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ENGINEERING DESIGN HANDBOOK

HELICOPTER ENGINEERING, PART TWO

DETAIL DESIGN

ARMY MATERIEL COMMAND, ALEXANDRIA, VIRGINIA

JANUARY, 1976

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HELICOPTER ENGINEERING

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HEADQUARTERS, US ARMY MATERIEL COMMAND **JANUARY 1976**

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DEPARTMENT OF THE ARMY
HEADQUARTERS US ARMY MATERIEL COMMAND
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ENGINEERING DESIGN HANDBOOK HELICOPTER ENGINEERING, PART TWO DETAIL DESIGN

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FOREWORD

The *Helicopter Engineering* Handbook forms a part of the Engineering Design Handbook Series which presents engineering data for the design and construction of Army equipment.

This volume, AMCP 706-202, *Detail Design*, is Part Two of a three-part Engineering Design Handbook titled *Helicopter Engineering*. Along with AMCP 706-201, *Preliminary Design*, and AMCP 706-203, *Qualification Assurance*, this part is intended to set forth explicit design standards for Army helicopters, to establish qualification requirements, and to provide technical guidance to helicopter designers, both in the industry and within the Army.

This volume, AMCP 706-202, deals with the evolution of the vehicle from an approved preliminary design configuration. As a result of this phase of the development, the design is described in sufficient detail to permit construction and qualification of the helicopter as being in compliance with all applicable requirements, including the approved system specification. Design requirements for all vehicle subsystems are included. The volume consists of 17 chapters and the organization is discussed in Chapter 1, the introduction to the volume.

AMCP 706-201 deals with the preliminary design of a helicopter. The characteristics of the vehicle and of the subsystems that must be considered are described. Design problems that may be encountered during the helicopter design are discussed and possible solutions are suggested. The documentation necessary to describe the preliminary design in sufficient detail to permit evaluation and approval by the procuring activity also is described.

The third volume of the handbook, AMCP 706-203, defines the requirements for airworthiness qualification of the helicopter and for demonstration of contract compliance. The test procedures used by the Army in the performance of those additional tests required by the Airworthiness Qualification Program to be performed by the Army also are described.

PREFACE

This volume, AMCP 706-202, *Detail Design*, is the second section of a three-part engineering handbook, *Helicopter Engineering*, in the Engineering Design Handbook series. It was prepared by Forge Aerospace, Inc., Washington, D.C., under subcontract to the Engineering Handbook Office, Duke University, Durham, NC.

The Engineering Design Handbooks fall into two basic categories, those approved for release and sale, and those classified for security reasons. The US Army Materiel Command policy is to release these Engineering Design Handbooks in accordance with current DOD Directive 7230.7, dated 18 September 1973. All unclassified Handbooks can be obtained from the National Technical Information Service (NTIS). Procedures for acquiring these Handbooks follow:

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Users of the handbook are encouraged to contact USAAVSCOM, St. Louis, MO, System Development and Qualification Division, with their recommendations and comments concerning the handbook. Comments should be specific and include recommended text changes and supporting rationale. DA Form 2028, Recommended Changes to Publications (available through normal publications supply channels) may be used for this purpose. A copy of the comments should be sent to:

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US Army Materiel Development
and Readiness Command
Alexandria, VA 22333

Revisions to the handbook will be made on an as-required basis and will be distributed on a normal basis through the Letterkenny Army Depot.

CHAPTER 1
INTRODUCTION

AMCP 706-202, Engineering Design Handbook, *Helicopter Engineering. Part Two, Detail Design*, is the second part of a three-volume helicopter engineering design handbook. The preliminary design (covered in AMCP 706-201) is developed during the proposal phase, at which time all subsystems must be defined in sufficient detail to determine aircraft configuration, weight, and performance. The detail design involves a reexamination of all subsystems in order to define each element thoroughly with the aims of optimizing the aircraft with regard to mission capability as well as cost considerations.

Detailed subsystem specification requirements are the basis for in-depth analysis and evaluation of subsystem characteristics and interfaces. Based upon complete system descriptions and layouts, performance, weight, and cost trade-offs are finalized. Periodic reviews of the design are conducted to evaluate maintainability, reliability, safety, producibility, and conformance with specification requirements.

Development testing may be required to permit evaluation of alternate solutions to design problems or to obtain adequate information for trade-off investigations. Appropriate consideration of human engineering factors often requires evaluation of informal mock-ups.

Weight control is an important element of the detail design phase. Subsystem weight budgets, prepared on the basis of the preliminary design group weight breakdown, are assigned at the initiation of the detail design phase. The continuing evaluation of compliance with the budget as an essential part of the management of the project and the assurance of compliance with weight guarantees of the helicopter detail specification are described in conjunction with the discussion of the Weight Engineering function in AMCP 706-201.

The requirements and procedures for airworthiness qualification and proof of contract compliance for a new model helicopter for the US Army are defined and discussed in AMCP 706-203, which is the third volume in this handbook series. Qualification is not time-phased, but is a continuing part of the acquisition program. A number of qualification requirements are integral parts of the detail design effort.

Design reviews by the procuring activity are required during the definition of subsystem configurations as well as during the final design of assemblies and installations. Evaluation of a full-scale mock-up of the complete helicopter is a major part of the design review process. The requirements for this review are described in detail in AMCP 706-203; but the construction and inspection of the mock-up must be completed at the earliest practical point of the detail design phase to permit the contractor to complete the design and manufacture of helicopters for test and for operational deployment, with reasonable assurance that the configuration is responsive to the mission requirements.

Also completed during the detail design phase are a variety of analyses necessary to substantiate the compliance of the physical, mechanical, and dynamic characteristics of subsystems and their key components with applicable design and performance requirements, including structural integrity. The analysis required during the design, development, and qualification of a given model helicopter are those specified by the applicable Contract Data Requirements List (CDRL).

This volume reviews the functions performed by the major helicopter subsystems and outlines the requirements applicable to the design and installation of each one. Principal documentation of the detail design phase is the final drawings of the helicopter in sufficient detail for procurement, fabrication, assembly, and installation. This volume, therefore, also includes discussion of materials and processes pertinent to the manufacture of helicopter components. This volume is intended to provide designers, engineers, relatively new to the helicopter technology, and program managers a general design guide covering all of the helicopter detail design specialties; however, it is not intended as a source of detailed design procedures for use by the experienced design engineer in his specialty.

Throughout this volume the mandatory design requirements have been identified with the contractual language which makes use of the word "*shall*". To assist in the use of the handbook in the planning or conduct of a helicopter development program, the word "*shall*" has been italicized in the statement of each such requirement.

Since the mission requirements for individual helicopters result in variations between subsystem configurations and performance requirements, the procuring activity will specify in its Request for Proposal (RFP) the extent to which the design requirements of this handbook are applicable to the acquisition of a given helicopter.

CHAPTER 2
MATERIALS

2-1 INTRODUCTION

This chapter addresses the properties of the various materials used in the construction of helicopters. These materials include ferrous and nonferrous metals, nonmetallic materials, composite structures, adhesives and sealants, paints and finishes, lubricants, greases, and hydraulic fluids. Among the ferrous metals are carbon steel, stainless steel, and alloy steels. The nonferrous metals include aluminum, magnesium and titanium alloys, beryllium, copper, brass, and bronze. Thermoplastic and thermosetting plastics, elastomers, woods, fabric, and fluoroplastics are reviewed. Composite structures, including filament laminates, fabric laminates, and filament wound, and honeycomb and sandwich construction are discussed as are the adhesives used for bonding of primary structure, honeycomb and composites, fabrics, rubber, elastomers, glass, and plastics. Sealing compounds, such as putty and paste, also are detailed. A discussion of paints and coatings, special finishes, plating, and tapes is included, as well as a review of the most commonly used lubricants and their applications.

The designer will find that a good working relationship with vendors will help him to keep abreast of new materials and processes with possible applications to helicopters. New materials are being introduced continually, and new processes alter the cost and performance relationships among older materials. Most of the materials are produced so as to conform to one or more Government specifications, and many manufacturers take steps to keep their products on the qualified-products lists, where such are required. Proper regard for and awareness of these concerns in design callouts will simplify procurement, fabrication, and qualification of hardware.

2-2 METALS

2-2.1 FERROUS METALS

2-2.1.1 General

This discussion provides a brief review of ferrous metals and their application to the construction of helicopters, as well as of some of the parameters governing the choice of a particular ferrous metal for specific use.

A more comprehensive discussion, as well as detail design data, will be found in Chapter 9, AMCP 706-100, which describes such items as material selection factors, standard mill products, and cost data. MIL-HDBK-5 is a source of mechanical properties data and provides additional detail design data. For technical data and information pertaining to wrought iron, carbon steels, and low-alloy steels, an excellent source is MIL-HDBK-723, which covers some of the more practical aspects of metal forming and joining. Finally, for design data and metallurgical details, the designer should consult the various American Society of Metals (ASM) Handbooks.

Because of weight considerations, it is desirable to restrict the heavier ferrous metals to those applications where very high strength, a high modulus of rigidity, high resistance to fatigue, and high modulus of elasticity are required.

The more expensive high-performance steels often are more economical in terms of weight, cost, and fabrication processes than are the lower-cost ferrous products. Applications for these materials include high-stress parts such as rotor drive shafts, masts, hubs, vertical hinges, flapping hinges, tie cables, tubular frames, and control cams, keys, gears, and hydraulic cylinders.

2-2.1.2 Carbon Steels

The carbon steels are a broad group of iron-base alloys having small amounts of carbon as their principal alloying element. Commonly, the carbon content falls between 0.03 and 1.2%. The American Iron and Steel Institute (AISI) code usually is used for designating steels. This system employs a four- or five-digit number to designate each alloy, with the first digits referring to the alloy and the last two digits giving the carbon content in points of carbon, where one point is equal to 0.01%. Thus, 1045 steel is one of a series of nonsulphurized carbon steels and has 0.45% carbon. Other carbon steel series are the 11XX series, which are resulphurized; the B11XX series, which are acid Bessemer resulphurized, and the 12XX series, which are rephosphorized. Low-carbon steels range from 0.05 to 0.30% carbon, medium-carbon steels range from 0.30 to 0.60% carbon, and high-carbon steels range from 0.60 to 0.95% carbon.

The machinability of low-carbon steels is poor. They tend to drag and smear and to build up on the cutting edges of tools, generating considerable heat and decreasing cutting efficiency. Medium-carbon steels machine better, although the cutting pressures are higher. High-carbon steels are too hard for good machining, but they are used where fine finish and dimensional accuracy are required. Hot- and cold-rolled steels machine better than do annealed steels, and the machining properties of the low-carbon steels are improved by the addition of sulphur, phosphorus, or lead.

The low-carbon steels have excellent forming properties, and can be worked readily by any of the normal shaping processes. Their ready formability is due to the fact that there is less carbon to interfere with the planes of slip. By the same token, the difficulty of working increases with increasing carbon content.

Plain carbon steel is the most readily welded of all materials. Low-carbon (0.15%) steel presents the least difficulty; as the carbon content increases to 0.30%, some martensite may form as a result of rapid cooling. If they are cooled too rapidly after welding, medium- and high-carbon steels may harden, but preheating to 600°F or post-heating to 1100°F will remove brittle microstructures.

The yield strength of low-carbon steels is on the order of 46,000 psi, while that of high-carbon steels is on the order of 150,000 psi. The modulus of elasticity in tension remains at 30 million for all of the carbon steels. Core (Brinell) hardness ranges from 149 for low-carbon to 400 for higher carbon.

2-2.1.3 Alloy Steels

Alloy steels are those that contain significant amounts of such alloying metals as manganese, molybdenum, chromium, or nickel, which are added in order to obtain higher mechanical properties with heat treatment, especially in thick sections. A family of extra-high-strength, quenched, and tempered alloy steels has come into wide use because these materials have yield strengths of more than 100,000 psi.

The alloy steels have relatively good resistance to fracture, or toughness. Weldability is good, and machinability and castability are fair. The alloy steels generally can be hardened to a greater depth than can unalloyed steels with the same carbon content. Many of the alloy steels are available with added sulphur or lead for improved machinability. However, resulphurized and leaded steels are not recommended for highly stressed aircraft parts because of drastic reductions in transverse properties. The alloy steels are somewhat more difficult to forge than are the corresponding plain carbon steels, and the maximum recommended forging temperatures are about 50% lower

Cold-forming, if performed, is done in the annealed condition because of the high strength and limited ductility of heat-treated materials. Notch toughness of alloys in the heat-treated condition is much better than that of the carbon steels. Corrosion resistance is about the same.

The AISI designation system is used for alloy steels also. This is illustrated by 4130 steel, which is an alloy steel containing chromium, molybdenum, and 0.30% carbon.

2-2.1.4 Stainless Steels

All stainless steels contain at least 10.5% chromium, from which excellent corrosion resistance is obtained. Apparently, a very thin, transparent, and tough film of oxide forms upon the chromium surface. This film is inert, or passive, and does not react upon exposure to corrosive materials. There are three broad types of stainless steels, as defined by the crystal structure: austenitic, ferritic, and martensitic.

Austenitic stainless steels, which have an austenitic structure at room temperature, are known as the 300 series (AISI). These materials have excellent ductility at very low temperatures, the highest corrosion resistance of all steels, and the highest scale resistance and strength at elevated temperatures. Austenitic steels are difficult to machine, but can be formed when care is given to the rate of work-hardening. They are not hardenable by heat treatment. Welding is done best in an inert atmosphere; because of the low thermal conductivity, care must be taken to avoid cracking. Carbide precipitation is minimized during welding by selecting one of the stabilized grades, e.g., 321 or 347.

Ferritic stainless steels are magnetic and have good ductility. Because of the low carbon-to-chromium ratio, the effects of thermal transformation are eliminated and the steels are not hardenable by heat treatment. They also do not work-harden to any great extent, are machined easily, and are formed readily. A general-purpose ferritic stainless is type 430.

Martensitic steels have a higher carbon-to-chromium ratio and are hardenable by heat treatment. They are characterized by good ductility, hardness, and ability to hold an edge. These steels are magnetic in all conditions, are tough and resistant to impact, and attain tensile strengths of up to 200,000 psi when hardened. Martensitic steels machine very well. Type 410 is the most widely used steel in this group.

2-2.1.5 Precipitation Hardening Steels

Precipitation hardening (PH) steels are those that harden at relatively low temperatures due to the precipitation of copper, aluminum, or titanium intermetallic compounds. They may be nonstainless or stainless. The best known is 17-4 PH, which is stainless by composition and is used for parts requiring high strength and good resistance to corrosion and oxidation at temperatures of up to 600°F. 17-4 PH is martensitic in nature, but other precipitation hardening steels may be austenitic. Forming properties are much the same as for stainless steels; forming must be accomplished before heat treatment, and allowance must be made for the dimensional changes that occur during the hardening process. Strength properties are lowered by exposure to temperatures above 975°F for longer than 0.5 hr. The heat-treating procedures are specified in MIL-H-6875.

2-2.1.6 Maraging Steels

The maraging steels are not treated in the references given in par. 2-2.1.1; hence they are discussed in somewhat greater detail here.

The term "maraging" is derived from the capability of the material for age hardening in the martensitic condition. The distinguishing features of the 18% nickel maraging steels are that they are designed to be martensitic upon cooling to room temperature after hot-working or annealing, and to be age-hardenable to ultra high strengths in that condition.

The 18% nickel maraging steels essentially are wrought alloys. The nominal yield strengths of four well-established grades are 200, 250, 300, and 350 ksi. The ability of these steels to transform into martensite upon cooling from elevated temperatures is imparted by their nickel content. The transformation, which begins at about 310°F and ends at about 210°F, is of the diffusionless or shearing type. The formation of martensite in these steels is not disturbed by varying the cooling rate within practicable limits. Hence, section size is not a factor in the process of martensite formation, and the concepts of hardenability that dominate the technology of the quenched and tempered steels are not applicable with the maraging steels.

The 18Ni maraging steels may be cut with a saw in the annealed or hot-worked condition. Alternatively, oxyacetylene and plasma arc torches may be used. Hot-rolled or annealed maraging steels can be sheared in much the same manner as can the quenched and tempered structural steels that have yield strengths in the vicinity of 110 ksi. In grinding, these steels behave in a manner similar to that of stainless steels, using a heavy-duty, water-soluble grinding fluid.

The maraging steels can be hot-worked to finished or semi-finished products by all of the standard methods of forming that are used for other steels. To avoid carburizing or sulfidizing, the metal should be free of oil, grease, and shop soil before heating. Fuel with low sulphur content is preferred. The metal can be press- or hammer-forged at temperatures ranging from 2300° down to 1500°F. Forging is completed at relatively low temperatures. The objective is to refine the grain structure, thereby enhancing the strength and toughness of the steel. A minimum reduction of 25% in thickness during the final forging cycle is recommended to produce optimum mechanical properties in the finished product. Hot bending, hot drawing, and hot spinning are accomplished at 1500°-1800°F.

Cold-forming operations are performed on the annealed material. Even in the annealed condition, the 18Ni maraging steels have yield strengths of up to 120 ksi, approximately four times those of deep-draw-body stock. The tensile elongations of these steels in the form of annealed sheet may be as little as 3-4%. These factors impose limitations upon forming the sheet metal by tensile stresses. On the other hand, these steels work-harden very slowly, making them well suited to forming methods dominated by shear. They can be cold-reduced by 80% or more, and shapes are formed readily by rolling or spinning. Flat-bottom cups can be deep-drawn to considerable depths. Rounded shapes are formed more readily by means of the flexible die process. Cold-rolled, solution-annealed material is preferred. Rolling and welding of sheet, strip, and plate are common methods of making cylindrical shapes.

In the annealed condition, the 18Ni maraging steels are machined quite easily. In the age-hardened condition, machining is difficult because of the hardness imparted by the aging process.

Although these steels have been welded by all of the common welding processes, the toughest welds are produced by the gas tungsten arc process or the electron beam (EB) process. For maximum toughness, the carbon, sulphur, silicon, phosphorus, and oxygen content must be kept at very low levels. It is good practice to avoid prolonged times at elevated temperatures, not to preheat, to keep interpass temperatures below about 250°F, to use minimum weld energy input, and to avoid conditions causing slow cooling rates.

Annealing is accomplished at 1500°F with air cooling. For improved combinations of strength and toughness, the steel may be double-annealed. The procedure is to heat the material to 1600°-1800°F, air cool to room temperature, reheat to 1400°-1500°F, and again air cool. Special furnace atmospheres are required in order to prevent carburization, sulfidation, or excessive oxidation.

Age-hardening is accomplished at 900°F, the time varying from 3 to 6 hr. Air is used commonly as the heat-treating atmosphere. It is advisable to maintain the temperature at all parts of the load to within ± 10° of the desired temperature.

The nominal mechanical properties of the age-hardened 18Ni maraging steels are listed in Table 2-1.

Additional data regarding high-strength 18Ni maraging steels will be found in Ref. 1.

2-2.2 NONFERROUS METALS

2-2.2.1 General

A brief review of nonferrous metals and their application to the construction of helicopters, as well as of some of the parameters governing the choice of one metal among many for a particular application, along with detail design data, is found in AMCP 706-100 and in MIL-HDBK-5.

Metals such as aluminum, magnesium, or titanium may be selected because of their relatively light weights. Other factors in material selection include corrosion resistance, thermal and electrical conductivity, lubricity, softness, cost and ease of fabrication, hardness, stiffness, and fatigue resistance. Usually, it is the sum of a number of factors that influences a designer to select the sequence of materials and fabrication processes that constitute a design item. This discussion is intended to provide data to assist the designer in formulating the required decision.

Comparative mechanical properties for representative nonferrous alloys are given in Table 2-2.

TABLE 2-1
MECHANICAL PROPERTIES OF 18Ni MARAGING STEELS

SERIES	350	300	250	200
ULTIMATE TENSILE STRENGTH, psi	365,000	294,000	260,000	210,000
0.2% YIELD STRENGTH, psi	355,000	290,000	255,000	206,000
ELONGATION, %	10.0	11.8	13.0	12.5
REDUCTION OF AREA, %	50.0	57.0	61.0	62.0
NOTCH TENSILE STRENGTH, (K_T=9.0), psi	330,000	420,000	390,000	325,000
CHARPY V-NOTCH, ft-lb	11.0	17.0	19.0	28.0
FATIGUE ENDURANCE LIMIT (10^8 CYCLES), psi	120,000	120,000	115,000	115,000
ROCKWELL "C" HARDNESS	57 60	52 55	47 51	41 45
COMPRESSIVE YIELD STRENGTH, psi	388,000	272,000	246,000	183,000

TABLE 2-2
COMPARATIVE MECHANICAL PROPERTIES FOR SELECTED NONFERROUS ALLOYS

PROPERTY	MAGNESIUM AZ91C-T4	ALUMINUM 2017	TITANIUM 4Al3Mo1V	COPPER CART. BRASS
YIELD STRENGTH, ksi	[illegible]	32	145	11
TENSILE STRENGTH, ksi	34	55	175	44
ELONGATION, %	7	12	5	66
MODULUS, 10^3 ksi	6.5	10.4	15.5	--
HARDNESS (Bhn)	55	45	200	--

2-2.2.2 Aluminum Alloys

MIL-HDBK-694 contains a comprehensive discussion of aluminum alloys, along with design data and a complete summary of standardization documents, including military, federal, and industry specifications. These specifications cover most of the uses of aluminum in detail and should be consulted before proceeding with design.

With few exceptions, aluminum alloys are designed either for casting or for use in wrought products, but not for both. Although some general-purpose alloys are available, compositions normally are formulated so as to satisfy specific requirements. The more widely used and readily available compositions are covered by Government specifications. Most are adaptable to a variety of applications.

The Aluminum Association has devised a four-digit system for wrought alloys in which the first number designates the major alloying element. Thus, 1 is pure aluminum, 2 is copper, 3 is manganese, 4 is silicon, 5 is magnesium, 6 is magnesium and silicon, and 7 is zinc. The last two digits are supposed to designate the aluminum purity, but the exceptions destroy the rule. However, the more frequently used alloys become familiar to the designer. The aluminum casting alloys usually are identified by arbitrarily selected commercial designations of two- and three-digit numbers.

Most aluminum alloys used for wrought products contain less than 7% of alloying elements. By regulation of the amounts and types of elements added, the properties of the aluminum can be enhanced and its working characteristics improved. Special compositions have been developed for particular fabrication processes, such as forging and extrusion. Wrought alloys are produced in both heat-treatable and nonheat-treatable types. The mechanical properties of the nonheat-treatable materials may be varied by strain-hardening or by a combination of strain-hardening and annealing.

The aluminum alloys specified for casting purposes contain one or more alloying elements; the maximum amount of any one element must not exceed 12%. Some alloys are designed for use in the as-cast condition; others are designed to be heat-treated in order to improve their mechanical properties and dimensional stability. High strength with good ductility can be obtained by selecting the appropriate composition and heat treatment.

The heat-treatment and temper designations for aluminum are long and complex. The designations most frequently stamped on products are: F-as fabricated; O-annealed; H-strain-hardened (many subdivisions); T2- (cast products only); T4-solution heat-treated and naturally aged, and T6-solution heat-treated and artificially aged. The heat treatment of aluminum alloys is detailed in MIL-H-6088. The processes commonly used are solution heat treatment, precipitation hardening, and annealing. A small amount of cold-working after solution heat treatment produces a substantial increase in yield strength, some increase in tensile strength, and some loss in ductility. Rapid quenching will provide maximum corrosion resistance, while a slower quench—used for heavy sections and large forgings—tends to minimize cracking and distortion.

Most forming of aluminum is done cold. The temperature chosen permits the completion of the fabrication without the necessity for any intermediate annealing. Hot-forming of aluminum usually is performed at temperatures of 300°-400°F, and heating periods are limited to 15-30 min. When nonheat-treatable alloys are to be formed, the temper should be just soft enough so as to permit the required bend radius or draw depth. When heat-treatable alloys are used, the shape should govern the alloy selected and its temper.

To a great extent, the choice of an alloy for casting is governed by the type of mold to be employed. In turn, the type of mold is determined by factors such as intricacy of design, size, cross section, tolerance, surface finish, and the number of castings to be produced. In all casting processes, alloys with high silicon content are useful in the production of parts with thin walls and intricate design.

The most easily machined aluminum alloy is 2011-T3, referred to as the free-cutting alloy. In general, alloys containing copper, zinc, and magnesium as the principal added constituents are machined the most readily. Wrought alloys that have been heat-treated have fair to good machining qualities.

The welding of many aluminum alloys is common practice because it is fast, easy, and relatively inexpensive. Welding is useful especially for making leak-proof joints in thick or thin metal, and the process can be employed with either cast or wrought aluminum or with a combination of both. The relatively low melting point, the high thermal conductivity, and the high thermal expansion pose problems. Preheating is necessary when welding heavy sections; otherwise, the mass of the parent metal will conduct the heat away too rapidly for effective welding. A rapid welding process is preferred in order to minimize distortion due to expansion and contraction. Molten aluminum bsorbs hydrogen easily, and this may cause porosity during cooling. Because they provide a protective inert-gas shield, TIG and MIG welding are common choices. TIG is an inert-gas shield-arc process with a tungsten electrode, and MIG is an inert-gas, shielded-metal-arc process using covered electrodes. A suitable flux, and mechanical (stainless steel brush) removal of the oxide film just prior to welding, are mandatory. Certain aluminum alloys — 2014, 7075, etc. — are extremely difficult to fusion weld (excluding spot welding) and normally would not be used in structural applications when welded. Brazing is somewhat more difficult, and soldering of aluminum is extremely difficult. The other joining processes include riveting and adhesive bonding, both of which are used extensively in aircraft structures.

Applications for aluminum in helicopters include the sheet-metal exterior surface of the fuselage, framing, stringers, beams, tubing, and other usages where the density, corrosion resistance, and ease of fabrication of aluminum give it an advantage over steel and where its higher strength and modulus properties give it an advantage over magnesium.

2-2.2.3 Magnesium Alloys

MIL-HDBK-693 provides a comprehensive discussion of magnesium alloys and their properties, and also describes design, fabrication, and performance data. Numerous Military and Federal Specifications covering specific shapes, forms, and processes also are summarized.

The outstanding characteristic of magnesium is its light weight. This is important in helicopter design, where payload ratio is a direct function of vehicle weight. Magnesium is two-thirds as heavy as aluminum and one-fourth as heavy as steel. The low density is most effective in relatively thick castings, where the increased rigidity of magnesium is an additional benefit. For this reason magnesium is used frequently in main rotor gearboxes, motor transmission casings, and many other load-bearing applications in helicopters. Most of the helicopter power systems have several hundred pounds of magnesium in their construction.

The ASTM (American Society for Testing Materials) nomenclature system is used exclusively in designating magnesium alloys. In this sytem, the first two letters indicate the principal alloy elements, while the numbers indicate the respective percentages. Thus, [illegible] aluminum, F rare earth, H thorium, K zirconium, L lithium, M manganese, Q silver, T tin, and Z zinc. By this designation, AZ91C is identified as an alloy of magnesium containing 9%

aluminum, 1% zinc, and having a "C" variation.

The heat-treat and temper designations for magnesium virtually are identical to those for aluminum. The temper designations used are those in ASTM B296.

There are four groups of magnesium casting alloys. The Mg-A and Mg-Z binary systems are designed for use at temperatures below 300°F and are of lower cost. The Mg-F and the Mg-H binary systems are designed for good strength in the 500°-800°F range. The choice of casting composition is dictated largely by certain features of the design, and by cost and method of production. For magnesium alloys, the important casting processes are sand, permanent mold, and die. The choice of a casting process depends upon the size, shape, and minimum section thickness of the part; and upon the tolerances, types of surface finish, number of pieces to be produced, and relative cost of finishing the part.

Magnesium alloys, both cast and wrought, have outstanding machinability. Greater depths of cut and higher cutting rates can be used with these metals than with other structural metals. Magnesium does not drag or tear, and it is not necesary to grind and polish the material in order to obtain an extremely fine finish. The chips from machining readily clear the work and the tools.

Because of its position in the electromotive series, magnesium is subject to more corrosion than are the other structural metals. The many corrosion problems associated with the use of magnesium severely limit its use in rotary aircraft. Magnesium alloys *shall not* be used for parts that are not readily accessible for inspection, application of protective finish, and replacement.

Magnesium cannot be welded satisfactorily to other metals, and welding of magnesium to magnesium can be accomplished reliably only by a skilled operator. The metal also cannot be soldered properly. Thus, electron beam (EB) welding is the most satisfactory welding process, although flux dip brazing also may be used; care must be employed in removing all of the flux because of the danger of corrosion. The best method of joining magnesium in thin sections is by adhesive bonding.

2-2.2.4 Titanium Alloys

MIL-HDBK-697 contains a comprehensive description of titanium alloys and their properties, and discusses design, fabrication, and performance. In addition, seven Military Specifications for specific forms of titanium will be found in Refs. 2 and 3.

Although titanium is relatively costly, its high strength-to-weight ratio, excellent corrosion resistance, and capability of performing at considerably higher temperatures than aluminum often give it advantages for particular applications, as in the hot structures and exhaust ducting for helicopter power systems. Indeed, increased payloads resulting from weight savings can more than offset the initial costs, and in the long run titanium may prove less costly for specific applications than lower-priced materials.

Titanium is available from the mills in various wrought shapes and in a wide range of alloyed and unalloyed grades including billet, bar, extrusions, plate, sheet, and tubing. The mill products can be grouped into three categories according to the predominant phase in their microstructure: Alpha, Alpha-Beta, and Beta titanium. There is no single accepted system for the designation or classification of titanium and its alloys as there are for other metals.

Titanium actually is easier to machine than the stainless steels because the effects of work-hardening are far less pronounced. Titanium requires low shearing forces, and is not notch-sensitive. Because of these properties, it can be machined to extremely low micro-inch finishes. On the other hand, the sharp angle of curl of the chip and the high friction cause the shearing point to heat rapidly. At elevated temperatures, titanium tends to dissolve anything within contact, and the cutting tool is dulled readily. Further, the carbides and oxides on the forged pieces are extremely abrasive to tools and must be removed by nitric-hydrofluoric acid treatment prior to machining. Overall, considerable knowhow is required for the economical machining of titanium.

Titanium assemblies are joined by spot, seam, flash, and pressure welding techniques. In fusion welding, the TIG process is used; heavy welding also requires inert gas on the bottom of the weld. EB welding is quite satisfactory.

2-2.2.5 Copper and Copper Alloys

A comprehensive discussion of copper and copper alloys and their properties, design and fabrication characteristics, and design and performance data is contained in MIL-HDBK-698.

The various forms of copper and copper alloys have found only limited use in helicopters. Their thermal and electrical conductivity properties are advantageous in inserts, studs, bushings, etc., where low load lubricity is desired. Beryllium copper is useful for springs and other applications where its good modulus, hardness, fatigue resistance, and ease of forming are advantageous. However, copper alloys are the heaviest of the common structural metals, and, therefore, have a weight disadvantage in airborne applications.

The various types of copper and its alloys are better known by name than by code number. The term copper is used when the material exceeds 99.4% purity. The principal alloying agent of brass is zinc, while tin is the principal alloying agent in bronze. The beryllium coppers have small percentages of beryllium, producing a remarkably hard, high-modulus, high-strength, nonsparking material.

The copper and copper alloys are cast readily in all of the various casting processes. The alloys are cold-formed easily, and are capable of being rolled, drawn, spun, and flanged. In hot-working they are rolled, extruded, pierced, and forged. The machinability of copper alloys is excellent. For sand castings, low speeds and coarse feeds are used for removing the scale in order to increase tool life. It is better to remove the scale by sand blasting and pickling.

The copper alloys are welded readily by all the welding processes, although their high thermal conductivity is a problem. They are adaptable to brazing and are the easiest of all metals to solder.

2-2.3 ELECTROLYTIC ACTION OF DISSIMILAR METALS

Dissimilar metals, as defined in MIL-STD-889, should not be used together in helicopter applications unless the mating surfaces are insulated adequately. When tape is used between two dissimilar metals, such as in the mounting of a magnesium gearbox to an aluminum airframe, the contractor must insure that there will be no loss of mounting torque as a result of normal usage and vibrations.

Metals can be grouped in four categories, as shown in Table 2-3.

Metals grouped in any one of the categories in Table 2-3 can be considered similar to one another, while those metals placed in different groups should be considered dissimilar to one another. The categorization does not apply to fasteners — such as rivets, bolts, nuts, and washers — that are component parts of assemblies and usually are painted prior to being used. Instead, the metals referred to are surface metals. For example, zinc covers all zinc parts, including castings and zinc-coated parts.

TABLE 2-3
GROUPING OF METALS AND ALLOYS
(MIL-STD-889)

GROUP I	MAGNESIUM AND ITS ALLOYS ALUMINUM ALLOYS 5052, 5056, 5356, 6061, AND 6063
GROUP II	CADMIUM, ZINC, AND ALUMINUM AND THEIR ALLOYS (INCLUDING THE ALUMINUM ALLOYS IN GROUP I)
GROUP III	IRON, LEAD, AND TIN AND THEIR ALLOYS (EXCEPT STAINLESS STEELS)
GROUP IV	COPPER, CHROMIUM, NICKEL, SILVER, GOLD, PLATINUM, TITANIUM, COBALT, AND RHODIUM AND THEIR ALLOYS; STAINLESS STEELS; AND GRAPHITE

Par. 2-2.2.2 details the use of aluminum alloys in helicopter construction. Aluminum alloys used in helicopters may contain copper or zinc as an essential constituent. In sheet form, these alloys are susceptible to corrosive action resulting in a loss of strength of the material, which becomes brittle without evidence of surface change. Aluminum alloys containing magnesium, magnesium and silicon, and chromium as the essential alloying constituents are much more stable under prolonged weathering conditions than are the aluminum alloys containing copper or zinc.

Cadmium behaves similarly to zinc as a coating metal in affording electrochemical protection of ferrous metals against corrosion. Cadmium plating thus can be used to put ferrous metals into the same group as aluminum alloys, giving them a similarity. A detailed discussion of coating processes can be found in par. 2-6. In general, when two dissimilar metal surfaces come in contact with one another, a corrosive action called galvanic action can take place. Coating metals are used as thin layers between dissimilar metals to prevent this type of corrosion.

Table 2-4 illustrates the position of metals with regard to their susceptibility or lack of susceptibility to galvanic action.

2-3 NONMETALLIC MATERIALS

2-3.1 GENERAL

This paragraph discusses the applications of the thermoplastic and thermosetting plastics, elastomers, fabrics, and transparent materials. Other materials — such as glass in light bulbs or optical piping, ceramics and mica in electrical insulation, and carbon and graphite in lubrication or electrical contacts — also play significant roles in helicopter construction.

The nonmetallic materials used in composite structures, reinforced plastics, and other composite materials are treated in par. 2-4; plastic materials used as sealants and adhesives are covered in par. 2-5. Comprehensive discussions and detailed design data will be found in existing documents. Among these are: MIL-HDBK-700, MIL-HDBK-17, the *Modern Plastics Encyclopedia*, published annually by McGraw-Hill, and the *Materials Selector Issue*, published annually by Reinhold Publishing Co.

The major disadvantage of plastics is their low modulus, which is in the order of a few hundred thousand psi compared to 10 or more million psi for metals. They also are more sensitive to heat, softening markedly at 400°F and below. On the other hand, plastics can be as strong as steel, be lighter than magnesium, and have better abrasion resistance than metals. Normal corrosion is not a problem. Although they are nonconductors for electricity and poor conductors for heat, they can be exceedingly tough and wear-resistant, and can be fabricated in a variety of ways. When judiciously selected and properly used, they often can perform better at lower cost than any other material.

TABLE 2-4
POSITION OF METALS IN THE GALVANIC SERIES

CORRODED END (ANODIC, OR LEAST NOBLE)

MAGNESIUM
MAGNESIUM ALLOY
ZINC
ALUMINUM 1100
CADMIUM
ALUMINUM 2017
STEEL OR IRON
CAST IRON
LEAD-TIN SOLDERS
LEAD
TIN
BRASS
COPPER
BRONZE
COPPER-NICKEL ALLOYS
TITANIUM
MONEL
SILVER SOLDER
NICKEL
INCONEL
CHROMIUM-IRON
18-8 STAINLESS
18-8-3 STAINLESS
SILVER
GRAPHITE
GOLD
PLATINUM

PROTECTED END (CATHODIC, OR MOST NOBLE)

SOURCE:
REFERENCE DATA FOR RADIO ENGINEERS
FEDERAL TELEPHONE & RADIO CO. 3RD EDITION

2-3.2 THERMOPLASTIC MATERIALS

Thermoplastic materials are those that soften when heated and harden when cooled. Typical of the thermoplastic family are the polyvinyls, acrylics, nylons, polycarbonates, and fluorocarbons. Often, these have linear micromolecular structures. Products of these materials usually are formed by extrusion or by injection molding, and they are available for manufacturing in the form of rods, tubes, contoured shapes, plate, sheet, and film and in a wide range of shapes, sizes, and thicknesses. The materials are machined, or shaped readily by thermoforming processes. Many items may be purchased in the finished form as produced by extrusion or injection molding; included are screws, nuts, bolts, inserts, grommets, straps, pins, knobs, handles, instrument facings, housings, boxes, conduits, electrical receptacles, covers, rails, runners, guides, snaps, and slides. Many of these items are supplied as off-the-shelf inventories in a variety of sizes.

Nylons and polycarbonates are known as the engineering plastics. Nylon, a polyimide, has high strength and high elongation, giving it a toughness that many applications depend upon. It has high modulus in flexure, good impact strength, a low coefficient of friction, and high abrasion resistance, as well as good fatigue resistance under vibration conditions. Its primary disadvantages, though not significant, are dimensional change with moisture absorption, and the need for incorporating carbon black in order to protect against ultraviolet degradation in outdoor use. Nylon is used in gears, arms and other contact applications, and in pressure tubing, belting, and wear pads.

The polycarbonates are aromatic esters of carbonic acid. They have excellent rigidity and toughness, high impact strength, and low water absorption. They are stable dimensionally under a wide range of conditions, are creep-resistant, and are transparent and stable in sunlight. Probably their major deficiency is that their fatigue resistance is lower than is desirable. Polycarbonates are used in shields, lenses, ammunition chutes, knobs, handles, etc.

The acrylic of interest here is polymethylmethacrylate, better known as Plexiglas. This plastic has crystal clarity, outstanding weatherability in optical properties and appearance, dimensional stability, good impact resistance, and a low water absorption rate. Its major deficiency is its low resistance to scratching. Its major use is as window glazing and for such applications as transparent aircraft covers; covers for signal lights, where its ease of coloring is advantageous; and in other optical and instrumentation applications. Its use as a window material is discussed in par. 2-3.5.

For helicopters, the polyvinyls are used largely in the form of sheeting simulating leather or upholstery fabric. These are very tough and wear-resistant. In the transparent form, they are used to make pockets and holders for documents and maps.

The fluorocarbon polymers have excellent thermal stability at continuous temperatures of 400°-550°F.

They virtually are inert to chemical attack, have excellent damping properties, and have outstanding electrical characteristics, such as high dielectric strength, low dissipation factor and radio frequency (RF) transparency. They are used widely in microwave components and high-frequency connectors, as well as in wire coatings, gaskets, and electrical terminals.

2-3.3 THERMOSETTING MATERIALS

Although there is a great diversity in the chemical makeup of thermosetting resins, they have one characteristic in common: once they are cross-linked, they do not soften under heat and cannot be formed by thermoforming processes. With the application of heat, thermosetting resins undergo a series of changes that are irreversible. The polymerization reaction that occurs results in such a high degree of cross-linking that the cured product essentially is one molecule. In many cases, this results in a highly rigid molecule of good thermal stability. The thermosetting resins usually are used with fillers and reinforcement.

Three of the most widely used of these materials are the epoxy, phenolic, and polyester resins. These are employed extensively with Fiberglas fabric, with chopped fiber in laminates, in sprayed forms, in filament-wound structures, in honeycomb sandwich structures, and in combinations with balsa wood or formed shapes.

The epoxy resins are based upon the reactivity of the epoxide group and generally are produced from bisphenol-A and epichlorohydrin. Epoxies have a broad capability for blending properties through resin systems, fillers, and additives. Formulations can be soft and flexible or rigid and tough. They are available as prepolymers for final polymerization in the form of powders and liquids with a wide range of viscosities. Some cure at room temperature, while others require curing at elevated temperatures. The powders may be transfer-molded by machine, and the liquids may be cast. More often, the liquid is used to impregnate materials for bonding.

The outstanding characteristic of epoxies is their capability to form a strong bond with almost any surface. For this reason, they are used widely in adhesive formulations. The molded products have high dimensional stability over a wide range of temperatures and humidities, excellent mechanical and shock resistance, good retention of properties at 500°F, and excellent electrical properties.

The phenolics are the oldest and the least expensive of the thermosetting plastics. The basic resin is manufactured by means of a reaction between phenol and formaldehyde. This resin is blended with dye, filler, and curing agents to make the molding powder, which is called the "A" stage powder. Powders such as these are molded for 2 min at 325°F at 1500 psi pressure. As the granules are warmed by the hot mold, the resin melts; the material flows and fills the cavity, further reacting and going through a rubbery "B" stage. With further cross-linking it reaches the "C" stage, at which it is hard and infusible.

Fillers used in typical phenolic molding powders are fibrous in nature; their interlocking fibers act to reduce the brittleness of the cured resin. Wood flour is used most commonly, while asbestos and graphite fibers form the conventional heat-resistant plastics. Paper and fabric fillers are used for high impact or shock-resistant phenolics. When the powders are used for lamination or in making composite structures, a solution of the resin in alcohol is used to impregnate the fabric and then "B" staged. The layers of impregnated fabric are laid together and then cured by heat and pressure.

The advantages of phenolics are their low material processing costs, dimensional stability, excellent load-bearing properties, excellent electrical characteristics, and good weathering properties. They are used in electrical components, receptacles, conduits, housings, etc.

The polyesters are plastics formed of chains produced by repeating units of a polyacid and a polyglycol. They may be aliphatic or aromatic. Familiar forms — fibers such as Dacron synthetic fiber or Mylar film — are the so-called linear polyesters. More important to applications in helicopters are the thermosetting resins. These are the three-dimensional or cross-linked polyesters that are formed by bridging unsaturated polyesters. In this form, the polyester is supplied as a syrupy liquid that — when mixed with a small amount of curing agent, applied to a fabric, chopped Fiberglas, or filament tow, and laid over a form — rapidly reacts so as to establish a rigid structure. Such structures are of particular use for radomes because of their RF transmission and excellent weatherability. They have high modulus and impact strength as well as excellent flexural and tensile properties. The polyesters may be cast in order to produce glazing materials.

Another highly cross-linked family of polymers consists of the urethanes. These are formed by the reaction of isocyanates with esters and unsaturates. In the process, carbon dioxide is evolved and forms a highly porous structure. The stiffness ranges from soft, flexible foams to highly rigid foams. The flexible foams are employed for cushioning and padding and to reduce noise and shock, as well as for thermal insulation. The rigid foams are used as light-weight stiffeners in structures.

2-3.4 ELASTOMERIC MATERIALS

MIL-HDBK-149 presents a comprehensive discussion of the technology of the elastomeric materials and their applications. From the standpoint of durability and performance, natural rubber remains in demand, and substantial quantities are used in blends with SBR, butyl, and other synthetic rubbers. Natural rubber is a sterospecific polymer of isoprene.

Its applications in pneumatic tires, bumpers, shock absorbers, etc., as well as in belting, gaskets, and seals, are well known. Substantial quantities continue to be employed in helicopters. Carbon black constitutes about 50% of the weight of these compositions.

A more advanced synthetic rubber is neoprene, a general-purpose synthetic made by emulsion polymerization of chloroprene. A notable characteristic of this rubber is its resistance to gasoline, fuels, lubricating oils, and other solvents, and its excellent resistance to weather-oxidations, ozone, and ultraviolet light. It has good tensile strength, tear resistance, abrasion resistance, and rebound characteristics, and excellent adherence to metal and fabrics. It provides average insulation and has excellent dielectric strength. In helicopters, it is used to coat radomes and the leading edges of the rotors for protection against abrasion by rain and dust. It also is used in boots on other leading edges and areas where wear is a factor; and in transmission belts, hoses, tires, seals, and electrical applications.

Another important family of elastomeric materials is the silicones, which are used in many diverse and seemingly unrelated applications. The silicones are organo-poly-siloxanes, having alternating silicon and oxygen atoms in the backbone of the chain. The silicone resins may be cast, extruded, or injection-molded so as to form shaped products. They are available in sheet or bulk form; as a range of pastes and liquids for use as adhesives, sealants, and coatings; and as powders for foaming. They are stable continuously at temperatures from −140° to +600°F, and intermittently to 700°F. They are weather-resistant, have high dielectric strength and a low dissipation factor, and are bonded easily to metals, ceramics, and plastics substrates. Aromatic solvents and chlorocompounds swell silicones, and they have higher gas permeability than do other rubbers. They are used as foaming agents, encapsulating resins, sealants, and in electrical applications.

2-3.5 WINDOW MATERIALS

Glazing materials and methods of attachment are discussed in detail in MIL-HDBK-17. That document also lists additional Military Specifications covering specific glazing materials, resins, cements, and processes for the design and fabrication of window systems.

The optical properties of greatest significance for aircraft glazing are surface reflection, index of refraction, absorption of light, and transmission of an undistorted image. The thermal properties of primary concern are the coefficient of expansion, the thermal conductivity, and the distortion temperature. The major physical properties are density and hardness, or scratch-resistance; the major mechanical properties are tensile and compressive strength and the modulus of elasticity. The ideal glazing will be strong enough to withstand structural and operational (wind and water) loads, hard enough to remain unscratched, optically clear after a life of operation, unchanged by thermal loads, and unaffected by the weather.

Although no materials possess all of these desirable characteristics, there are several that perform very well. The three glazing materials that are employed most often are glass, cast polyester, and cast acrylic (methylmethacrylate). Polycarbonate, an otherwise strong contender, has not yet been produced economically in large sheets with the required optical properties.

Monolithic glass is used in helicopters only when use temperatures exceed the performance temperatures of the laminated glasses and the polymeric materials. The lamination of glass with plastic improves the resistance to thermal and mechanical stresses, and minimizes the possibility of complete failure of a panel. Splintering of the glass is prevented, although the load-carrying capacity of the laminated glass is less than that of plate glass. The plastic interlayer is selected so as to provide the greatest ability to absorb impact energy. Polyvinyl butyral is the most common interlayer material for both glass and plastic laminates.

A new thermosetting, polyester-base, transparent sheet material has been developed under the trade name "Sierracin 880". It can be used for aircraft enclosures that operate at surface temperatures of up to 300°F, and is characterized by its two-stage cure. After forming and post-curing, the ultimate physical properties of this material are obtained. Sierracin 880 generally is used as a laminate with acrylic, and is described in MIL-P-8257.

The glazing material used most widely for helicopters is cast polymethylmethacrylate. In many aircraft, it constitutes a major portion of the fuselage walls. For window applications, the stretched, modified acrylic sheet is preferred, per MIL-P-25690. The modified material has slightly higher heat resistance than does heat-resistant polymethylmethacrylate, along with better resistance to crazing and solvents. When stretched to 60-100% biaxially or multiaxially, acrylic sheets develop increased resistance to crazing, higher impact strength, and improved resistance to crack propagation — without detrimental effects upon their other properties except for reduced abrasion resistance and laminar tensile and shear strengths. The sheets may be formed thermally to different contours. Laminated plastic glazing materials are made by bonding two or more layers of acrylic or polyester plastic sheet to a soft plastic interlayer by means of an adhesive. This process greatly improves the impact and structural strengths of the material. Laminated plastic glazing materials are defined in MIL-P-25374.

Differing thermal expansion rates of glazing materials, edge attachment materials, and metal airframes present one of the major problems in the design of window-glazing. For all types of glazing, an edge wrap is used in order to minimize the propagation of stresses originating in cracks and chips at the edges. The preferred wrap is two or more layers of polyester (Dacron) fabric, woven from twisted yarns and impregnated and bonded in place with epoxy resin. The wrap overlaps sufficiently on the glaze material and extends sufficiently beyond the edges to absorb the stresses of attachment closure and at the same time to distribute the load uniformly across the window. There are many closure designs, but the preferred enclosure will be designed so as to hold the glaze securely in a sliding grip in such a manner as to allow for reciprocal longitudinal motion — as the glaze expands and contracts — while always applying a comprehensive load endwise. This may be achieved by placing a compressible, neoprene-impregnated tube at the bottom of the closure channel. The closure will be attached rigidly to the airframe. The contacting areas between the closure and the edge wrap will be sealed with a flexible sealant, preferably one made of silicone.

2-4 COMPOSITE STRUCTURES

2-4.1 FIBERGLAS LAMINATES

Of all the fibers available for the reinforcement of plastics, glass is used by far the most widely. Of the various glass compositions, only two are important in aircraft construction: "E" and "S" glass. "E" glass is used extensively; "S" glass provides greater tensile strength and modulus, but is considerably more expensive.

The major advantages of glass-reinforced plastic (GRP) over isotropic structural materials (primary metals) include:

1. Formability and versatility. Large complex parts and very short production runs are practical. Because there are few limitations on size, shape, and number of parts, design freedom is maximized. In addition, the reinforcement can be oriented as desired in order to increase properties in specific directions.

2. Chemical stability. GRP is resistant to most chemicals, and does not rust or corrode.

3. Toughness. Good impact resistance is a feature of GRP.

4. Strength-to-weight ratio. Specific strength of GRP is very high. For example, unidirectional GRP has a specific strength about five times that of the commonly used steel and aluminum alloys.

5. Insulation. GRP is a good thermal and electrical insulator, and therefore, will transmit radar and radio waves.

6. Repairability. Minor damage can be patched readily and effectively.

On the other hand, GRP has certain disadvantages compared to other construction materials, namely:

1. Nonuniformity. Variations in material properties within a part and from part to part are inherent in most of the fabrication techniques.

2. Low modulus. Stiffness of GRP is relatively low.

3. Slow fabrication. Production rates are low in comparison with most metal-forming operations.

Thus, GRP construction is most advantageous for parts with complex shapes that would be difficult to form from metal, for parts with anisotropic strength requirements, or in applications where the conductivity or poor dent or corrosion resistance of metals present a problem.

Some typical GRP applications in helicopter construction are in canopies, covers, and shrouds (for formability, specific strength, dent resistance); rotor blades (for formability, anisotropic strength, and stiffness); control surfaces (for anisotropic properties, dent resistance, repairability); and antenna housings (for radio frequency transparency, formability). It is conceivable that an entire helicopter airframe can be constructed from GRP, as has been done on several small, fixed-wing aircraft.

2-4.1.1 Design Considerations

Design rules and procedures for reinforced plastics do not differ markedly from those for metals. Stress strain curves, however, are similar to those for wood (which also is a fiber-reinforced composite) in that there is no yield point.

As Part I of MIL-HDBK-17 contains considerable property data on specific materials, only generalities are considered here.

There are two essential ingredients in glass-reinforced plastics: glass fibers and resin. A finish, or coupling agent, that enhances adhesion between the glass and resin usually is used as a coating on the glass, and may be considered as a third component or as part of the reinforcement. The resin system generally has the limiting role in determining the chemical, thermal, and electrical properties of a laminate, while the type, amount, and orientation of the reinforcement predominate in determining the basic mechanical properties.

2-4.1.2 Resin Systems

Essentially all GRP laminates are made with thermosetting resins that, when mixed with suitable catalysts or curing agents, are permanently converted to the solid state. Reinforced thermoplastics (RTP), which contain short glass fibers, are a rapidly growing element of the injection-molding industry; such parts, however, are not considered laminates. Probably 95% of all GRP laminates are made from polyester, epoxy, or phenolic resins. For very-high-temperature service (above 500°F), silicone (MIL-R-25506 and MIL-P-25518) and polyimide resins are available. These, however, have no known applications in current helicopter technology.

2-4.1.2.1 Polyesters

These are by far the most widely used resins when the entire GRP industry is considered. They are low in cost, easily processed, and extremely versatile. Available types range from rigid to flexible; there are also grades that are fire-retardant, ultraviolet-resistant, and highly chemical-resistant. The upper temperature limit for long-term operation of general-purpose grades is 200°F, although temperature-resistant resins are available that are useful up to 500°F. These can be formulated for rapid curing at room temperature or with long pot-life for curing at elevated temperature. Thus, they commonly are used for wet layups but prepregs also are used frequently. Prepregs that cure by ultraviolet light also are available.

Disadvantages of polyesters include high shrinkage during cure, inherently tacky surface if cured in the presence of air, odor, and fire hazard in wet layup fabrication from styrene monomer and peroxide catalysts.

Requirements for general-purpose polyester laminating resins and laminates are contained in MIL-R-7575 and L-P-383, respectively; fire-resistant resins and laminates are detailed in MIL-R-25042 and MIL-P-25395.

2-4.1.2.2 Epoxies

Epoxies probably are the resins most frequently used for aircraft GRP laminates. Although about twice as costly as polyesters or phenolics, epoxy resins still are inexpensive for most applications. Mechanical, electrical, and chemical-resistant properties are excellent. Adhesion to most substrates is very good, and cure shrinkage and moisture absorption are low. Temperature resistance of general-purpose types is intermediate between that of polyesters and phenolics. Formulation and fabrication possibilities are extremely versatile. As with polyesters, epoxies can be formulated for uses such as wet layups or prepregs for room-temperature or elevated-temperature curing, and for fire-retardancy. The choice of curing agent plays a major part in determining curing characteristics, temperature resistance, chemical resistance, flexibility, etc. In addition, a variety of modifiers and fillers is available to provide specific qualities.

There are relatively few disadvantages with epoxy resins. However, because amine curing agents that are commonly used in room temperature curing formulations may cause severe dermatitis, skin contact must be avoided. MIL-R-9300 contains requirements for epoxy laminating resins, while requirements for epoxy laminates are covered in MIL-P-25421.

2-4.1.2.3 Phenolics

Phenolic resins are used primarily in GRP applications where an inexpensive material with heat resistance up to 500°F and/or nonflammability is required. Excellent electrical properties also are obtained. Because water is produced and released in the curing reaction, relatively high molding pressure is required in order to prevent porosity in phenolic laminates. Prepregs nearly always are used.

MIL-R-9299 and MIL-P-25515 cover the requirements for phenolic laminating resins and phenolic laminates, respectively.

2-4.1.3 Types of Reinforcement

Glass reinforcement is available in several basic forms, and in a wide variety of specific constructions within these basic categories. Those forms commonly used in GRP laminates include woven fabric, chopped fiber mat, and nonwoven continuous tapes or roving. Nearly all of these are derived from continuous filaments of 0.00023, 0.00028, or 0.00038 in nominal diameter. Numerous standard yarn constructions are available, with varying numbers of parallel filaments per strand, strands per yard, and twists per inch of the strands. Likewise, there is a multiplicity of fabrics woven from these yarns that vary not only in type and amount of yarns but also in the type of weave. MIL-Y-1140 is an excellent reference for definitions and requirements for the various yarns and woven fabrics.

The type of reinforcement selected will depend upon the mechanical property requirements, part shape, and applicable fabrication technique as discussed in the paragraphs that follow.

2-4.1.3.1 Nonwoven Continuous Filaments

This form of reinforcement offers maximum mechanical properties, but has minimum fabrication possibilities due to the difficulty of placing and aligning the reinforcement in complex shapes. A big advantage where this type of construction is practicable is that the fibers can be oriented in proportion to the stress in any given direction. Filament winding is the most widely used fabrication technique with nonwoven continuous filaments. This construction is covered more completely in par. 2-4.3.

2-4.1.3.2 Woven Fabric

This construction provides good mechanical properties and formability and is, therefore, the most commonly used reinforcement in aircraft fabrication. When wetted with resin, the cloth has considerable ability to stretch and conform to rather complex contours. Although intended specifically for polyester laminates, MIL-C-9084 fabric usually is specified for laminates made with all resins. Requirements for eleven basic fabrics and six subtypes are defined in the specification. Approximate thickness per ply ranges from 0.003 in. for 112 fabric to 0.027 in. for 184 fabric. (Still heavier fabrics, woven from rovings rather than yarns, are available in thicknesses of up to 0.045 in. per ply; these are covered in MIL-C-19663.) Most of these fabrics are balanced weaves, with nominally equal construction in the warp and fill directions; 181 fabric at 0.009 in. per ply is the standard balanced fabric upon which most test laminates and published properties data are based. Representing the extreme of unbalance is 143 fabric, which has a warp strength about 10 times as great as its fill strength. This approaches the nonwoven construction described previously, sacrificing some mechanical properties for improved drapability. Most high-strength, glass-fabric-based laminates are made from 181 and/or 143 fabrics.

As with MIL-C-9084, MIL-Y-1140 originally was intended for polyester laminates. However, its requirements also usually are specified for fabrics containing epoxy-compatible finishes.

2-4.1.3.3 Chopped Fiber

The third common form of reinforcement is chopped fiber mat, as defined by MIL-M-15617. Because the fibers are short and their orientation is completely random, this material is very conformable. For the same reasons, and also because of its high bulk — which limits the percentage of glass obtainable in a laminate — mechanical properties are lower than with roving or fabric. Continuous (swirl) strand mat is another variation and is particularly useful for deep contours. In both types of mat, the glass is held in place with a small amount of resin binder. Both types are available in weights ranging from 0.75 oz to 3 oz per ft^2, corresponding to laminated thicknesses of about 0.013 in. to 0.050 in. per ply. Mat reinforcement also is produced preimpregnated with resin; this is called sheet molding compound (SMC).

Chopped fiber parts can be fabricated by the spray-up or preform techniques described subsequently.

2-4.1.4 Fabrication Methods

As previously discussed, each of the common forms of glass reinforcement (roving, fabric, mat) can be purchased either dry or preimpregnated with the laminating resin, which is cured or dried partially to a solid or tacky condition. The latter, called prepregs, are advantageous in that they contain a controlled, uniform, and readily measurable amount of resin. They are, therefore, easier to lay up, because wet layup operations often are messy and odorous. Prepregs can be obtained with varying degrees of tack so as to suit the specific operation. And, because complete quality control tests can be made before the part is fabricated, the problems of incorrect weighing and mixing of the resin system are eliminated completely. In order to obtain reasonable shelf life, prepregs are formulated with curing agents or catalysts that require heat to cure (generally 250°-350°F for at least 1 hr). Under heat, the resin melts initially, and then converts chemically to a thermoset solid. Some pressure almost always is required in prepreg laminating in order to maintain good contact between plies of reinforcement. This pressure results in greater resin flow and, consequently, in higher glass ratios and better mechanical properties than are obtained with unpressurized wet-layup laminates.

Generally, epoxy resins and prepregs of roving, tape, or fabric are associated with components of higher quality, cost, and strength, while polyester resins and wet-layup (or SMC) processing of fabric or mat are used where maximum required properties do not justify the increased costs. Those fabrication methods applicable to construction of laminated

TABLE 2-5
PROCESS COMPARISON GUIDE FOR GRP LAMINATES*

PROCESS		RESIN	FIBERGLAS	NORMAL FIBERGLAS BY WEIGHT	MOLDING TEMP, °F	MOLDING PRESSURE, psi	SIZE OF PRODUCTS TO DATE
OPEN MOLD HAND LAYUP	CONTACT MOLDING	POLYESTER EPOXY	MAT FABRIC WOVEN ROV	30 45 40	70 TO 110	0	↑
	VACUUM BAG	POLYESTER EPOXY	MAT FABRIC WOVEN ROV	40 50	70 TO 110	12 TO 14	SMALL PROTOTYPES TO 1-PIECE BOAT HULLS, 80 ft LONG
	PRESSURE BAG	POLYESTER EPOXY	MAT FABRIC WOVEN ROV	45 60 55	70 TO 220	50	↓
	AUTOCLAVE	POLYESTER EPOXY	MAT FABRIC WOVEN ROV	45 60 55	70 TO 250	50 TO 100	ONLY LIMITED BY SIZE OF AUTOCLAVE
	SPRAYUP	POLYESTER EPOXY	CONTINUOUS ROVING	30	70 TO 110	0	BOATS TO 30 ft, ONLY LIMITED BY SIZE OF MOLD
CLOSED MOLD MATCHED-DIE MOLDING	PREFORM	POLYESTER EPOXY	CHOPPED	30	225 TO 300	100 TO 300	FROM SAFETY HELMETS TO 17 ft BOAT HULLS
	MAT	POLYESTER, PHENOLIC, MELAMINE, SILICONE, EPOXY	MAT	30	225 TO 350	100 TO 3000	FROM SMALL TRAYS TO 4×8 ft PANELS
	FABRIC	POLYESTER, PHENOLIC, MELAMINE, SILICONE, EPOXY	FABRICS, PREPREG WOVEN AND NONWOVEN	60	225 TO 350	100 TO 3000	PANELS UP TO 5 in. THICK, 4×8 ft

* FROM OWENS CORNING FIBERGLAS CORPORATION TECHNICAL BULLETIN 1-PL-1998-B

GRP helicopter components are discussed subsequently. General guides to molding processes, and resulting laminate properties, are shown in Tables 2-5 and 2-6, respectively.

2-4.1.4.1 Open Mold Hand Layup

This method consists simply of placing the required number of plies of reinforcement and resin over a single mold surface, and rubbing or rolling out the air. Curing then is accomplished by one of the following processes:

1. Contact molding. The laminate is allowed to cure without the application of pressure, usually at room temperature. Heat can be applied to accelerate the cure, but the contact process usually is employed for large parts and/or short production runs, and both heat and pressure may be impracticable. A strippable film, such as cellophane, sometimes is smoothed onto the exposed surface of the layup in order to provide a better finish.

2. Vacuum bag. A film (usually polyvinyl alcohol or nylon) is placed over the surface of the part and sealed at the edges, or the entire mold is placed in a bag. A vacuum is then drawn, resulting in the application of atmospheric pressure to the laminate. Even this relatively low pressure (15 psi) considerably improves the laminate quality by reducing entrapped air and resin-rich areas.

3. Pressure bag. In this case a rubber film (often contoured to the part shape) is placed over the layup and the mold is sealed with a pressure plate. Air or steam pressure of up to about 100 psi then is applied to the cavity.

4. Autoclave. In this variation of the pressure bag process, the entire assembly (mold, layup, rubber film) is placed in a steam autoclave and cured, normally at about 50-100 psi.

TABLE 2-6
GENERAL PROPERTIES OBTAINABLE IN GLASS REINFORCED PLASTICS*

	POLYESTER		EPOXY		PHENOLIC
PROPERTY	GLASS MAT, PREFORM, OR SHEET MOLDING COMPOUND	GLASS CLOTH	GLASS MAT	GLASS CLOTH	GLASS MATT OR CLOTH
SPECIFIC GRAVITY	1.35 TO 2.30	1.50 TO 2.10	1.8 TO 2.0	1.9 TO 2.0	1.70 TO 1.95
GLASS CONTENT, % BY WEIGHT	25 TO 45	60 TO 67	40 TO 50	65 TO 70	45 TO 55
TENSILE STRENGTH, psi	15,000 TO 25,000	30,000 TO 70,000	14,000 TO 30,000	20,000 TO 60,000	4,000 TO 60,000
COMPRESSIVE STRENGTH, psi	15,000 TO 50,000	25,000 TO 50,000	30,000 TO 38,000	50,000 TO 70,000	17,000 TO 40,000
FLEXURAL STRENGTH, psi	25,000 TO 40,000	40,000 TO 90,000	20,000 TO 26,000	70,000 TO 100,000	10,000 TO 95,000
IMPACT STRENGTH, IZOD 0.5X0.5 in. NOTCHED BAR (ft-lb/in. OF NOTCH)	8 TO 20	5 TO 30	8 TO 15	11 TO 26	8 TO 35
WATER ABSORPTION, 24 hr, 1/8 in. THICKNESS, %	0.01 TO 1.00	0.05 TO 0.50	0.05 TO 0.95	0.08 TO 0.70	0.1 TO 0.9
HEAT RESISTANCE CONTINUOUS, °F	300 TO 350	300 TO 350	330 TO 500	330 TO 500	350 TO 500
BURNING RATE	← SLOW-TO-SELF-EXTINGUISHING →				NONE
VOL RESISTIVITY AT 50% RH AND 73° F, ohms/cm³	10^{14}	10^{14}	3.8×10^{15}	3.8×10^{15}	7×10^{12}
ARC RESISTANCE, sec	120 TO 180	60 TO 120	125 TO 140	100 TO 110	20 TO 150

*COMPILED BY REINFORCED PLASTICS/COMPOSITES DIV, SOCIETY OF THE PLASTIC INDUSTRY, INC

2-4.1.4.2 Sprayup

In this method, continuous roving is chopped into 1- to 2-in. lengths and blown into a spraying stream of resin and catalyst that is directed against the mold. (A fast room-temperature-setting polyester generally is used.) The mixture is hand rolled to reduce air and level the surface. The resulting part is similar in construction to a chopped fiber mat hand layup. Although this process is very efficient for large components, it has limited application in aircraft construction due to poor uniformity of thickness and relatively low strength-to-weight ratios.

2-4.1.4.3 Matched Die Molding

Whenever closely controlled thicknesses are required, two mold halves are necessary. Matched die molding also is practical for high-volume production even where high-quality surfaces and close tolerances are not required. Pressures of up to 300 psi and temperatures to 350°F commonly are used. Prepreg fabrics and tapes usually are specified for aircraft applications requiring maximum strength-to-weight ratios. However, for complex shapes and volume production, chopped glass preforms (held together, like mat, by a small amount of resin binder) frequently are used.

2-4.1.5 Surface Finishes

For many applications, a smooth surface, free from air pockets and exposed glass fibers, is required. Such a surface may be specified in order to improve outdoor weathering characteristics, material-handling capabilities, human contact applications, or, simply, appearance. There are three different methods used to obtain a smooth, resin-rich laminate surface:

1. Veil mats. These consist of loose, nonwoven mats of glass or synthetic fibers. Thickness may range

from 0.001 to 0.030 in. They are so loosely constructed that resin content in the veil area is about 85% by weight.

2. Gel coats. This technique consists of spray-coating the mold surface with 0.010- to 0.020-in. layer of thixotropic (nonsagging) resin, which is allowed to set prior to laying up the glass reinforcement. Resilient resins usually are used so as to provide a compromise between scratch resistance and impact strength. Most gel coats are polyesters, but the method also can be used with epoxies.

3. Thermoplastic films. This method consists of laminating a film or sheet of weather-resistant and/or decorative plastic, such as polyvinyl fluoride or acrylic, to the GRP surface. This technique should be applicable to a variety of GRP processing methods with both polyester and epoxy resins, but it has not been used widely in the past. Recently, however, a process involving vacuum-forming of thermoplastic sheets — which then are reinforced by spraying the back side with chopped glass and polyester resin — has found wide acceptance, especially in the manufacture of large parts (up to 300 ft²).

2-4.2 FABRIC LAMINATES

Industrial laminates, also called high-pressure laminates, are reinforced plastics that are manufactured in standard, simple shapes such as sheets, rods, and tubes. Fabrication of parts from these materials generally is accomplished by standard metalworking operations, such as cutting, drilling, punching, and machining, in contrast to molding to the desired shape as discussed in par. 2-4.1. Where production quantities warrant mold costs, parts also may be custom-molded by procedures similar to those described in par. 2-4.1 for low pressure, closed-die laminates.

Industrial laminates are used for components of simple geometry requiring intermediate strength, lightweight, and nonmetallic characteristics. In helicopter construction, they frequently are used for wear surfaces, such as on conduits and pulleys for control cables and in electrical circuit boards.

Industrial laminates can be made with a number of mechanical, chemical, thermal, and electrical properties by varying the type and ratio of resins and reinforcements. Those combinations that presently are available commercially are shown in Table 2-7. In each case, the laminates are manufactured by stacking up sheets of the impregnated reinforcement (or by wrapping, in the case of tubes or rods) and curing them under heat and pressure. Very high pressure — ranging from about 200 to 2500 psi — are used, resulting in high-quality, void-free parts.

From the basic combinations shown in Table 2-7 more than 70 standard grades of laminates are derived. Of these, 32 grades are classified by the National Electrical Manufacturers' Association (NEMA). Descriptions of the NEMA grades and their applications are contained in Vol. 46, *Modern Plastics Encyclopedia*, as also are the properties of these laminates. The designer should consider grades, application, and producibility prior to final component design.

General characteristics resulting from the selection of the various reinforcements and resins are described subsequently.

TABLE 2-7

COMMON RESIN REINFORCEMENT COMBINATIONS OF THERMOSET LAMINATES

RESIN TYPE	PHENOL FORMALDEHYDE			MELAMINE FORMALDEHYDE			POLYESTER			EPOXY			SILICONE		
REINFORCEMENT	SHEET	TUBE	ROD	SHEET	TUBE	ROD	SHEET	TUBE	ROD	SHEET	TUBE	ROD	SHEET	TUBE	ROD
CELLULOSE PAPER															
COTTON FABRIC															
ASBESTOS PAPER															
ASBESTOS FABRIC															
NYLON FABRIC															
GLASS PAPER AND MAT															
GLASS FABRIC															

*FROM MODREN PLASTICS ENCYCLOPEDIA, [illegible]

NOTE:

MATERIALS HAVING [illegible] REQUIREMENTS

2-4.2.1 Reinforcement Selection

The common reinforcements used in high-pressure laminates are paper, cotton, nylon, glass, and asbestos. Attributes of these materials are:

1. Paper. The least expensive, and adequate for many purposes. Kraft paper has relatively long fibers and is the strongest type. Alpha cellulose offers improved electrical properties, machinability, and uniformity, while rag paper laminates have the lowest water absorption and intermediate strength.
2. Cotton. Better impact and compressive strengths than paper, and most grades are only slightly more costly than paper. Electrical characteristics, however, generally are not as good. The heavier fabrics have the best mechanical properties, while the fine weaves have good machinability.
3. Nylon. Low moisture absorption and excellent impact strength and electrical properties, as well as good resistance to chemicals and abrasion. However, nylon laminates have relatively poor creep resistance at elevated temperatures and are comparatively expensive.
4. Glass. Highest mechanical strengths by far. These materials also have superior electrical properties and heat resistance. Cost is relatively high.
5. Asbestos. Used in the form of paper, mat, and fabric. These laminates have excellent heat, flame, chemical, and abrasion resistance.

Costs range from low for those with a paper base to high for a fabric base. The designer should select the type of material best suited for the application, considering interface requirements and reliability, maintainability, producibility, and survivability.

2-4.2.2 Resin Selection

The resins used in the manufacture of industrial laminates include phenolic, epoxy, polyester, silicone, and melamine. The characteristics of each are:

1. Phenolic. The most widely used by far. These resins are inexpensive and have adequate properties (mechanical, electrical, thermal, and chemical) for many design applications.
2. Epoxy. Used especially where high resistance to chemicals and moisture is required. Mechanical properties and dimensional stability also are superior.
3. Polyester. Less common, but used for mechanical and electrical applications, especially where flame resistance is a requirement.
4. Silicone. Used primarily with glass fabric where heat resistance to 500°F is required. Arc resistance is excellent and moisture absorption is low. The very low dissipation factor of these resins at high frequencies is utilized in radar and radio insulators. Room temperature mechanical properties are comparatively low for glass laminates, and cost is high.
5. Melamine. Excellent arc resistance at moderate cost. Mechanical properties, and heat, flame, and chemical resistance qualities, also are good.

2-4.2.3 Special Types

In addition to the materials listed previously, there are two special types of industrial laminates that deserve mention:

1. Postforming grades. Made from resins that, although thermoset, will soften enough at elevated temperatures to allow the material to be molded into intricate shapes. Special paper or fabric reinforcement also is used, permitting considerable stretching.
2. Clad laminates. Clad, on one or both surfaces, with a variety of materials, including aluminum, copper, stainless steel, silver, magnesium, and various rubbers. The copper-clad laminates (generally glass/epoxy) are used widely as printed circuit boards.

2-4.2.4 Specifications

In addition to the NEMA Standards (Pub. No. LI 1-1965), the following Military and Federal Specifications are applicable to fabric laminates:

1. L-P-509, for sheets, rods, and tubes of various resins and reinforcements
2. MIL-P-79, for rods and tubes of paper/phenolic, cotton/phenolic, and glass/melamine
3. L-P-513, for paper/phenolic sheet
4. MIL-P-15035, for cotton/phenolic sheet
5. MIL-P-18324, for cotton/phenolic sheet for water- or grease-lubricated bearings
6. MIL-P-15037, for glass/melamine sheet
7. MIL-P-15047, for nylon/phenolic sheet.

2-4.3 FILAMENT COMPOSITION

This paragraph is concerned primarily with high-performance composites, consisting of plastics reinforced with nonwoven filaments of glass, boron, and high-modulus graphite. Because the fibers are nonwoven and, usually, untwisted, they can be packed to high fiber loadings. The fibers can be oriented along the axes of stress in proportion to design requirements, allowing efficient utilization of the outstanding properties of this type of reinforcement.

When the specific strength (tensile strength-to-density ratio) and specific modulus (Young's modulus-to-density ratio) of the metal alloys (aluminum, steel, titanium) commonly used in aircraft construction are compared, it is shown that they are nearly equal to 7 to 9 × 10^6 in. and 100 to 110 × 10^6 in., respectively.

Although S-glass offers a substantial improvement in both quantities, more important is the comparatively recent introduction of the exotic fibers, with boron and graphite being of primary commercial interest. Unidirectional composites made from these fibers have specific moduli in the 600 to 500×10^6 in. range. The specific strength of boron composites is comparable to that of glass, while graphite composites are somewhat lower in this property.

2-4.3.1 Types of Reinforcement

A summary of the properties of the previously mentioned filaments that are used in reinforced composites is contained in Table 2-8. The derivations and characteristics of these fibers are discussed subsequently.

TABLE 2-8
TYPICAL VALUES OF PHYSICAL AND MECHANICAL CHARACTERISTICS OF REINFORCEMENT FIBERS

MATERIAL	DENSITY, lb/in³	TENSILE STRENGTH, 10³ psi	SPECIFIC STRENGTH, 10⁶ in.	TENSILE MODULUS, 10⁶ psi	SPECIFIC MODULUS, 10⁶ in.
E-GLASS	0.092	500	5.4	10.5	114
S-GLASS	0.090	665	7.4	12.5	146
970-S-GLASS	0.092	850	9.2	15.3	156
BORON (ON TUNGSTEN)	0.093	500	5.4	60.0	650
GRAPHITE (RAYON)	0.061	300	4.9	50.0	820
GRAPHITE (PAN, TYPE I)	0.072	300	4.2	55.0	765
GRAPHITE (PAN, TYPE II)	0.063	400	6.3	40.0	635

2-4.3.1.1 E-glass

This glass was developed originally for its superior electrical properties.

Glass roving is manufactured by drawing the molten glass through resistance-heated platinum bushings at about 2400°F. From 51 to 408 (usually 204) filaments are gathered into a single strand, coated with a binder, and wound onto a drum at approximately 10,000 fpm. The coating bonds the filaments into a strand, protects them from abrading each other, and also serves as a coupling agent to improve the resin-glass bond. For use with epoxy resins, an 801 sizing usually is specified. Requirements for E-glass roving are contained in MIL-R-60346 under the Type I classification.

Standard continuous roving uses ECG 135 strands — where E designates the glass composition, C indicates continuous filaments, and G designates a filament diameter of 0.00037 in. — resulting in 13,500 yd of strand per lb. ECG 67.5 (408 G filaments per strand) and ECK 37 (408 K filaments of 0.00052 in. diameter per strand) rovings also are available. A roving package is made by winding a number of strands (or ends) under each tension onto a cylinder. The number of ends ranges from 8 to 120, with 60 being the most common quantity. Standard packages range from 7 to 35 lb nominal weight.

E-glass rovings are available widely, both dry and preimpregnated with a variety of resin systems. Prepreg tapes of unidirectional filaments up to 48 in. wide, having a nominal cured thickness of either 0.0075 in. or 0.010 in., also are available. These can also be purchased in two-ply bidirectional (0 deg, 90 deg) or three-ply isotropic (−60 deg, 0 deg, + 60 deg) forms.

2-4.3.1.2 S-glass

This composition, sometimes called S(994), was developed under Air Force contract for its high-strength properties. S-glass is available in the same forms (roving, tape, and prepeg) as is E-glass. The standard roving designation in this case is SCG 150, indicating that there are 15,000 yd of strand per lb due to the lower specific gravity of S-glass. The major deterrent to its wider use has been its cost, which is about 15 times that of E-glass. A commercial grade, S-2, containing most of the S-glass properties at somewhat lower cost, has been introduced.

Another development is 970 S-glass, which has 20% greater modulus and ultimate strength than S-glass.

The chemical compositions of various glass reinforcements are presented in Table 2-9. HTS-901 and HTS-904 are the epoxy-compatible sizings for S-glass, while 470 sizing is used with S-2 rovings. S-glass roving requirements also are contained in MIL-R-60346 under the Type III classification.

2-4.3.1.3 Boron Filaments

These products currently are made by vapor deposition of boron on very fine tungsten wire. Work is under way to develop boron filaments on glass or graphite substrates in order to reduce cost and total density substantially. In order to make handling practicable, the material usually is supplied in collimated prepreg tapes that are one filament thick and up to 3 in. wide. A Military Specification on boron filament prepreg is MIL-B-83369.

2-4.3.1.4 Graphite

A wide variety of filamentary carbon products is produced by pyrolysis of organic fibers. These products may be divided into two broad categories: low-modulus and high-modulus materials. Low modulus carbon and graphite are used frequently in

TABLE 2-9
NOMINAL COMPOSITION OF GLASS REINFORCEMENTS

TYPE	% BY WEIGHT					
	SiO_2	Al_2O_3	MgO	BeO	CaO	B_2O_3
E-GLASS	54.3	15.2	4.7	--	17.3	8.0
S AND S-2-GLASS	64.3	24.8	10.3	--	--	--
970-S-GLASS	62.0	19.0	9.6	9.4	--	--

woven form, which is produced directly from rayon fabric at a fraction of the cost of high-modulus graphite. However, these products, used primarily for high-temperature insulation and ablation, have no known applications in helicopter construction.

High-modulus graphite fibers are produced in a three- or four-step heating process. During the final step, graphitization, the fibers are held in tension, thereby imparting a high degree of orientation to the graphite crystals. Material developed in the U S uses rayon fibers and has an irregular (popcorn shape) cross section. Material developed in England is pyrolyzed from a polyacrylonitrile (PAN) precursor having a circular cross section. In either case, the average filament diameter is 0.0003 in.

The British PAN material is made in untwisted tows of 10,000 filaments, and is available in continuous lengths. The rayon-derived, high-modulus graphite used in the U S is made in continuous lengths from 2-ply yarns having 720 filaments per ply and 1.5 or 4 twists per in., depending upon the manufacturer. The greatest development activity in high-performance fibers is focused upon graphite. Due to the small filaments, it can be formed around radii as small as 0.05 in., a major advantage over boron fiber. It also is expected that the greatest potential for cost reduction and product improvement lies with graphite. Evidence of both was displayed recently in the commercial announcement of a 75 × 10^6 psi modulus fiber at $400 per lb and a 30 × 10^6 psi modulus fiber at $60 per lb. Laboratory quantities of 100 × 10^6 psi modulus fiber have been produced. Because new products and new manufacturers frequently enter the field, the data in Table 2-8 include only those products with which a significant amount of experience exists.

Graphite fiber can be produced in the same variety of forms as can glass fiber. Thus, in addition to yarn and tow, fabric, mat, and chopped fiber can be supplied. As with boron, however, the most practicable form for most applications is unidirectional prepreg tape. A Military Specification on high-modulus graphite fiber prepreg is MIL-G-83410.

Among the most serious disadvantages of graphite composites are poor abrasion and impact resistance. Thus, surface protection frequently is necessary. Also impeding the exploitation of this material until very recently has been its low interlaminar shear strength due to poor resin-fiber bonding. However, surface treatments have been developed that result in shear strengths above 10,000 psi.

2-4.3.2 Resins

While all of the resins discussed in par. 2-4.1 have been used in filament winding, epoxies are used almost exclusively for aircraft applications at normal operating temperatures. Where nonwoven, high-performance reinforcement is used, the best available resin system also should be chosen since the difference in resin cost represents a very small percentage of the total part cost. Phenolic and polyimide resins are used only where very-high-temperature operation is specified.

2-4.3.3 Manufacturing Processes

Structures of nonwoven reinforced plastics may be formed by filament winding, tape wrapping, automatic tape layup, or hand layup. Filament winding can be performed with glass rovings, graphite yarns, and boron single filaments. This process is practicable for cylinders and tanks with high hoop stresses; however, it is limited to hollow structures with convex surfaces. Normally, filament winding is accomplished by rotating the part on its axis as on a lathe. Parts also have been wound by revolving the spool of reinforcement around the fixed part. Generally, prepreg is used, but wet winding also is practiced. In the latter case, the reinforcement travels through a bath of high-viscosity (at ambient temperature) resin system that is heated in order to lower the viscosity for efficient wetting of the reinforcement. When the impregnated reinforcement is cooled to ambient temperature, the high viscosity is reattained. Latent curing agents must be used in order to obtain a reasonable pot-life for the heated resin system.

Tape winding is similar to filament winding, except that prepreg tapes of closely collimated reinforcement (generally 1/8 in. wide) are wound. A recent advance in fabrication technology is a numerically controlled tape-laying machine capable of applying prepreg tape (heated, if desired) at a controlled rate and pressure, and shearing it at the desired length and angle. Still another machine applies reinforcement in three dimensions by weaving fibers perpendicular to the normal laminate. This, of course, greatly increases interlaminar properties, which usually are limited to the capabilities of the resin.

Hand layup is still the most widely used method where winding is not practicable. This process is no different from conventional layup of glass mat and fabric, except that the fibers are nonwoven and oriented, and generally are preimpregnated with resin.

Filament-wound parts usually are cured under wrapping tension pressure only, although they may be autoclaved or vacuum-bagged. Parts that are laid up (rather than wound) may be cured by an appropriate method as described in par. 2-4.1, such as pressure bag, vacuum bag, autoclave, or matched die molding.

2-4.3.4 Applications

Nonwoven, oriented filament composites are in order wherever maximum strength and/or stiffness-to-weight ratios in specific directions are desired. They are not usually practicable where isotropy is required. Typical properties of these components are shown in Table 2-10. It is entirely possible to mix the various reinforcements in such proportions and orientations as are required in order to obtain almost any intermediate properties. The possible effects of different thermal expansion coefficients must be considered, however.

Both E- and S-glass filaments have been used in construction of the spar envelope, skins, trailing edge, etc., of rotor blades. Design studies have suggested the use of boron and high-modulus graphite in these same areas, as well as in rotor hubs, swashplates, drive scissors, transmission housings, drive shafts, airframe stiffeners, and entire fuselage sections. Boron hardware development presently is more advanced than that of graphite because the material was introduced earlier. However, graphite composites are expected to be useful in many of the same applications.

Considerable design and physical property information is contained in MIL-HDBK-17, Part I, and in Ref. 4.

2-4.4 HONEYCOMB AND SANDWICH CONSTRUCTION

Sandwich construction, as shown in Fig. 2-1 is a composite structure comprising a combination of alternating, dissimilar, simple or composite materials, assembled and fixed in relation to each other so as to obtain a specific structural advantage. They are made of three or more laminations of widely dissimilar materials that can be considered homogeneous when bonded together. The layers include the facings, the bonding agent, and the core. The primary functions of the core are (1) to separate the outer layers so as to obtain a high bending stiffness, (2) to support these outer layers (the facings) in order to prevent elastic instability when they are highly stressed, and (3) to carry shear loads.

TABLE 2-10
TYPICAL UNIDIRECTIONAL COMPOSITE PROPERTIES BASED ON COMMERCIAL PREPREGS*

		THICKNESS PLY, in.	FIBER CONTENT, % VOL	TENSILE STRENGTH, ksi	TENSILE MODULE, 10^6 psi	FLEXURAL STRENGTH, ksi	FLEXURAL MODULE, 10^6 psi	COMPRESSIVE STRENGTH, ksi	BEAM SHEAR, ksi	DENSITY, lb/in.^{3}	SPECIFIC STRENGTH, 10^6 in.	SPECIFIC MODULUS, 10^6 in.
E-GLASS		0.0075	61.3	185	7.6	200.0	7.0	99	8.00	0.0729	2.54	104
S-GLASS		0.0075	63.5	220	8.7	230.0	7.9	120	9.00	0.0722	3.04	120
BORON		0.0052	50.0	186-232	29.6-32.0	245.0	28.0	443-460	17.00	0.0765	2.45-3.05	390-420
HIGH MODULUS GRAPHITE	HMG-50	0.0082	57.0	120	25.5	116.0	25.0	90	7.55	0.0541	2.22	471
	THORNEL-50	0.0077	53.0	104	25.0	116.0	24.1	—	7.40	0.0538	1.93	465
	MORGANITE-I	0.0130	48.0	—	—	110.0	—	—	8.27	0.0604	—	—
	MORGANITE-II	0.0130	52.5	—	—	163.5	—	—	12.96	0.0552	—	—

*FROM 3M COMPANY'S SCOTCHPLY TECHNICAL DATA SHEETS FOR EPOXY PREPREGS; 1009-26 RESIN ON GLASS AND RESIN ON BORON AND GRAPHITE.

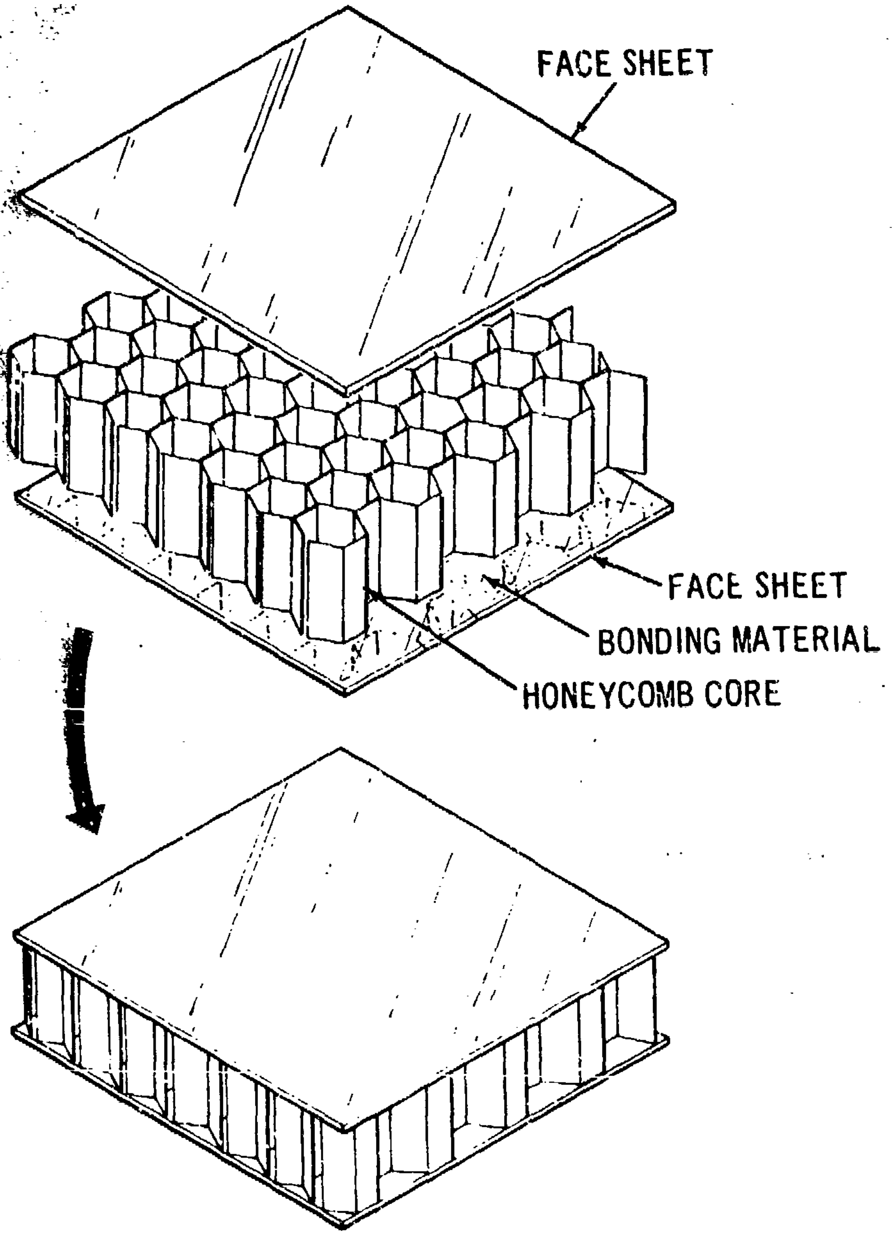


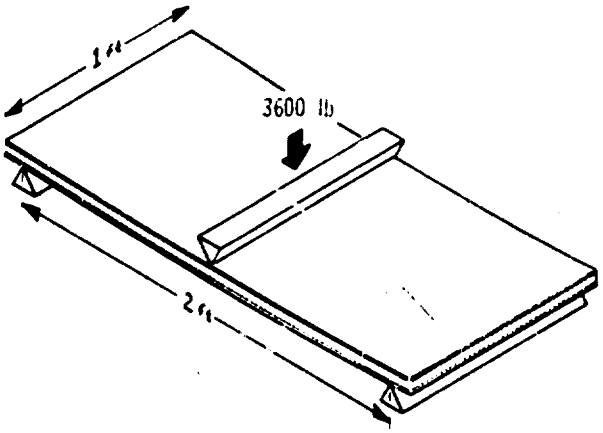
Figure 2-1. Sandwich Structure

Typical applications of sandwich construction in the airframe industry include access doors, bulkheads and partitions, jet engine shrouds, control surfaces, spoilers, and helicopter rotor blades.

Properly designed sandwich construction has many advantages; high strength-to-weight and stiffness-to-weight ratios are the most predominant. Secondary advantages include fatigue resistance, impact resistance, and aerodynamic efficiency.

A comparison of minimum-weight design for various structural systems and materials on an axially loaded cylinder proves the superiority of sandwich construction over monocoque and waffle construction. For a practical range of loading, waffle construction is approximately five times heavier and monocoque at least 10 times heavier (Ref. 5).

Honeycomb sandwich is the lightest possible material that can be used to achieve an optimum stiffness-to-weight ratio. A comparison of various materials (Fig. 2-2), based upon an equivalent deflection, suggests a 30% weight advantage when compared with a nested I-beam construction and an 80%



MATERIAL	DEFLECTION, in.	WEIGHT, lb
HONEYCOMB SANDWICH	0.058	7.79
NESTED "I" BEAMS	0.058	10.86
STEEL ANGLES	0.058	25.90
MAGNESIUM PLATE	0.058	26.00
ALUMINUM PLATE	0.058	34.20
STEEL PLATE	0.058	68.60
GLASS REINFORCED PLASTIC LAMINATE	0.058	83.40

Figure 2-2. Weight Comparison of Materials for Equal Deflection

advantage when compared with a flat aluminum plate.

Optimum fatigue resistance is a byproduct of sandwich application. The increase in flexural and shear rigidities of the construction, at no increase in mass, provides for an increase in the fundamental modes of excitation to higher octaves. In addition, the attachment of the core at the facing provides visco-elastic damping that prevents amplification of resonance. It is quite possible to design a sandwich structure for an infinite life under cycling loads, provided that the maximum loading is no more than 35% of the ultimate capability of the construction. A comparison of conventional sandwich structures (Fig. 2-3) in a sonic environment indicates that the sandwich can operate in excess of 500 hr at approximately 160 dB, (decibel) while skin-stiffened structure of the same weight will fail at less than 200 hr under 130 dB.

Aerodynamic efficiency of a sandwich structure is a consequence of the continuous, uniform support of the core materials. This characteristic is accepted widely in both aerospace and aircraft applications. Vertical support of the material in the sandwich construction is limited in span to the cell size, which is, in most cases, no more than 3/16 in.

Among the materials available for sandwich application are conventional honeycomb, foams, and balsa wood. Advantages of balsa wood and foams are limited, and their use usually is due essentially to a limited physical characteristic requirement rather than to an overall property consideration. Balsa wood is used predominantly in flooring applications, where the need for continuous support is provided by the fibers. Employment of foam cores in a sandwich construction is, essentially, a cost consideration. Both balsa wood and foam may produce adverse effects, and also may limit the environmental capabilities of the construction.

Honeycomb core construction represents by far the most efficient utilization of parent material. Conventional honeycomb cores, as illustrated in Fig. 2-4, are essentially hexagonal in shape and are manufactured from almost any material that can be made into a foil thickness. Properties of honeycomb cores can be predicted accurately, based upon the configuration and the parent material properties. The merits of one type over another are related to the properties of the foil material; the relative increase in efficiency is related directly to the increase in the property of the core.

Honeycomb core material can be made from metals — such as aluminum, stainless steel, and titanium — or from Fiberglas impregnated with resins such as nylon-phenolic and polyimide. Other types of core material include those made from Kraft paper and Dupont's Nomex* nylon-fiber-treated materials.

Fiberglas core material provides radar transparency and acts as a dielectric. It has low dielectric constants and a low loss tangent. Kraft paper core material is available in many varieties, and is used

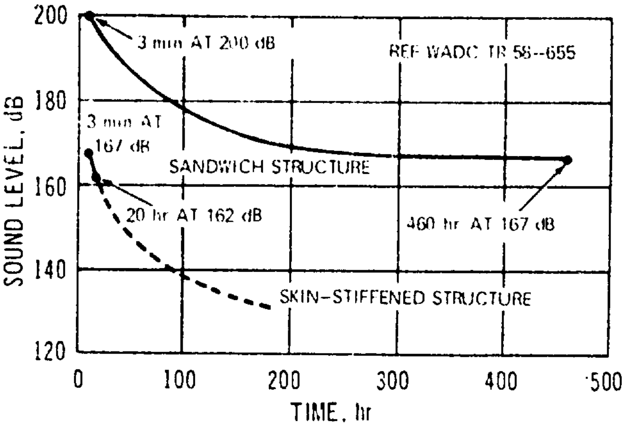


Figure 2-3. Comparative Sonic Fatigue Resistance of Conventional and Sandwich Structures

*Registered Trademark

when cost is a factor and/or thermal conductivity is of concern. Dupont Nomex nylon-fiber-treated core material, though recently developed, has thermal resistance and the properties required for aircraft flooring applications.

Employment of a honeycomb core material in a construction is an exact technique. Physical characteristics of the construction must be investigated thoroughly, and related to available core properties, prior to the firming of the design. In addition to the structural requirements, the environmental operating conditions must be explored.

Common honeycomb types (Fig. 2-4) include the conventional hexagonal shape, a rectangular flexible core, and the reinforced and square cell shapes. The rectangular core is, essentially, an over-expanded hexagonal core. The flexible core is a configuration departure in that it includes a free sine wave that allows the core material to assume compound curvature at no sacrifice in the mechanical properties of the foil material. Flexible core, unlike anticlastic hexagonal core, does exhibit characteristics of a synclastic material. Reinforced hexagonal core employs an additional flat sheet in the center of the hexagonal cell so as to favor a mechanical advantage in a specific orthotropic direction. The square cell core is a consequence of manufacturing ease, and is employed primarily where resistance welding techniques are required in order to develop the core material.

Although a predominant use of honeycomb core material is for constant thicknesses (flat, single and compound curvature applications), it also is used for such components as airfoil sections.

The mechanical properties of the core material in a sandwich construction also must be considered. The core, whether isotropic or orthotropic, may be considered as a continuum spacer for the membranes (the facings). Typical properties of balsa wood cores are presented in Figs. 2-5 and 2-6. Figs. 2-7, 2-8, and 2-9 illustrate typical properties of hexagonal aluminum core material. Several different alloys are presented. Table 2-11 is a presentation of the properties of typical rigid foams.

The term sandwich construction describes the close attachment between face and core material in this type of structure. Should this attachment be weak, or

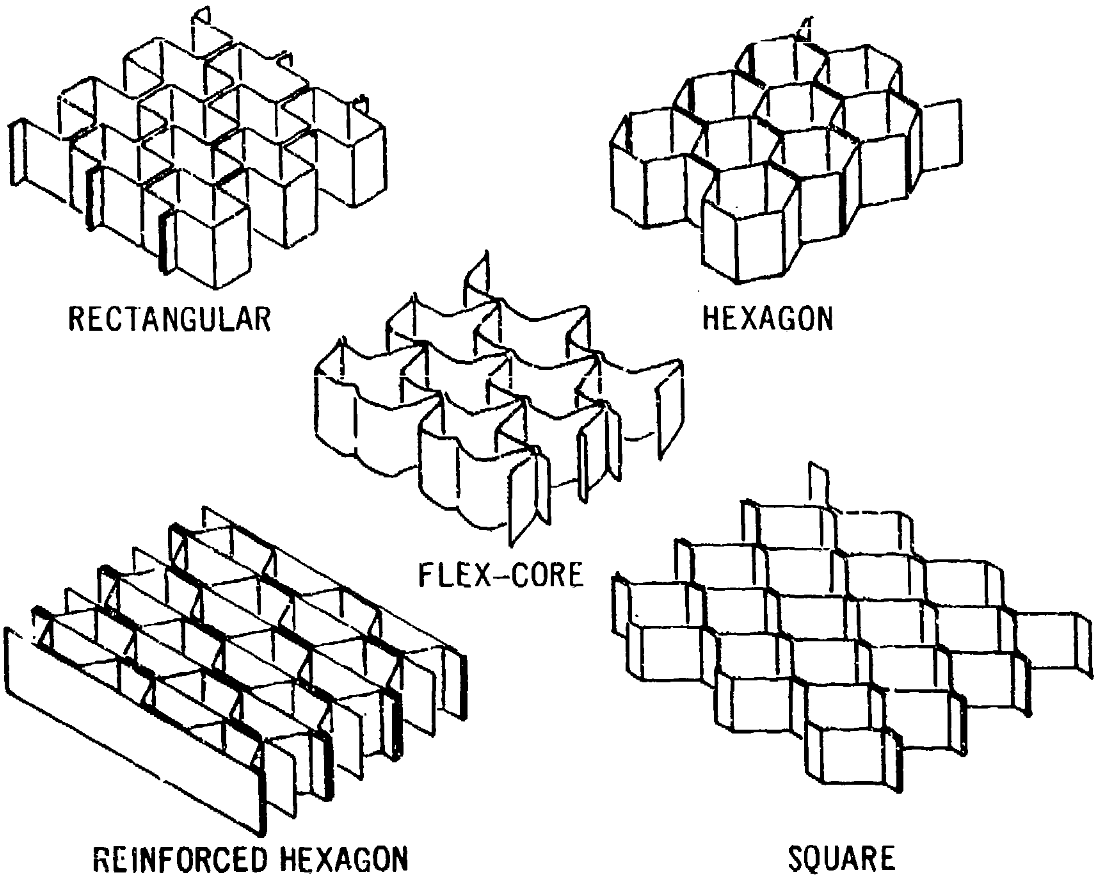


Figure 2-4. Common Honeycomb Configurations

absent, the construction is no longer a sandwich. Attachment of the core to the facings is necessary, and must be of sufficient strength to develop the full mechanical properties of the sandwich construction. For example, if the construction is loaded to its limit, then failure is expected to appear either in the facings, or in the core, or simultaneously in both. However, it cannot appear in the attachment between core and facings. It is most important for the designer to investigate the properties of the bonding agent so as to assure compliance with these requirements.

Adhesives of various types and properties currently are available to satisfy every sandwich requirement. Table 2-12 contains a partial listing of common adhesives currently in use. Laboratory shear bond strengths at room temperature of aluminum-to-aluminum bonds with various types of adhesives are presented in Table 2-13. Useful temperature range and strength properties of structural adhesives after exposure are listed in Table 2-14.

The process of applying adhesive to facing or core material must not be ignored. For the adhesive to be efficient, it must be applied to joining surfaces that are free from oxides and contaminants, and its application must take place under controlled en-

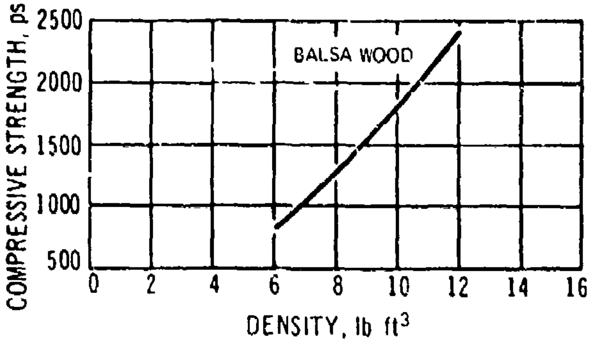


Figure 2-5. Properties of Balsa Wood — Compressive Strength vs Density

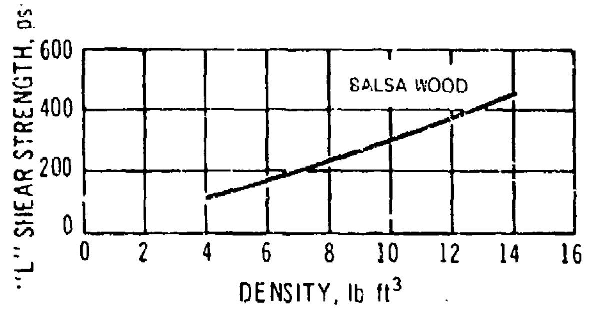


Figure 2-6. Properties of Balsa Wood — "L" Shear Strength vs Density

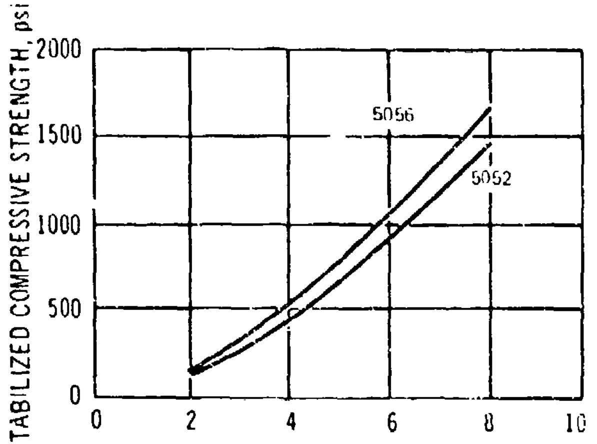


Figure 2-7. Typical Stabilized Compressive Strength

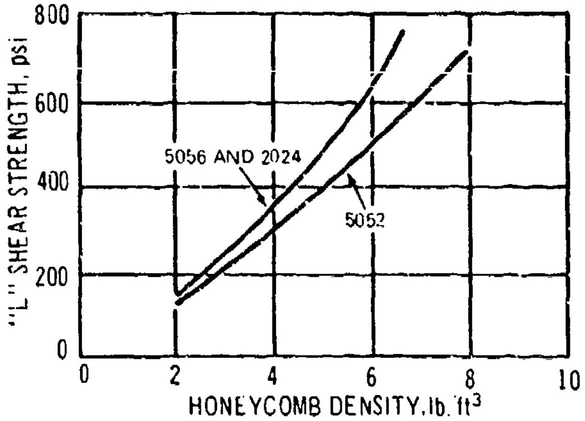


Figure 2-8. Typical "L" Shear Strength

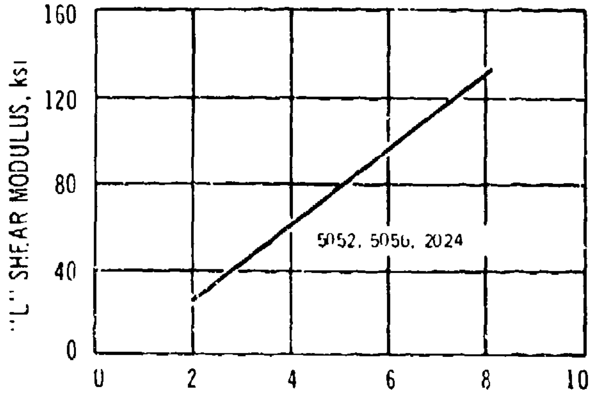


Figure 2-9. Typical "L" Shear Modulus

TABLE 2-11. PROPERTIES OF RIGID FOAMS

		DENSITY, lb/ft³		COMPRESSIVE STRENGTH, psi		SHEAR STRENGTH, psi		THERMAL CONDUCTIVITY, Btu/(hr-ft²(°F/in.))		MAX TEMP, °F	
	DENSITY	MIN	MAX	MIN	MAX	MIN	MAX	MIN	MAX	SHORT TERM	FULL TIME
URETHANE	CO_2 BLOWN	1.0-1.[illegible]	65.0	10.20	18,000	10.15	2,100	0.21	1.00	600	450
	FREON BLOWN	1.5	15.2	15.25	1,590	10.20	200	0.11	0.37	350	250
	FROTH	1.5	4.0	15.00	100	11.00	65	0.11	0.16	350	250
POLYSTYRENE	EXTRUDED	1.3	4.5	10.0	140	15.0	95	0.24	0.33		175
	MOLDED	0.5	10.0	8.0	200	13.0	90		0.77		175
	SELF EXPANDED		10.0	45.0	120			0.24			
EPOXY	PRE-FORMED		38.0	50	6,000			0.65			500
	PACK-IN-PLACE	15.0	25.0	600	3,000			0.24	0.80		600
	FOAM-IN-PLACE	2.0	8.0	13	110	38.0	360	0.11	0.50	450	350
SILICONE	HEAT ACTIVATED POWDER	12.0	18.0	25.7	325			0.30		650	
	ROOM TEMP LIQUID	3.0	20.0	8.0	115			0.16	0.30	650	
PHENOLIC	LOWEST DENSITY	0.3	1.0	1	5	5.0		0.22			
	LOW DENSITY		2.0	5	25	14.0		0.20			250
	MEDIUM DENSITY		4.0	40	100	30.0		0.20			250
	HIGH DENSITY	15.0	22.0	300	1,100	90.0		0.30			200

vironmental conditions. The elapsed time between preparatory cleaning for bonding and the application of adhesive must be held to a minimum. Process control during application and throughout the bonding of the construction is vital for the development of the specified property for the sandwich. Adhesive manufacturer recommendations must be adhered to methodically.

Design considerations for sandwich structural components are somewhat similar to those for homogeneous material. The main difference is the inclusion of the effects of the core material. The basic design concept requires the spacing of strong, thin facings far apart in order to achieve a high stiffness-to-weight ratio. The lightweight core material having this property also will provide the required resistance to shear and the strength to stabilize the facings to their required configuration. Sandwich is analogous to an I-beam; the flanges carry direct compression and tension loads in a similar manner as do the facings of the sandwich, and the web carries the shear loads as does the core material. The departure from typical procedures for sandwich structural elements is the inclusion of effects for shear properties on deflection, buckling, and stress. Because the facings are used to carry loads in a sandwich, prevention of local failure under edgewise, direct, or flatwise bending loads is as necessary as is prevention of local crippling of stringers in the design of sheet-stringer construction.

Structural instability of a sandwich construction can manifest itself in a number of different modes. Various possibiliti[illegible] illustrated in Fig. 2-10.

Intercellular buckling (face dimpling) is a localized mode of instability that occurs when the facings are very thin and the cell size is relatively large. This effect can cause failure by propagating across adjacent cells, thus inducing face wrinkling. Face wrinkling is a localized mode of instability that exhibits itself in the form of short wave length in the facing; it is not confined to individual cells of cellular type cores, and is associated with a transverse straining of the core material. A final failure from wrinkling usually will result either from crushing of the core, tensile rupture of the core, or tensile rupture of the core-to-facing bond. If proper care is exercised in selection of the adhesive system, the tensile bond strength will exceed both the tensile and compressive strengths of the core failure.

Shear crimping often is referred to as a local mode

TABLE 2-12.
COMMON ADHESIVES IN CURRENT USE

ADHESIVE TYPE	TYPICAL SYSTEMS TRADE DESIGNATION	MANUFACTURER
NITRILE PHENOLIC	AF 30 METLBOND 402	3M COMPANY NARMCO
VINYL PHENOLIC	FM 47 AF 31 METLBOND 105	AMERICAN CYANAMID 3M COMPANY NARMCO
EPOXY PHENOLIC	AEROBOND 422 HT 424 HYLOC 422	ADHESIVE ENGINEERING AMERICAN CYANAMID HYSOL
UNMODIFIED EPOXY	HP 304 HYLOC 901 B-3 METLBOND 547	HEXCEL HYSOL NARMCO
MODIFIED EPOXY 250 CURE	AF 126 FM 123 HYSOL 9601 PLASTILOCK 717 RELIABOND 711 & 393-1 HP 103	3M COMPANY AMERICAN CYANAMID HYSOL BF GOODRICH RELIABLE MFG HEXCEL
MODIFIED EPOXY 350 CURE	METLBOND 328 AF 120, AFL L FM 1000, FM 61 RELIABOND 398-420 HP 104	NARMCO 3M COMPANY AMERICAN CYANAMID RELIABLE MFG HEXCEL
EPOXY POLYAMIDE	EC 2216 HP 316 HYLOC 901 B-1	3M COMPANY HEXCEL HYSOL
POLYAMIDE	FM 34 FF 951	AMERICAN CYANAMID HEXCEL
MODIFIED URETHANES	HP 300,310,322 URALITE 412	HEXCEL RESOLIN
CORE SPLICING ADHESIVES	FM 37 AF 3206 RELIABOND 370B HP 901	AMERICAN CYANAMID 3M COMPANY RELIABLE MFG HEXCEL

TABLE 2-13.
SHEAR BOND STRENGTH OF ADHESIVES

ADHESIVE TYPE	BOND SHEAR STRENGTH, psi*
NITRILE PHENOLIC	3500
VINYL PHENOLIC	4200
EPOXY PHENOLIC	3400
UNMODIFIED EPOXY	3100
MODIFIED EPOXY-250 CURE	4500
MODIFIED EPOXY -350 CURE	3300
EPOXY POLYAMIDE	5500
POLYIMIDE	3300

*AVERAGE VALUES AT ROOM TEMPERATURE. TEST SPECIMENS ALUMINUM TO ALUMINUM, LAP JOINTS.

TABLE 2-14.
USEFUL TEMPERATURE RANGE AND STRENGTH PROPERTIES OF STRUCTURAL ADHESIVES*

ADHESIVE TYPE	USEFUL TEMP RANGE, °F	TYPICAL VALUES LAP SHEAR, psi	PEEL STRENGTH
NITRILE PHENOLIC	 350	3500–4500 3300–1700	GOOD TO EXCELLENT
VINYL PHENOLIC	-67 225	2000–3000 100–1000	FAIR TO GOOD
EPOXY PHENOLIC	70 400	1300–5000 200–1900	POOR TO MEDIUM
UNMODIFIED PHENOLIC	-67 500	1300–3000 800–3000	POOR TO MEDIUM
MODIFIED EPOXY 250 CURE	250 180	1500–2500 1000–1900	GOOD
MODIFIED EPOXY 350 CURE	-67 300	3000–3500 1500–2500	GOOD
EPOXY POLYAMIDE	-67 180	4000–5000 2800–3300	GOOD
POLIMIDE	UP TO 600	3300	POOR

*192 HOUR EXPOSURE

of failure, but it actually is a special form of general instability for which the buckle wave length is very short due to a low transverse shear modulus of the core. The phenomenon of shear crimping occurs quite suddenly, and usually causes the core to fail in shear. General instability for configurations having no supplementary stiffening except at the boundaries involves overall bending of the composite wall coupled with transverse shear deformations. Whereas intercellular buckling and wrinkling are localized phenomena, general instability is of a widespread nature. Premature general buckling normally is caused by insufficient sandwich thickness or by insufficient core shear rigidity.

The basic design principles of sandwich construction can be summarized as follows:

1. Sandwich facings *shall* be at least thick enough to withstand design stress under design loads.
2. The core *shall* be thick enough and have sufficient shear rigidity and strength so that overall sandwich buckling, excessive deflection, and shear failure will not occur under design loads.
3. The core *shall* have a high enough modulus of elasticity, and the sandwich great enough flatwise tensile and compressive strengths, so that wrinkling of either facing will not occur under design loads.
4. For cellular honeycomb cores, where dimpling of the facings is not permissible, the cell size spacing *shall* be small enough so that dimpling of either wall into the core spaces will not occur under design loads.

In addition, selection of materials, methods of sandwich assembly, and material property used for design *shall* be compatible with the expected environment where the sandwich is to be used. For example, facing-to-core attachment *shall* have sufficient flatwise tensile and shear strength to develop the required sandwich strength in the expected environment. Included as environment are effects of temperature, water and moisture, corrosive atmosphere and fluids, fatigue, creep, and any condition that may affect material properties. Additional characteristics — such as thermal conductivity, dimensional stability, and electrical continuity of sandwich material -- should be considered in arriving at an effective design for the intended task.

2-4.5 ARMOR MATERIALS

There are available a variety of armor materials and material combinations that can be used for passive protection of helicopters. Armor types, with appropriate Military Specification references and a relative comparison of cost, availability, machinability, weldability, formability, and multiple-hit capability, are summarized in Table 2-15. Table 2-15 also lists the areal densities, and provides a comparison of strength, hardness, shock and vibration, and resistance to corrosion for the various types of armor

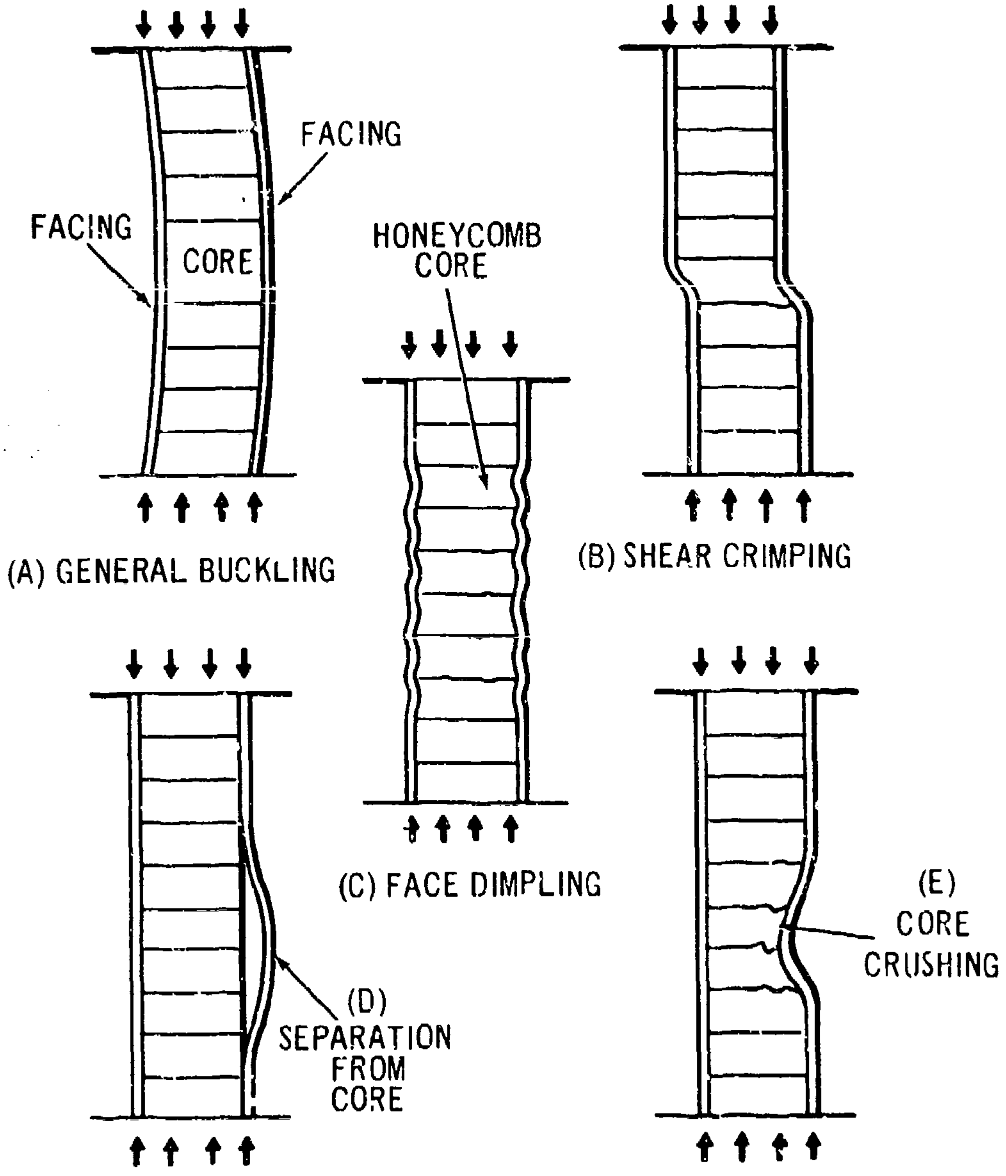


Figure 2-10. Modes of Failure of Sandwich Composite Under Edgewise Loads

TABLE 2-15.
ARMOR MATERIAL DESIGN DATA AND PHYSICAL CHARACTERISTICS

MATERIALS			RELATIVE COST	AVAILABILITY	MACHINE ABILITY	WELD ABILITY	FORM ABILITY	MULTIPLE HIT CAPABILITY	AREAL DENSITY, lb ft² in. THICK	STRENGTH	HARDNESS	SHOCK, VIBRATION RESISTANCE	RESISTANCE TO CORROSION	TEMPERATURE RESISTANCE	
														TO 200°F	TO 450°F
SPECIFICATION TYPES	STEEL	WROUGHT HOMOGENEOUS MIL-S-12560 ①	LOW	GOOD	GOOD	GOOD	GOOD	FULL	40.8	HIGH	MEDIUM	GOOD	POOR	GOOD	GOOD
		ROLLED HOMOGENEOUS, JAN-A-256 ①	LOW	GOOD	GOOD	GOOD	GOOD	FULL	40.8	HIGH	MEDIUM	GOOD	POOR	GOOD	GOOD
		NON-MAGNETIC, ROLLED, JAN-A-434	LOW	GOOD	GOOD	GOOD	GOOD	FULL	40.8	HIGH	MEDIUM	GOOD	POOR	GOOD	GOOD
		WROUGHT, HIGH HARDNESS, MIL-S-46100 ①	LOW	GOOD	FAIR	POOR	POOR	FULL	40.8	HIGH	HIGH	FAIR	POOR	GOOD	POOR
		CAST, MIL-S-11356 ①	LOW	GOOD	GOOD	FAIR	POOR	FULL	40.8	HIGH	MEDIUM	GOOD	POOR	GOOD	FAIR
		FACE HARDENED MIL-A-00784 ①	LOW	FAIR	FAIR	POOR	POOR	FULL ②	40.8	HIGH	HIGH	FAIR	POOR	GOOD	POOR
		ROLL BONDED, DUAL-HARDNESS, MIL-S-46099 ①	HIGH	GOOD	POOR	POOR	POOR	FULL ②	40.8	HIGH	HIGH	POOR	POOR	GOOD	POOR
	LIGHT METALS	2024-T4 ALUMINUM ALLOY, MIL-A-7169	LOW	GOOD	GOOD	POOR	GOOD	FULL	14.4	MEDIUM	LOW	GOOD	FAIR	GOOD	FAIR
		7039-T6 (HEAT-TREATABLE, WELDABLE) ALUMINUM ALLOY MIL-A-46063	LOW	GOOD	GOOD	GOOD	GOOD	FULL	14.4	MEDIUM	LOW	GOOD	FAIR	GOOD	FAIR
		2219-T81 ALUMINUM ALLOY, MIL-A-46118	LOW	GOOD	GOOD	POOR	GOOD	FULL	14.8	MEDIUM	LOW	GOOD	FAIR	GOOD	FAIR
		5083-5456 (WELDABLE) ALUMINUM ALLOY, MIL-A-46027	LOW	GOOD	GOOD	GOOD	GOOD	FULL	13.7	MEDIUM	LOW	GOOD	FAIR	GOOD	FAIR
		WELDABLE TITANIUM ALLOY, MIL-T-46077	HIGH	FAIR	FAIR	GOOD	FAIR	FULL	23.0	HIGH	MEDIUM	GOOD	GOOD	GOOD	GOOD
		TITANIUM ALLOY, MIL-A-23556	HIGH	FAIR	FAIR	GOOD	FAIR	FULL	23.0	HIGH	MEDIUM	GOOD	GOOD	GOOD	GOOD
		MAGNESIUM-LITHIUM ALLOY, MIL-A-21648	MEDIUM	GOOD	GOOD	GOOD	GOOD	FULL	7.6	MEDIUM	MEDIUM	FAIR	FAIR	GOOD	GOOD
	ORGANIC	DORON, MIL-A-17856	LOW	GOOD	N/A	N/A	N/A	LIMITED	10.6	LOW	LOW	GOOD	FAIR	FAIR	FAIR
	TRANSPARENT	GLASS, MIL-G-5485	HIGH	GOOD	N/A	N/A	N/A	LIMITED	12.7	LOW	HIGH	POOR	GOOD	FAIR	FAIR
		GLASS FACED PLASTIC, MIL-A-46108 ③	HIGH	GOOD	N/A	N/A	N/A	LIMITED	16.0	LOW	HIGH	POOR	GOOD	POOR	FAIR
	CERAMIC-ORGANIC	CERAMIC-FACED COMPOSITE, MIL-A-46103	HIGH	GOOD	N/A	N/A	N/A	LIMITED	VARIABLE	LOW	HARD FACED	FAIR	GOOD	FAIR	FAIR
EXPERIMENTAL TYPES	STEEL	HIGH-HARDNESS, HY-80 ①	LOW	FAIR	POOR	POOR	POOR	FULL	40.8	HIGH	HIGH	FAIR	POOR	GOOD	POOR
		HIGH-HARDNESS, 9 Ni-4Co ①	LOW	FAIR	POOR	POOR	POOR	FULL	40.8	HIGH	HIGH	FAIR	POOR	GOOD	POOR
		AUSFORMED, 5M21 ①	LOW	FAIR	POOR	POOR	POOR	FULL	40.8	HIGH	HIGH	FAIR	POOR	GOOD	POOR
		AUSFORMED, HY-TUF ①	LOW	FAIR	POOR	POOR	POOR	FULL	40.8	HIGH	HIGH	FAIR	POOR	GOOD	POOR
		CARBURIZED AND AUSFORMED, 5M21 ①	LOW	GOOD	POOR	POOR	POOR	FULL	40.8	MEDIUM	HIGH	FAIR	POOR	GOOD	POOR
		DUAL-HARDNESS, ROLL BONDED, H-11 & 5M21 ①	HIGH	GOOD	POOR	POOR	POOR	FULL	40.8	HIGH	HIGH	FAIR	POOR	GOOD	POOR
	LIGHT METALS	7075-T6 ALUMINUM ALLOY	LOW	GOOD	GOOD	FAIR	GOOD	FULL	14.4	MEDIUM	LOW	GOOD	POOR	GOOD	FAIR
		CAST ALUMINUM	LOW	GOOD	GOOD	FAIR	POOR	FULL	14.4	MEDIUM	LOW	GOOD	POOR	GOOD	FAIR
		6A1-4V (ELI GRADE) TITANIUM ALLOY	HIGH	FAIR	FAIR	GOOD	FAIR	FULL	23.0	HIGH	MEDIUM	GOOD	GOOD	GOOD	GOOD
		6A1-4V (SPECIAL GRADE) TITANIUM ALLOY	HIGH	POOR	FAIR	GOOD	FAIR	FULL	23.0	HIGH	MEDIUM	GOOD	GOOD	GOOD	GOOD
		4A1-4V TITANIUM ALLOY	HIGH	FAIR	FAIR	GOOD	FAIR	FULL	23.0	HIGH	MEDIUM	GOOD	GOOD	GOOD	GOOD
		BERYLLIUM PLATE	HIGH	FAIR	GOOD	FAIR	GOOD	FULL	9.7	HIGH	HIGH	GOOD	GOOD	GOOD	POOR
	TRANSPARENT	LEXAN	LOW	GOOD	N/A	N/A	N/A	LIMITED	6.3	LOW	LOW	FAIR	FAIR	POOR	FAIR
		PLEXIGLAS	LOW	GOOD	N/A	N/A	N/A	LIMITED	6.3	LOW	LOW	FAIR	FAIR	POOR	FAIR
	COMPOSITES	METAL-CERAMIC-ORGANIC COMPOSITES ④	VERY HIGH	POOR	N/A	N/A	N/A	LIMITED	VARIABLE	LOW	VARIABLE	FAIR	POOR	FAIR	FAIR
	CERAMIC-ORGANIC	B_4C-DORON	VERY HIGH	POOR	N/A	N/A	N/A	LIMITED	VARIABLE	LOW	HARD FACED	FAIR	GOOD	FAIR	FAIR
		Al_2O_3-DORON	HIGH	GOOD	N/A	N/A	N/A	LIMITED	VARIABLE	LOW		FAIR	GOOD	FAIR	FAIR
		SiC-DORON	VERY HIGH	POOR	N/A	N/A	N/A	LIMITED	VARIABLE	LOW		FAIR	GOOD	FAIR	FAIR

① MAGNETIC
② TEN (10) CALIBERS FROM PREVIOUSLY DAMAGED AREA
③ NOT RESISTANT TO FLAME
④ MAGNETIC CHARACTERISTICS DEPEND UPON THE METAL IN THE COMPOSITES

TABLE 2-16.
FABRICATION DATA FOR LIGHTWEIGHT ARMOR MATERIALS

	HIGH-HARDNESS STEEL	DUAL-HARDNESS HEAT-TREATED STEEL	DUAL-HARDNESS AUSFORMED STEEL	ALUMINUM OXIDE PLASTIC COMPOSITE	BORON CARBIDE PLASTIC COMPOSITE	K-T SILICON CARBIDE PLASTIC COMPOSITE
MAX PLATE SIZE	0.250 THICK-33.5X32 in., 1.250 THICK-72X56 in.	65X96 in.	26X96 in., UP TO 2000 lb	MONOLITHIC TILE 28 in. DIAG 5X12 ft MAX PANEL	MONOLITHIC TILE 15X15 in., 4X6 ft PANELS HAVE BEEN MADE	MONOLITHIC TILE 6X6 in., 7X18 in. PANELS UP TO 2 in. THICK
MIN RADIUS OF CURVATURE (FOR CAL .30 AP THREAT)	10X THICKNESS	APPROX 2X THICKNESS IN THE ANNEALED CONDITION	8 TO 9 in. DEPENDING ON THICKNESS	2 TO 6 in.	6 in.	SMALL
EXTENT OF COMPOUND CURVATURE	7 in. DEEP DISHES HAVE BEEN EXPLOSIVELY FORMED	VERY SMALL RADIUS IN BOTH DIRECTIONS IN ANNEALED COND	NO EXPERIENCE	6 in. RADII	NOT AVAILABLE	
BALLISTIC EFFECT OF CURVATURE	NONE	SLIGHT IMPROVEMENT	NONE			CURVED SHAPES MAY BE BETTER THAN FLAT PANELS
TOOLS REQUIRED FOR CUTTING	TORCH CUTTING	GAS OR PLASMA ARC TECHNIQUES		DIAMOND TOOLS		
WELDING PROCEDURES REQUIRED	AUSTENITIC STAINLESS STEEL (MIG OR SUBMERGED ARC) OR LOW HYDROGEN FERRITIC ELECTRODES	ANNEALED CONDITION HARDEX HEAT TREATABLE ELECTRODE GIVES BALLISTIC JOINT HEAT-TREATED CONDITION-STAINLESS STEEL ELECTRODE WITH 700° PREHEAT	MIG METHODS	CAN'T BE WELDED		
DRILLING POSSIBLE	REQUIRES CARBIDE TIPS, SPOT HEATING WITH TORCH BEFORE DRILLING MAY LESSEN DIFFICULTIES	WITH SPECIAL CARBIDE TOOL IF HEAT-TREATED	USING MASONARY TYPE BITS	YES	WITH SPECIAL CERAMIC DRILLS	
RECOMMENDATIONS FOR ATTACHMENT TO STRUCTURE	THROUGH BOLTING			THREADED INSERTS IN FIBERGLAS OR FLANGE TYPE PERIPHERAL SUPPORT	VARIOUS TYPES OF BRACKETERY AND THREADED INSERTS IN BACKING	
BOLTING COMPRESSION CONCENTRATIONS	NOT IMPORTANT			CERAMIC MUST BE RELIEVED OF COMPRESSION UNDER BOLT HEAD WITH THROUGH BOLTING		
PANEL JOINING METHODS	WELDING OR MECHANICAL JOINTS			TILES JOINED ON CONTINUOUS PLASTIC BACKING, PANELS MAY BE APPLIED TO A FRAME USING METAL DOUBLER STRIPS AT JOINT		CERAMIC IS MOUNTED TO BACKING MATERIAL OF REQUIRED SIZE AND SHAPE
BALLISTIC JOINTS BETWEEN PANELS	CAN'T BE ACHIEVED BY WELDING UNLESS MATERIAL THICKNESS IS INCREASED IN THE HEAT-AFFECTED ZONE	BY WELDING WITH HARDEN ELECTRODE IN ANNEALED CONDITION	CAN'T BE ACHIEVED BY WELDING UNLESS MATERIAL THICKNESS IS INCREASED IN THE HEAT AFFECTED ZONE	WITH VARYING DEGREES OF EFFICIENCY MATERIAL THICKNESS MUST BE INCREASED AT JOINT		

materials. Table 2-16 summarizes fabrication data for some of the available lightweight armor materials that can be considered for installation on helicopters.

2-4.5.1 Available Materials

Materials that can be considered for use in armor design, and their properties, include:

1. Aluminum alloy. Produces less splash from bullet impacts than other armor materials in common use, is exceptionally effective against yawed and high-obliquity impacts, and is nonmagnetic.

2. Titanium. Nonmagnetic and resistant to sea water corrosion.

3. Homogeneous steel. Rolled from a steel alloy with the toughness and percent of elongation necessary to achieve a good resistance to both puncture and spalling. It is designated magnetic.

4. Nonmagnetic steel armor. Certain fully annealed austenitic steels are not magnetic. If these steels are cold-worked, however, they become magnetic.

5. Hard-faced steel armor plate. Composed of a hard surface overlaying a softer backing material of tougher steel. It is somewhat more effective, on a weight basis, against solid shot than is face-hardened armor plate, and can be fabricated, by special techniques, to a curve. It is magnetic.

6. Ballistic nylon. Provides excellent protection from fragments and tumbled projectiles. Ballistic nylon pads or quilts can be considered for replacement of insulation and sound-attenuating blankets. The ballistic level of nylon cloth and/or felt con-

figured with fasteners and/or attachments should be established and/or verified by gunfire tests for each configuration.

7. Ceramic. Built up of various materials, each intended to perform a particular function in defeating the projectile; for example, a glass-fiber-reinforced plastic to absorb the energy of impact, faced with a layer of ceramic tile (aluminum oxide, Al_2O_3; silicon carbide, SiC; boron carbide, B_4C; titanium diboride, TiB_2, etc.) to shatter the projectile. On a weight basis, some of these composites compare favorably with standard steel armor plate for stopping solid shot. However, they have poor capabilities for stopping multiple hits, and produce many secondary fragments when struck. Ceramic is the bulkiest of the materials listed here, and usually is the most expensive.

8. Ceramic-faced. The ceramic facing may be applied before or after the armor metal has been shaped or formed.

9. Transparent. Composed of glass or clear organic polymers, either alone or in combination (MIL-G-5485, MIL-A-7168, MIL-A-46108).

In general, ceramic armor exhibits the lowest weight per unit area for protection against armor-piercing ammunition (cal .30 and .50). Metallic armor exhibits substantially better multihit capability, although the probability of a small panel of aircraft armor taking a multiple bullet hit from a high-firing-rate gun is remote. Metallic armor for aircrew seats may become competitive on a weight basis when the armor is used simultaneously as support or structure.

2-4.5.2 Design

For design strength and rigidity requirements, refer to MIL-A-8860 and AMCP 706-170.

2-5 ADHESIVES AND SEALANTS

2-5.1 BONDING AGENTS

There are literally hundreds of proprietary adhesive formulations suitable for various aircraft bonding applications. Some of these may be used in bonding a wide variety of materials, while others are usable only for highly specialized purposes. Adhesives generally are categorized under the two broad classifications of structural and nonstructural types.

2-5.1.1 Structural Adhesives

This category of materials is used for bonding primary structures that are subject to large loads. Typical ultimate bond shear strengths are several thousand psi. Structural adhesives usually are formulated from thermosetting resins that, when mixed with a suitable curing agent, react to form an infusible and insoluble solid. Depending upon the type of curing agent, the conversion may occur within a few minutes at room temperature, or, at the other extreme, it may require heating up to about 350°F to effect a cure within a reasonable time. The latter type of material, due to its low reactivity at low temperatures, can be premixed and stored (often, under refrigeration) as a one-component system until used.

Nearly all applications of structural adhesives require fixturing in order to hold the components being bonded in contact during cure; this is because at some point during the cure cycle the adhesive goes through a fluid flow stage.

Most structural bonds are made with tape or film adhesives. These are usually from 0.005 to 0.015 in. thick, and may be unsupported or supported on thin, open-weave fabrics (carriers) of glass, nylon, or other fibers.

Film adhesives have two important advantages:

1. Uniformity. Variations in thickness and composition are minimal. Because both shear and peel strengths are sensitive to bondline thickness, control of this variable is desirable. Although bondline thickness also is affected by curing pressure variations, film adhesives — particularly those having flow restricted by carriers and/or high-melt viscosities — can reduce thickness variations appreciably. In addition, film adhesives eliminate the weighing errors and inadequate mixing that are possible with two-part liquid adhesives. Quality control checks can be made on each roll of film before production parts are bonded; this is not practicable to perform on each batch of most liquid adhesives due to limited pot-lives.

2. Ease of assembly. Film adhesives are available in a wide range of tacks, varying from dry to very sticky. Complicated parts can be assembled simply by cutting the film to the shape of the desired bondline and laying it on the first surface. The second surface then is placed in position, and is held by the adhesive tack until bonding pressure can be applied. Films that are not tacky at room temperature are tacked readily by momentary contact with a hot iron at strategic locations. Parts of many layers may be laid up in this manner and bonded at one time. Adhesive waste also is minimized when films are used because there is no excess material left to set up in the mixing container.

Most film adhesives require curing temperatures of 250°-350°F, and, therefore, have long shelf lives. Cold storage usually is advised, however, although some types are stable for many weeks at room temperature. A few types are available that cure at lower temperatures, including room temperature; these must be stored at temperatures well below 0°F.

The other common physical form for structural adhesives is the two-part liquid mixture. These materials consist of two components that react, when mixed, to form a thermosetting solid. Usually, they are 100% nonvolatile. Many cure at room temperature in a few hours or days; others require heat to cure. Frequently, they are in the form of high-viscosity pastes containing inert fillers and/or thixotropic agents. In contrast to most film adhesives, however, these uncured pastes usually become fluid when heated. Pot-lives, like cure times, depend upon the rate of chemical reactivity, which is influenced greatly by temperature. Thus, adhesives that cure rapidly at room temperature may have only a few minutes of pot-life, while those requiring high-temperature cures have pot-lives varying from hours to months.

Less common structural adhesive forms include one-component pastes and powders, and all require elevated-temperature cures.

Essentially all structural adhesives of interest for helicopter applications are based upon either epoxy or phenolic thermosetting resins. Because these materials are brittle inherently, they usually are made flexible with elastomers or thermoplastic resins in order to improve peel strengths. Polyurethane adhesives also show promise, as they can be formulated with both strength and flexibility. To date, however, they have not been used widely in structural aircraft applications.

Epoxies are the most versatile and widely used structural adhesives. They have excellent adhesion, low creep, low shrinkage during cure, and 100% nonvolatility. The liquid or paste types have either low peel properties or limited temperature resistance, and are less adaptable to modification (for improvement of these particular characteristics) than are the film types. The latter can be modified with tough thermoplastics such as nylon or polyvinyl acetal resins. Primers (low-viscosity solutions of adhesive dissolved in solvents) are available for use in conjunction with modified epoxy-film adhesives. Their primary function is to protect prepared metal surfaces from contamination and oxidation since epoxy films have adequate wetting and adhesive characteristics without primers.

Phenolic adhesives used in the aircraft industry always are modified with an elastomer or another resin. Although they can be produced in liquid form, they now are used predominantly as films. Vinyl (polyvinyl formal or butyral) phenolic adhesives were the first materials used for aircraft metal bonding. Rubber-phenolic adhesives include those modified with neoprene or nitrile rubber, the latter presently being the most widely used type of elastomer-phenolic structural adhesive. Epoxy-phenolics are used primarily because of their outstanding temperature resistance. Being very rigid, they have good shear strength and creep resistance but poor peel and impact properties. Because of the poor wetting and flow characteristics of the elastomer-phenolic films, a coating of liquid primer on the substrates usually is advised. As with all phenolic condensation reactions, gases are evolved during cure, necessitating relatively high bonding pressures. The actual pressure required to contain these volatiles is a function of the temperature rise rate; 100 psi is a typical recommendation when the bondline is heated rapidly.

All of the adhesive types discussed previously may be used for metal-to-metal bonding. The selection will depend upon the relative importance of such factors as shear strength, peel strength, temperature resistance, chemical resistance, fatigue and creep properties, fabrication method, and cost. Generally, a modified epoxy or nitrile-phenolic film adhesive is chosen for primary structural applications, while a paste-type epoxy and simple contact tooling may suffice for secondary structures with less-critical requirements. The requirements for several classes of structural adhesives are covered completely in MMM-A-132 and MMM-A-134. Although there is some overlapping, MMM-A-132 is concerned mainly with film adhesives while MMM-A-134 generally has less stringent requirements which are met by the liquid- and paste-type epoxies.

Cured, reinforced-plastic composites can be bonded to themselves or to metals with the same adhesives and techniques used for bonding metals. In addition, adhesive prepregs can be used either for an entire layup or as a single-ply bonding layer between a conventional reinforced-plastic layup and the substrate. These materials consist of a structural grade of reinforcement impregnated at a high resin content with a resin formulation having good adhesion qualities. Reinforced plastics can be bonded to metals and other substrates by employing between the substrate and layup a layer of conventional film adhesive that is cured simultaneously with the laminate. This procedure is advantageous in that it precludes any mismatch of mating surfaces, a problem that always exists to some extent with preformed parts. While this technique has been found effective with a number of adhesive and laminating resin combinations, such materials must be selected carefully for compatibility with both chemical reactions and curing temperatures and pressures.

Most of the epoxy adhesives also are suitable for bonding facings to honeycomb core in applications where good flow and wetting ability, and low curing pressure are required. Some of the phenolic-based adhesives also may be used for sandwich construction, although most are not recommended for this purpose due to poor filleting action and the evolution of volatiles during cure. When phenolic adhesives are used in sandwich bonding, the core either is perforated or pressure is released just prior to reaching the final cure temperature. MIL-A-25463 contains requirements for adhesives for bonding sandwich. It defines two classes: Class 1, for facing-to-core bonding only; and Class 2, for bonding facing to core and inserts, edge attachments, etc. Because most adhesives suitable for sandwich construction also can be used for metal-to-metal bonding, nearly all sandwich adhesives are qualified to both MIL-A-25463, Class 2, and to MMM-A-132. The adhesive prepregs described previously also can be used in fabricating sandwich panels with reinforced-plastic facings. One disadvantage of this procedure, however, is that a relatively porous laminate is obtained due to the lack of laminating pressure between cell walls of the core.

Typical bond strengths obtainable from several of the common types of adhesives are given in Table 2-17.

2-5.1.2 Nonstructural Adhesives

These adhesives are used primarily to bond interior accessories made of a variety of materials, including plastics, rubbers, metals, and fabrics. Because a joint failure would not be catastrophic in these applications, consideration of the highest possible adhesive strengths is not paramount; and other factors, such as cost and convenience, can be given equal attention.

The adhesives preferred in these applications generally are based upon solutions or dispersions of various elastomers and thermoplastics. They set up through solvent evaporation rather than by chemical cure, and therefore do not require temperature or pressure for curing. Because initial tack often is adequate to hold in position the parts being bonded, even clamping fixtures frequently are unnecessary. On the other hand, because these adhesives remain thermoplastic, they lack the temperature and chemical resistance of the thermosetting structural adhesives.

Where somewhat stronger or more temperature- and chemical-resistant bonds are required, semistructural adhesives, such as the two-part epoxies and urethanes, may be used with room-temperature curing.

Cements based upon a solution of the polymer being bonded are used frequently for bonding noncrystalline thermoplastics such as acrylics, cellulosics, polycarbonates, polystyrenes (including ABS), and vinyls to themselves. The dissolved polymer gives body to the cement, while the solvent softens the adherends, effecting a weld or bond when the solvent evaporates and the plastic rehardens.

Transparent acrylic plastics also may be bonded with two-part adhesives, consisting of methylmethacrylate monomer and a catalyst, which have excellent strength and transparency. MIL-A-8576 defines three types of two-part acrylic adhesives. Type I contains solvent and is covered in MIL-P-5425. Types II and III are without solvent and may be used for bonding plastics as covered in both MIL-P-5425 and MIL-P-8184.

For bonding of dissimilar materials, flexible films or fabrics, rubbers, or other such materials, elastomer-based adhesives are preferred; they offer good adhesion to many materials and better peel strengths, and can elongate so as to accommodate different thermal expansion coefficients. Elastomeric adhesives may be dissolved in a suitable organic solvent or dispersed in water; tackifying resins, antioxidants, plasticizers, and reinforcing fillers are usual

TABLE 2-17.
TYPICAL PROPERTIES OF COMMONLY USED STRUCTURAL ADHESIVES

CHEMICAL TYPE	PHYSICAL FORM	CURE TEMP, °F	SHEAR STRENGTH, psi ①				T-PEEL ① STRENGTH, lb/in.			SANDWICH PEEL ② in.-lb/3in.WIDTH		
			-67°	75°	180°	250°	-67°	75°	180°	-67°	75°	180°
MODIFIED EPOXY	FILM	250	5100	5200	2900	1000	20	30	25	55	60	35
NYLON-EPOXY	FILM	350	6700	6500	3400	2200	60	100	90	9[illegible]	170	45
EPOXY-PHENOLIC	SUPPORTED FILM	350	3200	3500	3300	2900				35	33	31
NITRILE-PHENOLIC	FILM	350	4100	4200	2400	1800	10	40	20	NA	NA	NA
NEOPRENE-PHENOLIC	SUPPORTED FILM		3500	1900	1100			15		17	36	12
EPOXY (GEN PURPOSE)	2-PART PASTE	75-200	1500	3000	800		2	3	4	NA	NA	NA
EPOXY (HIGH TEMP MOD)	1-PART PASTE	250	2000	2000	2500	3000	2	2	2	NA	NA	NA
EPOXY (HIGH PEEL MOD)	2-PART PASTE	75-200	2000	2500	400		2	25	2	NA	NA	NA

① ALUMINUM ADHERENDS TESTED PER MMM-A-132 AT THE INDICATED TEMPERATURES.
② ALUMINUM CORE AND FACINGS TESTED PER MIL-A-25463 AT THE INDICATED TEMPERATURES.

components of the formulation.

MMM-A-1617 covers requirements for adhesives based upon natural rubber, neoprene, and nitrile rubber. Adhesives based upon natural or reclaimed rubber are suitable for bonding such items as rubber and fabrics to metals in applications where oil and fuel resistance is not a problem. Neoprene- and nitrile-based adhesives generally have greater peel strengths in the same applications, as well as good resistance to oils and fuels. The neoprene type usually is best for bonding neoprene and most other rubbers and rigid plastics, and has the best heat resistance. Nitrile rubber adhesives are preferred for bonding nitrile rubber, vinyls, and other flexible plastics.

Silicone rubbers should be bonded to themselves or to other substrates with silicone adhesives, such as those described in MIL-A-46106 or MIL-A-25457. No heat or pressure is required.

Contact adhesives are a special type of elastomer-based adhesive having high immediate strength upon contact of the two coated adherends, but they do not permit any repositioning. They are covered by MMM-A-130.

Other special-purpose adhesive specifications include MMM-A-121, MMM-A-122, MMM-A-189, MIL-A-24179, and MIL-A-21366.

2-5.1.3 Processing Operations

Process and inspection requirements for structural adhesive bonding are contained in MIL-A-9067. Factors to be considered include type of surface preparation, control limits and methods of surface treatment, solutions, clean-room layup area requirements, prefitting of parts, adhesive storage controls, handling of cleaned parts, application of primer and adhesive, tooling concepts, temperature and pressure controls, secondary bonding of subassemblies, rework, and destructive and nondestructive verification testing.

Equal in importance to the selection of an optimum adhesive system is the selection of the best surface preparation for the adhesives and adherends being used. Some suggestions are given in MIL-A-9067. Other recommended sources are ASTM No. D 2561 for metals, ASTM No. D 2093 for plastic surfaces, and Ref. 6. With some metals, such as aluminum, the surface treatment is practically universal; while with other metals, such as stainless steel, it is advisable to evaluate different treatments with each combination of alloy, condition, and adhesive. Significant batch-to-batch variations in a given type of alloy may be noted. For most reinforced plastics, a wet-sanding treatment is recommended to obtain a water-break-free surface. When nonstructural adhesives are used in bonding, a careful solvent wiping and/or sanding treatment will suffice for many materials.

2-5.1.4 Design of Bonded Structures

Adhesive joints should be designed so that they are stressed in the direction of maximum strength. Thus, the adhesive should be placed in shear while minimizing peel and cleavage stresses. Maximum bond area and uniform thickness should be provided for, and stress concentrations should be avoided where possible. Scarfing and bevelling are two methods that sometimes can be used to reduce the cleavage-stress concentrations at the edges of lap joints.

Test methods for sandwich constructions are described in MIL-STD-401, while numerous other test methods for adhesives are contained in FTMS No. 175.

2-5.2 SEALING COMPOUNDS

There is a degree of overlapping between sealants and adhesives; most sealants must adhere in order to be effective, while an adhesive generally seals the joint that it bonds. In addition, many sealants are formulated from the same basic polymers that are used in adhesive compositions. Sealants are related particularly to the elastomeric adhesives, and many of the qualitative comparisons made in the previous paragraph apply to sealants as well as adhesives. In order to form trowelable pastes, sealants are formulated with higher viscosity and lower tack than are the elastomeric adhesives. Lower-viscosity sealants also are available and are suitable for dipping, brushing, and even spraying. These materials, however, are classified more properly as coatings.

Commercial sealants are manufactured from a variety of polymers, including polysulfide, urethane, silicone, neoprene, acrylic, butyl rubber, chlorosulfonated polyethylene, and polymercaptan. In addition to the base elastomer, a typical sealant formulation may include curing agents, accelerators, plasticizers, antioxidants, solvent thinners, and inorganic fillers or reinforcing agents.

Sealants may be one- or two-component types. All of the latter cure into tough, thermoset elastomers. The one-component sealants are subdivided into three categories: nonhardening putties that remain permanently soft; solvent-release types that become semihard through evaporation of a volatile ingredient, and types that cure by reaction with atmospheric moisture. Properties of the latter, after curing, are similar to those of the cured two-part sealants. (One further form of "sealant" is the cured elastomeric tape or extrusion. Because these must be held in place mechanically, they more properly might be called gaskets.)

All effective sealants must have a high ultimate elongation and a low modulus in order to accommodate expansion and contraction of the joint being sealed. Most commercial sealants have these qualities. They vary widely, however, in their degree of recovery, ranging from near 0% recovery (or 100% plastic flow) for a permanently soft putty to nearly 100% recovery for a cross-linked (cured) elastomer. This property is important because a low-recovery sealant, once compressed, must accommodate subsequent joint expansion entirely by its elongation, or it will fail. A compressed, high-recovery sealant will return, as the joint expands, to its original dimension before it begins to elongate in tension.

Of the various chemical types of sealants, only polysulfides, urethanes, and silicones are currently of importance in the aircraft industry. These are all high-recovery elastomers when cured.

Polysulfides are most commonly used in helicopters, where they act as both sealants and aerodynamic fairing compounds. They have excellent adhesion characteristics and resistance to solvents and fuels, weathering and aging, and temperatures up to 250°F. MIL-S-7124 and MIL-S-8802 describe the two-part elastomeric sealing compounds with increasingly severe requirements for adhesion and resistance to temperature and fuels. MIL-S-8784 compounds are formulated purposely with very low adhesion for such uses as fuel tanks access doors. A grade for sealing electrical components is described in MIL-S-8516. One-part, noncuring, polysulfide putties also are available. A material of this type is defined by MIL-S-11030; it is intended for sealing optical instruments, but is useful for various purposes.

Silicone sealants have outstanding environmental resistance because they are unaffected, relatively, by temperatures ranging from cryogenic to more than 500°F, and by moisture, ozone, and ultraviolet radiation. However, because they are the most expensive sealants, they are used only where these excellent properties are required. Some types also have very good electrical characteristics, and are used to seal electrical systems. MIL-S-23586 covers silicone sealants for electrical applications, and MIL-A-46106 describes a general-purpose, room-temperature-curing adhesive-sealant for both mechanical and electrical requirements. As ordinary silicones have relatively poor fuel and oil resistance, fluoro-silicone sealants should be used where these properties are required. Both one- and two-component materials are in common use. The former cure by absorption of atmospheric humidity, and, therefore, cure very slowly in confined areas or in thick sections. Primers usually are recommended to permit maximum adhesion to metals.

Although of totally different chemical composition, polyurethane sealants have many similarities to the silicones. Both two-component and one-component moisture-curing types are common. Primers (often silicone-based) are recommended, but, in this case, primarily for retention of adhesion in humid or water-immersion situations. These sealants exhibit complete recovery after extended outdoor exposure. They also are useful in cryogenic applications where they are surpassed only by the silicones, and in electrical applications. Polyurethanes also have excellent oil resistance, and greater abrasion resistance than any other sealants. One problem is loss of adhesion upon exposure to ultraviolet light.

An area where sealing compounds frequently are used is in edge-sealing of honeycomb sandwich panels. Because joint expansion and contraction are not major considerations in this instance, relatively rigid sealants usually are employed. These are essentially the same materials as the epoxy (and occasionally urethane) paste adhesives discussed previously, except that microballoon (hollow microspheres of glass or plastic) fillers frequently are used to produce a lightweight, closed-cell structure. Sandwich panels also may be sealed with an edge wrapping of Fiberglas prepreg.

Viscous sealants may be applied with a variety of equipment, ranging from a putty knife to a completely automatic mixer-dispenser system. Fluid sealants (coatings) may be brushed or sprayed. One-component sealants supplied in cartridges can be applied from manual- or air-operated guns, or these sealants can be dispensed directly from pails or drums by air-powered or hydraulic pumping equipment. Two-part sealants can be weighed and mixed by hand or by metering-mixing equipment that dispenses the components according to preset ratios. Frozen cartridges of premixed sealant also are available commercially; these must be stored at −40°F until just prior to use.

2-6 PAINTS AND FINISHES

2-6.1 PAINTS AND COATINGS (ORGANIC)

MIL-F-7179 prescribes in detail the manner in which the external and internal surfaces of a helicopter are to be finished. Other helpful documents are TB 746-93-2, MIL-STD-171 (MR), and AMCP 706-100.

Helicopters require a Type I protection, i.e., protection against severe deteriorative conditions. For most surfaces, this involves one coat of wash primer (MIL-C-8514), one coat of primer (MIL-P-23377), and two top coats of a topcoat for example, TT-E-516 or MIL-C-81773. Preparation of the surface for painting will differ with the type of metal and with the surface (external or internal).

For this handbook, exterior surfaces are defined as all visible surfaces of an end-item that is housed within the helicopter and all visible surfaces of the helicopter, including all portions of the system that are exposed to the airstream. Interior surfaces are the nonvisible surfaces of an end-item that is housed within the fuselage of the craft.

Prior to painting, aluminum surfaces usually are finished with Anodize MIL-A-8625 or Alodine 1200 (MIL-C-5541), and magnesium with Dow 17 or HAE (MIL-M-45202).

Nonstainless steels are phosphate-treated (MIL-P-16232), stainless steels are passivated (QQ-P-35), and Fiberglas surfaces are sanded and cleaned with naphtha (TT-N-95).

The first coat of paint applied is the wash primer. The term designates a specific material that combines the properties of an inhibitive wash coat, or metal conditioner, with those of the conventional anticorrosive primer. The essential components of wash primers are phosphoric acid, chromate pigment, and polyvinyl butyral resin. Wash primers can be formulated that are effective equally over iron, steel, aluminum, treated magnesium, copper, zinc, and a wide variety of other metals. The advantages of wash primers include ease of application and rapid drying, useful range of temperature and humidity, application to a variety of metals, effectiveness in preventing underfilm corrosion, and good adhesion as a base for subsequent coatings. The most frequently used wash primer is that defined in MIL-C-8514, a smooth-finish, spray-type, pretreatment coating furnished in two parts: resin component and acid component. The materials must be mixed prior to use.

The primer, which must conform to MIL-P-23377, is used over the wash primer. It is compatible with the usual acrylic-nitrocellulose lacquer top coat, as well as with the alkyd top coats (TT-E-516) and urethane (MIL-C-81773). The two-component, epoxy-polyamide system has high chemical and solvent resistance and unusual weatherability. It is spray-applied. This specification also provides for an additional class of materials suitable for use under air-pollution regulations. The availability of classes of coatings meeting air-pollution regulations is becoming increasingly important, and this factor should be kept in mind by the designer.

The top coats most often specified for Army aircraft are the nitrocellulose and acrylic-nitrocellulose lacquers, which contain a wide range of pigmentation. They are preferred because of the ease in removing them with solvents when it is necessary to change camouflage or color schemes or when repainting is required. They also are applied with a spray in volatile solvents (MIL-L-19537). TT-E-516 describes another suitable top coat, and one that also meets air-pollution regulations. This coating is a styrenated phthalic alkyd resin combined with the necessary amounts of driers and volatile solvents. The mixture contains 50% resin solids, including small percentages of antioxidants, wetting agents, and stabilizers. A wide range of coloring pigments is available, and these are present in amounts of 24-45% of the total solid content.

There are special paint formulations for camouflage, battery compartments, high-temperature areas, walkways, and antiglare applications. Rain-erosion-resistant coatings (MIL-C-7439) are used on the leading edges of the rotor and on radomes. There are special formulations for high visibility, and paints for lettering and marking. Rubber, both natural and synthetic, and transparent surfaces such as glass and plastic windows, are not painted.

Particular attention must be directed to assemblies in which dissimilar metals are joined. It generally is required that each of the mating surfaces *shall* be finished with the minimum number of coats required for interior surfaces. Where magnesium is one of the metals to be joined to a dissimilar metal, the metals *shall* be separated by MIL-T-23142 tape or MIL-S-8802 sealant. The tape *shall* extend not less than 0.25 in. beyond the joint edge in order to prevent moisture from bridging between the dissimilar metals. All holes and countersinks that attaching parts pass through should be primed, and all joining bolts, screws, and inserts should be wet-primed when inserted. Preferably, all steel nuts, bolts, screws, washers, and pins should be cadmium-plated.

2-6.2 SPECIAL FINISHES

In addition to the finishing of surfaces with organic coatings as described in the previous paragraph, there are a number of special finishes for metal which serve to provide the desired protection without further application of organic coatings, or that are used to provide a suitable base for the application of organic finishes. Many of these processes involve the development of a durable, corrosion-resistant, oxide layer on the surface of the metal. Although the development of this surface oxide film may or may not involve the use of an electrical current, the chemical effect is similar and the process is called anodizing.

For aluminum, there are many different finishes, some of which are used to provide a base for paint and some of which provide protection without further painting. The processes all involve chromates as an oxidizing ingredient, and all have proprietary compositions. The performance of these methods of treatment is governed by MIL-C-5541. The reagents may be applied by spraying, dipping, or swabbing; generally, the metal is dipped in a sequence of baths and rinses that create clean, fine-grained oxide — uniformly distributed over the surface — with no coarse grains and no untreated areas. Class 1A treatments must withstand exposure to salt spray for 168 hr and are unpainted or followed by wash prime and prime treatments. Class 3 coatings are similar to Class 1A coatings, except that the electrical resistance is low.

For magnesium, there are two primary anodizing treatments. One of these is the Dow 7 treatment (MIL-M-45202) and the other method is the HAE treatment (MIL-M-45202), both of which involve electrolytic anodizing of the metal surface in order to build up a fairly thick layer of a complex aluminum-manganese-oxide phosphate. Neither is suitable for outdoor exposure without further coatings. The preferred method involves pretreating, priming, and an epoxy-polyamide finish.

Parts made of corrosion-resistant steel are passivated in order to develop their corrosion-resistant qualities. This process serves to remove the "active" centers on the surface and to leave a thin, durable, transparent layer of oxide that prevents further corrosive or oxidative attack on the metal. Passivation is accomplished by immersing the parts in an aqueous solution consisting of nitric acid and sodium dichromate. The temperature of immersion varies from 70° to 155°F, depending upon the alloy involved and the intended operating temperature. Prior to treatment, it is essential to wash parts carefully in an alkaline solution in order to remove all of the particles of iron that may have accumulated on the surface, as these would develop rust stains during the treatment. The passivation process is detailed in QQ-P-35.

Ferrous surfaces that are not to be painted usually are treated with black oxide. The resulting deposit is a hard, durable, oxide surface that is attractive and is somewhat resistant to corrosion and to wear. The process is applicable to both nonstainless and stainless steels; it involves immersing the previously cleaned part in an alkaline, or alkaline-chromate, oxidizing solution, followed by warm and then cold water rinses, a final chromic acid dip, and drying in warm air. The process is defined in MIL-C-13924.

Ferrous metals that are to be painted are given a phosphate coating in accordance with MIL-P-16232. These coatings are of two types: Type M, which has a phosphate base; and Type Z, which has a zinc phosphate base. Type M coatings are more resistant to alkaline environments than are Type Z coatings. When they are applied properly, the reaction forms a mixed-metal phosphate coating on the surface of the ferrous metal that is close-grained, fine, and free of powder and coarse grains and that seals the surface well. The treated surface is more resistant to corrosion and provides a firm base upon which to apply wash prime and prime coatings.

In all of the foregoing treatments, the metals employed, especially the ferrous metals, are subject to the absorption of hydrogen from the solutions, and the hydrogen serves to embrittle the metal. It is desirable to promote the diffusion of hydrogen from the metal by heating at 210°-225°F for 8 hr in order to provide hydrogen embrittlement relief.

Still another class of finishes frequently employed is the flame-sprayed type. In this technique, metals, silicon dioxide, titanium dioxide, alumina, or other oxides and ceramic materials are fed into a plasma or oxyacetylene flame — either as strands or powder — where they are vaporized and deposited on any substrate that will condense and hold them. By this means, similar or dissimilar metals can be applied to metallic or nonmetallic surfaces. Ceramic materials can be applied in order to provide abrasive surfaces, wear-resistant surfaces, or flame-resistant coatings. MIL-M-6874 covers the flame-spraying of metals. Worn metal surfaces can be built up and subsequently machined in order to repair shafts, pistons, pins, etc. One useful application has been the flame-spraying of titanium dioxide coatings upon the metal exhaust skirts of jet engines for oxidation protection.

2-6.3 PLATING

Another method of applying attractive, durable, and abrasion- and corrosion-resistant coatings to metals and plastics is metal plating. The platings of most interest in helicopter design are copper, nickel, chromium, and cadmium. Except for electrical components, where electrical conductivity is important, copper plating is used only to provide a base for the wear-resistant nickel and chromium platings. Chromium and nickel platings are used to provide hard and wear-resistant surfaces for such objects as snap fasteners, strap holders, handles, knobs, seat arms, instrument parts, and other items where painting would not be satisfactory or economical. Cadmium and zinc plating are employed almost exclusively to provide galvanic protection against corrosion. Cadmium is the preferred coating for ferrous metal items such as nuts, bolts, screws, inserts, and pins used in assembly, particularly where dissimilar metals are employed.

A treatment of metal plating can be found in Ref. 7. In this process, four methods are used extensively: electrolytic, chemical reduction or electroless, vacuum vapor deposition, and molten metal dip.

In electrolytic plating, the item to be plated is cleaned so as to provide an oil- and dirt-free surface, and then is connected as the cathode in an electrolytic cell. The anode is made of the plating material, and when an electrical current is passed through the electrolyte, the metal is deposited upon the surface of the item being plated. In some cases, the item first is coated with a thin layer of copper, which adheres readily to the base metal and forms a firm surface to which the plating metal (either nickel or chromium) can attach firmly. There are a great many proprietary electrolytic solution formulations and processes. Considerable skill is required to obtain a uniform, fine grain and bright coat, and much care must be exercised to assure cleanliness, avoidance of poisons, prevention of treeing, blistering and cracking, and avoidance of coarse-grained platings. Applicable Federal Specifications are QQ-N-290, QQ-C-320, and QQ-P-416.

Electroless or chemical-reduction plating depends upon the generation of activated atoms of the metal to be deposited adjacent to the activated metal surface upon which the plating is to be deposited. There are proprietary chemical compositions and reduction processes for most metals used in plating, and also for most metal and plastic base materials. There is less danger of hydrogen embrittlement with this process. The electroless method is suitable for localized plating, or for use in the field or shop where laboratory or process line facilities are not available. It also is effective for some of the more difficult jobs, such as nickel-plating of magnesium surfaces, where the nickel coating serves to make the surface more wear-resistant and less impact-sensitive. MIL-C-26074 covers the plating of electroless nickel.

Vacuum deposition plating is conducted in a vacuum chamber. The items to be plated are racked at some optimum distance from the source of the plating metal, and are rotated so as to obtain a uniform coating. The source of the plating metal may be a hot wire or a molten pool of the metal. The metal is heated electrically to a temperature at which it vaporizes from the surface. Under the influence of an electrostatic field applied between the metal source and the items being plated, the atoms of the metal become ionized and are attracted to the surface. This process produces an exceptionally bright, coherent, and tenacious coating. It is used frequently for metalizing plastics, and, because the danger of hydrogen embrittlement is negligible, it also is used for the plating of high-strength steel parts for high-stress applications. It is the preferred method of cadmium-plating high-strength bolts and nuts and other fasteners, and MIL-C-8837 details the requirements for this application.

Galvanized steel products are made by dipping the cleaned, preheated steel in molten zinc. Cadmium-plated parts also are made in this manner, and earlier tin coatings were applied by the dip process. However, the dip coating of steel with cadmium and tin has been superseded by the more economical and more precisely controlled electrolytic processes. MIL-T-10727 covers electroplating and hot dipping of tin.

In both the electrolytic and electroless processes, hydrogen embrittlement is a significant hazard. Diffusion of the hydrogen into the metal under the electrolytic forces is greater than in the case of electroless deposition. The danger increases with the strength and modulus of the plated material. Thus, it is necessary to program a hydrogen embrittlement relief heating cycle in order to promote the diffusion of hydrogen from the basic metal. The optimum time and temperature will depend upon the nature of the coating material and of the base metal, as well as upon the scheduling requirements for the part. Generally, the plating specification will require hydrogen embrittlement relief. Embrittlement from the vacuum plating process and from the molten metal dip process is minimal.

2-6.4 TAPES

Tapes of varying composition, texture, thickness, and width are used in a variety of ways in helicopter design. Fabric tapes may be woven or nonwoven, impregnated or nonimpregnated, and made from many of the advanced plastic materials. The tapes may or may not have adhesive on one or both sides (pressure-sensitive tapes).

One application for tapes is in the marking of helicopters. Decals conforming to the requirements of MIL-P-38477 may be used in lieu of paint for all external and internal markings within the size limits specified. They may be pressure-sensitive, adhesive-backed, and scored, and are applied over previously finished surfaces.

Antislip tapes are used on walkways, steps, and similar areas, while high-visibility tapes are employed in warning insignia. Caution should be used to insure that antislip tape edges are not exposed to airflow that can cause the tape to peel. The use of

Mylar and polymide tapes in electrically driven motors has made possible the construction of light, powerful motors that cool readily. Tapes also are used for binding wiring harness, sealing access ports, and sealing cavities for foam-in-place filling. Teflon tapes are used to provide low-friction surfaces for low-load-bearing areas subject to sliding contact. Finally, masking tapes are used in the finishing, repair, or repainting of surfaces.

2-7 LUBRICANTS, GREASES, AND HYDRAULIC FLUIDS

2-7.1 GENERAL

Although this chapter is devoted to materials, the subject of lubricants and hydraulics cannot be divorced from the total system. The designer cannot simply choose any oil or hydraulic fluid for a particular use. A lubrication system is designed for delivering oil to the mechanisms to be lubricated. Selection of the oil to be used is as much a function of the filtering, cooling, and pumping properties of the system parts as it is of the mechanisms to be lubricated. Similar considerations are applicable to the selection of hydraulic fluids. Even with greases, the matter of service and maintenance and the provision of adequate fittings, bushings, and seals must be considered in relation to the lubrication needs and the greases selected.

These are dynamic rather than static uses of materials. The wide range of lubricating materials and their applications are indicated in Table 2-18. Even where the system is not dynamic and maintenance is not a problem — as with the use of sealed bearings, self-lubricating bearings, dry film lubricants, and materials of high lubricity such as Teflon — the frictional and wear characteristics of the parts involved must be analyzed and balanced as a system rather than through treating each contact friction area as an isolated case.

2-7.2 DESIGN OF LUBRICATION SYSTEMS

The requirements for design of lubrication systems for turbine engines installed in helicopters are discussed in Chapter 3; requirements for transmission and drive system lubrication are discussed in Chapter 4. As described in Chapter 4, the contractor *shall* prepare a detailed lubrication list for approval prior to inclusion of such lubrication data in any technical order, handbook, or other maintenance document scheduled for delivery to the Government. Listing of a lubricant and a corresponding lubrication interval for a given application constitutes assurance by the contractor that such equipment will perform as required for the specified interval when so lubricated.

2-7.3 GREASES

A detailed discussion of the many approved greases is included in MIL-HDBK-275. For purposes of illustration, the applications for four types of greases are discussed.

1. Helicopter oscillating bearings. A suitable grease for use in bearings having oscillating motions of small amplitude — such as helicopter rotor head bearings — is described in MIL-G-25537, and is appropriate for equipment that must operate at ambient temperatures of −65° to 160°F. It also should be used for ball or roller bearings operating at high speeds or high temperatures.

2. Ball and roller bearings. These bearings may be lubricated with a grease described in MIL-G-25013. This grease is intended for use in the temperature range of −100° to 450°F, and is designed particularly for those high-temperature ball and roller bearing applications where soap thickeners may not be used. The speed factor or DN value of the bearing may not exceed 200,000. This grease is not to be used for sliding metal-on-metal uses, such as journal bearings, spiral gears, and gear trains

3. Gears and actuators. A general-purpose grease for equipment requiring a lubricant with high load capacity is described in MIL-G-81322. This grease is useful in the temperature range of −65° to 350°F, and is compatible with rubber.

4. Pneumatic systems. Another frequently required grease is described by MIL-G-4343. This grease is intended for use in pneumatic systems as a lubricant between rubber seals and metal parts (under dynamic conditions), and has proven satisfactory in service at pressures to 2000 psi. It is suitable for use with MIL-P-5516 rubber, but should not be used with other types of rubber without prior testing for compatibility.

2-7.4 DRY FILM AND PERMANENT LUBRICANTS

Because of the high level of sand and dust contamination frequently encountered in helicopter operation, it is advantageous to use nonwetting lubricants that do not cause the dust to adhere. Included among such lubricants are plastic materials of high lubricity, such as Teflon and nylon, molybdenum disulfide in a plastic matrix, and porous metal (usually bronze) bearings filled with a nonvolatile lubricant, such as silicone. These usually are employed in low-load applications, such as in instrumentation, and in low-speed sliding applications.

Molybdenum disulfide frequently is applied as a dry-film lubricant in a phenolic or epoxy bonding agent to provide a secure bonding to the metal base. MIL-L-46010 describes such a heat-cured, solid-film lubricant, and MIL-L-46147 an air-cured solid film lubricant which are intended to reduce wear and prevent galling and seizure of metals. Dry-film lubricants may be used on steel, titanium, aluminum, aluminum alloys, and other metals. They are useful where conventional lubricants are difficult to apply

TABLE 2-18. HELICOPTER LUBRICANTS AND HYDRAULIC FLUIDS

ITEM	MIL SPEC	USE	CONTACT SURFACE
GREASE	MIL-G-4343	PNEUMATIC SYSTEMS	RUBBER-TO-METAL, DYNAMIC
	MIL-G-27617	PLUG VALVES +400°	FUEL OIL SYSTEMS
	MIL-G-6032	PLUG VALVES	GASKETS, VALVES—GASOLINE AND OIL RESISTANT
	MIL-G-21164	HIGH LOAD STEEL SURFACES	STEEL-TO-STEEL, SLIDING
	MIL-G-23827	BALL, ROLLER, NEEDLE BEARINGS, GEARS, ACTUATOR SCREWS	METAL-TO-METAL, WIDE TEMP RANGE, HIGH LOAD
	MIL-G-25013	BALL AND ROLLER BEARINGS, -100° TO +450°F	TANGENTIAL METAL, ROLLING CONTACT
	MIL-G-25537	HELICOPTER ROTOR HEAD BEARINGS, OSCILLATING BEARINGS	METAL-TO-METAL, SLIDING- SMALL AMPLITUDE
	MIL-G-81322	WHEEL BEARINGS, GEARS, ACTUATOR SCREWS	METAL-TO-METAL, ROLLING AND SLIDING
LUBE OIL	MIL-L-3572	WINDSHIELD WIPERS	METAL-TO-METAL, LOW LOAD, SLIDING
	MIL-L-3918	INSTRUMENT, JEWEL BEARINGS	STEEL AND JEWEL PIVOTS, TO -40°F
	MIL-L-6081	TURBINE ENGINES	METAL-TO-METAL, DYNAMIC
	MIL-L-7808	TURBINE ENGINES	METAL-TO-METAL, DYNAMIC
	MIL-L-6085	AIRCRAFT INSTRUMENTS AND ELECTRONIC EQUIPTMENT	METAL-TO-METAL, FERROUS AND NONFERROUS
	MIL-L-6086	GEARBOXES, EXTREME PRESSURE	METAL-TO-METAL, HARDENED
	MIL-L-23699	HELICOPTER TRANSMISSIONS, GEARBOXES	METAL-TO-METAL, HARDENED, TO -40°F
	MIL-L-22851	AIRCRAFT PISTON ENGINES	METAL-TO-METAL
	MIL-L-27694	TACHOMETER GENERATORS, GYROMOTORS, GIMBALS	METAL-TO-METAL, LOW LOAD, HIGH SPEED
CORROSION PREVENTIVE	MIL-C-6529	AIRCRAFT ENGINES	MINERAL BASE OILS, 50 hr MAX OPERATION
LUBRICANT	MIL-L-8362	TIRE-TUBE, MOUNTING, DEMOUNTING	METAL-RUBBER
SOLID FILM	MIL-L-8937	CAMS, TRACKS, ROLLERS, SPHERICAL BEARINGS	LOW-LOAD SLIDING CONTACT, SAND AND DUST ENVIRONMENT
THREAD COMPOUND	MIL-T-5544	THREADS, SS BOLTS, PIPING, MOUNTINGS	ANTISEIZE, GRAPHITE, TO 400°F
COMPASS LIQUID	MIL-L-5020	MAGNETIC COMPASS	METAL-TO-METAL, METAL-TO-JEWEL
DAMPING FLUID	MIL-S-81087	SERVO SYSTEMS, CRANK CASES, GEARBOXES, FLUID TRANSMISSIONS, ELECTRONIC INSTRUMENTS AND EQUIPMENT	METAL-TO-METAL, SILICONE, -100° TO +500°F
HYDRAULIC FLUID	MIL-H-5606	LANDING GEAR SHOCK STRUTS	FOR SYNTHETIC SEALING MATERIAL
	MIL-H-6083	PRESERVATIVE OIL TESTING AND STORAGE	NONOPERATING FLUID
	MIL-H-27601	HIGH TEMPERATURE SYSTEM, -40° TO +550°F	MINERAL OIL, PETROLEUM BASE
	MIL-H-81019	ULTRA-LOW TEMPERATURE SYSTEM, AUTOPILOTS, SHOCK ABSORBERS, BRAKES, SERVO CONTROLS	PETROLEUM BASE FOR SYNTHETIC SEALING MATERIALS
	MIL-H-83282	AUTOPILOTS, SHOCK ABSORBERS, BRAKES, SYSTEMS USING SYNTHETIC SEALING MATERIALS	FIRE RESISTANT SYNTHETIC HYDROCARBON BASE

or retain, or where other lubricants may be contaminated easily with dirt and dust. They generally are suitable for sliding motion applications, such as in flap tracks, hinges, and cam surfaces, but may not be used with oils and greases.

2-7.5 HYDRAULIC FLUIDS

The design, installation, and data requirements for hydraulic systems are covered by MIL-H-5440 and are discussed in detail in Chapter 9. Two types of systems are defined: Type I, which is designed for the −65° to 160°F range; and Type II, which is used in the −65° to 275°F range. Two classes of systems are defined: 1500 psi, which has a cutout pressure of 1500 psi at the main pressure controlling device; and 3000 psi, which has a cutout pressure of 3000 psi. As in the case of the lubricating oils, selection of the hydraulic fluid is integral with design of the system. Pumps, motors, flight control actuators, heat exchangers, flexible connectors, packings, fittings, filters, accumulators, and electrical interconnects must be defined carefully and their characteristics shown to be compatible with the fluid selected.

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CHAPTER 3
PROPULSION SUBSYSTEM DESIGN

3-0 LIST OF SYMBOLS

A_i = area of inlet, inside diameter, in.^{2}
A_m = maximum inlet frontal area, in.^{2}
P_a = ambient pressure, psia
V_{lmax} = maximum velocity along leading edge surface, fps
V_i = inlet velocity, fps
V_∞ = free stream velocity, fps
W_f = fuel flow, lb/hr
ΔP = inlet pressure loss, psi
δ = $P_a/14.7$, dimensionless

3-1 INTRODUCTION

The propulsion system is defined during the preliminary design of the helicopter after the engine or engines have been selected, and their location in the airframe has been chosen. This chapter is concerned with the detail design of the various possible propulsion system configurations.

The propulsion system consists of the engine or engines, air induction subsystem, exhaust subsystem, fuel and lubrication subsystems, starting subsystem, controls, transmission subsystem, auxiliary power unit (if applicable), and infrared radiation suppression subsystem. It also *shall* include cooling and fire protection subsystems. The air induction subsystem also *shall* include inlet protection.

In the paragraphs that follow, engine installation considerations, propulsion control requirements, fuel and lubrication subsystem requirements, compartment cooling, accessories, and APU (auxiliary power unit) design requirements are discussed. The transmission subsystem is discussed in Chapter 4.

3-2 ENGINE INSTALLATION

3-2.1 GENERAL

Many different engine installation arrangements are possible: front drive, rear drives, side by side for multiengine helicopters, etc. Engine locations may be selected so as to control the overall CG of the helicopter to obtain the most effective power train geometry. All the subsystems *shall* function well individually and together as an integral propulsion system.

Engine installations usually fall within one of three categories: the completely submerged installation, the semi-exposed installation, and the exposed installation.

3-2.1.1 Submerged Installation

The submerged engine installation places the engine completely within the airframe. This arrangement, an example of which is shown in Fig. 3-1, requires careful consideration to insure adequate accessibility for maintenance.

Removable firewall sections often are used, but this requires attention to the detail design of seals and securing devices to insure that the fire protection of adjacent components and crew stations will not diminish with repeated removal of the firewalls. Although it may be difficult to service submerged engines because of limited accessibility, their location often makes it possible for maintenance personnel to work from ground level or from the cabin floor, eliminating the need for built-in service platforms. Because the engine may be located deep within the airframe, engine air induction and exhaust ducts usually are greater in length and more complex than for any other configuration. However, this inherent inefficiency may be offset by the external cleanliness of the aircraft.

3-2.1.2 Semiexposed Installation

The semiexposed installation usually nestles the engine between the main gearbox and the top of the fuselage. The engine is mounted directly on the airframe. This method, an example of which is shown in Fig. 3-2, requires only superficial structure to complete the engine enclosure.

The height of the engine in this type of installation usually requires built-in service platforms. These platforms often are built into the cowling, so that the work platform automatically is available when the cowl is opened. When this is impossible and separate platforms are provided, the engine cowling can be much lighter because it need not be structural.

The closeness of the rotor to the engine installation requires that careful consideration be given to the cowl-locking mechanism. It should be easily operable by one man and capable of being inspected from ground level for adequate security.

The semiexposed installation lends itself well to the use of either front- or rear-drive engines because the engine can be located either forward or aft of the main gearbox. The rear-drive engine may result in very short engine air induction subsystems that can be anti-iced efficiently by most heating systems. Front-drive engines may result in more complex induction subsystems, and, because of their com-

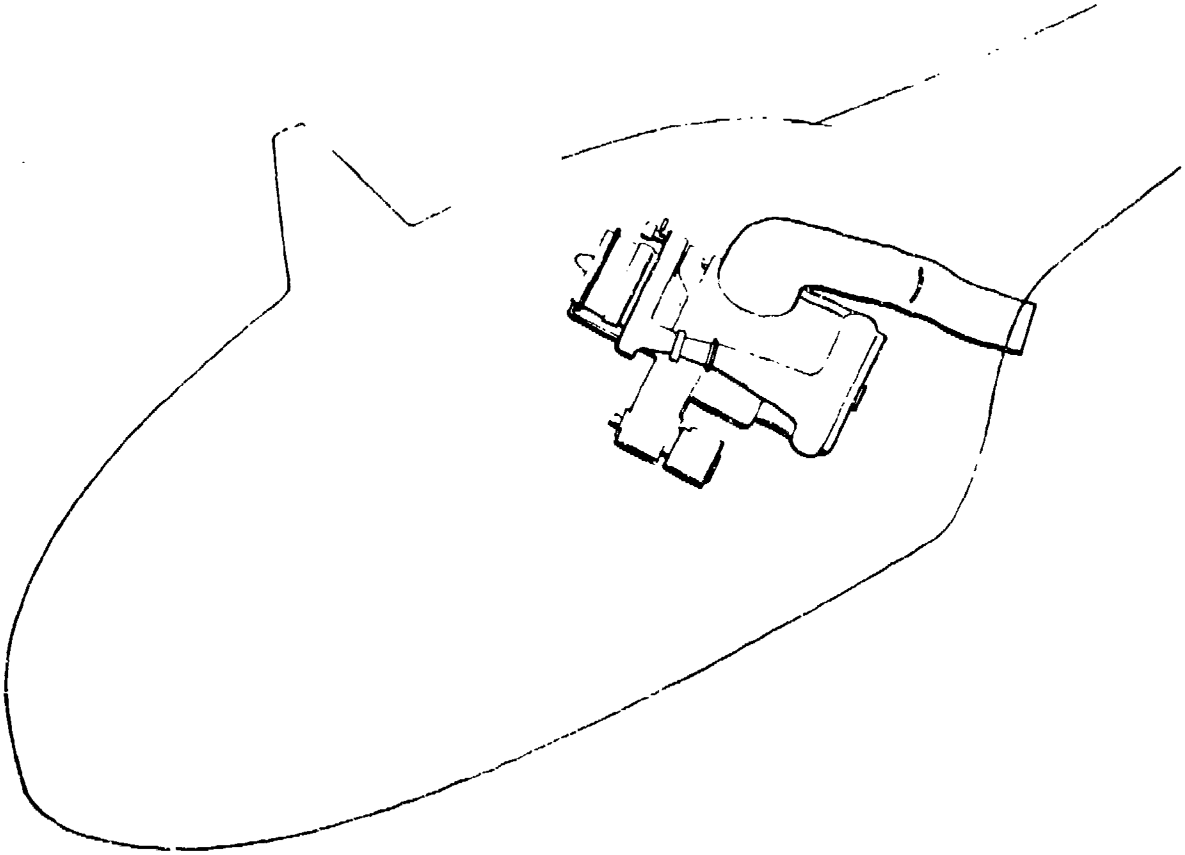


Figure 3-1. Submerged Engine Installation (Example)

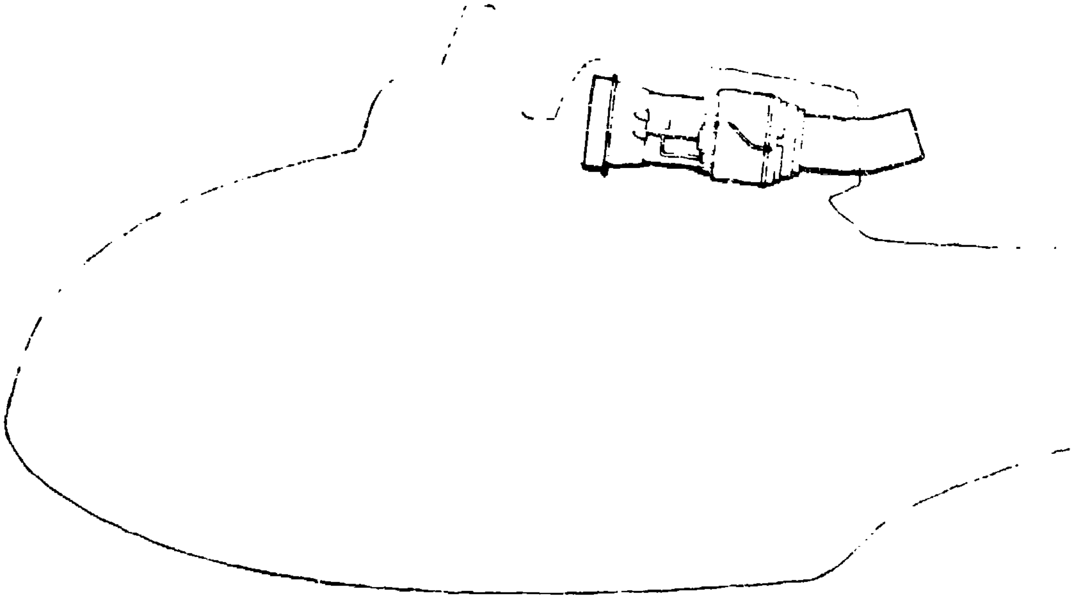


Figure 3-2. Semiexposed Engine Installation (Example)

plicated shapes, may require more complex anti-icing subsystems.

Twin-engine helicopters often employ the semi-exposed configuration. This enables a single housing unit to enclose all engines, and provides good accessibility to all engines provided they are spaced sufficiently far apart. Additional accessibility can be obtained by making interengine firewalls removable. If this is done, the designer must pay particular attention to the seals on the removable sections. Labyrinth seals are superior to other sealing methods because they can be separated repeatedly; however, they do not seal as well as other methods.

Interchangeability between engine installation components is desirable, and usually can be obtained for the exhaust ducts, engine mounts, and engine control installations. Careful design will enable many detail parts to be used on opposite assemblies with considerable savings in initial and maintenance costs, along with increased aircraft availability.

3-2.1.3 Exposed Installation

In the exposed installation, engines are located outside the airframe and are exposed on all sides. This arrangement, an example of which is shown in Fig. 3-3, commonly is used with a streamlined nacelle that provides environmental protection and reduces aerodynamic drag. The externally mounted engine arrangement provides the best accessibility, provided adequate service platforms or other convenient work areas are available.

The nacelles sometimes are attached directly to fuselage frames. Access to the engines usually is provided by removable panels. The engine mounts may be incorporated into the basic nacelle structure. A feature of this type of nacelle construction is that many external surfaces may consist of hinged panels, which when open, expose the engine. If desired, the panels may be opened individually for servicing any localized section of the engine. When the panels are in the closed and locked position, they can become loadsharing structural members of the nacelle. The panels also may be designed to serve as a work platform. This arrangement requires a minimum amount of fixed structure and results in maximum accessibility of the engine.

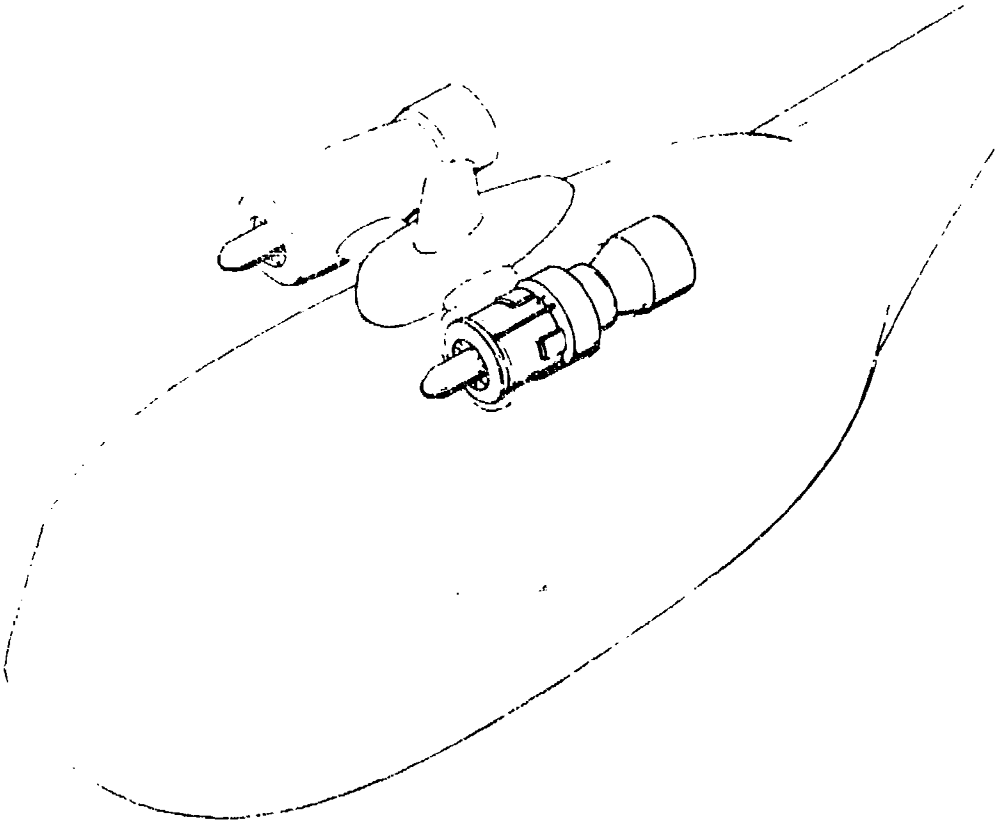


Figure 3-3. Exposed Engine Installation (Example)

3-2.1.4 Design Checklist

The following items are applicable to each of the preceding types of engine installations and, as such, are basic design objectives:

1. A properly designed engine enclosure *shall* be:

a. Aerodymanically clean

b. Sized and proportioned to the engine and its related subsystems

c. Fastened to the airframe, not the engine. This eliminates problems with metal fatigue associated with engine vibrations.

d. Arranged in such a fashion that the major portion may be opened quickly for inspection and minor repairs, or removed entirely for major maintenance tasks and cowling repairs

e. Adequately ventilated to prevent accumulations of gases, and designed so that accumulations of dirt, waste, or fuel may be observed without removal of cowl sections

f. Properly drained so that no fuel is trapped in any ground or flight attitudes. Any fuel likely to leak into the engine compartment must be drained clear of the helicopter through an appropriate drain system.

2. Appropriate firewalls must be provided to contain fires within the engine cowling or nacelle.

3. Daily maintenance aids should be incorporated if the configuration will permit them. These include work platforms, inspection access doors, and supports to hold cowling or nacelle panels in an open position to ease maintenance operations and protect equipment against accidental damage.

4. Maximum interchangeability of parts *shall* be incorporated into the design.

5. It should be easy for an observer at ground level to determine that the cowling is secured properly.

6. The enclosure should have easily-accessible provisions for fire extinguishing, by ground personnel, during engine starts.

7. All portions of the cowling that might be subjected to exhaust gas impingement and to exhaust flames in the event of an exhaust subsystem failure, *shall* be corrosion-resistant steel, titanium, or other equivalent temperature-resistant alloy material. The material selected *shall* also be determined by the heat transfer analysis considering the engine heat rejection.

8. Cowlings *shall not* interfere with any parts of the engine, its operation, its accessories, or its installation.

9. Cowlings *shall* be designed to provide adequate cooling of the engines and engine accessories during flight and ground operations.

3-2.2 ENGINE MOUNTING

The engine mounts *shall* be designed to withstand the loads resulting from the engine torque, thrust, and gyroscopic couple in combination with all applicable ground, flight, and inertia loads. In addition, engine mounts *shall* withstand transient torque and crash load conditions. The engine mounts and supporting structure *shall* withstand the inertia torque resulting from sudden stoppage of the turbine rotors combined with the flight loads for 3.0 g flight. Torque decay time histories *shall* be determined by analysis of the engine characteristics, but in no case *shall* stoppage be considered as occurring in more than 3.0 sec.

Engine mounting requirements are specified in MIL-E-8593 and *shall* be followed.

Turboshaft-engine-powered helicopters may require critical alignment of high-speed shafts. It is good practice to design a high degree of accuracy into the mounts and supporting structure, and thus eliminate the need for adjustment on installation. This requires more intricate tooling during manufacturing, but insures positive shaft alignment.

For multiengine configurations, interchangeability is desirable and can be achieved by designing the engine mounts so that common detail parts can be assembled to result in opposite assemblies.

The various types of engine mountings may be described briefly as follows:

1. A three-point-suspension type that incorporates a gimbal or ball joint

2. A mounting that cantilevers the engine from the gearbox. Few engines can be cantilever-mounted; consequently, this method will not be discussed.

The front mount of the three-point suspension may be either a single- or two-point configuration. The gimbal arrangement likewise may be a single- or two-point support. The ball joint arrangement, on the other hand, must use the two-point support to obtain torsional restraint for the engine. When the three-point support is used with either the gimbal or ball joint, it must provide engine freedom for thermal expansion in all directions. This is accomplished by providing lateral, axial, and vertical restraint at the two laterally disposed points, vertical restraint at the single point, and torsional restraint through the two laterally disposed points.

Other configurations using the three-point arrangement may place the gimbal or ball joint support alongside the drive shaft instead of concentrically as is commonly the case. Because the engine support and drive shaft no longer are concentric, a simple "trailer hitch" arrangement may be used advantageously for cost savings and to provide for easier drive shaft and coupling inspection and overall maintenance. With this configuration, it is imperative that at least one of the drive shaft couplings be capable of providing adequate axial displacement.

The positive gimbal or ball joint may be replaced by an elastomeric element that supplies vibration isolation in addition to the flexibility of mechanical joints. Isolation mounting systems are discussed in the paragraph that follows.

3-2.3 ENGINE VIBRATION ISOLATION

Chapter 8, AMCP 706-203, specifies that an engine vibration survey *shall* be conducted to determine the engine vibration environment in the helicopter. In addition, a test plan *shall* be prepared, and ground and flight tests conducted to verify that the engine vibration environment is satisfactory.

Successful helicopter design will require a flow of data among the engine manufacturer, the airframe manufacturer, and the procuring activity. This flow and the required data are defined in Chapter 8, AMCP 706-203.

As pointed out in AFSC DH 2-3, a mounting subsystem *shall* be designed so that the natural frequencies of the engine(s), when installed in the helicopter, do not exceed a certain limiting frequency in those modes of motion that may be energized by the vibratory-forcing functions generated during the operation of the helicopter. The natural frequencies *shall not* exceed 70% of the lowest frequency of the forcing function.

3-2.4 FIREWALLS

To provide for isolation of fires, zones that contain both combustible material and a source of ignition must be defined and *shall* be separated from the rest of the aircraft by firewalls. The firewall must withstand a 2000°F flame for 15 min. Sources of ignition may be hot engine surfaces or electrical connections. High pressure ratios and increased cycle temperatures have made virtually the entire engine surface an ignition source. Consequently, the practice of defining the entire engine compartment as a fire zone has evolved.

Stainless steel, at least 0.015 in. thick, is the most commonly used firewall material. However, in applications employing a structural firewall, improvements in weight and cost-effectiveness may be realized by the use of titanium or other suitable material. In such applications the structural requirements usually are predominant, and the material thickness required is easily capable of providing the necessary fire protection.

Firewalls provide the most effective protection when they are kept free of sharp protuberances such as angles, clips, and brackets. This allows the fire-extinguishing system to operate more efficiently.

Engine installations incorporating nacelles usually require that only the interface to the airframe be fireproof. This area, therefore, should be kept to a minimum to achieve minimum firewall weights.

Side-by-side engine installations require a common center firewall, which can be made removable to enhance engine accessibility. When this is done, care must be taken to insure a tight-fitting, rugged seal. All-metal seals appear most attractive for this application. Pliable seals, either butted or lapped, eventually deteriorate, thereby reducing the firewall integrity. The seal provides the removable section with a certain amount of inherent support, facilitating removal or installation. Side-by-side engine installations are not desirable due to vulnerability and survivability considerations.

On each face of firewalls, and immediately adjacent thereto, use should be made of materials of a type that will not ignite as a result of heat transfer from flame on the opposite side of the firewall. Combustible fluid-carrying lines that traverse a firewall *shall* be equipped with shutoff valves.

3-2.4.1 Fire Detectors

Three basic types of detection systems are used: infrared, continuous wire, and spot (thermal sensors).

The infrared or surveillance fire-detection system provides extensive fire zone coverage.

Continuous wire fire-detection systems are of two types: those in which the resistance across a eutectic salt filling an annular space between two conductors is monitored continuously, and those in which increasing pressure of a gas trapped within a sealed line pneumatically actuates a switch. Each of these types is routed throughout the fire zone in the areas where temperature changes caused by fire are likely to occur. The continuous-wire element is subject to vibration and maintenance damage, which can result in false fire alarms. However, continuous-wire systems are not vulnerable to false alarms from sunlight.

The spot type of fire detector, or thermal sensor, actuates a switch to trigger the master fire-warning circuit. This type inherently is more rugged than continuous-wire detectors, but has very limited coverage. As a result, spot detectors in reasonable numbers can be used only in fire zones of limited volume, such as combustion heater compartments.

MIL-D-27729 covers volume surveillance types of flame and smoke detection systems. MIL-F-7872 covers continuous-type fire and overheat warning systems. MIL-F-23447 covers radiation-sensing (surveillance type) fire warning systems.

3-2.4.2 Fire Extinguishing

Almost all recent helicopter designs use high-rate-discharge fire-extinguishing systems. Most systems use vaporous extinguishing agents propelled by a dry charge of high-pressure nitrogen. More recently, some extinguishers have used pyrotechnics as the propellant agent. Inert agents, such as bromotrifluoromethane or dibromodifluoromethane, often are used because of their good extinguishing properties and low toxicity. Furthermore, the low boiling point of the agents facilitates vaporization and distribution within the fire zones.

An effective fire-extinguishing system is one that will, by test, demonstrate 15% by volume agent concentration within the fire zone for a duration of at least 0.5 sec. The system must meet the requirements specified in MIL-HDBK-221.

3-2.5 ENGINE AIR INDUCTION SUBSYSTEM

Two basic tools are useful in the aerodynamic design of engine air induction subsystems. These are:

1. Analog field plotter, which uses an electrically conductive paper and is based on the fact that LaPlace's partial differential equation is identical for an electrical field and an inviscid fluid. This technique yields local streamlines, velocity potential lines, and surface velocities, and is well suited to two-dimensional problems.
2. Potential flow digital computer program, which uses the technique of superposition of sources and sinks to yield the same results as the analog field plotter, but with greater accuracy.

Basic criteria for the aerodynamic design of the air induction subsystem duct, which must satisfy the requirement of the engine model specification, are:

1. The air induction subsystem *shall* prevent any erratic or adverse airflow distribution at all operating conditions and attitudes.
2. The air induction subsystem *shall* have minimal aerodynamic losses. A 0.5-1.0% pressure loss should be attainable in most air induction system designs. When a particle separator is installed (see par. 3-2.5.2), the pressure loss will be higher but should not exceed 2.0-2.5%. Each 1% of pressure loss results in 1.5-2.0% power loss.
3. The air induction system *shall* meet the minimum acceptable engine inlet distortion limits as prescribed by the engine specification. The local total pressure should not differ from the average by more than 5.0%.

Items 1, 2, and 3 are interrelated and pertain mainly to pressure gradients determined by the duct area distribution, duct wall radii of curvature, and changes of duct wall curvature. To insure that these requirements are met, the duct pressure gradient is made favorable for the flow by decreasing the cross-sectional area of the duct along its length and by contouring the walls of the duct to polynomial equations.

3-2.5.1 Air Induction Subsystem Design

The external lip profile is established by fitting an external cowl contour (usually a NACA Series I or an elliptical shape) from the lip tangent point to the inlet envelope boundaries. The inner lip shape usually has an elliptical contour, and the design parameters are given in Ref. 1. Class A Kuchemann-Weber circular intakes described in Ref. 2 yield design parameters similar to those given in Ref. 1. Ref. 2 also suggests a desired design range of inlet velocity to free stream velocity ratio, i.e.,

$$0.4 \leq V_i / V_o \leq 0.65 \qquad (3\text{-}1)$$

where

V_i = inlet velocity, fps
V_o = free stream velocity, fps

Beyond this range, the possibility of leading-edge velocity peaks, and hence flow breakdown at the nose, increases greatly.

Ref. 3 shows that a certain minimum frontal area is needed to keep the external maximum velocity within limits. This criterion is satisfied when

$$\frac{A_m}{A_i} \geq 1 + \frac{(1 - V_i / V_o)^2}{(V_{max} / V_o)^2 - 1} \qquad (3\text{-}2)$$

where

A_m = maximum inlet frontal area, in.^{2}
A_i = area of inlet, inside diameter, in.^{2}
V_{max} = maximum velocity along leading edge surface, fps

In addition, pressure measurements should indicate that variations in inlet total pressure, evaluated in terms of a distortion index, as defined in the engine model specification, are within the required specified values.

3-2.5.2 Inlet Protection

The engine air induction subsystem should be designed, to the maximum practicable degree, so that foreign objects from external sources will not enter induction subsystems. The level of protection required for the engine air induction subsystem is defined during preliminary design. Various engine air particle separators (EAPS) are described in Chapter 8, AMCP 706-201.

An engine air inlet sand and dust protection device, if installed, *shall* meet the criteria specified in Chapter 8, AMCP 706-201.

3-2.5.3 Anti-icing

An induction anti-icing subsystem *shall* be designed to anti-ice satisfactorily the induction system when operating in the icing environments specified in MIL-E-8593. The anti-icing subsystem *shall* function from sea level to the helicopter service ceiling.

If an ice detector is installed in the induction system, the detector *shall* conform to MIL-D-8181.

Any failure of the anti-icing control shall result in the anti-icing subsystem remaining in or reverting to the anti-icing ON mode.

Engine air induction systems can be anti-iced either electrically or by the use of engine bleed air. The former (electrically) can use a nonmetallic duct in which thermoelectric heating elements are embedded. The latter type (bleed air) has used a metallic duct that is formed into a double-skin heat exchanger adjacent to the area requiring thermal protection. These bleed air heat exchangers have been made with and without fins.

The decision between electrical and hot air systems is made for each helicopter on the basis of the required helicopter mission and results of trade-off studies in which the airframe/transmission/engine match is considered to determine the system which results in the lowest aircraft penalty.

3-2.5.3.1 Electrical Anti-icing

Typically, electrically anti-iced helicopter engine air induction subsystems require a variation in local power density from 4.0 to 16.0 W/in.^{2} to account for local variations in surface velocity and moisture impingement rate. Relatively large amounts of electrical power are necessary, which results in a substantial investment in generator power and helicopter weight.

Electrical systems are relatively easy to design and test. The surface temperature of the air induction duct normally is held to 40°F (4.4°C) for the atmospheric design condition the anti-icing system is required to meet. Calculation procedures are contained in Ref. 4.

3-2.5.3.2 Bleed Air Anti-icing

Hot-air-type anti-icing subsystems use compressor bleed air, which must be adequate in quantity and temperature to meet all requirements throughout the power and environmental spectra. An advantage of these systems is that the related power penalty is applicable only on a cold day, when anti-icing is required. In most cases, the helicopter will not be power-limited on a cold day and, therefore, will suffer only a fuel consumption penalty from the use of compressor bleed air.

The bleed air system presents some mechanical problems, however. Fins, if used, must be brazed properly to the outer skins; otherwise, the local fin thermal effectiveness will be zero. A double-skin heat exchanger without fins requires small gap heights whose tolerances become very important. Manufacturing and assembly capabilities also become important design considerations.

In a bleed air anti-icing subsystem, engine bleed air is ducted from the compressor bleed port to a solenoid shutoff valve and then into the intake manifold at the leading edge of the induction subsystem inlet. The flow then impinges on the inlet leading edge, providing the greatest heat transfer at the external flow stagnation point where the thermal load is highest. The flow then passes through heat exchangers along the inner and outer lips of the air intake. The gap height (minimum gap = 0.080 in.) along the induction system flow passages is tapered so that the external skin temperature is maintained close to 40°F. The air then is discharged overboard through discharge slots located at the rear of the outer lip. A thermal switch should be used to monitor duct skin temperature. This switch actuates a warning light if the skin temperature drops below 40°F while the system is in operation. Actuation of the light indicates either a subsystem failure or icing conditions more severe than the subsystem capacity.

3-2.5.3.3 Anti-icing Demonstration

The capability of the anti-icing subsystem must be demonstrated by test. The test requirements are described in Chapter 9, AMCP 706-203.

3-2.6 EXHAUST SUBSYSTEM

The design of the engine exhaust system *shall* meet the following objectives:

1. Minimize pressure loss to reduce engine power loss. Losses usually can be held to a pressure loss of 1% or less. A 1% pressure loss generally results in an approximate 1% power loss.
2. Prevent loss of tail rotor efficiency due to hot exhaust gas flowing through the tail rotor
3. Prevent loss of power due to heating of the inlet air and/or reingestion
4. Prevent overheating of the adjacent structure due to impingement by exhaust gases
5. Provide maximum possible thrust recovery.

Exhaust system assemblies usually are welded or furnace brazed. The brazed assembly offers high resistance to metal fatigue because the strength of the material is not affected appreciably by the brazing operation. Brazing has both advantages and disadvantages. It requires higher initial tooling expenditures, has a potentially lower unit price, and usually is more difficult to repair. By contrast, welds require less tooling and are repaired more easily, but are more prone to metal fatigue because of the metallurgic change in the weld area.

To reduce pressure losses and thus obtain maximum efficiency, the exhaust system should not make abrupt cross-sectional changes. Duct bends should be gradual and an adequate diffusion angle must be maintained. In multiengine helicopters, savings in initial costs and greater availability of spare parts make interchangeable exhaust ducts desirable.

3-2.6.1 Exhaust Ejectors

Engine compartment/engine component cooling is discussed in par. 3-6. However, engine installations requiring positive compartment cooling often use exhaust ejectors as air pumping devices. Although many ejector configurations are possible, two commonly are used for this purpose.

The first configuration provides an annulus for momentum exchange at the downstream end of the exhaust duct. The nacelle that forms the outer surface of the annulus extends beyond the exhaust duct to provide an adequate mixing length for this ejector. Care must be taken to prevent damage to the nacelle through exhaust gas impingement. High-temperature liners often are used in this area. The exhaust duct is of conventional design, mounted directly onto the engine.

The second configuration locates the momentum-exchange annulus at the upstream end of the exhaust duct, the engine serving as the inner ring of the annulus and the duct as the outer ring. This method allows longer mixing lengths and more precise alignment of the annulus rings, resulting in higher operating efficiencies. Two basic designs are possible. One separates the exhaust duct from the engine, thereby reducing metal fatigue of the exhaust duct, which is supported by the nacelle or airframe structure. The other uses standoffs to mount the exhaust duct on the engine. The latter must be used when the engine installation requires a exhaust-duct-imposed load to change engine natural vibration frequencies.

Generally, the selection of a method for providing the necessary cooling airflow will have been made during preliminary design. As discussed in AMCP 706-201, par. 8-2.4, this selection requires examination of the weight penalty and power loss associated with the alternative means. Design considerations pertinent to the integration of an exhaust ejector into the cooling system are reviewed in par. 3-6. Included are references to design procedures and requirements for design, documentation, and demonstration.

3-2.6.2 Infrared (IR) Radiation Suppression

If the missions assigned to a helicopter include the possibility of exposure to IR seeking weapons, an IR radiation suppression subsystem may be required. The extent and type of suppression required will have been defined by the procuring activity and included in the preliminary design. As discussed in Chapter 8, AMCP 706-201, the IR suppression subsystem may be a part of the helicopter or be a separate kit. In either case, the suppression of IR radiation requires reducing the temperature of the heat sources.

The paragraphs that follow discuss only passive countermeasures to IR weapons. Active countermeasures also may be required but, as discussed in Chapter 8, AMCP 706-203, the qualification of such systems by US Army Aviation Systems Command (USAAVSCOM) is limited to the interface between the helicopter and the subsystem.

The principal heat sources are the engine hot parts, exhaust duct, and exhaust plume. However, other heat sources also may produce significant amounts of energy in the IR frequency band. The radiation from other sources (e.g., heat exchanger outlets and solar reflection from windshields) also may have to be reduced to bring the total IR signature of the helicopter within prescribed limits.

3-2.6.2.1 IR Suppression Requirements

The IR suppression requirements for a new Army helicopter will be provided by the System Specification. Typically, the requirement will be stated in the example that follows.

> **IR SUPPRESSION REQUIREMENT**
> **(example)**
>
> The maximum total IR radiation signature of the helicopter *shall* be suppressed to levels not to exceed 3 (2 desired) W/sr in the 3-5 micron bandwidth. The total IR radiation signature is comprised of direct (i.e., visible hot metal parts), indirect or reflected, and exhaust plume radiation. Radiation from the engine and tailpipe and from all secondary sources, such as heat exchanger outlets and fuselage areas washed by exhaust, are included. The suppressed IR radiation signature requirements *shall* be based upon the Army Hot Day Atmosphere (i.e., 125°F at sea level) and *shall* apply with the engine operating at intermediate power and with the helicopter at the gross weight that results in the highest exhaust gas temperature. The signature *shall* be evaluated at lower hemisphere, upper hemisphere, and coplanar viewing angles and the required level of suppression *shall* apply to the viewing angle that results in the maximum signature.

Requirements for IR suppression sufficient to defeat specific threats will be provided in a classified supplement to the system specification. The maximum acceptable signature will be described in more detail, with maximum allowable limits being prescribed for the energy radiated at each of several wavelengths within the IR spectrum.

A comprehensive treatment of military IR technology is given by AMCP 706-127 and AMCP 706-128 (*Infrared Military Systems, Parts One and Two*). Methods for the estimation of IR radiation, with or without suppression, are given by this two-part handbook. Additional references pertinent to IR suppressor design will be provided by USAAVSCOM upon request. However, compliance with suppression requirements will be demonstrated by a IR signature survey (see par. 8-9.2, AMCP 706-203).

3-2.6.2.2 Exhaust Suppressor

As noted previously, the principal sources of IR radiation from a helicopter are the visible hot parts of the engine and the engine exhaust. Therefore, the most important part of the IR suppression subsystem generally is the exhaust suppressor. Depending on the missions assigned to the helicopter, the suppressor may be an integral part of the helicopter or it may be designed as a kit to be installed only when the helicopter is in a combat role in which engagement by IR seeking weapon(s) is probable. Further, the exhaust IR suppressor may be developed by the engine contractor and provided as an engine accessory, or the helicopter manufacturer may be responsible for the development of the suppressor and of the installation.

Design of an exhaust IR suppressor is a complex process. The alternatives available are discussed in Chapter 8, AMCP 706-201, and the trade-offs among the possible power losses and weight increases also are discussed. The methods of cooling a suppressor are described and the possible use of low emissivity coatings also is discussed. Further explanation of the heat transfer processes, including pertinent equations and values for the applicable material properties and other constants, is given in Ref. 4.

Typically, the first requirement of an exhaust suppressor is to shield the hot engine parts from external view. The shield in turn is itself heated by impinging exhaust gas. The effectiveness of the shield as an IR radiator then must be reduced by cooling or by control of the emissivity of the surface of the shield, or both. However, the temperature of the exhaust plume can be reduced only by cooling and this really can be accomplished only by dilution: by mixing cooler air with the plume. Depending on the specific suppression requirements, the amounts of cooling air may be quite large and the power required to provide the necessary airflow also may become large. Ref. 5 describes a study of advanced engine/transmission/airframe integration concepts capable of meeting a stringent IR suppression requirement with minimum impact upon propulsion system weight and power required for specified mission performance.

Whether an exhaust IR suppressor is made an integral part of the engine installation or a removable kit, the installation must be sound structurally and must not affect adversely the vibration environment of the engine or of the helicopter. Further, the installation of an exhaust IR suppressor on a helicopter *shall not* result in a reduction of engine or engine compartment cooling below minimum required levels under any normal operating condition. Also, the installation *shall not* reduce appreciably the power available from the engine, or increase appreciably either the power required from the engine or the SFC based on power required. Typically, a power loss not in excess of 3% of the power required to hover at the specified hot-day condition (e.g., 4000 ft, 95°F) is acceptable, although the loss allowable in a given case will depend upon the suppression levels specified. In any case, the installation of any device to suppress the IR signature *shall not* prevent the compliance of the helicopter with the performance requirements of the system specification.

3-3 PROPULSION CONTROLS

Control of a helicopter propulsion system requires consideration of the characteristics of the helicopter rotor as well as the engine. This control system is defined during preliminary design, and the requirements are discussed in detail in Chapter 8, AMCP 706-201. The detail design engineer must interface the control system design with the engine manufacturer to assure that the operation and performance defined in preliminary design are attainable. Accordingly, the continuing relationship between engine and airframe manufacturers is paramount during the detail design phase. Detailed discussion regarding this interface relationship is provided in Chapter 8, AMCP 706-203.

3-4 FUEL SUBSYSTEM

3-4.1 GENERAL

The fuel subsystem consists of tanks, refueling/defueling features, fuel feed and vent lines, fuel pumps, valves, fuel gaging components, and associated items such as fuel tank compartment structure, drainage provisions, and protection materials.

A typical fuel subsystem is illustrated diagrammatically in Fig. 3-4.

Definition of the fuel subsystem occurs during the preliminary design and is discussed in Chapter 8, AMCP 706-201. Detail design of the fuel subsystem *shall* be in accordance with MIL-F-38363, except as modified by the procuring activity. The fuel subsystem *shall* be crashworthy in accordance with Ref. 6 and MIL-STD-1290. Design and installation features required for crashworthiness are described in Chapter 8, AMCP 706-201. All components and structural areas (e.g., tanks, vents, and certain dry bay areas) in which air/fuel mixtures may be exposed to possible ignition sources, *shall* be adequately protected from fire or explosion. MIL-F-38363 also contains requirements for the environmental conditions under which the fuel subsystem *shall* operate satisfactorily. Unless the system specification provided by the procuring activity prescribes otherwise, the ranges of ambient air temperature and fuel temperature and the maximum water content of the fuel *shall* be as specified in MIL-F-38363.

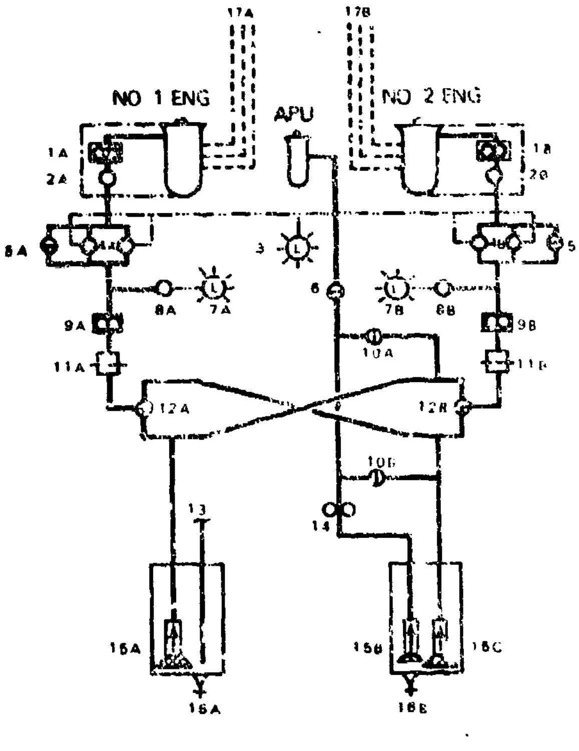
For maximum reliability, it is desirable that the operation of fuel-feed subsystems be functionally independent of the operation of other helicopter subsystems, specifically the electrical, hydraulic, and pneumatic subsystems. Feed systems must deliver to the engine(s) under all possible design operating conditions of the helicopter, both ground and flight, the necessary fuel flow at the pressure specified by the engine model specification. Where possible, suction fuel-feed subsystems should be used in lieu of pressurized systems for increased safety and reliability.

In the paragraphs that follow the principal design requirements applicable to key components and functions of the fuel subsystem are described and discussed. Reference should be made to MIL-F-38363 for additional requirements applicable to the materials, hardware, and components used in a fuel subsystem as well as to the installation of the subsystem in the helicopter. The specification also describes the data that must be provided to permit the procuring activity to evaluate the fuel subsystem design for a new helicopter.

3-4.2 FUEL SUBSYSTEM COMPONENTS

3-4.2.1 Fuel Tanks

Main fuel tanks normally consist of one or more bladder-type cells interconnected to form a tank of specific capacity. Cells *shall* be constructed in accor-



1AB — ENGINE DRIVEN FUEL PUMP
2AB — FUEL FILTER
3 — FILTER IMPENDING BY-PASS WARNING LIGHT (AIRFRAME MOUNTED DUAL FILTERS)
4AB — AIRFRAME MOUNTED DUAL FUEL FILTERS
5AB — DUAL FILTER BY-PASS VALVE (ELECTRICAL)
6 — APU SHUTOFF VALVE (MANUAL)
7AB — BOOST PUMP FAILURE WARNING LIGHT
8AB — BOOST PUMP PRESSURE SWITCH
9AB — ENGINE-DRIVEN BOOST PUMP
10AB — SYSTEM PRIME VALVE (MANUAL)
11AB — FIREWALL QUICK DISCONNECT
12AB — SELECTOR VALVE (MANUAL)
13 — HEATER SUPPLY LINE
14 — WOBBLE PUMP
15ABC — CHECK VALVE WITH SCREEN
16AB — SUMP DRAIN VALVE (MANUAL)
17AB — ENGINE DRAINS

LEGEND

——— FUEL SUPPLY
—·—·— ELECTRICAL
—··—··— ENGINE SUPPLIED COMPONENT ENVELOPE
- - - - - ENGINE DRAINS

Figure 3-4. Typical Fuel Subsystem

dance with MIL-T-27422. The type, protection level, and class as defined by the specification *shall* be appropriate to the application (e.g., Type I for self-sealing, Type II for nonself-sealing). The tank configuration and installation *shall* comply with MIL-F-38363 and MIL-T-27422. These specifications contain the requirements for liquid-tight structure surrounding self-sealing tanks and the installation of backing boards to protect flexible self-sealing tanks.

To prevent overflow each tank *shall* contain expansion space equal to no less than 3% of the total fuel volume of the tank when the helicopter is in a normal ground attitude. Gravity filler openings *shall* be so located that all tanks can be filled without overfilling into the expansion space. For pressure refueling systems the level shutoff valve *shall* prevent filling of the expansion space. Each filler opening cap *shall* be in accordance with MIL-C-38373. All tanks *shall* be provided with a low-point drain for fuel sampling and defueling purposes.

External fuel can be contained in tanks complying with MIL-T-7378 or MIL-T-18847. The installed location should permit service personnel standing on the ground to inspect visually and service the tanks. Inflight jettisoning should not affect the helicopter adversely. To minimize combat turnaround time, all external tanks should be readily removable and replaceable without helicopter disassembly.

3-4.2.2 Fuel Tank Vents

Each fuel tank *shall* be vented to the atmosphere through lines whose capabilities are compatible with the performance of the helicopter, without producing tank pressures detrimental to the helicopter structure or to the tank.

If a pressure refueling system is required, the venting capacity of each tank also *shall* be sufficient to discharge the maximum rate of fuel flow, without excessive tank pressure in the event that the refueling system shutoff valves fail in an open position. Traps must be avoided, and the subsystem should be operable with the helicopter afloat if amphibious operations are required. If vent valves are used to prevent spillage, they *shall*, as required by MIL-STD-1290, close when the helicopter is in a position of extreme attitude. The vent lines may be designed to prevent spillage without valves by traversing three directions, etc. See Ref. 6 for vent line design details.

3-4.2.3 Fuel Gaging

In accordance with MIL-F-38363, a fuel-gaging system that meets the requirements of MIL-G-26988 *shall* be provided. It *shall* be installed in accordance with MIL-G-7940. System indication of total fuel quantity and of the quantity in each main tank *shall* be continuous. Design of the gaging system and fuel cell interface must preclude gaging system puncture of the fuel cell during crash conditions.

Each main tank *shall* contain a device independent of the fuel gaging system to provide a low-fuel warning. The quantity of fuel remaining at the moment of actuation of the low-fuel warning must be sufficient to allow the engine to operate for 0.5 hr at maximum-range power unless otherwise specified.

3-4.2.4 Refueling and Defueling

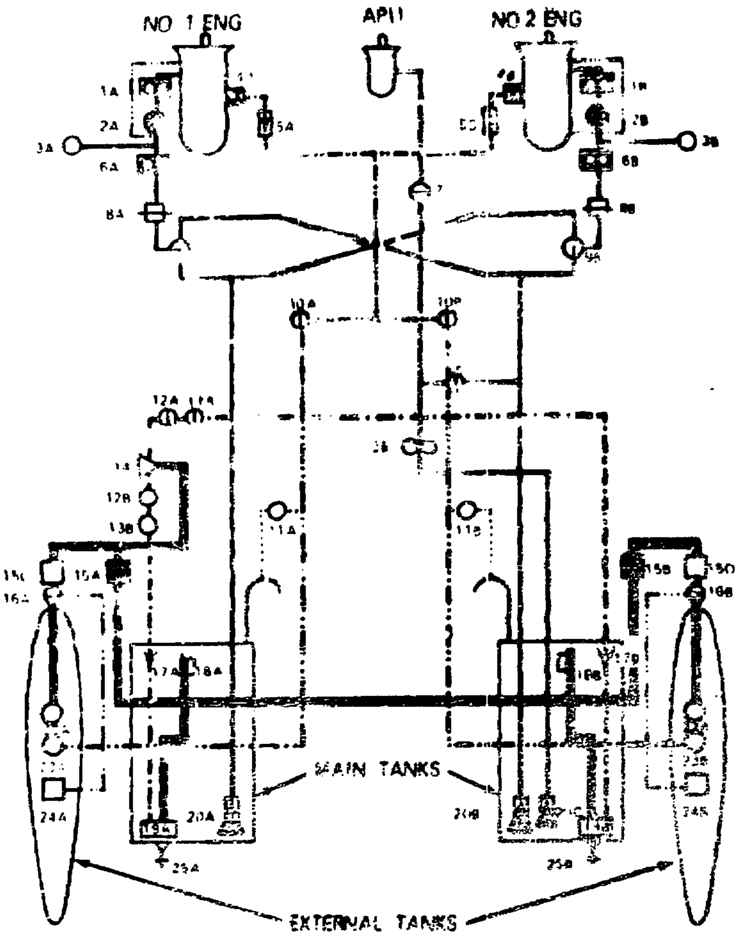
The refuel/defuel features for any given fuel subsystem *shall* be specified by the procuring activity. However, the design criteria for such features *shall* be in accordance with MIL-F-38363.

All helicopters *shall* be capable of being refueled on the ground, using a gravity refueling system, without external power being applied. Unless the tanks are too small for the rate to be practical, or the procuring activity has specified another rate, the fuel system *shall* be capable of being refueled at a continuous rate of 200 gpm without any operations other than removing the filler cap and connecting the fueling nozzle bonding plug being required. During the tank topping portion of the refueling, the flow rate may be less than 200 gpm.

If the internal fuel capacity is 600 gal or more, a pressure refueling system *shall* be required (MIL-F-38363). In this case, it *shall* be possible to refuel all tanks from a single connection, the fuel lines being such as to allow the fuel level in all tanks theoretically to reach the full position simultaneously. On the other hand, the subsystem design must be such that it is possible to fill selectively any individual tank or to avoid filling any given tank. Operation of shutoff valves, plus the precheck system, must not depend upon the use of external power. To prevent excessive surge pressure within the refueling system, shutoff valve closure time may be controlled, or appropriate pressure relief valves incorporated. Additional system requirements are given in MIL-F-38363.

Defueling of tanks *shall* be possible using the pressure refueling system when installed. When the defueling adapter (MIL-A-25896) is not used also for refueling, positive means, such as a check valve, *shall* be installed to prevent refueling through the adapter. A typical pressure refueling system is illustrated in Fig. 3-5.

A sump *shall* be provided in that portion of each fuel tank which is lowest when the helicopter is in the normal ground attitude. A drain valve for fuel sampling and for removal of sediment and water *shall* be provided in each tank sump (MIL-F-38363).


NO 1 ENG
APU
NO 2 ENG
1A
2A
3A
6A
8A
5A
1B
2B
3B
6B
8B
9A
7
10A
10B
12A
11A
14
12B
13B
15C
16A
15A
11A
11B
15B
15D
16B
17A
18A
18B
17B
20A
20B
24A
24B
25A
25B
26
MAIN TANKS
EXTERNAL TANKS

1AB	ENGINE-DRIVEN FUEL PUMP
2AB	FUEL FILTER, ENGINE MOUNTED
3AB	BOOST PUMP PRESSURE SWITCH
4AB	RESTRICTER AIR
5AB	CHECK VALVE (AIR)
6AB	ENGINE-DRIVEN BOOST PUMP
7	APU SHUTOFF VALVE
8AB	FIREWALL QUICK DISCONNECT
9AB	SELECTOR VALVE
10AB	VALVE SHUTOFF PRESSURE REGULATOR & PRESSURE RELIEF)
11AB	VALVE VENT (ELECTRICAL)
12AB	PRECHECK VALVE (MANUAL) TURN TO CLOSE, NORMALLY OPEN
13AB	PRECHECK VALVE (ELECTRICAL) ENERGIZE TO CLOSE, NORMALLY OPEN
14	REFUELING CONNECTION (GROUND OR AIR TO AIR)
15ABCD	FLOW SENSORS
16AB	VALVE, AUX TANK SELECTION AND PRESSURE REFUELING SHUTOFF VALVE (ELECTRICAL)
17AB	HIGH LEVEL SENSOR
18AB	SURGE RELIEF VALVE
19AB	PRESSURE REFUELING SHUTOFF VALVE
20ABC	CHECK VALVE WITH SCREEN
21	ENGINE PRIME VALVE
22AB	DROP VALVE, FUEL
23AB	DROP VALVE, AIR
24AB	FLOAT SWITCH
25AB	SUMP DRAIN VALVE
26	WOBBLE PUMP

LEGEND

▬▬▬	PRESSURE REFUELING & DEFUELING
─────	VENT
─ ─ · ─	PRECHECK & SHUTOFF
─ ·· ─	BLEED AIR
─ ─ ─	ELECTRICAL
━━━	FUEL SUPPLY

Figure 3-5. Typical Fuel Subsystem With Pressure Refueling

3-4.2.5 Fuel-dumping

If inflight fuel-dumping is required for rapid reduction of helicopter gross weight, the discharged fuel *shall* not impinge upon or re-enter the helicopter, or be discharged into either an area of static electricity discharge or the engine exhaust plume. Unless the fuel tanks can provide a significant head for gravity dumping, pumps *shall* be used to achieve the desired dumping rate. The tank vent line capacities must be reviewed to prevent fuel cell collapse while dumping.

3-4.2.6 Engine Feed System

The feed system, whereby fuel is delivered to the engine(s), *shall* be designed in accordance with MIL-F-38363. In general, fuel must be available on an uninterrupted basis without continuous attention of the crew. The feed system must allow normal performance of the engine(s) under all attitudes normal for the helicopter on the ground, and in both steady and maneuver flight at all altitudes up to and including the service ceiling.

Typically, the fuel feed system consists of a main tank or tanks feeding directly to the engine(s) and such transfer tanks as are necessary to meet the fuel capacity requirements of the helicopter within the weight and space limitations of the configuration. On multiengine helicopters a separate, independent, main tank and feed system *shall* be provided for each engine. These independent systems also must be so designed that fuel from any tank can be fed to any or all engines.

Two independent and isolated methods *shall* be provided to move fuel out of each tank, except only one method need be provided for jettisonable external tanks. Each method of moving the fuel *shall* meet the flow requirements for the particular tank under all required engine and helicopter operating conditions (altitude and attitude, including maneuvers). Fuel *shall* be provided to the engine at conditions specified in the engine specification. The suction feed capability *shall* be determined for fuel temperatures specified in MIL-F-38363 or as specified by the procuring activity.

When the system includes transfer tank(s) as well as main tank(s), the normal sequencing of intertank feed must maintain the CG of the fuel system within acceptable limits throughout the range of fuel loads from full to empty at all required flight conditions.

To insure that the particle sizes of contaminants in the fuel do not exceed the limits given in MIL-E-5007, it may be necessary for the engine feed system(s) to include filters or strainers. If so, the strainers *shall* be in accordance with MIL-S-8710 and the installation *shall* be in accordance with MIL-F-38363.

Additional requirements and guidelines applicable to the design and installation of feed systems that will be exposed to enemy ground fire and for which self-sealing tanks are required also are given by MIL-F-38363.

3-4.2.7 Fuel Drains

The fuel subsystem design *shall* include adequate drainage provisions in accordance with MIL-F-38363. Sufficient drains are required so that all low points of the system can be drained. All nacelles, bladder tank cavities, dry bays, and pockets and traps in the structure where fuel may collect *shall* be drained to the exterior of the helicopter. Drain holes or drain tubes *shall* be 0.38 in. diameter minimum, except the drains for structure surrounding self-sealing tanks *shall* be 0.50 in. diameter minimum. Fuel drains *shall* not be interconnected with drain lines carrying other liquids. The installation of fuel drains *shall* be such that under no operating condition will drainage re-enter the helicopter or come in contact with either the engine exhaust gas wake or wheel brakes. In those isolated cases when the passage of a fuel line through the occupied portion of the helicopter cannot be avoided, any joints in the line within the occupied area *shall* be shrouded and drained.

3-4.2.8 Controls and Instrumentation

The controls for the fuel system *shall* be grouped in the cockpit in a functional manner. A simplified diagram of the fuel system *shall* be inscribed on the panel so the function of each switch is indicated clearly.

As a minimum, presentation of the following performance data for the fuel system *shall* be provided in the cockpit:

1. Fuel quantity, each main tank and total
2. Low fuel-level warning for each main tank
3. Bypass warning for each fuel filter
4. Low fuel-inlet pressure warning for each engine
5. Indicator lights for electrically operated fuel shutoff valves.

As additional auxiliary fuel systems are added, appropriate controls and instrumentation *shall* be provided.

3-4.3 TESTING

Substantiating the capability of the fuel system to fulfill its functional requirements during all phases of aircraft operation is required by MIL-F-38363. Such verification *shall* occur in three phases:

1. Component testing
2. Fuel system simulator testing
3. Ground and flight testing.

In addition to testing the complete fuel subsystem, all components must be qualified in accordance with MIL-F-8615, and must be so qualified or have passed safety of flight tests approved by the procuring activity before ground and flight tests are conducted. Fuel subsystem demonstration requirements are described in Chapter 9, AMCP 706-203.

3-5 LUBRICATION SUBSYSTEM

Engine lubrication subsystems may be an integral part of the engine, thereby eliminating various connections to the airframe and conditions that may lead to oil contamination when changing engines. Lubrication oil may be contained in an engine-mounted oil tank and cooled by a heat exchanger.

Engines lacking an integral lubrication subsystem require the addition of an oil reservoir, lines, instrumentation, and a cooler — if an engine heat exchanger is not available — to cool the oil. Airframe-mounted oil reservoirs *shall not* be located in the engine compartment.

A lubrication subsystem integral to the engine will have been tested completely during engine qualification. Component testing of nonintegral subsystems will be necessary in accordance with MIL-O-19838 to substantiate proper rates of oil flow, pressure, temperature, and deaeration. Engine lubrication subsystem demonstration requirements are specified in Chapter 9, AMCP 706-203.

3-6 COMPARTMENT COOLING

Airflow through the engine compartment is required to prevent the engine; engine-mounted accessories; other components, equipment, or fluids within the compartment; and/or surrounding structure from exceeding maximum allowable temperature limits. The maximum allowable temperatures normally will be given in the applicable engine or equipment specification but additional limits may be prescribed by the system specification. Temperatures must be kept below the allowable limits under all operating conditions, both ground and flight, prescribed for the helicopter for all ambient air conditions between the hot and cold atmospheres (temperature as functions of altitude) given as limits by the system specification. Further, the maximum compartment or component temperature limits *shall not* be exceeded following engine shutdown from any operating condition with ambient air conditions anywhere within the prescribed limits.

Heat rejection requirements for the engine and its components will be provided by the engine specification. The amounts of heat rejected by other accessories or equipment installed within the engine compartment may be provided in the applicable equipment specifications. In the case of transmissions and gearboxes, developed by the helicopter manufacturer, the heat rejection rate must be calculated, based on design values for gear-mesh and bearing efficiencies, and later confirmed by test (see Chapter 4). When the heat rejection rates are known, surface temperatures of individual heat-producing components can be calculated on the basis of free air convection at the surface. Ref. 4 contains a section which treats each of the fundamental heat transfer mechanisms — i.e., conduction, convection, and radiation — in considerable detail. The equations and calculation procedures for both steady-state and transient heat transfer problems are given, together with tables and charts of values of the physical properties of materials needed in the calculations. Should the information be inadequate for a given problem, an extensive list of references also is provided.

The quantity of cooling airflow required for adequate cooling of the engine compartment also must be determined analytically by heat-transfer calculations. This flow usually must be obtained by forced convection during operation of the engine, with the residual heat remaining at shutdown being dissipated by free convection. The calculation of heat balance within the compartment is complex, with consideration of all three heat-transfer mechanisms being required. In the design of the cooling subsystem, it is necessary to assure that the airflow over large surfaces such as the engine is such that the temperatures are approximately uniform. Large temperature differentials can result in differential expansion and hence warping of the engine case. Such a condition, which can cause excessive loads on engine bearings and hence premature engine failure, must be avoided.

As mentioned in par. 3-2.6.1, an engine exhaust ejector is a convenient means for pumping compartment cooling air during operation. The design of an ejector cooling system *shall* be coordinated with the design of the engine exhaust system, with care being taken that the installation does not cause excessive power loss or adversely affect engine operation by producing an unacceptably high pressure at the engine exhaust. In any case, the engine exhaust system *shall* meet the requirements given in par. 3-2.6 (this handbook) and Chapter 8, AMCP 706-201. Procedures for the design of an ejector, or jet pump, are given in Ref. 4. For additional ejector design information see "Performance of Low Pressure Ratio Ejectors for Engine Nacelle Cooling", AIR 1191, *Society of Automotive Engineers*, November 1971. Procedures for design, including determination of the power requirements for fans also are given in Ref. 4.

Compliance of the engine compartment cooling subsystem must be demonstrated by test. In addition to the other temperature limits, all surfaces exceeding 400°F *shall* be protected from fuel spillage. This can be accomplished by use of drip fences or by cooling the surfaces to a temperature below 400°F. The requirements for a propulsion system temperature survey are given in Chapter 8, AMCP 706-203. A propulsion system temperature demonstration, described in Chapter 9 of AMCP 706-203, may require further testing in addition to the temperature survey.

3-7 ACCESSORIES AND ACCESSORY DRIVES

Helicopter design requirements result in minimum engine accessory drive requirements. Except for the engine starter and tachometers, most accessories are driven directly from the main gearbox. This is to take advantage of the ability of the helicopter to autorotate in the event of engine failure. If engine failure occurs, accessory power does not fail because the main gearbox continues to turn and supply power. Installation of helicopter accessories is discussed in Chapter 4.

3-8 AUXILIARY POWER UNITS (APU's)

3-8.1 GENERAL

The requirement for an APU will be established during preliminary design. The paragraph describes design and installation requirements for APU's and refers to pertinent qualification requirements.

Emphasis is placed upon the single-shaft APU configuration because of its wide use. Due to the trend in helicopter design toward such items as pneumatic main-engine starting, air conditioning, avionic cooling, IR radiation suppression, purging of main engine inlet-protection systems, air supply for anti-icing, and the availability of air-driven accessory motors (boost pumps, etc.) additional emphasis is placed upon the bleed air type of APU. New helicopter designs have favored this type because of lower overall system weight, despite the lower energy-transmission efficiency of the pneumatic main engine-starting system. The bleed air type of APU usually incorporates an integral gearbox capable of driving small electrical generators and pumps and, therefore, provides emergency system power of all types.

Several APU configurations can be selected to supply pneumatic power (combined with small amounts of shaft power). Four configurations are compared in Table 3-1 as to geometric shape, weight, cost, and reliability. Selection of configuration based upon shape will depend upon helicopter space available. Weight and volume are comparable; but cost, reliability, maintainability, and life cycle costs favor the single-shaft combination bleed APU type. Modern single-stage centrifugal compressors can produce 4.0:1 and higher pressure ratios with a wide range of flow between choke and stall.

Although the driven compressor may produce higher pressures, variable inlet guide vanes or diffuser vanes may be necessary to obtain the required flow range.

3-8.2 APU INSTALLATION DETAILS

In considering the detail design for installation of the APU, all interfaces with the helicopter *shall* be treated. These include APU mounting; inlet, exhaust and bleed air ducting; compartment cooling, and appropriate APU subsystems.

3-8.2.1 Method of Mounting

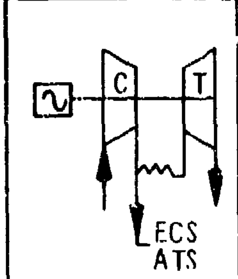
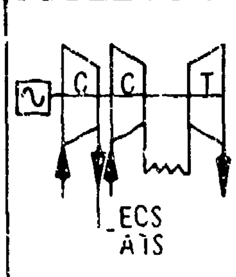
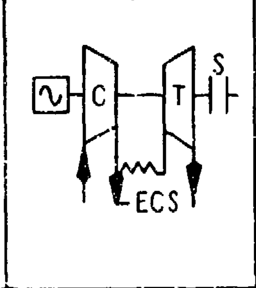
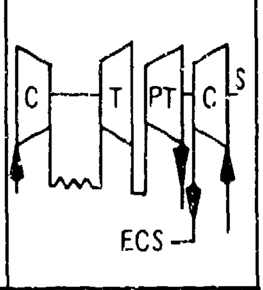
The APU mounting subsystem *shall* be capable of withstanding all flight maneuver forces, providing for thermal growth, isolating vibration (when required), and being arranged for ease of maintenance (especially for rapid installation and removal).

Driven equipment usually is APU-mounted, thus removing the problem of alignment. In some cases, however, a straight-through external drive shaft is used to transmit shaft power into an auxiliary or combining gearbox. In such cases, a flexible coupling must be provided.

The APU often can be supported rigidly. This is done conveniently by means of a three-point support arrangement. Two pin-type mounts on each side of the APU limit vertical, axial, and horizontal motion, but provide lateral freedom for thermal expansion. A single gimbal then is used to support vertical loads, while giving thermal freedom radially and axially.

In most helicopter applications, vibration isolation has not been required because the APU is able to withstand flight loads and vibrations without shock mounts. Experience has shown that frequencies up to 500 Hz are significant, but that APU susceptibility is highest from 5 to 100 Hz. The APU must be capable of withstanding the complete aircraft vibration spectrum in both operating and nonoperating modes. When the APU is not operating, normal loads are not available to stabilize parts in position. Hence, it is possible for external vibration to cause unusual motion of internal parts, thus resulting in excessive wear and premature failures. APU components of particular interest in this regard are the combustor liner assemblies, the aerodynamically located internal

TABLE 3-1
APU TYPES FOR MAIN ENGINE STARTING ENVIRONMENTAL CONTROL, AND ELECTRICAL SUPPLY

	SINGLE-SHAFT COMBINATION BLEED	SINGLE-SHAFT DRIVEN COMPRESSOR	SINGLE-SHAFT COMBINATION BLEED + TORQUE CONVERTER	TWO-SHAFT PT DRIVEN COMPRESSOR
				
LENGTH	100%	107%	110%	110%
DIAMETER	100%	85%	100%	84%
WEIGHT	100%	97%	100%	100%
COST	100%	125%	125%	160%
RELIABILITY	100%	91%	70%	73%

C-COMPRESSOR
T-TURBINE
ECS-ENVIRONMENTAL CONTROL SYSTEM

ATS-AIR TURBINE STARTER
PT-POWER TURBINE
∿-ALTERNATOR
S-STARTER

seals, the gear train, and the nonpreloaded bearing assemblies. Because some installations do not require airborne APU operation, considerable time may be accumulated in this nonoperating mode.

In addition to the requirements of MIL-P-8686, the APU qualification test *shall* include vibration tests in operating and nonoperating modes. These *shall* be established based upon the vibration spectrum at the APU mounting points and not merely upon the frequencies of the main rotors.

The trend in maintenance philosophy is toward minimum scheduled maintenance of the APU, but with the design adapted for rapid APU removal in case of malfunction. Hence, quick-removal connections *shall* be used, with airframe components and support structures arranged so that the unit can be removed without removal of other equipment.

Minimum system weight usually will dictate that the APU should be located near the main engines. A multiengine aircraft, with air ducting required to start each engine, suggests a submerged APU position for minimum weight. Pod or surface locations can be used when fuselage space is critically limited. Also to be considered is the advantage of a minimum length of high-temperature exhaust ducting, even with some sacrifice in inlet or bleed duct lengths.

3-8.2.2 Inlet Ducting

An inlet air collector or muff often is designed and provided by the APU manufacturer. The airframe ducting then runs between the fuselage surface and the collector connection. Ducting pressure losses will affect APU available power directly. A typical relationship is illustrated in Fig. 3-6. For any output shaft power level, a correction factor is given showing the power loss for each unit of pressure loss in the airframe ducting. Values are given for both inlet and exhaust losses. The APU manufacturer may prescribe limits on these losses in a form such as is shown in Fig. 3-7. If the sum of inlet and exhaust losses falls below the curve, the requirements have been met.

The APU air collector design may be critical, especially when transsonic compressors are used. Entry conditions to the inducer will have important effects on compressor efficiency. Collector design can be compromised by the location of accessories, but an attempt should be made to maintain a uniform total pressure distribution around the compressor inlet.

FUEL FLOW PARAMETER CORRECTION:

$$W_f/\delta \text{ CORR.} = W_f/\delta \left(\frac{P_a - \Delta P \text{ INLET}}{P_a}\right), \text{ lb/hr}$$

WHERE

W_f/δ CORR. = FUEL FLOW PARAMETER CORRECTED FOR INLET LOSS, lb/hr
W_f/δ = FUEL FLOW PARAMETER WITH ZERO LOSSES, lb/hr
P_a = AMBIENT PRESSURE, psia
ΔP INLET = INLET PRESSURE LOSS, in. H_2O
δ = $P_a/14.7$

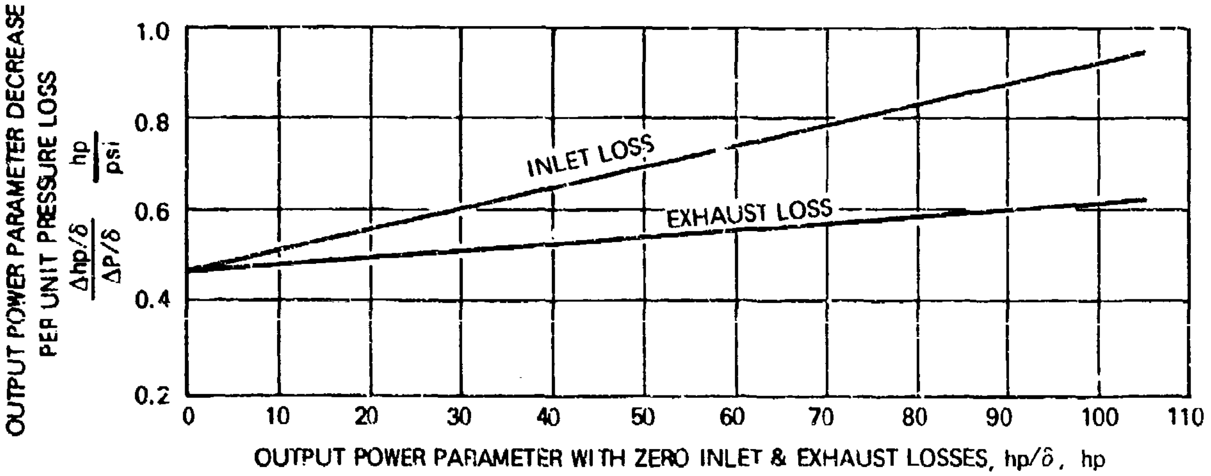


Figure 3-6. Performance Corrections for Duct Losses

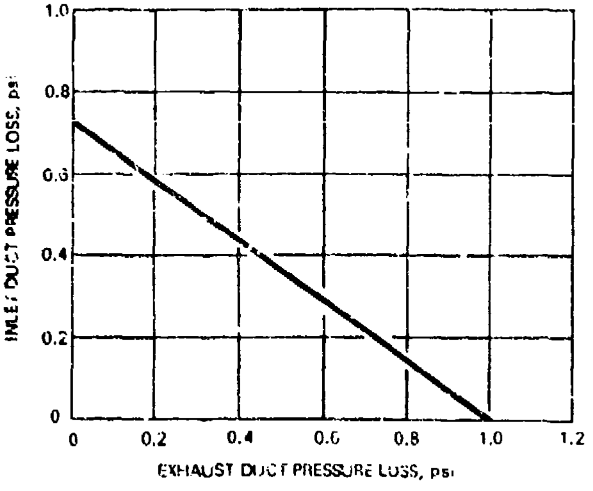


Figure 3-7. Allowable Combined Inlet and Exhaust Duct Pressure Losses

possibly stabilized and directed by splitters or baffles. The APU will have channels to provide controlled acceleration of air from the collector into the inducer inlet. The airframe ducting should include a straight section of several diameters in length leading into the APU collector. This will reduce the effects of ram air stratification and cavitation in the ducting.

If a device is incorporated to protect the inlet against dust or foreign object damage (FOD), this may affect the velocity profile and the resultant pressure losses must be coordinated with the APU manufacturer so that model specification performance is maintained. One technique is to use directional changes of the total duct system for separation of larger particles (>200 microns).

An intake screen *shall* be used for FOD protection. This may be located at fuselage entry, APU collector entry, or compressor entry. The discussion of the engine air induction subsystem in par. 3-2.5 is equally applicable to the APU inlet.

3-8.2.3 Exhaust Ducting

Exhaust ducting design will be related to APU compartment cooling arrangements. The gas-exit velocity can be used as the primary jet for an ejector. Secondary air is taken from the APU compartment. Also, the ducting is a substantial heat source and should be located to minimize effects on compartment temperature.

If compartment ventilation must be limited due to fire hazards, high-temperature bellows can be used to seal ducting. Also IR radiation suppression may be needed for the APU exhaust. Compartment and duct cooling methods used on the main engines will apply equally here and are discussed in pars. 3-2.6 and 3-6.

3-8.2.4 APU Bleed Air Ducting

In the case of a bleed air type of APU, a collector arrangement will be integral with the unit. The APU control valve can be mounted directly on the APU bleed collector flange. If installation space limitations exist, or valve weight will overload the APU flange, the valve may be line-mounted.

A flexible bellows can be used to minimize flange loads, thus accommodating vibration, thermal growth, and installation tolerances. A quick-disconnect type of connection is desirable for maintainability. Attention must be paid to duct pressure loads. APU model specification performance is based on pressure, flow, and temperature at the APU bleed collector flange.

3-8.2.5 Cooling

Surface temperature limits will be specified in the model specification. During APU operation, the compartment cooling system *shall* cool the compartment adequately. Temperature transients should not exceed the limits specified in the model specification.

APU firewalls are identical to the main engine requirements discussed in par. 3-2.4.

The reduction drive, accessories, and lubrication system represent a considerable heat source. Heat can be dissipated into compressor inlet air or into the APU compartment. An oil (to air) cooler may be needed to maintain system oil temperatures within acceptable limits.

A typical small bleed air APU may reject about 150 Btu/min to the lubrication system at sea level pressure and 130°F. An exhaust ejector could provide a cooling airflow rate of about 5 lb/min. Compartment cooling is discussed further in par. 3-6.

3-8.3 APU SUBSYSTEMS

Inasmuch as APU subsystems are similar to those previously discussed in this chapter for the main engine(s), the paragraphs that follow discuss only those characteristics peculiar to the APU.

3-8.3.1 Electrical Controls

APU electrical controls are categorized as sequencing, protective, and load, or output, controls. Some components are APU-mounted and others are airframe-mounted. Electrical power to activate the system may be provided by the aircraft system or by an APU-driven generator. APU controls *shall* be qualified concurrently with qualification of the APU in accordance with MIL-P-8686.

3-8.3.1.1 Sequencing Controls

The most advanced APU sequencing is done with solid-state electrical equipment. A speed signal is obtained from the gearbox or power section by means of a frequency signal from a magnetic pickup, tachometer generator, output alternator, or mechanical speed switch. A frequency-sensitive sequencer then actuates relays for starting and APU acceleration.

The start is initiated by a switch that actuates the APU starting system. This may be electrical or hydraulic (par. 3-8.3.5). The APU will begin rotation without fuel or ignition to provide a momentary air purge of the air and gas passages. Fuel and ignition subsystems must be actuated at the lowest possible speed, perhaps 5% or below, to insure good starts at both cold and hot extremes of ambient temperature. Actuation can be accomplished by a fuel pressure switch, a time delay relay, or a speed signal. Some combustion systems require separate start and main (or run) fuel systems. At 10-20% speed, the main fuel valve will be opened. A third speed point can be used to turn off the starter. At approximately 90% speed, start fuel and ignition will be turned off. This signal, with a time delay relay, also can arm the aircraft load circuits. A fifth sequencing point at 110% speed provides for protective shutdown.

3-8.3.1.2 Protective Controls

Protective controls are required to confine malfunctions to the APU and, thereby, to protect the helicopter.

Protective devices may include overspeed, exhaust overtemperature, and compartment overtemperature subsystems. A simple thermocouple sensor in the APU tailpipe, feeding to the solid-state circuit control (sequencer), provides APU overtemperature protection. Similarly, thermocouples within the compartment can signal the sequencer as fire protection.

Built-in test equipment should be used only to the extent necessary to indicate an APU failure. Indicating lights *shall* be used for pilot advisory purposes. Audio annunciators can be adapted for the APU if desired.

3-8.3.1.3 Output Controls

The electrical load-control circuit usually is armed by the 90% speed sequencing point and a time delay relay. Acceleration is rapid from 90 to 100% speed for most shaft power APU's (1 sec or less). The bleed air APU, however, may have a bypass fuel valve to reduce the acceleration time between 90 and 100% (about 4 sec). The bleed air APU may use a load-control valve to regulate bleed air as a function of exhaust gas temperature. A thermocouple in the APU tailpipe can signal a solid-state circuit component, which, in turn, gives a modulating signal to the load-control valve. The valve will open so that maximum continuous exhaust gas temperature is maintained. Thus, maximum bleed air available from the engine is obtained. Large shaft power requirements require careful examination of transient operation, because a rapid response may be needed to avoid overtemperature shutdown. If the compressor is marginal on stability, the load control valve may be scheduled to bleed small amounts of air during transients to prevent compressor stall.

Output load controls for electrical, hydraulic, or direct shaft power are regulated by aircraft system components.

3-8.3.1.4 Electrical Control Location

APU-mounted control components include all driven equipment: sensors for temperature, pressure, and speed; valves and ignition components; and other control components capable of withstanding compartment temperatures. Solid-state sequencing controls, power supply, load-control valve controls, miniaturized relays, and malfunction indicators *shall* be airframe-mounted so as to limit temperature to 200°F or below. If required, some of these components can be mounted on the air inlet collector or other ducting where temperatures can be limited by heat transfer to incoming air.

3-8.3.1.5 Electrical Power Requirements

If the aircraft can provide small amounts of electrical power, the APU system is simplified. About 4 A, 24 V is adequate to operate most APU relay systems. Battery systems suffer from ambient temperature limitations in that the battery must be kept warm (0°F or above) to permit the −65°F APU starts required by MIL-P-8686.

If no batteries are available, an APU-driven ignition and control generator can supply start and run sequencing power. Such a system requires a stored-energy APU start, such as hydraulic or pneumatic. Voltage buildup must be very rapid so that system sequencing can begin early (5%) to achieve low self-sustaining speed and good cold starting.

3-8.3.2 Fuel System Controls

The APU fuel system controls must provide for automatic starting, acceleration, and rated speed governing throughout the operating envelope defined in the model specification.

Techniques for scheduling fuel may be mechanical, electronic, or fluidic. The electronic system is attractive when multiple-sensing inputs for precise speed control, protective circuits, etc., are required. In most cases, however, the APU system requirements are simple and are handled best by a mechanical governor with acceleration fuel flow scheduled by compressor discharge pressure.

The APU fuel system consists of the fuel supply (common with the main engine(s)), airframe-mounted boost pump, and fuel lines (with shutoff and check valves) connected to the APU. An inlet filter of large capacity, but small micron rating, provides clean fuel at the APU fuel pump.

3-8.3.2.1 Rated Speed Governing

The APU speed-regulation requirements generally are satisfied by droop governing, as explained for the main engine in Chapter 8, AMCP 706-201. A speed band (droop) of 2-4% through the load range is adequate for frequency control at 400 Hz AC power.

Provided bleed airflow and pressure requirements are met, the speed band is not critical for a bleed air APU. Speed recovery and stability will be specified in the APU model specification. If required, isochronous governing to hold speed within a narrow band (±0.25%) can be accomplished by a null system. Topping speed adjustments should be provided to account for installation differences and deterioration of the efficiency of the APU or driven equipment between overhauls.

3-8.3.2.2 Filtering Requirements

Filtering has been a problem on some military helicopter installations. Combat situations have resulted in fuel contamination beyond original expectations. Servicing of components in the field also can introduce contamination, which must be considered.

To keep required maintenance at a minimum, APU filters should be of extra-large capacity, or of self-purging configuration. A high percentage of fuel control, valve, and nozzle failures results from contamination damage. Sources of contamination include line or component contamination during servicing and wear products generated internally in pumps and moving parts.

Screens or micronic filters *shall* be placed at valve and nozzle entrances to prevent passage of particles left during assembly, installation, or field maintenance. APU fuel pump inlet and outlet filters *shall* be standard equipment and of the throwaway type. The rated micron size of filters should be as large as possible, considering orifices and jet and internal tolerances, so that capacity requirements can be reduced.

3-8.3.3 APU Lubrication Subsystem

The lubrication subsystem generally is self-contained within the APU, unless an external oil cooler is required. The trend is toward a completely sealed oil system requiring no scheduled maintenance. Filters are included internally and, with current units, are serviced at specified intervals. With adequate seals and filtering of buffer air (if used), external contamination virtually can be eliminated. Slightly larger filters, with bypass valves, should eliminate the need for change between overhauls. Oil consumption rates generally are low enough so that oil level checks may be eliminated for long periods.

As with fuel subsystems, field maintenance of APU lubrication systems introduces more problems than it solves. Reliability factors for lubrication pumps, filters, relief valves, and jet or mist supply systems are high.

Altitude operating requirements are defined in the APU model specification. Maximum temperature limits also are specified in the APU model specification.

Qualification testing of the lubrication system is concurrent with APU qualification in accordance with MIL-P-8686.

3-8.3.4 APU Reduction Drive

The APU-reduction-drive design is established primarily by the driven equipment and accessories required. The bleed air APU usually is designed to deliver a small percentage of its total output shaft horsepower. Drive pads will be provided for fuel controls, control generators (if needed), and all driven accessories such as electrical generators and hydraulic pumps.

The turbine nozzle and diffuser matching within the APU can be changed to trade off shaft power capability for bleed performance within the limits of stall. For the shaft horsepower APU, output may be concentrated at one or two larger pads. If APU power is applied directly into an aircraft combining gearbox, fewer APU pads will be needed. In this case, required helicopter accessory outputs can be obtained from the auxiliary gearbox.

3-8.3.5 APU Starting

The APU starting subsystem *shall* be fully automatic, using the sequencing systems previously described. The starter energy level must exceed the APU drag (resistive) torque by sufficient margin to provide the required acceleration. The starting torque requirements will be described in the APU model specification.

Electrical starting is satisfactory if starter and battery energy levels are chosen properly. The minimum-weight subsystem requires the smallest battery consistent with adequate breakaway torque, including consideration of initial voltage drop. This will result in a longer starting time to rated speed (15 to 20 sec), but battery is adequate to provide some torque to very high speeds (80-90%). The basic limitation is that cold-day starting (−65°F) is not practicable without warm and oversized batteries. It will be found that when APU size increases above 300 hp, battery size becomes excessive. Also, the slower start makes the APU more sensitive to the fuel acceleration schedule. Achieving successful starts at both −65°F and +130°F without adjustments may require a compensation mechanism.

The hydraulic method of APU starting is satisfactory, especially when −65°F starting is required. Hydraulic starter motors should be sized for high initial torque to give rapid acceleration. This will tend to result in a lower accumulator volume requirement, and to reduce sensitivity to fuel schedule variations. A typical subsystem will have an initial cranking torque of 50% more than the highest APU resistive torque. Ideally, the motor should incorporate an overrunning clutch so that no drag is induced onto the APU when accumulator fluid is expended. During −65°F starting, the APU may self-sustain at 40-50% speed. A few foot-pounds of drag from a motor can necessitate additional accumulator volume so that starting torque continues to 55 or 60% speed.

As previously discussed, hydraulic starting can be arranged by using a battery for control power or by incorporation of an APU-driven ignition and control (permanent magnet) generator.

3-8.4 RELIABILITY

Reliability characteristics are specified in the APU model specification. For helicopters using the APU for inflight emergency power, starting reliability is of major importance. Typical requirements for APU starting failure rates are shown in Table 3-2.

TABLE 3-2
APU RELIABILITY

NUMBER OF STARTS	ALLOWABLE FAILURES
0-500	0
501-775	1
776-1050	2
1051-1390	3

APU operating life may be demonstrated to any given model specification requirement. A test plan may be chosen from MIL-STD-781. Typically, if a mean time between failures (MTBF) of 1500 hr is specified, a test with seven APU's, each operated to 300 hr without failure, would be recommended. Other combinations of run time and number of engines may be chosen, e.g., three APU's with each running 750 hr without a relevant failure.

High installed reliability can be obtained by specifying a time between overhauls (TBO) close to the specified MTBF. However, this results in a considerably higher life-cycle operating cost. Lowest cost is obtained by using the remove for failure (RFF) philosophy, when APU's are repaired or removed only in event of a malfunction.

A failure mode and effect analysis (FMEA) for the APU *shall* be specified to show the consequences of each probable failure. This will assist in reliability prediction, choice of scheduled maintenance intervals, and the determination of RFF versus TBO philosophies.

Several problem areas have been experienced by operation of the APU in the helicopter. These areas include vibration, recirculation of exhaust gases, FOD, etc. They point up the importance of careful delineation, in the quality assurance provisions of the APU model specification, of design and test conditions that simulate the helicopter environment.

The vibration environment is primary. The profile of amplitude and frequencies at the APU installation must be defined. The amplitude and frequencies of the APU-generated vibration also should be specified to anticipate airframe structural problems.

APU air inlet temperature limits are specified, and the APU installation must be designed to prevent recirculation of main engine or APU exhaust gas. Careful attention must be given to this problem because it is impractical to attempt to limit helicopter operation in undesirable wind directions. Inlet duct locations either must be remote from exhaust outlets, or safety shutdown sensors must be provided.

Similar arguments apply concerning FOD, sand, or dust ingestion. Generally, high inlet duct locations are preferable because concentration of dust in hover is stratified vertically.

Many APU service problems can be traced to fuel system components. Some of these result from helicopter fuel system contamination. Additional emphasis should be given to filtration, and to prevention of main tank contamination by proper fuel-handling methods.

A major source of reliability problems arises from maintenance requirements either scheduled or unscheduled. For example, more contamination is introduced into oil systems (causing excessive wear and early bearing failure) through frequent oil level checks, oil additions, and oil changes than through seals and vents during normal running. APU design should stress minimum scheduled maintenance, throwaway filters and components, sealed systems, and automatic controls requiring no adjustment. This approach not only will increase reliability, but also will decrease net life-cycle APU costs.

3-8.5 SAFETY PROVISIONS

Good APU safety design must include provisions to prevent a failure from causing helicopter damage, and, if possible, to permit mission completion in event of a failure. Thus, the APU installation *shall* be designed so that fire, APU rotor failure, and crash damage are contained within the APU compartment.

APU-rotor containment is an important safety consideration, and can be handled in several ways. Structure can be designed to withstand and hold a rotor tri-hub burst at overspeed trip conditions (assuming a fuel control failure), but this causes an undesirable weight penalty. Alternatively, rotor integrity can be demonstrated by means of specialized tests to measure stress levels under operating conditions. Further, systems can be arranged to guarantee that blade failures occur first (e.g., stress grooves), but that smaller mass blade failures can be contained within the casing structure.

Fuel and ignition sources *shall* be separated by means of the compartment design. One philosophy is to put the entire APU into a fireproof compartment. A second philosophy seeks to prevent fire by confining fuel sources and by segregating the hot section with a bulkhead. The APU controls and oil sump should be housed in fireproof containers. Electrical subsystem ignition sources should be routed or housed away from fuel lines. APU inlet air should be ducted from outside the helicopter to prevent recirculation in case of compartment fire. Fire detectors and fire extinguishing equipment *shall* be used to protect against fire within the compartment (see par. 3-2.4).

All APU fuel system components *shall* be crashworthy. Fuel lines *shall* be made of flexible hose with steel-braided outer sheath, with the minimum number of couplings. At bulkheads, the hose should be run through uncut, using frangible hose stabilizer fittings. When lines go through a firewall, self-sealing, breakaway couplings *shall* be used. All line supports should be frangible. Lines *shall* be 20-30% longer than necessary to accommodate structural displacements. Routing *shall* be along the heavy basic structure, but away from electrical components (unless electrical systems are shrouded).

Drain lines for combustor, fuel pump, gearbox, vents, etc., *shall* be connected with frangible fasteners, and made of low-strength materials.

Preferred design calls for engine-mounted fuel boost pumps with suction fuel supply. In the event that tank-mounted boost pumps are required due to fuel subsystem configuration, they are acceptable if mounted with frangible attachments. Electrical lead wires must be 20-30% longer than necessary, and shrouded to minimize crash damage.

Self-sealing, breakaway couplings *shall* be used at all connections. Filters and valves *shall* withstand 30-g loads applied in any direction. Electrically actuated valves can be bulkhead-mounted, with wiring on one side and the valve and fuel lines on the opposite side.

APU oil tanks and coolers also *shall* withstand 30-g loads, and must be mounted away from impact areas. They *shall* be located within the compartment, but away from hot sections and inlet air ducting to prevent ingestion of spilled oil. Oil filters *shall* be integral with the APU.

Batteries and electrical accessories *shall* be located high enough in the fuselage to remove them from possible fluid spillage areas. They *shall* be compartmentalized with flexible fire resistant panels. Extra wire length is needed and *shall* be supported with frangible connections. The basic structure *shall* withstand 30-g loads applied in any direction.

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CHAPTER 4
TRANSMISSION AND DRIVE SUBSYSTEM DESIGN

4-0 LIST OF SYMBOLS

A_f = axial force, lb
a = bearing outer ring bore, in.
A_i = portion of total time, %
B_{10} = life at which 10% of a bearing population fails, cycles or hr
b = Hertzian contact band semiwidth, in.
b = bearing OD, in.
C = capacity of bearing for life of 10^6 cycles with 90% probability of survival, lb
C_C = case convection cooling coefficient, hp/°F
C_h = load inclination factor (helical gear), dimensionless
C_p = elastic coefficient, $(psi)^{1/2}$
c = liner (steel) OD, in.
c_p = specific heat of oil, Btu/lb-°F
D = gear diameter, in.
D = material factor (bearings), dimensionless
D_{BC} = bolt circle diameter, in.
D_{PD} = stud pitch diameter, in.
D_b = involute base circle diameter, in.
D_i = inside diameter, in.
D_{mi} = spline minor diameter, in.
D_o = outside diameter, in.
D_p = pitch diameter, in.
D_r = gear root diameter, in.
D_1 = major diameter of spline, in.
D_2 = outside diameter of spline tooth member, in.
d = pinion pitch diameter, in.
d = track of braked wheels, ft
d = light alloy section OD, in.
E = modulus of elasticity (Young's modulus), psi
E = processing factor (bearings), dimensionless
E_D = energy dissipation rate, Btu/in.^{2}-min
E' = combined modulus of elasticity, psi
e = pitch plane misalignment, in./in.
F = flow rate, gpm
F = face width of gear tooth, in.
F = lubrication factor (bearings), dimensionless
F_{bs} = breakaway slip force, lb
F_e = effective face width, in.
F_m = average effective face width, in.
f = coefficient of friction, dimensionless
f_b = frequency of backward travelling wave, Hz
f_f = frequency of forward travelling wave, Hz
f_i = inner race curvature, % of ball diameter
f_o = outer race curvature, % of ball diameter
f_o = static resonant frequency, Hz
G = EHD material parameter, dimensionless
G = speed effects factor (bearings), dimensionless
G = lengthwise tooth stiffness constant, psi
G_S = specific weight of oil, lb/gal
H = misalignment factor (bearings), dimensionless
h = oil film thickness, μin.
h_E = EHD oil film thickness, μin.
I = stress index modifier, dimensionless
I = moment of inertia, slug-ft^2
J = geometric shape factor, dimensionless
K = Hertz stress index, psi
K_f = stress concentration factor, dimensionless
K_i = inertia factor, dimensionless
K_l = life factor, dimensionless
K_m = misalignment factor, dimensionless
K_o = overload factor, dimensionless
K_r = reliability factor, dimensionless
K_s = size factor, dimensionless
K_t = temperature factor, dimensionless
K_v = dynamic load factor, dimensionless
k = conversion constant
k = contact line inclination factor, dimensionless
k = geometry factor, dimensionless
L = gear face width, in.
L = design life or scheduled removal time (TBO), hr
L_A = adjusted life, hr
L_{CD} = gear center distance, in.
L_2 = life for 2% failure of a bearing population, hr
L_{10} = life for 10% failure of a bearing population, hr
M = mechanical advantage, dimensionless
M = moment, in.-lb
m = profile contact ratio, dimensionless
m_g = gear ratio, dimensionless
m_p = contact ratio factor, dimensionless
m_o = modified contact ratio (spiral bevel gear), dimensionless

N = number of teeth (gear or spline)
N = number of bolts or studs
N_g = number of teeth on gear
N_p = number of teeth on pinion
n = number of discrete values
n = number of radial nodes
n = rotational speed, rpm
n_{cr} = critical speed, rpm
n_{op} = normal operating rotational speed, rpm
n_p = pinion rotational speed, rpm
P = load, lb
P_L = power loss, hp
P_{LC} = power loss to oil cooler, hp
P'_{LC} = power loss to oil cooler (cold condition), hp
P_b = base pitch, in.
P_d = diametral pitch, in.$^{-1}$
P_d = transverse diametral pitch (measured at large end of bevel gear), in.$^{-1}$
P_f = friction power loss, hp
P_i = power input to transmission, hp
P_l = power loss, % of transmitted
P_m = mean transverse diametral pitch (bevel gear), in.$^{-1}$
P_p = oil pump loss, hp
P_t = fastener tension loading, lb
P_w = gear windage power loss, hp
p = pump discharge pressure, psig
p_c = circular pitch, in.
p_n = normal circular pitch (helical gear), in.
Q = torque, lb-ft or lb-in.
Q_B = brake torque, lb-ft
Q_S = skid torque, lb-ft
Q_f = flange torque capacity, lb-in.
Q_s = stud torque, lb-in.
R = mean transverse pitch radius, in.
R = reliability (for 1-hr mission), dimensionless
R_t = distance from pitch circle to point of load application, in.
r_g = radius of curvature of gear tooth, in.
r_p = radius of curvature of pinion tooth, in.
S = probability of survival, dimensionless
S = rms surface finish, μin.
S_{at} = allowable endurance limit stress, psi
S_{br} = bearing stress, psi
S_c = compressive (Hertz) stress, psi
S_{cf} = compressive (Hertz) stress at failure, psi
S_k = hoop stress, psi
S_h = bursting stress, psi
S_τ = shear stress, psi
S_t = tensile stress, psi
S_{ts} = torsional shear stress, psi
T = temperature, °F
T_A = ambient air temperature, °F
T'_A = ambient air temperature (cold condition) °F
T_c = critical temperature, °F
T_c = circular tooth thickness, in.
T_i = initial temperature, °F
T_S = average external surface temperature, °F
T'_S = average external surface temperature (cold condition), °F
U = EHD speed parameter, dimensionless
V_1 = rolling velocity of faster of two bodies in contact, in./sec or fps
V_2 = rolling velocity of slower of two bodies in contact, in./sec or fps
V_T = total rolling velocity (V_1 + V_2) of two bodies in contact, in./sec or fps
V_T = $(V_1 + V_2)/2$, in./sec or fps
V_s = sliding velocity, in./sec or fps
W = load, lb
W = helicopter weight carried on braked wheels, lb
W = EHD load parameter, dimensionless
W_d = dynamic load, lb
W_f = failure load, lb
W_t = gear tooth load, lb
W'_t = effective gear tooth load $(K_o W_t + W_d)$, lb
w = running load, lb/in.
Y = modified Lewis form factor (spur gear), dimensionless
Y_c = modified Lewis form factor (helical gear), dimensionless
Y_k = modified Lewis form factor (bevel gear), dimensionless
Z = total transverse length of line of action, in.
Z_t = scoring geometry factor, dimensionless
Z'_t = modified scoring geometry factor (spur gear), dimensionless
α = pressure viscosity coefficient, in.^{2}/lb
α = linear coefficient of thermal expansion, in./in.-°F
β = contact angle, deg
γ = fraction of theoretical contact (splines), dimensionless
Δ = increment, as ΔT, deg F or $\Delta(D_p/2)$, in.
δ = mean CLA surface roughness, μin.
δ = incremental growth, in./(in.-°F)
δ = deflection, compression, or protrusion, in.
η = efficiency, dimensionless or %; subscripts c and f, p, and t denote coarse and fine pitch, pump, and transmission, respectively
λ = ratio of oil film thickness to surface roughness, dimensionless
λ = failure rate, hr^{-1}

μ_c = dynamic viscosity, lb-sec/in.^{2}
μ = Poisson's ratio, dimensionless
ν = helical tooth lead line inclination, deg
ξ_f = EHD parameter, dimensionless
ρ = density, slug/ft^3 or lb/in.^{3}
σ = standard deviation, dimensionless
ϕ = gear tooth pressure angle, deg
ϕ_n = normal pressure angle (helical gear), deg
ϕ_t = transverse operating pressure angle, deg
ψ = gear tooth helix or spiral angle, deg
$\dot{\Omega}$ = angular acceleration or deceleration of rotor, rad/sec^2
ω = rotational speed, Hz

4-1 INTRODUCTION

4-1.1 GENERAL

The proper use of this chapter as an aid in the achievement of satisfactory transmission and drive system detail design requires a clear understanding of several basic concepts. The contents reflect the past in that prior applications are used to generate an experience and information base. The suggested analytical approaches and techniques for design and for the basic selections of gear, bearing, and shafting configurations represent the present state-of-the-art. However, the technology as discussed does not exclude the future in that areas of uncertainty and limitations of knowledge are emphasized wherever they appear applicable. The mechanical drive system designer must practice his skill from a moving or living technology base.

The use of geared transmission systems predates recorded history. A relatively sophisticated differential gear-drive system was employed in the Chinese "South Pointing Chariot" circa 2600 B.C. (Ref. 1), and geared drives still represent the most efficient method of power transmission. This chapter is intended to encourage rather than to subvert new and unconventional approaches to old problems. The sole limitation upon incorporation of the unconventional in helicopter drive systems is that the basic rules of nature (laws of mechanical physics) are relatively inviolable and should be treated with respect.

There is no perfect or unique design solution to a given power transmission requirement, and all known designs have been compromised by the individual requirements of the aircraft into which they must be integrated. Optimization of a design cannot be viewed within the context of the power transmission task alone; thorough trade-off studies must consider such factors as suspension, layout, airframe support structure, rotor control systems, aircraft weight and balance, space limitations, and locations of engines. Almost any known drive system could be made lighter, more efficient, and less costly if it were not necessary to consider interface effects. However, the design optimization techniques addressed in this chapter are limited to the components of the power transmission subsystems without consideration of the possible overriding effects peculiar to a given aircraft configuration.

The designer must be aware of the significance of the total Army environment. Major subsystem components such as gearboxes, driveshafts, or hanger bearing assemblies probably will be subjected to rough treatment during shipping, handling, and removal from or installation on the helicopter. The consistent use of sophisticated or special tools and torque wrenches simply will not occur, even though specified by the designer. Extremes in temperature, humidity, sunlight, precipitation, and sand and dirt contamination will occur during both flight operation and lengthy periods of outdoor parking. Pressure cleaning equipment — employing steam, water-soap solutions, or other solvents — will be used on the helicopter and its drive subsystem; these cleaning solutions may be more than 100°F hotter or colder than the components creating the further risk of thermal shock. Exposure to such hostile envionments will occur repeatedly for long periods of time, but must not compromise the mission availability of the helicopter. Improper maintenance, tool drops, and reverse or improper installation where possible also will occur. Component design must be tolerant and forgiving of such treatment wherever practicable.

4-1.2 REQUIREMENTS

General requirements are applicable to all drive system configurations. There also are specific requirements that vary according to the aircraft configuration and intended mission (par. 7-1.1, AMCP 706-201) and general requirements peculiar to particular configurations or arrangements of engines and rotors (par. 4-1.3).

4-1.2.1 General Requirements

Certain requirements are common to all Army helicopters regardless of configuration or intended usage. The desired level of attainment of these requirements and their relative importance generally are specified in the appropriate prime item development specification (PIDS). Because the transmission and drive system represents a significant portion of the total complexity and cost of the helicopter, these common requirements must be considered during detail design. Such requirements — without consideration of their relative importance — include performance, reliability, maintainability, and survivability.

4-1.2.1.1 Performance

The contribution of the drive system to helicopter performance can be defined in various ways. However, the following factors predominate:

1. Weight
2. Efficiency
3. Size
4. Noise level.

4-1.2.1.1.1 Subsystem Weight

Weight of the transmission and drive system is minimized by attention to compact size together and with the use of high-strength materials for dynamic components that make maximum use of the material properties available within the limits of permissible failure rates and reliability requirements. Superior quality control in processing and the skillful use of low-density material for forged or cast housings also are necessary. The latter is important because gearbox housings comprise from 20 to 60% of total transmission weights in current designs.

Excessively large gear ratios per stage generally add weight. The pinion size is determined primarily by the torque transmitted and is relatively independent of the gear ratio; but the weight of the gear member increases roughly as the square of the ratio. Large ratios per stage usually also are inefficient.

Considering the size of support structure and bearings, as well as of the gears themselves, the gear types order themselves by increasing weight and power loss (for approximately equal gear ratios) in the following manner:

1. Concentric drives: epicyclic or planetary devices
   a. Simple — same input-output rotation
   b. Star — reversing rotation.
2. Parallel-axis drives: spur, helical, and herringbone
3. Intersecting-axis drives: spiral bevel and hypoid.

This ordering reflects the effects, in addition to the intended speed change per stage, of additional torque vector translations and rotations. The concentric drive does not alter the torque axis; the parallel drive requires a translation of output vector with respect to input, and the intersecting-axis drive introduces a new coordinate through the rotation of the output vector with respect to input.

The selection of drive types should follow the given ranking, beginning at the output drive. Also, the largest reductions should be taken close to the final output, with smaller amounts of torque being carried as the distance from the output increases. The weight of any gear reduction stage is proportional to the second or third power of the torque. Therefore, worthwhile weight savings can be made by using drives in the order ranked — concentric drives near the output, parallel-axis types at intermediate or combining stages, and intersecting-axis types nearest the engine if drive direction changes are required.

There are occasional instances where these rules may not apply; e.g., high reduction ratio may be undesirable at secondary-level power outputs, such as tail rotor or auxiliary propellers, because of the long distance from the helicopter CG. Although the total drive subsystem weight may be less with a lower torque being carried by shafting, it may become difficult to obtain a satisfactory CG location due to the larger moment of the extra weight of the higher ratio final reduction stage.

The specific weights of current helicopter main gearboxes in range from 0.30 to 0.50 lb/hp for reduction ratio of 15:1 to 74:1 (Fig. 4-1). The next decade undoubtedly will see this index dropping near the 0.25 level.

4-1.2.1.1.2 Transmission Efficiency

The requirement for efficient power transmission is of such importance that gear types other than those listed in par. 4-1.2.1.1.1 seldom are considered seriously for application to the main power drive

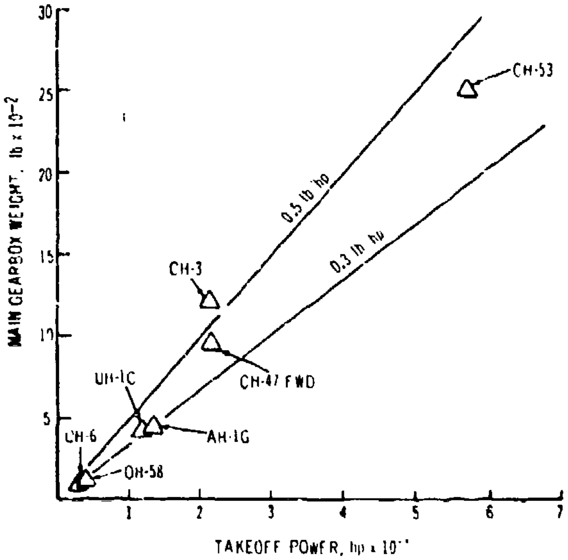

Figure 4-1. Helicopter Main Gearbox Weight vs Takeoff Power

train for many reasons. For example, with a 1% increase in power loss the life cycle cost for an assumed fleet of 1000 medium helicopters would be increased by $100,000 per helicopter by the extra fuel necessary to perform a constant mission. This is based upon a 6000-hr life, a specific fuel consumption (SFC) of 0.65 lb/hp-hr, and a fuel cost of $0.016/lb. Further, the average helicopter lift capability ranges from 5 to 10 lb/hp depending upon rotor disk loading and operational variables. Therefore, a medium helicopter of 2000 hp suffers a useful load reduction of from 100 to 200 lb with a 1% additional power loss. Thus, when comparing an alternative gear system of 1% lower efficiency, the basic gearbox weight would need to be reduced by more than 100 lb to compensate for the power loss.

The reverse drive efficiency of a gearbox also must be watched carefully, as excessive use of recess action gearing can create a problem in autorotation. For instance, Ref. 2 describes high-ratio recess action systems that operate at high efficiency as speed reducers but become virtually self-locking when operated as speed increasers as in autorotation.

A helicopter of 15,000 lb gross weight making an autorotational descent at 2000 ft/min is using its available potential energy at the rate of about 900 hp. Drive system windage, flat pitch tail rotor drag, and a few minimum accessory loads require approximately 100 hp. A 95% reverse drive efficiency leaves about 795 hp, a reasonable level to sustain the prescribed descent. However, sudden yaw control requirements conceivably could boost the reverse drive requirement momentarily to 200 hp. In such a case, either the rate of descent would increase sharply or some kinetic energy would be borrowed from the main rotor, with a slight reduction of rotor speed being compensated by an increase in descent velocity. However, if the reverse drive efficiency were 50%, the tail rotor and accessory load would extract 400 hp from the main rotor, requiring an almost 50% increase in rate of descent to sustain safe main rotor speeds.

The power losses of a typical high-speed twin-engine-drive main gearbox operating at constant speed might vary as shown in Fig. 4-2. At full speed there is 25-hp windage loss with zero power transmission, and then the power loss due to friction is added as the power transmitted is increased until a total loss of 60 hp is reached for the full twin-engine power input. The slightly downward inflected curve shape is rather typical for most modern, heavily loaded, high hardness gears and antifriction bearing systems, although in some instances a virtually straight line may be observed. Clearly, it is improper to speak of a gearbox having loss characteristics, such as those shown in Fig. 4-2, as possessing a given efficiency apart from a specific power rating. The transmission losses shown result in efficiencies as follows:

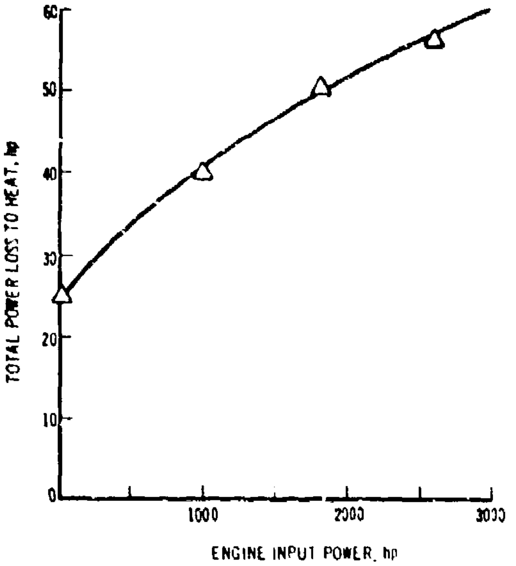

Figure 4-2. Power Loss to Heat vs Input Power — Typical Twin-engine-driven Gearbox

Input Power, hp	Efficiency, %
100	74.0
500	93.4
1000	96.0
1500	96.9
2000	97.4
2500	97.8
3000	98.0

The windage losses are influenced strongly by oil viscosity, the amount of oil supplied to the various gear meshes and bearings, and the oil scavenging characteristics of the transmission. Small gear-to-housing clearances, poor drainage paths, and excessive oiling should be avoided. Good estimations of gear windage losses P_w may be obtained from Eq. 4-1 (Ref. 3)

$$P_w = \frac{n^3 D^5 L^{0.7}}{100 \times 10^{15}}, \text{hp} \tag{4-1}$$

where

D = gear diameter, in.
L = gear face width, in.
n = rotational speed, rpm

This equation represents an application of basic propeller theory (Ref. 4) and is based upon air density at standard sea level conditions which is 0.00238 slug/ft³. However, for MIL-L-7808 oil at normal operating temperatures ρ = 1.748 slug/ft³. Consequently, if the helicopter designer can estimate or experimentally determine the oiliness of the transmission atmosphere, an average ρ may be employed. Ref. 4 suggests a 34.25:1 air-oil ratio in which case Eq. 4-1 can be expressed as

$$P_w = 2.18\left(\frac{n^3 D^5 L^{0.7}}{10^{16}}\right) \quad \text{, hp} \tag{4-2}$$

Methods of gear mesh oiling also affect windage losses. Differences of virtually 200% have been reported to exist between in-mesh and out-of-mesh oiling for a large, single-reduction gear set (Ref. 5).

The torque or load-sensitive contribution of the transmission to power loss is due almost entirely to the rolling/sliding load-carrying bearing and gear elements.

Good first-order approximations for antifriction bearing losses at moderate speeds are given in Ref. 6 and with a little greater precision in Ref. 7. However, where very accurate determinations are required or where high-speed applications exist, the prediction of these losses requires an understanding of the more complex factors involved (Ref. 8). For optimization studies and examination of the effects of external deflections and asymmetrical loading, there is no substitute for a good digital computer program of the basic equations such as is presented in Ref. 9.

In general, the significant power losses in bearings occur in regions of appreciable contact zone slip. Consequently, simple, well-guided cylindrical roller bearings are the most efficient for most applications. Radially loaded ball bearings are lower in efficiency, while angular contact (thrust loaded) ball bearings exhibit significantly greater friction losses. When the latter are used in very high-speed operations, their power loss characteristics deteriorate sharply as excessive centrifugal and gyroscopic forces affect ball kinematics. Standard tapered roller bearings (conical rolling elements) are the least efficient of these four types, largely because of the heavily loaded cone rib that is in sliding contact with the large end of the conical roller. A new type of angular contact cylindrical roller (Ref. 10) may offer advantages when fully developed. In this type, the cone rib load is theoretically reduced, but increased sliding results at one or both of the race-roller contacts. Experience is insufficient for evaluation of its relative heat generation, but indications are that it will prove to be more efficient than the standard tapered roller and that its losses will approach those of the angular contact ball bearing. The plain journal bearing is unsuitable for heavily loaded power transmission applications (see par. 4-2.4.2), although it may have satisfactory application in lightly loaded accessory uses; the power loss with this type of bearing is at least twice as great as for any of the previously mentioned types (Ref. 11). The hybrid boost bearing (hydrostatic plain thrust bearing in series combination with the stationary or rotating ring of an angular contact ball thrust bearing) has been evaluated experimentally (Ref. 12), but no helicopter experience is known. The experimental work indicated a friction torque for the hybrid bearing of roughly twice that for the simple ball thrust bearing.

The accuracy of the calculated losses for bearings is dependent upon proper installation and application design. Excessive preload, due to installation and/or thermally induced, can easily result in doubling the friction torque and also will have detrimental effects upon fatigue life and reliability.

The power loss in gearing is quite an involved and controversial subject. However, the specific weight rankings listed earlier in this paragraph hold true for basic efficiency also. There are reasonable ranges of gear ratio for which specific types of gear drives are best suited; when the upper limits are exceeded, the resultant power loss is apt to increase to a level where superior overall efficiency may be obtained by reverting to two stages of lower ratio. The suggested optimum ratio ranges are:

1. Concentric — epicyclic, 3:1 to 5:1
2. Parallel axes — straight spur, 1:1 to 2.5:1
   — single helicals, 1:1 to 5:1
   — double helicals, (herringbone), 1:1 to 10:1
3. Intersecting axes — spiral bevel. 1:1 to 3.5:1.

A number of assumptions are included in these suggested ranges. The lower limit for the epicyclic assumes a simple (nonreversing) design. The planet gears become excessively small so that the sun driver tends to act like a speed-increasing drive with a long, inefficient arc of approach. At these low ratios the epicyclic system becomes planet-bearing-capacity limited in that it is difficult to fit sufficiently large bearings to carry the necessary load if the gear teeth are stressed to satisfactory levels.

The upper limit for the epicyclic represents a reasonably designed, four-planet idler system whose weak point is the tendency for pitting of the sun gear. Because the design is sun-pinion-diameter limited (to acceptable Hertzian stress levels) and the planet gears are four times the diameter of the sun, the system is rapidly becoming inefficient from a weight standpoint. The carrier structure, ring gear, and ring gear housing (if used) are excessively heavy. If used in a final drive stage, the low speed of the plane bearings (9/4 × rotor speed) is not conducive to the formation of adequate oil film thicknesses for good performance even though there is virtually unlimited space for high-capacity planet bearing designs.

The upper limit of the single helicals is based upon the assumption that the helix angle ψ will be no greater than 15 deg in order to minimize the thrust component of the tooth loading. The extended upper limit given for herringbone designs is based upon the use of high helix angles (ψ = 35 deg); thus, maximum advantage is taken of the thrust component cancellation that permits attainment of very high face contact ratios which in turn permit some reduction in the profile contact ratio with an attendant reduction in sliding velocities. The upper limit for the spiral bevel gear ratio reflects the use of approximately a 90-deg intersection of axes and is based upon increased losses due to excessive sliding velocities in both the art of approach and the recess. The lower limit for spiral bevels would be applicable for overhung mountings at a 90-deg intersection of axes. Satisfactory full-straddle-mounted, 90-deg-axis systems cannot be accomplished below an approximate ratio of 1.4:1 for 90-deg axes.

In all cases the lubrication is limited to low-viscosity synthetic turbine oils with no unusual additives.

The specific efficiencies obtained in gear meshes are basically considered to be represented by analogy to classic physical concepts. The friction power loss of P_f of sliding bodies in contact is given by Eq. 4-3.

$$P_f = WV_s f/550, \text{ hp} \tag{4-3}$$

where

W = load, lb

V_s = sliding velocity, ft/sec

f = coefficient of friction, dimensionless

Minimization of power loss would simply seem to require attention to minimization of V_s and f. However, the apparent quantity f varies in a very complex manner with the intensity of load [expressed as a compressive (Hertz) stress S_c], the sliding velocity, and factors descriptive of the lubrication regime and the lubricant itself.

Lubricant regimes are classified loosely (from thin lubricant film separation to thick) as boundary, elastohydrodynamic (EHD), and hydrodynamic. There is no abrupt delineation between these regimes, but the following approximation is useful. In this system of definition, the boundary regime includes the subregime of microelastohydrodynamics (Refs. 13 and 14). Fig. 4-3 represents the salient characteristics and interdependent variables influencing f throughout the range of specific film thicknesses. The ratio λ is introduced to give physical meaning to these regimes in terms of the roughness or surface finish of the active tooth profiles according to

$$\lambda = \frac{h}{\delta}, \text{ dimensionless} \tag{4-4}$$

where

h = oil film thickness, μin.

δ = mean centerline average (CLA) surface roughness, μin.

Region III of Fig. 4-3 can be neglected for helicopter transmission components. The entire region is defined by classical hydrodynamics; and the properties of lubricant viscosity, sliding velocity, and load interact to build a supporting lubricant film that completely separates the load-carrying mechanisms. The observed friction is primarily dependent upon the viscosity of the supporting film.

Region II actually extends (on a submicroscopic scale) into Region I, but the true importance of this region is that it represents a transitional phase that only has become defined with engineering significance within the past decade. The pressure distributions within the loaded gear surfaces are considered basically Hertzian, but the film thickness is dependent upon the additional factors of elasticity of metals and the property of greatly increased lubricant viscosity under the Hertzian conjunction pressures. The observed film thickness is known to increase with increasing entrainment or sum of rolling

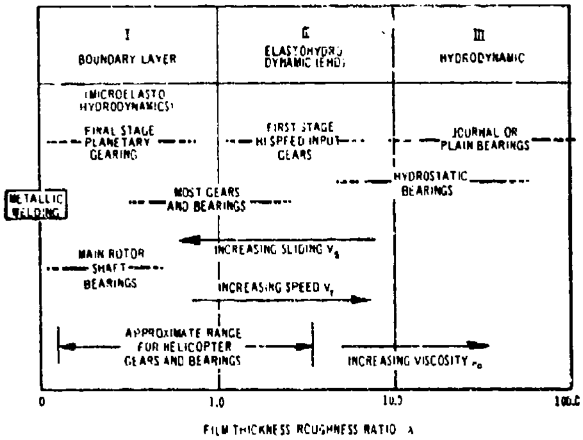
I
BOUNDARY LAYER
II
ELASTOHYDRODYNAMIC (EHD)
III
HYDRODYNAMIC
(MICROELASTOHYDRODYNAMICS)
FINAL STAGE PLANETARY GEARING
FIRST STAGE HI SPEED INPUT GEARS
JOURNAL OR PLAIN BEARINGS
HYDROSTATIC BEARINGS
METALLIC WELDING
MOST GEARS AND BEARINGS
INCREASING SLIDING V_s
MAIN ROTOR SHAFT BEARINGS
INCREASING SPEED V_r
APPROXIMATE RANGE FOR HELICOPTER GEARS AND BEARINGS
INCREASING VISCOSITY μ_o
0
1.0
10.0
100.0
FILM THICKNESS ROUGHNESS RATIO λ

Figure 4-3. Lubrication Regimes

velocities $V_T (V_T = V_1 + V_2$, where V_1, and V_2 are the velocities of the bodies in contact) of the loaded bodies, the lubricant viscosity at the conjunction inlet, and the pressure viscosity coefficient of the lubricant, and to decrease slightly with increasing load and V_s. The largest values of this region represent full separation of the loaded bodies, while the lower values permit some metal-to-metal contact of the asperities of roughness peaks. The most widely accepted expression in use today for EDH film thickness h_E (Ref. 15) is

$$h_E = 2.65 \left(\frac{G^{0.54} U^{0.70}}{W^{0.13}} \right) \times 10^{-6} \text{ , } \mu\text{in.} \qquad (4\text{-}5)$$

where the three EHD dimensionless parameters are

$G = \alpha E'$ (materials)

$U = \dfrac{\mu_0 \bar{V}_T}{E'R}$ (speed)

$W = \dfrac{w}{E'R}$ (load)

and

E' = combined modulus of elasticity, in.
$\bar{R}$ = mean transverse pitch radius, in.
$\bar{V}_T$ = mean rolling velocity $(V_1 = V_2)/2$, in./sec
w = running load, lb/in.
α = pressure viscosity coefficient, in.^{2}/lb
μ_0 = dynamic viscosity, lb-sec/in.^{2}

A physical sense for h_E is shown in Fig. 4-4. Eq. 4-5 is isothermal, in that it does not treat the effects of material heating in the conjunction, but is believed to be reasonably accurate up to V_s/V_T values of at least 0.3.

The observed friction in the EHD regime is primarily due to viscous shear of the lubricant in the high-pressure field of the conjunction.

Much experimental data exists to relate friction values to certain dimensionless parameters. Most take the form shown in Fig. 4-5 (Ref. 15). Such relationships hold for constant values of surface roughness and lay, and for specific lubricant types. The most important conclusion from these data is simply that friction is relatively low — on the order of 0.02 to 0.04 — for components operating in Region II. A more detailed analysis that considers the thermal aspects of EHD solutions as applied to simple involute gears may be found in Ref. 16.

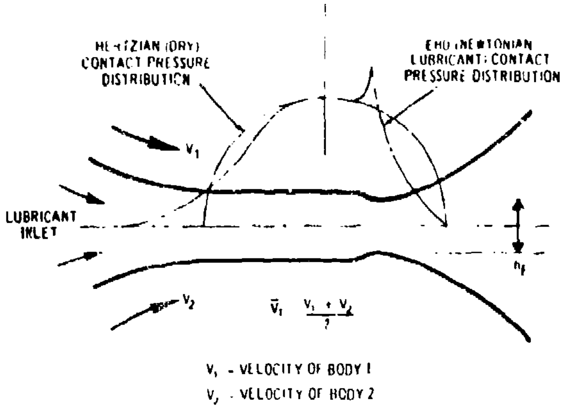
HERTZIAN (DRY) CONTACT PRESSURE DISTRIBUTION
EHD (NEWTONIAN LUBRICANT) CONTACT PRESSURE DISTRIBUTION
V_1
LUBRICANT INLET
h_f
V_2
$\bar{V}_T = \frac{V_1 + V_2}{2}$
V_1 - VELOCITY OF BODY 1
V_2 - VELOCITY OF BODY 2

Figure 4-4. Elastic Body Contact Pressure Distribution and Interface Contour

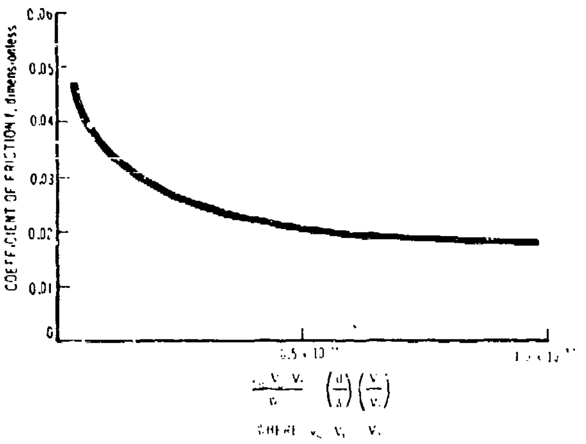

Figure 4-5. Friction Coefficient vs EHD Parameters — Region I and II

Region I, defined as boundary layer lubrication, represents conditions that predominate in the lower-speed components of helicopter gearboxes. In this region, f may be influenced significantly by interaction of asperities in the rubbing load-carrying elements, be they gears or bearings. The thinnest of films represented in this region may be monolayers of lubricant products that either adsorb or adhere to the exterior molecular surface of the metal. The variables influencing friction include the chemical composition and the interaction of the metal and lubricant combination and the roughness, lay, and texture of the surfaces with respect to their rubbing directions.

It is at the lower speeds that very noticeable differences exist in observed friction between the arc of approach and the arc of recess of involute gearing. Fig. 4-6 depicts a very low speed measurement of this phenomenon involving a spur gear set of minimum attainable profile contact ratio (CR) (Ref. 17). The very low CR is employed to study the extremes of these approach and recess portion effects without introducing the data confusion that normally would occur in the zones of double tooth pair contact. Fig. 4-6 also illustrates the dramatic improvement in *f* that results when the contract ratio is increased. The torque Q and pitch diameter D_p being held constant, the higher contact ratio was achieved by changing the diametral pitch P_d and the number of teeth N. Although the experiments cited were conducted in boundary lubrication conditions that yielded much higher *f*-values for teeth of coarse pitch, it is interesting to note that the *f*-value obtained for gears of finer pitch was in the range of values expected for EHD lubrication. This illustrates the importance of using gearing of relatively fine pitch to obtain maximum efficiency. In addition, it should be noted that the apparent differences in friction between the arcs

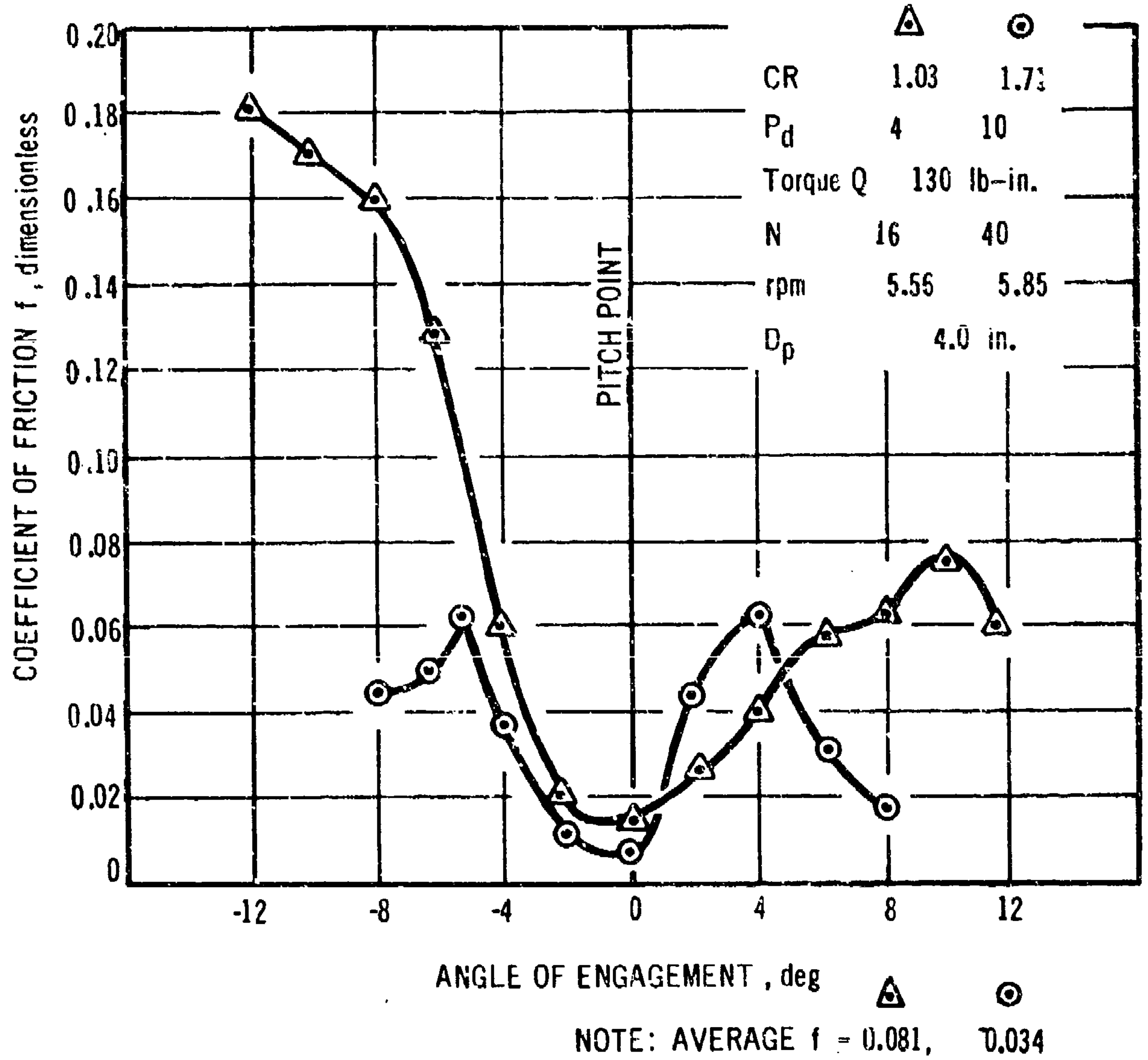

Figure 4-6. Angle of Engagement

of approach and recess are masked completely by the averaging effect found in the zones of double tooth pair contact.

The basic trends of friction change in approach and recess action are still valid in lubrication Region II. Fig. 4-7 (Ref. 18) represents experimental data taken at considerably higher values of surface sliding velocity, load, and lubricant viscosity.

Although the use of a digital computer program to examine the many instantaneous contact conditions that occur as a pair of gear teeth rolls through mesh is the more precise method of calculating efficiency and studying detail design variations, excellent results may be obtained by using average values for f. The percent power loss P_l of a gear mesh is expressed (Ref. 19) as:

$$P_l = \frac{f}{M} \times 100\ ,\% \qquad (4\text{-}6)$$

where

M = mechanical advantage, dimensionless

Values of M for various combinations of pinion and gear tooth members are listed in Ref. 19. For the coarse pitch gear (P_d = 4) of Fig. 4-6, the P_l is:

$$P_l = \frac{0.081}{4.6} \times 100 = 1.76\% \qquad (4\text{-}7)$$

and for the fine pitch gear set (P_d = 10) the P_l is:

$$P_l = \frac{0.034}{8.5} \times 100 = 0.40\% \qquad (4\text{-}8)$$

Their corresponding efficiencies η are then simply $\eta = 100 - P_l$ and we find:

$$\begin{aligned} \eta_c &= 98.24\% \\ \eta_f &= 99.60\% \end{aligned} \qquad (4\text{-}9)$$

where the subscripts c and f indicate coarse and fine pitch, respectively.

The frictional differences noted for approach and recess zones of involute action are characterized, with respect to the driving member, by the rolling and sliding contact motions being in opposite directions to one another in approach but in the same direction during recess motion.

The sensitivity of friction to the lay and textural features of the mating members in lubrication Region I is shown clearly in Fig. 4-8 (Ref. 20). These data represent the results of experiments conducted on a geared disk test machine with 3.0 in. diameter, 14.0 in. crown radius, case carburized and ground, consumable electrode vacuum arc remelted AISI 9310 steel disks. The circular ground data were taken with disks having a circumferential finish of 8 μin. and an axial finish of 16 μin., while the cross-ground disks

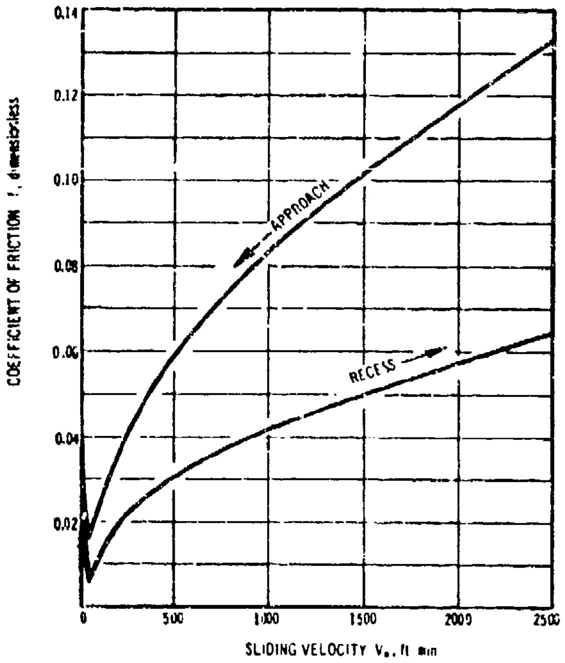

Figure 4-7. Coefficient of Friction vs Sliding Velocity

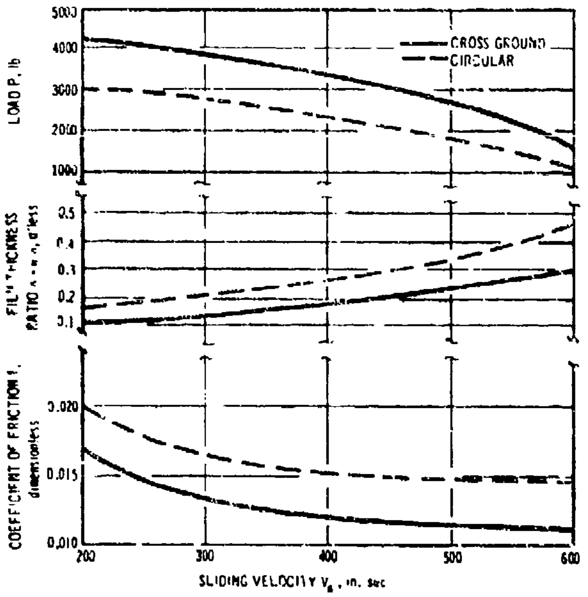

Figure 4-8. Effect of Surface Texture and Lay on Friction and Scuffing Behavior

had a circumferential finish of 16 μin. and an axial finish of 8 μin. Consequently, both types had identical reduced finish number δ-values and hence virtually identical λ-values, but the cross-ground data exhibited markedly lower *f*-values. The surface hardness was Rc60-63 in both types, and the cross-ground disks were prepared using grinding techniques normally used for spur gear tooth manufacture. A constant ratio of $V_s/V_T = 0.556$ was represented, and the lubrication was jet supplied MIL-L-7808 at 190°F. Therefore, a particular point on a gear mesh where $V_s = 55.6\%$ of V_T is represented on this figure as a linearly increased gear speed (rpm) by moving from left to right with increasing V_s. Unfortunately, these data do not reflect a constant load, but rather the limit load as defined by scuffing.

The designer should be aware that under fixed lubricant condition there is little he can do to control λ at low speeds aside from refining the surface finish.

Additional power loss sources of a transmission system include the accessories and the oil pump. Accessory power requirements are fixed by the individual helicopter requirements, the exact type of accessory involved, and the demand or duty cycle required. Determination of accessory power requirements is discussed in Chapters 7 and 9. Oil pump loss P_p is estimated adequately by the simple equation:

$$P_p = \frac{Fpk}{\eta_p}\text{, hp} \qquad (4\text{-}10)$$

where

F = oil flow rate, gpm
p = discharge pressure at pump outlet, psig
k = conversion constant for units, 5.83×10^{-4}
η_p = efficiency of pump (generally from 0.5 to 0.9), dimensionless

For a 20-gpm system with a regulated discharge of 60 psi, the pump outlet pressure would be 120 psig under typical conditions. For an assumed pump efficiency of 0.5, the loss would be:

$$P_p = \frac{(20)(120)(5.83 \times 10^{-4})}{(0.5)} = 2.78\text{ hp} \qquad (4\text{-}11)$$

4.1.2.1.1.3 Size

Compact gearbox size is important in the achievement of low subsystem weight because the housing or casing that encloses the dynamic components contributes a significant proportion of the total system weight. However, compaction should not be emphasized to the point of causing excessive oil churn and windage losses to the detriment of efficiency. Ref. 21 suggests that side clearances of 0.5 in. or greater between gears and casing walls result in negligible losses due to oil churn. The required clearance between casing wall and gear outside diameter increases with increasing values for such variables as arc of conformity, speed of gear rotation, oil viscosity, amount of oil jetted on the gear mesh, and the amount of run-off or drainage oil in the location at question. There are no formulas for calculating satisfactory diametral clearances, but some successful design applications have employed values of approximately 0.5 in. for 2000-fpm pitch line velocities and 3.0 in. for 25,000-fpm velocities for 180 deg of conformity and kinematic viscosities below 10 centistokes. Even at these clearances, it frequently becomes necessary to provide scrapers or some means to retard vortex generation and localized recirculation of the oil. When wet sump systems are employed, sufficient vertical space must be provided to keep the operating oil level (including the aerated or foam layer) below the gears and bearings.

4-1.2.1.1.4 Noise Levels

The fourth performance criteria of low noise level has received increased attention of late. Transmission noise is usually of importance only in relation to crew and passenger comfort levels, while rotor and engine noise are the principal contributors to the aural detectability of the helicopter. The fundamental gear-meshing frequencies, which range from 40 to 22,000 Hz in present-day gearboxes, are the primary sources of noise. Refs. 22 and 23 identify the magnitude of the problem for two helicopters. The overall helicopter configuration and the resulting number and location of gearboxes dictate the areas affected by noise; e.g., a tandem-rotor helicopter may have its forward transmission located above the crew compartment, resulting in less favorable noise conditions in the passenger area, while the reverse may be true for a single-rotor configuration.

Gear noise may emanate from the gearbox as a result of forced or resonant vibration of the housing or cases. It then reaches the crew or passengers either through direct airborne paths (windows, access panels, or door seals) or through airframe structural pathways connected to the gearbox mounting system. It is far more efficient and desirable to combat such noise at its source rather than to rely solely upon the use of insulating and soundproofing coatings or blankets in the crew or passenger compartments. The latter measures generally add more weight than would be needed to make comparable improvement in the problem at its source. Sound insulation also increases maintenance man-hours due to the need for removal of the material during airframe inspections. Also, soundproofing efforts often are defeated when torn or oil-soaked materials are removed and never replaced.

The use of elastomeric isolation mounting devices at the gearbox and hanger bearing supports is highly effective in reducing structural noise. Airborne noise should be minimized by eliminating any housing or case resonance through use of proper wall thickness, shape, or internal gear or bearing quill attachment methods. While it is virtually impossible to calculate these conditions with sufficient accuracy in the design stage, they are relatively easily measured during initial component testing, and corrective redesign then can be undertaken. Modifications in the shape of involute profiles so as to change drastically the fundamental and harmonic noise content have been investigated analytically (Ref. 23). However, the slight variation in profile required to achieve theoretical improvements was judged beyond the present manufacturing state of the art. Some methods for approximate analytical prediction of resonant performance for relatively simple structural housing shapes are advanced in Ref. 24.

It is not certain that the elimination of all interacting vibratory and resonant behavior in the various gear meshes is entirely beneficial; i.e., some sacrifice in the efficiency of the lower speed (boundary lubrication regime) meshes may result. The effects of axial lubrication upon the reduction of tooth-meshing friction is reported in Ref. 25.

Although the engineering field of gearbox noise generation is imprecise as yet, there exists considerable general knowledge that can be of practical benefit to the designer. For example, it is known that high contact ratio gearing and finer pitch sizes produce less noise than their opposite counterparts. Similarly, helical gears are quieter than straight spurs; spiral bevel gears are quieter than straight bevel or Zerol gears because of their greater inherent contact ratios, reduced dynamic increments or waste loads, and increased smoothness of operation.

In addition, increased gear tooth backlash and clearance can help to maintain subsonic air ejection velocities from high-speed meshing teeth (Ref. 26). Viscous films between stationary bearing rings and housings can provide sufficient damping to reduce vibration and noise propagation. Coulomb or dry friction devices have been successful in damping resonant modes in gear rims and webs, as have high hysteresis materials clad or bonded to shafts and webs.

4-1.2.1.2 Reliability

A complete general discussion of reliability concepts is contained in Chapter 12, AMCP 706-201. This paragraph, therefore, deals with specifics as related to mechanical power transmission components of the transmission system.

Concept definitions and numerical values for quantitative reliability indices generally are specified in procurement documents and, with increasing frequency, in PIDS. For transmission and drive system design there are two types of indices:

1. Values for such characteristics as mission reliability, flight safety reliability, and system reliability for the entire helicopter (usually for a given mission and operational environment). Typical values and methods of expression might be, respectively, 0.90 to 0.99 for one hour of mission time, one failure per 20,000 flight hours, and 0.70 to 0.80 probability per mission hour of no system failures requiring unscheduled maintenance. Because the reliability of the helicopter is a composite of the reliabilities of the individual subsystems, individual reliability levels must be assigned as targets for the design effort. As an example, assume the Request for Proposal (RFP) specified values of 0.98 and 0.9999 (one failure per 10,000 hr) for mission and safety criteria, respectively. The allowable apportionment for the drive system would be dependent upon the complexity and type of helicopter, but typical values for this apportionment could well be 0.999 and 0.9999, respectively. Techniques for design to these requirements will be addressed in par. 4-2.

2. Values for subsystems and components for such characteristics as reliability after storage and mean time between removals (MTBR). Typical values are a maximum of 10% degradation in mean time between failures (MTBF) after storage for six months in approved environment or containers and 1500 hr MTBR, including both scheduled and nonscheduled removals. This type of index is directly applicable to individual subsystems, including the transmission and drive system. The MTBF values subject to the 10% maximum degradation limit are those specified implicitly or explicitly in Item 1, i.e., the 0.90 to 0.99 mission reliability carries the reciprocal meaning of a 10 to 100 hr MTBF, the one failure in 20,000 hr for flight safety is a statement of 20,000 hr MTBF, and the 0.70 to 0.80 probability of zero system failures per hr implies a 3.3 to 5.0 hr MTBF. The MTBF levels corresponding to the drive subsytem apportionment in Item 1 are 1000 hr (0.999) and 10,000 hr (0.9999), respectively.

It is important that the designer understand that the MTBR and MTBF criteria discussed previously interrelate in a unique fashion with the subsystem design reliability when a finite time between overhauls (TBO) is selected. On the assumption that all transmission and drive system failures are of sufficient magnitude (and detectability) to force a mission to be aborted, and further that no TBO (scheduled removal) requirement is imposed, then it follows that the MTBF ≈ MTBR. The imposed requirement of a 1500-hr MTBR would necessitate a failure rate $\lambda \leq 0.00067$ or a one-hr mission reliability of 0.99933 for the transmission and drive system. The relationships satisfied in the preceding statements are:

$$\lambda = \frac{1}{MTBF}\ , \text{hr}^{-1}$$

and using Maclaurin's form of Taylor's formula (4-12)

$$R = \exp(-\lambda_t) \approx 1 - \lambda$$

where

R = reliability (for a 1-hr mission), dimensionless

However, if a 2500-hr TBO level were to be assigned, the 1500-hr MTBR could be satisfied only by a higher reliability number (lower failure rate). The relationship may be expressed for 1500-hr MTBR, as:

$$\lambda_{MTBR} = \sum \lambda \leq \lambda_{TBO} + \lambda_{MTBF}$$

$$\sum \lambda = \lambda_{MTBR} = \frac{1}{1500} = 0.00067 \qquad (4\text{-}13)$$

$$\lambda_{TBO} = \frac{1}{2500} = 0.00040$$

Therefore, $\lambda_{MTBF} \leq \sum\lambda - \lambda_{TBO} \leq 0.00067 - 0.00040 \leq 0.0027$

Hence, the new MTBF must be ≥ 3704 hr and its corresponding reliabilty $R \geq 0.99973$.

It is paradoxical that such factors as flight safety reliability and the increased cost of overhaul of a badly degraded gearbox (extensive secondary failures) may fix the TBO interval at a level that in turn requires a significant increase in the required MTBF to achieve a specified MTBR.

Relationships defined in this manner are tacitly assumed to fit a simple exponential failure distribution. This not only simplifies the arithmetic involved, but enables the designer to think directly in terms of the inverse relationships between the number of detail components comprising the subsystem and their intrinsic failure rate requirements. Although many individual gearbox components are better represented by other distributions — e.g., Weibull, gamma, and lognormal — the averaging effect on the subsystem as a whole is such as to invite cancellation effects, leaving a net MTBF value defined with sufficient precision to express design reliability requirements.

The "fit" of a given set of gearbox failure data to the exponential assumption reveals certain characteristics of the system in question. If the operating time of a group of gearboxes is sufficiently long, a definite wear-out trend will be observed as a nonconstant or increasing failure rate. One or more time-sensitive (wearing) components eventually will begin to dominate the failure picture as the end of their useful life is approached. Generally, those components operating in deep boundary layer lubrication regimes will be the first to influence the picture. In such instances the wear will progress to a state wherein conditions become favorable for the occurrence of a failure mode indigenous to that new set of operating conditions established by the particular wear state. Fig. 4-9 gives an example of a well-developed or debugged gearbox with relatively low TBO that satisfies the random failure characteristics accurately defined by the exponential distribution (Ref. 27).

Fig. 4-9 represents a gearbox with a 350-hr scheduled TBO, an MTBF of 502 hr, and a 1-hr reliability of 0.9980. However, due to the 350-hr scheduled removal frequency, the resulting MTBR is 204 hr.

Fig. 4-10 from Ref. 28 represents the distribution evidenced by a well de-bugged gearbox with relatively high TBO. In this case the upward inflection or concavity of the data points reveals the strong influence of component wear-out as the TBO level is approached.

Similarly, Fig. 4-10 represents a gearbox with an 1100-hr scheduled TBO, an MTBF of 507' hr, and a one-hr reliability of 0.9998. In this case the scheduled removal frequency results in an MTBR of 904 hr.

Obviously, design MTBR and MTBF values are dependent upon the designer's knowledge of representative failure modes and his ability to assign reasonably accurate failure rates. Extensive test and service experience with analogous components and subsystem elements is essential in arriving at realistic predictions. There is little published literature to aid the designer, but the selection of components with known lower generic failure rates always should be the objective. It should be recognized that generic failure rates (indicative of the intrinsic reliability characteristic of any component) cannot truly exist apart from the environment in which the component functions. Some insight into these environmental influences is given in par. 4-2.1.1.

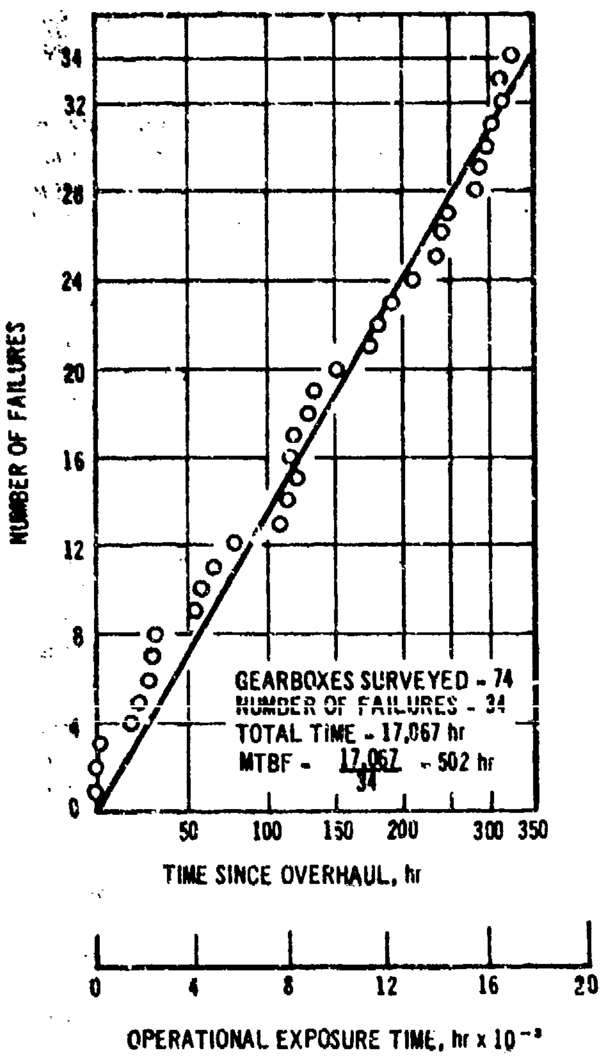

Figure 4-9. Number of Failures vs Hours Since Overhaul — MTBF ≈ 500 hr

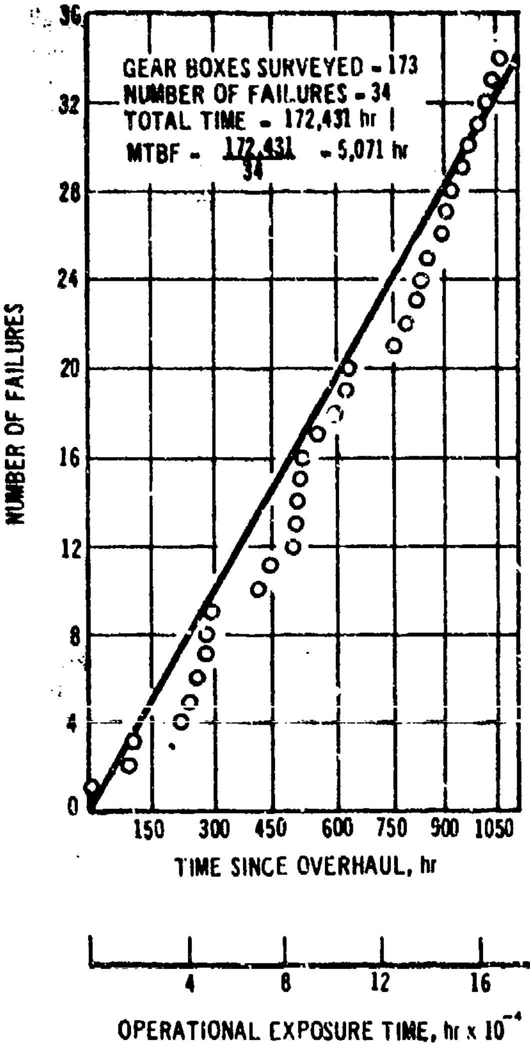

Figure 4-10. Number of Failures vs Hours Since Operation MTBF ≈ 5000 hr

Satisfactory failure rate estimates for rolling element bearings may be derived by several methods. One simple but sufficient value is:

$$\lambda = (1 - S)/L \text{, hr}^{-1} \quad (4\text{-}14)$$

where

S = probability of survival, dimensionless

L = design life or scheduled removal time (TBO), hr

The value for S is read from the curve in Fig. 4-11 corresponding to the ratio of the design life L to the life B_{10} at which 10% of the bearing population will fail. The B_{10} value for a given bearing is determined from the bearing manufacturer's data for the root mean cube RMC load. The applicable RMC load is based upon the profile of the design mission; i.e., for a mission profile of n discrete values of bearing load P_i each occurring for a percentage a_i of the total operational time:

$$RMC = \left(\frac{\sum_{i=1}^{n} a_i P_i^3}{100}\right)^{1/3}\text{, lb} \quad (4\text{-}15)$$

where

$\sum a_i = 100$

Although in many instances the life-load exponent more correctly can be taken as 4, the cube value is generally recommended for determinations of failure rate λ.

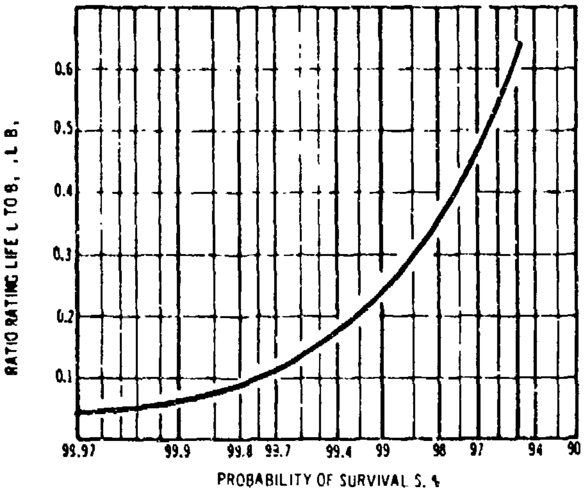

Figure 4-11. Probability of Survival vs L/B_{10} Ratio

Gear failure rates can be determined similarly from design stress levels. Properly designed gears in helicopter applications will exhibit pitting as the life-limiting failure mode (par. 4-2.2.1). The life-stress relationship for gear teeth is far more complex than for bearings because the specific sliding values generally are far higher, more types of metals and heat treatments are prevalent, and the elastohydrodynamic and chemical effects of the lubricant are known with less precision.

Available stress-life curves more often than not are based upon the mean pitting, or spalling, endurance of an unknown statistical sample (Ref. 29). Intensive research is underway by many organizations (ASME Research Program on the Relationship of Lubrication and Fatigue in Concentrated Contact, for example) that should result in the preparation of more meaningful stress-life pitting endurance charts that consider lubrication regimes, materials and metallurgy, and sliding speeds. The AGMA data of Ref. 29 reflect use of a stress-life exponent of between 9 and 10, whereas values of 5 have been reported (Ref. 30) for operation in Lubrication Regime I (Fig. 4-3). However, in the absence of well-documented, statistically-significant, test data, the AGMA data should be taken as representative of most gear applications. Use of the *RMC* value of the Hertzian or compressive stress in the contact area to obtain the mean spalling life from Fig. 4-12 is satisfactory, although the quartic mean level has been shown to offer excellent correlation in Lubrication Regime I.

There are many suitable techniques for reducing this life to a value of failure rate λ. One rather simple method uses standard Weibull paper to reduce the mean life to the level L_2 at which 2% of population will have failed by employing a dispersion exponent, or slope of 5.5. This rather steep slope is representative of typical helicopter gear performance where excellent quality control generally results in lower population dispersion. Very steep slopes frequently typify Lubrication Regime I. The life value L_2 for 98% reliabilty may be read from Fig. 4-13, a Weibull plot. The resultant failure rate λ is then:

$$\lambda = 0.02/L_2 \tag{4-16}$$

The mean value may be taken as the 50% or median rank for such steep slopes without loss of significant accuracy in using Fig. 4-13.

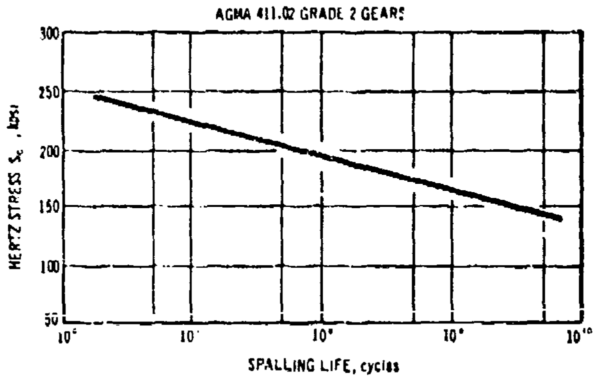

Figure 4-12. Spalling Life vs Hertz Stress

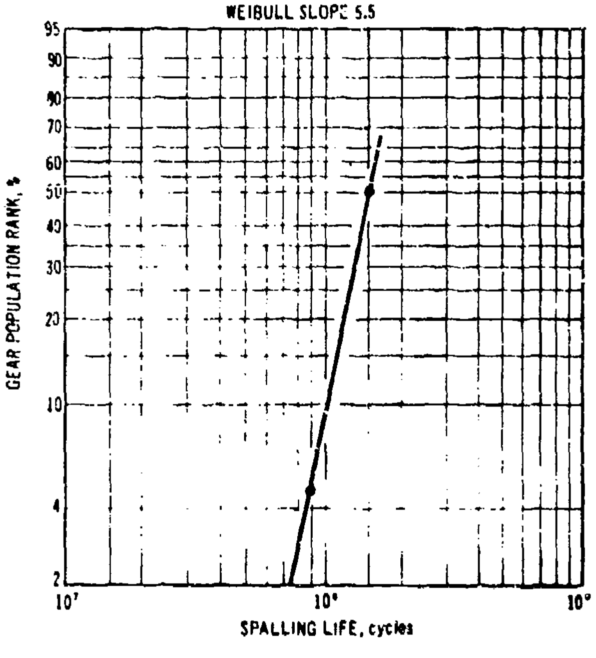

Figure 4-13. Weibull Plot — Spalling Life vs Gear Population Rank

4-1.2.1.3 Maintainability

A general discussion of maintainability may be found in Chapter 11, AMCP 706-201. This discussion, therefore, treats considerations relating specifically to the transmission and drive system.

The basic concept of maintainability often is expressed as a requirement for a specific or maximum number of maintenance man-hours per flight hour (MMH/FH). Army helicopters of a few decades ago exhibited values as high as 35 MMH/FH while helicopters in the present Army inventory have values ranging from 0.5 for the OH-58 (Ref. 31) to 6.5 MMH/FH for the CH-54 (Ref. 32). Small helicopters as a rule show better maintainability values than the larger, more complex machines. However, the values for any given size of helicopter may vary by 300% depending upon design variables. Current RFP requirements are in the range of 5 MMH/FH for medium-sized, twin-engine helicopter for organizational, direct support (DS), and general support (GS) maintenance levels combined. A value so stated must be apportioned in turn (par. 4-1.2.1.2) to the various subsystems to establish individual design goals.

Achievement of satisfactory maintainability levels is dependent upon two factors:

1. High component reliability (par. 4-1.2.1.2)
2. Ease of maintenance.

Maintenance is generally thought of as comprising two categories: nonscheduled (due to random failure or accident), and scheduled (due to time change of wear-out components, interim servicing, and inspections). However, all maintenance concerned with component change is discussed herein as a group.

The generic failure rates of many external components in the drive system are such that the components may be ranked in order of required frequency of removal or adjustment. Components such as hydraulic and electrical accessories, rotor brakes, shaft seals, external hanger bearings, and drive shaft couplings require relatively frequent inspection or maintenance and must be designed for ease of removal and installation. Accessibility is the key criterion Such components must not be located too close to one another, and adequate wrench clearances must be provided for standard tools.

Subsystem components such as gearboxes are generally not maintained at the field or direct support level and, therefore, they must have simple and accessible attachments with structural clearances adequate to permit easy removal and replacement. The use of integral guide pins or tapered dowels is recommended in any case where heavy components must be aligned for the installation of mounting screws, bolts, or clamps. It also should be emphasized that true leveling of the helicopter is seldom achieved for component change at the direct support level. Therefore, when heavy components necessitate the use of hoisting devices, extra care must be taken in the design of alignment devices and structural clearances so as to reduce maintenance effort.

External shaft seals always should be assembled in easily removable housings or holders to permit bench changing of the seal element. Squareness, alignment, and cleanliness practices all are critical to the proper performance of a seal and are difficult to achieve when the seal element must be changed in place. The shaft upon which the seal operates also must be easily removable because good practice requires that the shaft be slipped into the previously installed seal, allowing the use of adequate shaft lead chamfers to minimize the danger of seal damage. It also is desirable to have the shaft engaged with the driving spline or some other guiding device prior to making contact with the seal to prevent excessive side loading of the seal.

The attachment of all external components should be such that one man can remove all fasteners and similar items. Two examples of *poor* design that require unnecessary manpower are:

1. Bolt-and-nut fasteners through structure where one man cannot reach wrenches on both elements. Tapped holes on nut plates are proper solutions even though a larger number of cap screws may be required because their rigidity or strength may be lower than that of a bolt-nut joint.
2. Components that one man must hold while another installs fasteners. Possible solutions include the use of guide pins, longer pilot flanges, slotted clearance holes, retaining pins or clips, or other friction devices to secure the component temporarily.

Components that require a specific orientation to function correctly should be designed so that they can be installed only in that position, if possible. When this is not practicable, as in the case of some Government-furnished electrical accessories, decals may be used at the pad location to provide installation instructions. Examples of one-way components are seal housings with drain fittings, hydraulic pumps that require line connection fitting orientation, and bearing hangers.

Components that require tight-fitting pilot bores or similar devices should be provided with jacking pads for removal. One man can operate two or three jack screws (tightening each one a little at a time), whereas their omission might necessitate the use of two men to pry simultaneously on both sides of a component (and possibly a third man to catch the component when it breaks free).

Proper performance of scheduled maintenance tasks such as inspection and servicing is dependent to some extent upon accessibility and convenience. Inspections that are convenient and of a go-no-go nature are likely to be performed on time and with accuracy; those that require considerable quantitative judgment may be missed or interpreted incorrectly.

For example, the presence of vital fluids in all gearboxes, transmissions, and other reservoirs should be discernible from ground level without opening of complex cowlings. Min-max oil levels should be used to eliminate the need for topping off, and the minimum level should be exactly one or two quarts below the maximum whenever possible to discourage the practice of saving half a quart of oil in an open can. The minimum level should be such as to allow completion of several additional hours of operation at the maximum likely oil consumption rate so as to eliminate the need for adding oil when the level is near minimum.

While accurate values of maintenance times for transmission and drive system components are not available for all current Army helicopters, published data are helpful in identifying present troublesome areas. Table 4-1 presents maintenance workload factors relative to drive subsystems (Ref. 33). The rankings given relate primarily to other maintenance factors on the specific helicopter listed, and are not to be interpreted as relating the workload on one helicopter model to that of another.

TABLE 4-1. US ARMY HELICOPTERS — TRANSMISSION AND DRIVE SYSTEM ONLY — MAINTENANCE WORKLOAD (Ref. 23)

PROBLEM TITLE	WORK LOAD RATING				
	VERY HIGH	HIGH	MED.	LOW	VERY LOW
OH-58A					
TAIL ROTOR DRIVE SHAFT			X		
INPUT DRIVE SHAFT			X		
OH 6					
TAIL ROTOR DRIVE SHAFT			X		
MAIN GEARBOX			X		
UH 1					
INPUT QUILL OIL SEAL			X		
INPUT DRIVE SHAFT			X		
AH 1G					
INPUT QUILL OIL SEAL	X				
CH 47					
SYNCHRONIZING DRIVE SHAFT		X			
OIL PRESSURE TRANSDUCERS					X
CH 54					
OIL COOLER ASSEMBLY		X			
MAIN GEARBOX			X		
INPUT CARBON SEALS		X			
ROTOR BRAKE SEAL ASSY			X		
ROTOR BRAKE SUPPORT ASSEMBLY				X	
ROTOR BRAKE DISK				X	
ROTOR BRAKE PUCKS			X		

4-1.2.1.4 Survivability

Survivability in transmission and drive system operation may be defined as the capability to sustain damage without forced landing or mission abort and to continue safe operation for a specified period of time, usually sufficient to return to home base or, as a minimum, to friendly territory. The damage may occur from either internal component failure due to wear, fatigue, or use of a deficient or inferior component; or a hit by hostile forces. The current Army requirements generally define the period of time for safe operation after damage as a minimum of 30 min at conditions within the maximum power and load envelope, except in the case of total loss of the lubrication subsystem; the acceptable maximum power level for safe operation upon loss of lubrication is generally reduced to that required for sustained flight at the maximum range speed at sea level standard condition.

Survivability following internal component failure can be enhanced through such detail design practices as identification of primary failure modes, and using configurations and arrangements to assure kindly failure modes and to limit failure progression rates. Attention also must be given to the elimination or retardation of secondary failures caused by primary failure debris, and to providing for positive failure detection long before a critical condition is reached. Safe operation with this type of damage normally can be achieved for durations of 30 to 100 hr. Design techniques applicable to this goal are discussed in pars. 4-2.1, 4-2.4, and 4-4.3.

Survivability in cases of combat hits is considered coincident with the reduction of vulnerability. Normal practice is to design a complete helicopter survivability-vulnerability program plan in accordance with ADS-11. This plan will include many elements peculiar to the transmission and drive system. A good program plan requires the active participation of the responsible transmission and drive system design activity to assure practicable approaches with minimum penalties in drive system performance, weight, and cost.

Reduction of helicopter detectability and the defeat of specified ballistic threats are important elements in vulnerability reduction. With regard to detectability, the primary area of concern in the case of the drive system is noise (par. 4-1.2.1.1.4). While the lower frequency noise levels are basically associated with the rotor and/or tail rotor and propeller, the higher frequencies are generally attributable to the transmission and drive, and propulsion systems and their accessories.

Typical Army requirements specify a maximum sound pressure level for helicopter hover and fly-by at a specific distance from the flight path. Design goals for appropriate frequencies and sound pressures are given in Table 4-2. Noise level survey requirements are described in Chapter 8, AMCP 706-203. Design techniques to secure external as well as internal gearbox noise reduction are discussed in par. 4-1.2.1.1.4.

Complete defeat of ballistic threats must be accomplished for smaller caliber ordnance, and damage minimized as much as possible for the larger calibers. Depending upon requirements peculiar to the mission, the drive system components must be capable of withstanding a single ball or armor-piercing 7.62-mm bullet at 2550 fps, aligned or fully tumbled, striking at any obliquity at any point in the system. The 75-deg solid angle of the upper hemisphere (with respect to the helicopter) normally is excluded. The larger ordnance for which damage minimization should be considered is 23-mm high explosive incendiary (HEI).

The specific techniques available to the designer to meet the stated requirements include:

1. Redundancy
2. Design configuration
3. Self-sealing oil sump materials
4. Emergency lubrication considerations
5. Armor.

4-1.2.1.4.1 Redundancy

Redundancy is typified by multiple engine configurations. In these configurations all individual drive subsystem components between the engines and the collector gear in the main gearbox or combining gearbox usually are excluded from the survivability requirement by nature of their functional duplication, provided that:

1. No single projectile can kill all duplicated power paths
2. A single power-path kill cannot cause secondary failure of the duplicated power paths due to fragmentation of the first.

These two criteria can be satisfied by:

1. Physical separation of the drive paths sufficient to reduce the impingement angle within which a single projectile can produce a multiple kill
2. Sufficient size and strength of the killed-path component to attenuate the projectile velocity below the kill threshold for the second path component
3. Use of structure between the paths to confine a fragmented or loose component to its immediate locale
4. Use of armor to confine fragments or prevent projectile impact.

4-1.2.1.4.2 Design Configuration

Configuration is by far the most important and efficient technique for reducing vulnerability. Many slight configuration changes can increase survivability greatly without serious compromise of efficiency, weight, or cost.

For example, case hardened gears with tough, fracture-resistant core structure have surprisingly good tolerance to ballistic damage. Spiral bevel gears and planetary gears, used effectively throughout the drive train of small and medium helicopters are invulnerable to the 7.62-mm threat. Planetary ring gears may be penetrated so that the planet idler gears cannot mesh at a particular segment, but the remaining gears pick up the overload necessary to continue normal power transmission. The relatively high contact ratios and coarser pitch of spiral bevel gears are factors that make them particularly resistant to failure from loss of a single tooth segment. Narrow-face spur gears (less than 0.5 in.) can be a problem, and, therefore, it is desirable to use greater face widths in all primary power paths.

Experience with 12.7-mm ammunition is less extensive than with the 7.62-mm projectiles, but the same general observations hold true with a slightly larger scale of reference. Gear rims, webs, and integral outer-race sections of planetary idlers should be proportioned such that a ricochet entering the mesh will deform, fracture, or crack the gear teeth rather than the tooth supporting structure. Extensive observation of main rotor gearboxes damaged by 7.62-mm ball and armor-piercing (AP) ammunition have

TABLE 4-2. EXTERNAL NOISE LEVEL

EXTERNAL NOISE LEVEL

FREQUENCY, Hz		SOUND PRESSURE LEVEL, dB PERCEIVED
BAND	CENTER*	
OVERALL		90
22.4-44.7	31.5	86
44.7-89.2	63	85
89.2-178	125	85
178-356	290	86
356-709	500	88
709-1,410	** 1000	85
1,410-2,820	2000	81
2,820-5,630	4000	76
5,630-11,200	8000	72

*NOMINAL CENTER FREQUENCY (DETERMINED EMPIRICALLY)
**DRIVE SYSTEM FREQUENCIES

shown the digestive capabilities of conventional helicopter gearing to be quite adequate to discharge the spent bullet into the oil sump without functional failure of the power transmission system.

Integral gear shafts, quill shafts, and external interconnect or tail rotor drive shafts must be of sufficient diameter to withstand edge hits from fully tumbled bullets without failure. In thin-wall aluminum shafts operating with a minimum of 20% margin on first whirling critical speed, an external diameter of 3.0 in. is sufficient to defeat the 7.62-mm threat while a 4.0 in. diameter is necessary to defeat the 12.7-mm threat. Steel shafting may be considerably smaller depending upon the wall thickness employed. Of course, in event of damage the remaining portion of the shaft must have sufficient strength to transmit the required maximum torque.If the column buckling torque is 300% or more above this torque, a simple shear-stress calculation of the remaining post-impact area is sufficient. However, when the buckling margin is less, it is generally necessary to conduct real or simulated ballistic tests to demonstrate the adequacy of the design. Note that it is undesirable to increase diameters excessively since vulnerability is then increased for fuzed round threats.

Most ball and roller bearings are fabricated from through-hardened steels and, consequently, usually will fracture through the outer ring when struck by a bullet at near-zero obliquity. The surrounding case and liner structure serve to expend some of the kinetic energy of the bullet; however, conventional thicknesses of these structures generally are insufficient to prevent fracture of the bearing ring. Orientation of the rolling elements of the bearing at the instant of impact has much to do with the ring fracture mode. A zero obliquity hit between rolling elements frequently will discharge a double-fractured "pie-slice" ring segment into the bearing, while an aligned hit often produces a single outer ring fracture and frequently fractures the rolling element as well. When the design allows, a space between the outer gearbox wall and portion of the housing supporting the bearing liner and ring is effective in reducing bearing damage. This space provides a place for the spalling debris from the initial impact to expand and eject,

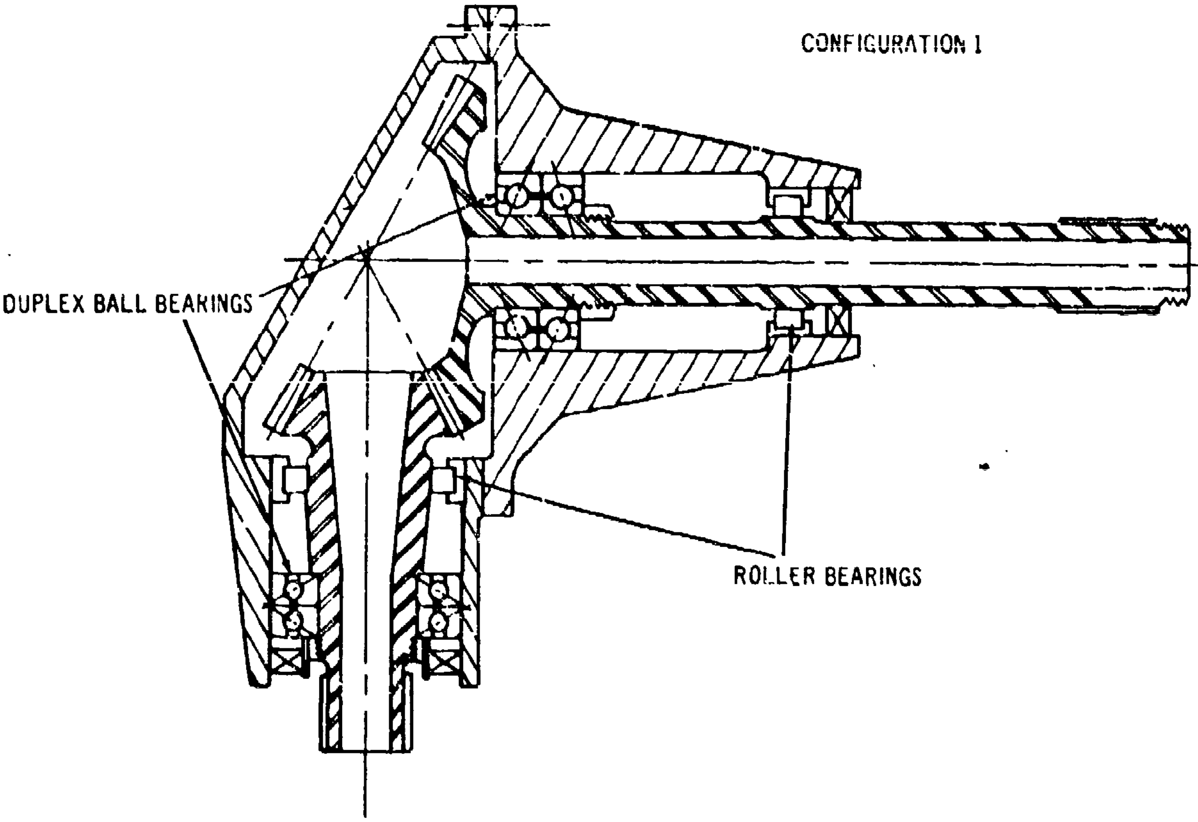

Figure 4-14. Typical Tail Rotor Gearbox — Vulnerable

thereby reducing the impact shock on the bearing ring.

Some insight into useful design techniques may be gained from examining three configurations of a simple, spiral bevel gear, 90-deg tail rotor gearbox as used on most helicopters with single main rotors. Fig. 4-14 is a schematic representation of a typical minimum-weight design featuring overhung pinion and gear mountings, designated as Configuration 1. The pinion and gear are both in overhung mountings supported by duplex ball and cylindrical roller bearings. A single 7.62-mm hit on any one of the four bearings probably would not result in instant functional failure; the bearing would continue to operate for some time because the considerable driving torque would break up and eject the relatively frangible rolling elements and cage of the damaged bearing. However, direct hits on either the pinion cylindrical roller bearing or the duplex ball bearing supporting the gear would soon result in excessive loss of gear mesh position. As the effective radial clearance of either of these bearings increased with the resulting rapid bearing deterioration, the operating backlash of the gear teeth similarly would increase while the depth of tooth engagement would decrease correspondingly. The probability of the gear teeth skipping mesh or breaking off upon application of significant yaw control would be great. Hits on the outboard bearings would yield far lower probability of gear mesh failure.

Configuration 2 is shown schematically in Fig. 4-15. The bevel gear set is identical to that of Configuration 1. Pairs of the same types of bearings used in Configuration 1 are now used to straddle mount both pinion and gear members. In this configuration the increasing radial clearance in any bearing sustaining a hit will result in less deterioration of the gear mesh, with a corresponding decrease in the probability of gear failure upon sudden yaw control input. A gearbox of this configuration designed for tail rotor

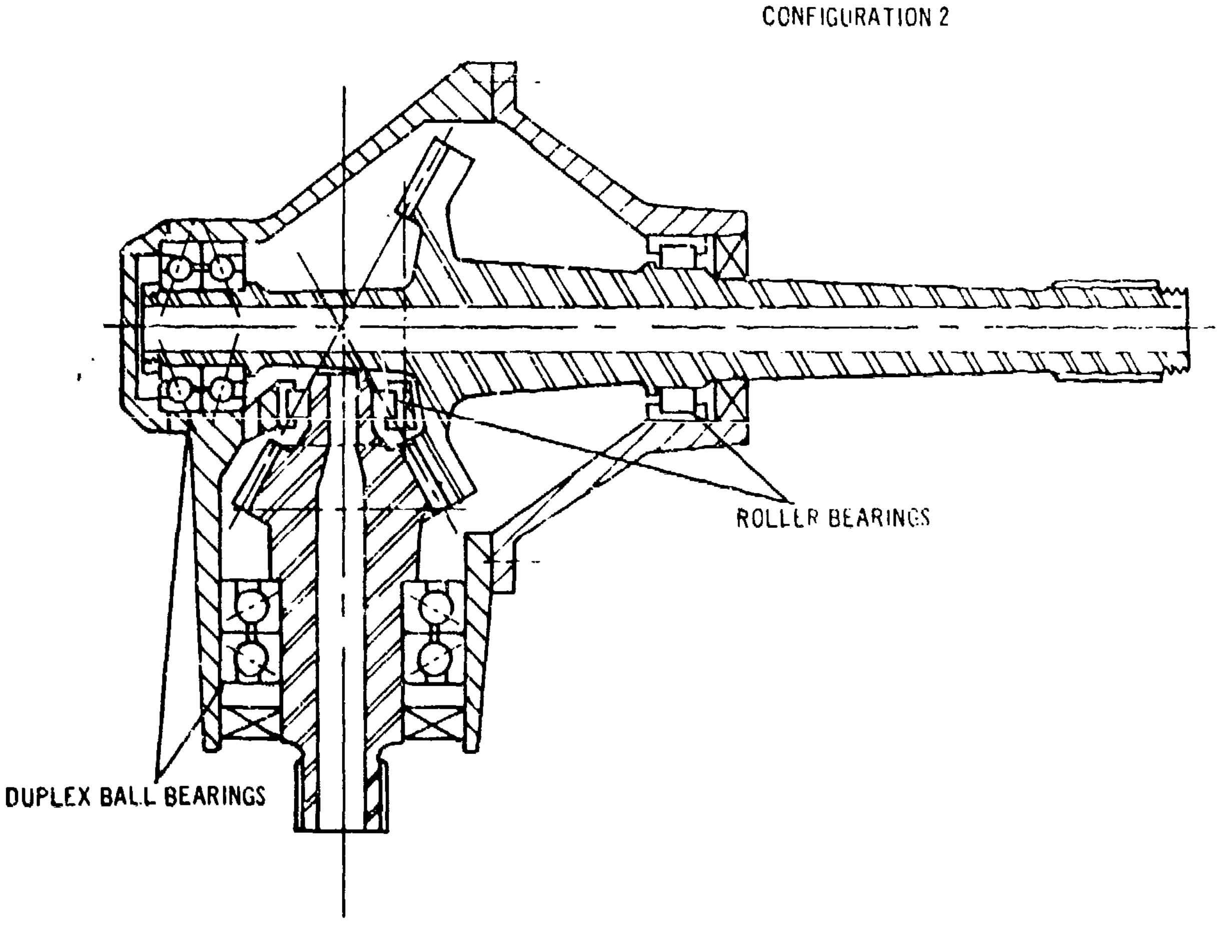

Figure 4-15. Tail Rotor Gearbox — 7.62 mm Proof

steady hover power of more than 150 hp would be judged capable of the required 30 min operation subsequent to a 7.62-mm bullet impact. However, the probability of functional failure after receiving a 12.7-mm hit would be quite high unless the bearing and gear components were inordinately large.

Configuration 3 is shown schematically in Fig. 4-16. This configuration has been arranged to defeat 12.7-mm threats with far less weight penalty than would be incurred by oversizing the elements of Configuration 2. The overhung mounting of Configuration 1 and the straddle mounting of Configuration 2 have been combined in this redundant or composite system. Both pinion and gear members are supported by two conventional cylindrical roller bearings and one duplex ball bearing pair. Emergency thrust shoulders are incorporated on the shafts adjacent to the integral roller bearing inner raceways. Sufficient axial clearance should be provided between the roller elements and the inner race thrust shoulders or flanges to preclude contact under normal operation conditions (including extreme cold when the light alloy housings have contracted relative to the steel shafts). However, upon functional failure of either duplex ball bearing, emergency axial location is provided by these thrust flanges. Use of a three-bearing system permits total functional loss of any one bearing without seriously compromising the operating parameters of the gear mesh, however, bearing alignment becomes more critical. As a result, one bearing of the three-bearing system must be designed with greater internal clearance than normal. The spiral bevel gear set shown in Fig. 4-16 has been enlarged slightly relative to the gear set shown in the prior two configurations to decrease vulnerability to 12.7-mm hits directly in the gear elements.

While numerous other configurations and types of bearings can be used to accomplish the same objectives, the logic used to provide inherent survivability remains unchanged. Similar principles should govern the design of the entire drive subsystem. Their application, of course, becomes more involved as the complexity of the gearbox design increases.

All shaft couplings, joints, hanger bearings or

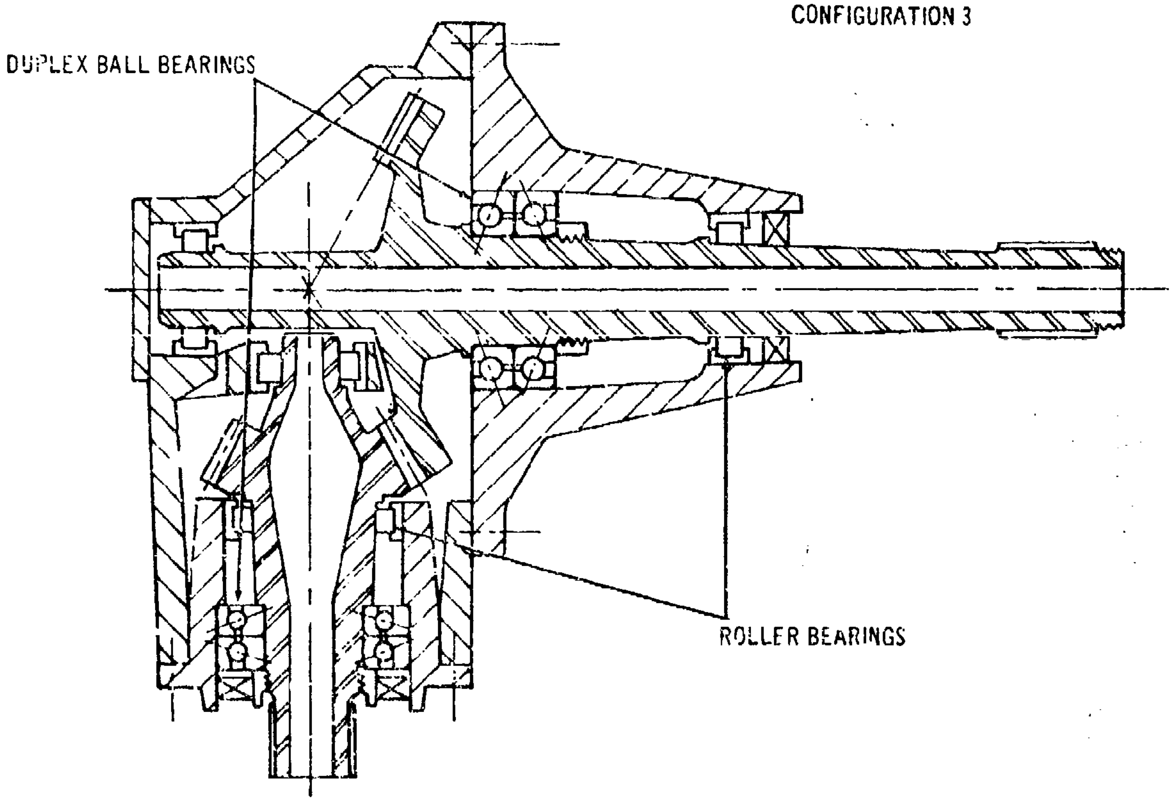

Figure 4-16. Tail Rotor Gearbox — 12.7 mm Proof

pillow block housings, tail rotor and intermediate main gearboxes, and input/output quill assemblies must be joined, retained, or mounted with a sufficient number of redundant fasteners to preclude loss of function from a single projectile. For well separated attachment points, four fasteners often suffice. However, rotating shaft joints and couplings often require six or more fasteners. Frequent use of flanges, ribs, and abrupt section changes in castings, housings, and similar structures provide effective stoppage of crack propagation, while enhancing heat rejection to the atmosphere. Internal ribs in oil sump areas are desirable because of the possibility of cracking by hydraulic ram effect in the oil as well as by projectile impact.

4-1.2.1.4.3 Self-sealing Sumps

Another design technique involves the use of self-sealing materials in the gearbox oil sump area. The most efficient material now available is defined as Type II in MIL-T-5579. This rubberized self-sealing compound originally was developed for fuel cells and can be fabricated to defeat either the 7.62-mm or the 12.7-mm threat. Another excellent defense material for 7.62-mm threat is a cast urethane coating approximately 3/8 in. thick. With the latter material, the design of the sump should be relatively simple, as in a casting cope where the drag may be withdrawn without use of breakaway core prints. The cast coating contracts after pouring and high residual compressive stress results. This prestressed resilient coat then shrinks to close completely the hole left by the piercing bullet. The coating is relatively dense (2.0 lb/ft^2 for the 7.62-mm threat) and also serves as an excellent heat insulator and noise and vibration damper. Its density is such that the surface area to be coated should be kept to a minium to reduce the attendant penalty on sizing of the oil cooling system. Flat shallow oil pans and sumps often provide the most efficient configurations.

The various shaft seals must be designed so that a direct hit cannot cause all the oil to leak from a gearbox. This may be accomplished by using nonrubbing labyrinth or slinger seals in series with the conventional contacting face- or lip-type of seal and by limiting the oil flow rate at the inboard seal face to a minimum.

Where external oil coolers and lines are used, current specifications often require the use of emergency oil shut-off valves to divert the oil directly back to the transmission lubrication distribution system in the event of a cooler or line hit, thus preventing total loss of oil. One such device is defined in Ref. 34.

4-1.2.1.4.4 Emergency Lubrication

The preferred method of reducing vulnerability is to assure fail-safe or emergency lubrication in the event of total loss of the normal lubricant supply. This capability must allow continued safe operation for 30 min at minimum cruise power at mission gross weight. Oil dams, wicks, and other means of retaining a minimum oil supply in the critical bearing areas are simple techniques to employ. Ball and roller bearing cages may be fabricated from sacrificially wearing, self-lubricating composite materials (Ref. 35) such as polyimides, Teflon-filled Fiberglas matrices, and silver-plated, high-temperature steel. The US Army Ballistic Research Laboratories (BRL) has demonstrated composite idler gears that wear off on meshing drive gears, thus providing a form of gear tooth lubrication (Ref. 36).

Ref. 37 reports a successful application of a grease developed specifically for helicopter gear and bearing lubrication. However, the normal lubricants (MIL-L-7808 or MIL-L-23699) serve the equally important functions of reducing friction and cooling. Ref. 38 describes the requirement that thermal equilibrium for the new "dry" running condition be established to achieve 30-min safe operation after loss of the cooling oil. The equilibrium can be established only by maintaining adequate running clearances and backlash in the bearings and gears in the presence of the thermal gradients that exist in the "dry" condition, with its altered frictional heat sources and modified conduction, radiation, and convective heat rejection paths. The emergency "friction reducing" lubricants can be of value in sustaining safe operation only in such a case. If a gear or bearing loses running clearance, a rapidly degenerative sequence of events results in catastrophic failure. Loss of operating clearance results in abnormally high heat generation because the gear teeth and bearings operate under interference conditions with attendant overloads. This heat generation in turn produces an increase in the thermal gradients, resulting in a further increase in overload and interference until the bearings seize or the gear teeth get so hot that they undergo plastic failure. Specific methods of preventing such occurrences are discussed in par. 4-4.4.3. The general recommended design procedure (Ref. 38) is as follows:

1. Design for minimal frictional losses commensurate with available manufacturing ability.

2. Calculate frictional losses for the "dry running" regime. An average friction coefficient $f = 0.16$ is suggested for the first approximation. Use this value with the mean values for sliding velocity, and load in the gear meshes and bearing contact areas.

3. Construct a thermal map with probable steady-state "dry running" temperature gradients.

4. Redesign all gear and bearing elements to provide some clearance under the mapped gradients. Added clearance should be provided at high-rate frictional heating sources to accommodate transient conditions. For example, relieving clearance will not be provided by expansion of the gear case until the increased heat generated by dry operation has heated the case.

5. Use materials with adequate hot hardness and friction properties for thermally vulnerable components.

6. Provide self-lubrication of bearings where possible. Methods include the use of suitable cage materials or the use of appropriately located wick devices.

7. Re-calculate bearing lives and gear stresses for the increased clearance conditions occurring during operation in the normal lubrication regime. Adjust design parameters accordingly; i.e., increase face widths or pitch of gear members, along with bearing capacities, as required.

4-1.2.1.4.5 Armor

In some cases it may be appropriate to employ armor plate to protect the vulnerable component. This design technique is the least preferred because it adds weight, increases maintenance task times, and penalizes the full-time payload.

In such applications, integral armor is preferred over parasitic or bolt-on armor. Not only is the weight penalty slightly less with integral armor, but the pitfalls of increasing payload at the expense of armor removal will be eliminated. For most applications dual-hardness steel armor will be the most efficient type to integrate because it can be rolled, welded, bolt-fastened, or integrally cast. Design of armor installations is discussed in detail in Chapter 14.

4-1.2.2 Drive System Configurations

The specific requirements for the drive system are dictated by the detailed configuration layout and vehicle requirements. AMCP 706-201 sets forth the evolution of an approved preliminary design configuration, which will include detailed requirements for transmission subsystem power input and output drives; i.e., the speeds, powers, location, and relative orientation of these drives. Typical configuration requirements for existing Army helicopters are discussed further in the paragraphs that follow.

4-1.2.2.1 Single Main Rotor Drive System

The majority of helicopters in current use are of the single main rotor configuration and are powered by either one or two engines. A shaft-driven, single-lifting-rotor helicopter always employs antitorque reaction and thrust device to counteract main rotor driving torque and to provide yaw control for helicopter maneuverability. A shaft-driven tail rotor located at the aft end of the tailboom and arranged as either a pusher or a tractor propeller is used most commonly.

The tail rotor shaft is driven through a 90-deg bevel gear set that in turn is driven from the main rotor gearbox by a long driveshaft or series of connected driveshafts. The power takeoff from the main rotor gearbox for the tail rotor is geared to the main rotor drive downstream of the output side of the free-wheeling or overrunning clutch located between the engine(s) and the main rotor gearbox; this arrangement permits full yaw maneuver capability during auto-rotation or engine-out operation.

Accessory drive requirements vary extensively and are dependent upon the primary vehicle mission and helicopter size. These drives may be mounted on the main gearbox or isolated in an accessory gearbox that is driven by a shaft from the main gearbox. Secondary accessory elements may be driven from the tail rotor driveshaft. Table 4-3 identifies certain configuration characteristics for the single main rotor helicopters in current Army use.

It should be noted that accessory requirements increase with the size of the helicopter. Light observation helicopters (LOH's) have few accessory requirements and possibly no drive redundancy. In general, these helicopters may be flown safely without hydraulic boost of the flight controls, and the battery suffices for emergency electrical supply in the event of failure of the engine-driven generator.

Utility helicopters (UH) frequently require redundant hydraulic pump and electrical generator drives due to the magnitude of the rotor control loads and the increased electrical loads attendant upon the larger amounts of instrumentation, electronics, and mission-essential equipment.

Cargo helicopters (CH) often must have auxiliary power unit (APU) for ground operation and check-out of electrical and hydraulic subsystems. It is common practice to employ an independent accessory gearbox driven through overrunning clutches from both APU and main rotor gearbox to permit accessory operation from either power source.

4-1.2.2.2 Multilifting-rotor Drive Systems

Multilifting-rotor helicopters have been designed and tested in many configurations — such as fore and

TABLE 4-3. HELICOPTER DRIVE SUBSYSTEMS — SINGLE MAIN ROTOR

HELICOPTER DESIGNATION	ENGINE OUTPUT		MAIN ROTOR GEARBOX REDUCTION STAGES				TAIL ROTOR AND ACC'Y DRIVE DATA			
	SPEED, rpm	POWER, hp	SPUR	SPIRAL BEVEL	PLAN-ETARY	M.R. SPEED, rpm	T.R. SPEED, rpm	ACC'Y DRIVES	INT'MED. G.B. RATIO	T.R.G.B. RATIO
	SINGLE ENGINE									
OH-6	6,000	312	NONE	2	NONE	456	2,041	△1	NONE	0.67:1
OH-58	6,180	312	NONE	1	1	354	6,130	△2	NONE	2.35:1
UH-1H AH-1G	6,600	1,400	NONE	1	2	324	4,301	△3	1:1	2.6:1
	TWIN ENGINE									
UH-1N	6,600	1,800	△7	1	2	324	4,302	△3	1:1	2.59:1
CH-3	18,966	2,500	△8	1	1	203	3,030	△4	1:1	2.44:1
CH-53	13,600	7,560	NONE	2	2	185	3,010	△5	1.31:1	2.91:1
CH-54	9,000	7,900	NONE	2	2	185	3,020	△6	1.22:1	2.91:1

NOTES:

△1 ONE ACCESSORY PAD ON MAIN GEARBOX, 4,200 rpm, FOR TACHOMETER GENERATOR.

△2 ONE ACCESSORY PAD ON MAIN GEARBOX, 4,200 rpm, TACHOMETER GENERATOR AND HYDRAULIC PUMP IN SERIES.

△3 4 OR 5 PADS ON MAIN GEARBOX; 2 OR 3, 4,200 rpm FOR TACHOMETER GENERATOR AND 1 OR 2 HYDRAULIC PUMPS; 2, 6,500 rpm OR 1 EACH 6,600 AND 8,000 rpm, DC GENERATOR, ALTERNATOR, COOLING FAN DEPENDING ON CONFIGURATION.

△4 2 AC GENERATOR, 8,000 rpm; 3 HYDRAULIC PUMPS; 3,700 rpm; 2 LUBE PUMPS, 2,500 AND 3,700 rpm; AND TACHOMETER GENERATOR, 3,900 rpm.

△5 ACCESSORY G.B. POWER TAKEOFF, 6,020 rpm; SERVO PUMP, 4,500 rpm, TACHOMETER GENERATOR, 4,200 rpm.

△6 2 ACCESSORY GENERATORS, 8,000rpm; 4 HYDRAULIC PUMPS, 2 4,300, 1 EACH 3,700 AND 3,200 rpm; AUXILIARY SERVO PUMP, 3,700 rpm.

△7 ENGINE COMBINING GEARBOX APPROXIMATELY 5:1; 1 SPUR AND 2 HELICAL STAGES

△8 1 SPUR AND 1 HELICAL STAGE.

aft disposed, laterally disposed, coaxial, and quadrilateral main rotor arrangements. All of these layouts feature counter-rotation of even numbers of main rotors to cancel the torque reactions and hence eliminate the requirement for nonlifting antitorque devices. All multirotor helicopters require rotor synchronization, which usually is accomplished by interconnect shafting between the individual main rotor gearboxes, or by dual-output reversing reduction gearing in the case of the coaxial configuration. In instances where separate engines are located at each main rotor gearbox, the cross-shafting supplies power to each rotor for engine-out operation. In any instance, the interconnect drive is essential to safety of flight, requiring reliability comparable to that of the main rotor mast and thrust bearing. The interconnect drive is located downstream from the engine free-wheeling clutches.

The only multilifting-rotor helicopter in current use by the US Army is the tandem-rotor CH-47. This helicopter features twin engines of 2650 hp at 15,160 rpm. The engines are located in outboard nacelles high on either side of the aft third of the fuselage. They drive directly into 90-deg reduction gearboxes that drive into a combining gearbox also with 90-deg shaft angle spiral bevel gears. The combining box is an integral part of the interconnect synchronizing drive to the forward and aft rotor gearboxes. These rotor gearboxes each feature a single spiral bevel and two planetary reduction stages with final output at 230 rpm. The accessories are all located at the aft main rotor gearbox and consist of oil cooler blower, two electrical generators, and two hydraulic pumps.

4-1.2.2.3 Compound Helicopter Drive Systems

Compound helicopters are those that use auxiliary propulsion devices other than the main lifting rotors in the forward flight mode. The majority of such designs have featured a single main lifting rotor, an antitorque rotor, and either turbojet engines or shaft-driven propellers for auxiliary propulsion.

The only compound helicopter to undergo development test for Army use has been the AH-56. It was powered by a single 3450-hp engine driving directly into the main rotor gearbox. A spiral bevel gear stage, a compound planetary, and a simple planetary provide the reduction gearing for the main rotor. A spur takeoff located at the engine input to the main rotor gearbox drove a shaft running along the top of the tailboom. This shaft drove the pusher propeller at the end of the tailboom directly; and through a 90-deg shaft angle spiral bevel gear set also drove the antitorque rotor. Accessories were mounted at the main rotor gearbox and consisted of two hydraulic pumps and a high-speed generator.

4-1.3 TRANSMISSION DESIGN AND RATING CHARACTERISTICS

All elements, components, and subassemblies of the transmission and drive system are subject to varying degrees of wear, abuse, fatigue, and other environmental hazards. In many instances, standard components will provide acceptable performance for a given drive system design at a savings in cost, ease of procurement, logistics, and maintenance over specially designed components. However, the designer must have a thorough understanding of the likely failure modes of standard components (pars. 4-2.1 and 4-2.2) and the pertinent life-load or life-environment relationships.

It is customary to specify an input torque limit for a helicopter main rotor gearbox. Indicated to the pilot by a torquemeter red-line, this limit may be lower than the sea-level-standard rating of the engine(s).

Depending upon such factors as helicopter type and design mission, the red-line torque usually is specified as a continuous rating, or, in rare instances, a 5-min rating. A 5-min limit may be specified for emergency operation only. A time limit is imposed because sufficient cooling capacity is not available for extended operation, lubricants may be degraded, or simply to achieve longer life of drive system components. Sufficient cycles will be accumulated at the 5-min rating during the service life of the drive subsystem to require the same bending fatigue gear design, i.e., infinite life, as would be required for a continuous rating at the same red-line limit. Although a 5-min drive system rating does not usually impair the operational capability of a helicopter with a typical speed-power relationship (Fig. 4-17), current Army specifications include a continuous drive sytem rating. A typical requirement would be a continuous rating of the main transmission equal either to 120% of the power required to hover out-of-ground-effect (HOGE), zero wind, at the density altitude defined by 4000-ft pressure altitude and 95°F, or to 100% of the intermediate power rating of the engine(s) at sea level and 95°F. The effects of power ratings upon life, overhaul, and selection of standards are discussed in the paragraphs that follow.

4-1.3.1 Power/Life Interaction

The mechanical failures of interest to the drive subsystem designer usually exhibit a definite relationship between life and power. The life-limiting failure mode of primary concern for a developed and serviceable gearbox is pitting or spalling of the gears and bearings (par. 4-2.1). However, the life/power relationship for this mode of failure is not reckoned with easily due to the many factors that govern the

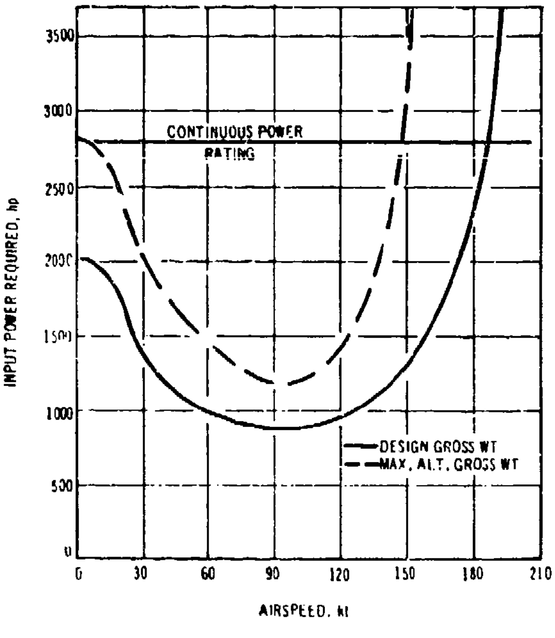

Figure 4-17. Typical Speed-Power Function

relationship. The metal chemistries, heat treatments, lubricants, loads, specific sliding ratios, velocities, temperatures, geometric shapes, surface textures and roughnesses gearbox deflections, and lubricant chemistry (including water content, and other contaminants) all influence the life of the surfaces in contact, or more properly, in conjunction. It is not unusual to observe dramatic life differences between two supposedly identical gearboxes when but one of the given variables is changed by manufacturing scatter, operating variability, system wear, or environmental factors.

In a complex system of gears and mixed bearing types, it is generally acceptable to use Miner's rule of cumulative damage in a simplified form for life prediction. A representative root-mean-cube power level is calculated from the assigned mission profiles using Eq. 4-15. The value of compressive, or Hertz, stress S_c corresponding to the *RMC* power or load is then calculated, and the life determined from an applicable S-N curve.

The Hertz-stress/life relationship varies significantly (Fig. 4-18). Each function shown results from data representing a particular set of design and operating variables. The wide variance among these functions emphasizes the danger in the use of a Hertz-stress/life function without consideration of the assumptions and test conditions.

Because calculated Hertz stress is an exponential function of load, little generality is lost by assuming exactness for the *RMC* life-load relationship and selecting an appropriate classic or modified stress-life function to predict the life of any particular conjunction whose variables are most nearly represented by the chosen function.

As an example of the selection and application of an *RMC* power, consider the three-mission profile spectra shown in Fig. 4-19. The UH-1H and AH-1G power histograms were constructed from flight recorder data (Ref. 43). The third histogram was constructed using the fatigue spectrum supplied with a recent Army RFP for a helicopter with a mission role

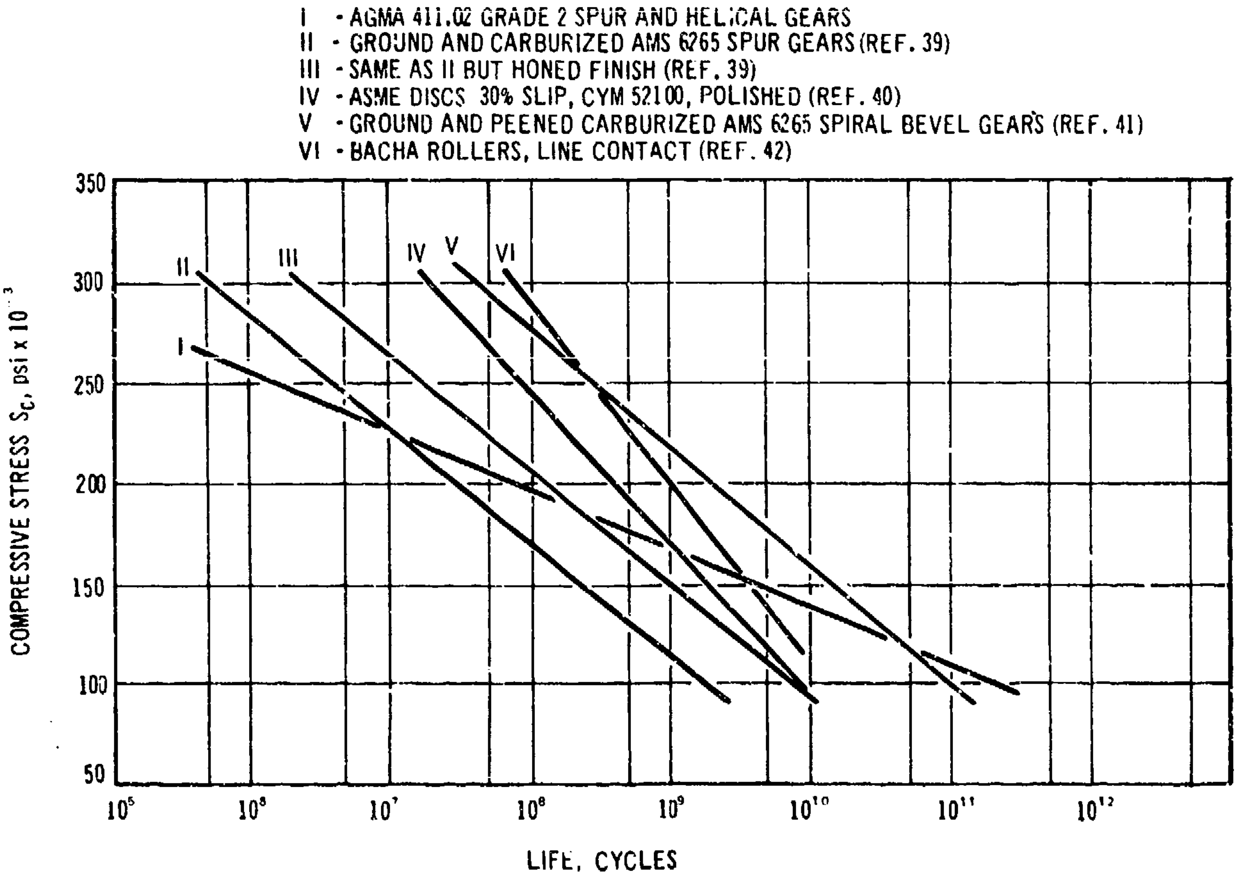

Figure 4-18. Gear Stress vs Life Curves

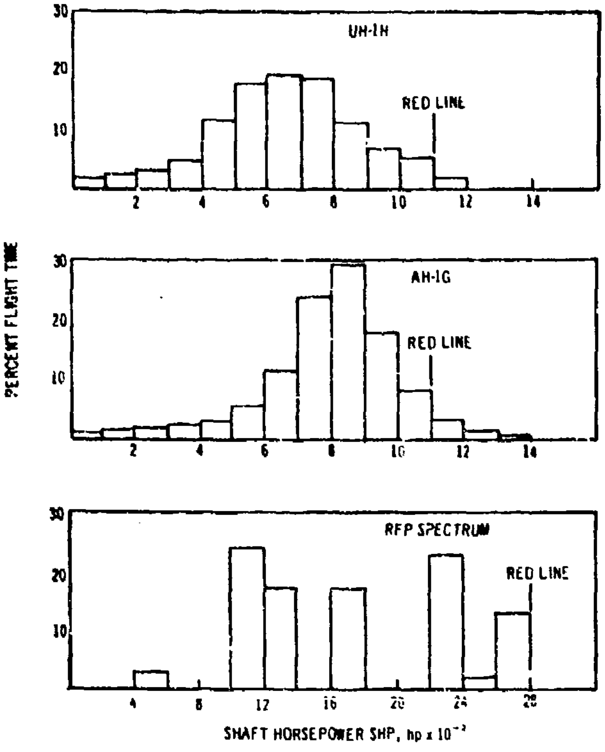

Figure 4-19. Shaft Horsepower Spectra Histograms

similar to the AH-1G but powered with twin advanced technology engines. The red-line and flight profile powers corresponding to this fatigue spectrum are taken from Fig. 4-17. The *RMC* powers for the three spectra are: UH-1H, 714 hp; AH-1G, 827 hp; and RFP, 1939 hp; representing 65%, 75%, and 69%, respectively, of the red-line powers for the three helicopters. However, because the sea level standard intermediate power ratings of the engines for the three helicopters are 1400 hp, 1400 hp, and 3000 hp, respectively, the *RMC* powers represent 51%, 59%, and 65%, respectively, of installed engine power.

On the assumption of no changes in lubrication state with advancing wear, the stress-life functions of Fig. 4-18 predict differences in the expected service lives of the same transmission system used in UH-1H and AH-1G helicopters based upon their respective *RMC* powers. For purposes of comparison, assume that the red-line power corresponds to a maximum stress S_c = 200,000 psi in a particular gear mesh. Because the Hertz stress in a spur gear is proportional to the square root of the load, which in constant speed operation is proportional to the transmitted power, the stress under *RMC* and red-line power will be related by the square root of the power ratio. Thus the stresses under the power for the UH-1H and AH-1G, respectively, will be:

$$(S_c)_{UH\text{-}1H} = (0.65)^{1/2}(200{,}000) = 161{,}200\text{ psi} \quad (4\text{-}17)$$

and

$$(S_c)_{AH\text{-}1G} = (0.75)^{1/2}(200{,}000) = 173{,}200\text{ psi} \quad (4\text{-}18)$$

Curve I of Fig. 4-18 indicates predicted lives of approximately 2.0 × 10^9 and 6.5 × 10^8 cycles for the stress levels of Eqs. 4-17 and 4-18, respectively. However, Curve II indicates values of 1.5 × 10^8 and 9.0 × 10^7 cycles, respectively. Thus, the life predicted for the AH-1G (Eq. 4-18) by Curve I is 7.22 times the life predicted by Curve II. Also, with Curve I the life predicted for the UH-1H (Eq. 4-17) is 3.0 times that for the AH-1G, while from Curve II the ratio is only 1.67. Clearly, it is essential that a stress-life (S-N) curve used represent accurately the design and operating conditions if a reasonably accurate life prediction is to be achieved.

The apparent large increase in life at equal values of stress for the spiral bevel gear in comparison with the straight spur gear (Curve V vs Curve III) can be explained best by the difference in the assumptions used in the calculation of the contact stresses. The spur gear analysis is based upon a cylindrical contact assumption wherein the ratio of peak to mean compressive stress is $4/\pi$ or 1.27324. The sprial bevel gear analysis is based upon an elliptical contact assumption wherein the ratio of peak to mean compressive stress is 1.5. Although neither assumption is really valid, the ratio of the peak stress for the bevel gear to that for the spur is 1.178 for equal bearing or contact areas. This margin accounts for roughly half the stress difference between the two curves at a selected life of 1.3 × 10^8cycles. Additional gain can be attributed to the shot peening process that was applied to the spiral bevel sets (Curve V) but not to the otherwise comparable spur gears (Curve III).

4-1.3.2 Transmission Overhaul Life Rating

The various gearboxes, driveshaft assemblies, and bearing hangers that comprise the typical drive subsystem of Army helicopters in the past may have had widely differing times between overhaul (TBO). Main rotor gearbox TBO's ranged from 500 to 1200 hr, tail rotor gearbox and bearing hanger TBO's were as high as 1600 hr, and driveshaft assembly and accessory gearbox TBO's ranged from several hundred hours to unlimited intervals based upon conditional overhaul.

Specifications for next-generation US Army helicopters call for much higher (3000-4500 hr) MTBR without dictating TBO values. However, using the relationships of par. 4-1.2.1.2, application of a 2000-hr TBO requires attainment of a 6002-hr MTBF to achieve the 1500-hr MTBR (par. 4-1.2.1.2). Although this MTBF concerns only failures of sufficient importance to cause gearbox removal, it cannot be attained easily.

The ultimate design goal is conditional removal without scheduled TBO levels. Achievement of this goal requires the use of reliable and thorough diagnostic techniques (par. 4-2.4.2) and failure modes with low rates of progression so operation can continue at least short-term without compromise of safety of flight.

The question of a cost-effective overhaul time, one that balances the increased cost of overhaul due to possible extensive secondary damage and corrosion against the loss of residual useful life, is beyond the scope of this document. A cost analysis of TBO based upon direct and indirect operating cost of the drive subsystems of light, medium, and heavy twin-engine helicopter is reported in Ref. 44.

4-1.3.3 Transmission Standards and Ratings

The use of available standards in detail design is encouraged for many reasons, not the least of which is cost reduction. Available standards can contribute to lower costs if it becomes unnecessary to prepare special specifications; conduct qualification tests; procure special tooling; and otherwise compound procurement, storage, and supply activities. The standards available include military (AN, MS, NAS, AND, Federal Specifications, MIL Standards, and AMS) and commercial (AGMA, AFBMA, SAE, AISI, and ANSI). However, the limitations and ratings of standards must be thoroughly understood to prevent their misapplication.

These following are some instances in which it is better to select a nonstandard part:

1. Excess cost or nonavailability (many published military and commercial standards never have been manufactured)
2. Insufficient strength or inadequate properties (published standards for parts such as studs may not provide the required static and fatigue strength or corrosion resistance)
3. Inadequate quality control for the criticality of the application (many published standards include an inspection sampling frequency that is inadequate for critical applications).

Some recommended uses of commercial standards are discussed in the paragraphs that follow.

AGMA (American Gear Manufacturers Association) design standards and specifications for gear tooth bending, scoring risk, case hardening practices, and gear precision classifications are excellent design starting points. However, experience accumulated through development and field tests will suggest further sophistications and modifications.

Many useful standards and specifications are published by the Society of Automotive Engineers (SAE). The smaller size bearing locknuts and washers are useful, but for larger bearings the SAE parts generally are too heavy. The thread specification series also is ideal for use with special bearing or spline locknuts because the series includes sizes compatible with standard bearing bores. The 30-deg pressure angle involute spline and serration standards will suffice for most spline applications and lend themselves well to inspection with simple gages. In special instances, where greater precision is required to improve load sharing among the teeth or to improve positioning or location accuracy for the mating members, a standard spline can be modified by reducing the involute profile, lead, and spacing tolerances.

Whenever possible, bearings should be in accordance with the standard AFBMA (Anti-Friction Bearing Manufacturers Association) metric envelope dimensions, using the Aircraft Bearing Engineers Committee (ABEC) and Roller Bearing Engineers Committee (RBEC) precision grades. Departures from standard envelopes may be necessary for very light series, large bore bearings; but the common bore size, width, and outside diameter increments and tolerances should be retained to facilitate use of standard inspection equipment by the bearing manufacturer. Cylindrical roller diameters and lengths will vary among suppliers and may not follow recommended values. However, individual rollers with one of two crown radius or drop values are usually available from all aerospace grade suppliers. All ball bearing suppliers furnish balls in 1/32-in.-diameter increments and occasionally in 1/64-in. increments. Standard grade tolerances in microinches govern sphericity; e.g., grade 5 implies 5 μin. sphericity.

Many special steels, frequently called "tool-steels", using consumable electrode vacuum remelt technology, are finding increasing use in helicopter gearbox applications. The chemistries of these steels are identified only by AISI (American Iron and Steel Institute) specifications. It is frequently necessary to add special limits on trace elements and inclusions to these specifications to make them comparable to some of the commonly used AMS (Aerospace Material Specifications) grades.

4-1.4 QUALIFICATION REQUIREMENTS

Qualification requirements are described in AMCP 706-203. However, there are a number of qualification requirements that must be considered integral to the drive subsystem design process. The confidence level for passing qualification tests with a minimum of redesign and retest is increased significantly by rigorous attention to these requirements during all phases of component detail design. This paragraph treats the requirements for:

1. Component and environmental performance
2. Development testing
3. Overstress testing
4. Life substantiation testing

as they affect detail design.

4-1.4.1 Component and Environment

Many components of the typical drive system must be qualified initially through individual testing. Such lubrication system components as scavenge and pressure pumps, filters, pressure switches and transmitters, temperature switches and transmitters, chip detectors, level transmitters, jets, pressure regulators, and monitors are best defined by specification control or source control drawings. Qualification tests are classified as functional, structural, or environmental; and care must be exercised by the designer in designating applicable qualification and quality assurance requirements. Structural and endurance tests are destructive by nature and therefore are limited either to prototype qualification or to random sampling in production. Functional and environmental test areas may be specified as a quality assurance requirement, with the sampling rate up to 100%. In addition to obtaining certification of performance to specification from the supplier, it may be appropriate to perform functional tests as a part of receiving inspection. The applicable qualification and acceptance requirements must be set forth by the designer and incorporated into the subject drawing or specification. The need for completeness and accuracy in creation of the specification control drawing cannot be overemphasized.

If the designer cannot identify or anticipate the characteristic failure mode of a component, it will be necessary to establish thorough environmental test procedures for conducting the functional or endurance qualification. Once the characteristic failure mode has been established, it is often possible to increase the effectiveness of the quality assurance testing by concentrating on a particular index of performance while eliminating those test factors that reveal no useful information. For example, the critical condition for the functional test of an oil pump may be established by reducing inlet pressure to simulate limit altitude operation, allowing checks for cavitation as well as volumetric flow efficiency. Similarly, a hermetically sealed pressure switch might be subjected to a wide range of vibratory frequencies and amplitudes during functional or endurance testing in the presence of an environment involving elevated temperature and 100% relative humidity.

Attention to detail of this nature can save time and cost during subsequent testing or field evaluation.

4-1.4.2 Development Testing

In the broad sense, development testing may include both design support testing and the evaluation of prototype hardware on bench test rigs. These functions are an extension of detail design whereby early confirmation of design assumptions or errors is achieved, and necessary modifications of the initial design are identified. A thorough initial development test program may include:

1. Static tests of castings
2. Deflection tests
3. Gear contact tests
4. Assembly and disassembly tests
5. Lube system debugging
6. Incremental load and efficiency tests
7. Thermal mapping tests.

4-1.4.2.1 Static Casting Tests

The designer and/or structure analyst will predict critical sections of the castings based upon the assumed load data. A test fixture capable of applying and reacting these loads in a manner analogous to the intended helicopter use will be designed and employed. Stress coating and examination techniques should be used to pinpoint the orientations and relative locations of the maximum fiber stresses. Strain gages with suitable temperature-corrected bridges or crack wire then should be applied at the locations of these maxima, and the casting should be loaded in increments to failure. Recorded data will demonstrate compliance with stated requirements and must be correlated with analytical predictions so that accurate safety-of-flight decisions can be made based upon subsequent flight load survey data or when material discrepancy and review report action is required with respect to production hardware.

4-1.4.2.2 Deflection Tests

Deflection tests often are used to obtain data relating to hot and cold static torque and external load deflection. These data are often required in connection with gear tooth contacts and for verification of spiral bevel gear development. Useful information also may be acquired regarding planetary gear compliance and contact misalignment. Deflection test data permit accurate determination of planet load sharing and analysis of cumulative tooth spacing error; adequacy of bearing element and mounting rigidity and clamp nut torque levels also may be investigated. Compatibility of input and output shaft deflections with seal requirements may be evaluated in addition.

Multiple dial indicator gages customarily are used with incremental load applications to obtain the required deflection data. It is generally necessary to perform extensive housing modifications, usually in the form of strategically located drilled holes, to permit suitable measurements. Application of red-lead paste to gear tooth members, followed by slow rotation of the drive system under the designated torque or braking load, is used to obtain witness contact patterns.

4-1.4.2.3 Contact Tests

The red-lead paste technique described in par. 4-1.4.2.2 supplies information only on gear contacts. A more sophisticated contact test that also permits detailed bearing study and yet does not need extensive housing rework uses copper plating and gas oxidation. For this test, contacting elements (gear pinions and inner and outer bearing races) are flash-copper-plated (≈0.0001 in. maximum thickness) and the gearbox then is assembled without oil. The assembled gearbox is mounted in a manner that simulates the helicopter installation and each shaft seal location is vented to permit air escape. The input shaft is rotated slowly to permit rolling elements to assume their proper locations, and then static torque equivalent to design rating power is applied. A reducing gas, such as H_2S, is slowly bled into the gearbox, preferably through an orifice near the upper gearbox surface. All exposed copper surfaces are oxidized to a black tarnish while the Hertzian contacts remain untarnished. After the gearbox is purged completely with fresh air, the unit may be disassembled for detailed examination and evaluation. All contacts should be carefully compared with design assumptions.

4-1.4.2.4 Assembly and Disassembly

It is essential that the designer evaluate ease of assembly and disassembly of the gearbox, suitability of standard and special tools, absence of physical interferences, and opportunities for incorrect assembly. Special attention should be given to suitability of torque values specified for nuts, safetying provisions, and absence of thread galling or seizing. External line, hose, and electrical connections should be examined for proper fit and location.

4-1.4.2.5 Lubrication System Debugging

Early in the gearbox bench testing process, attention must be given to troubleshooting the lubrication systems. Proper oil jet distribution should be verified. Use of transparent (acrylic or Plexiglas) windows and covers wherever possible is helpful. Adequate oil scavenging should be verified. Design changes to incorporate internal baffles, oil scrapers, and intercompartmental venting are not uncommon.

4-1.4.2.6 Incremental Loading and Efficiency Tests

Immediately following lubrication system testing it is generally desirable to proceed with incremental power step testing, with disassembly and inspection taking place between each step. Intervals of 25%, 50%, 75%, 100%, and 125% of design power rating are recommended. Operation for 2 to 5 hr at each step is desirable to achieve definitive wear-track markings at thermally stabilized conditions. Visual inspection of the wear patterns of all gear meshes should be made after each step paying special note to the rate of tooth pattern fillout in order to verify use of proper initial tooth shapes. The use of black oxide on gears and bearing rings between each load step will assist in accurate visual inspection.

It may be convenient to schedule efficiency measurements simultaneously with load increment testing. One rather involved but satisfactory method of accurate efficiency determination requires the external application of insulating material to the entire gearbox housing and subsequent measurement of the oil temperature drop and flow rate across the oil cooler (Ref. 45). Power loss to the cooler P_{LC} is

$$P_{LC} = \frac{F G_S c_p \Delta T}{42.4}\ , \text{ hp} \qquad (4\text{-}19)$$

where

F = oil flow rate, gpm

G_S = specific weight of oil, lb/gal (a function of temperature and aeration)

c_p = specific heat of oil, Btu/lb-°F (a function of temperature and aeration)

ΔT = temperature differential between oil out of transmission and oil out of cooler, deg F

With the assumption that the insulation is effective in preventing cooling convection, P_{LC} is the only power loss from the transmission. In this case the transmission efficiency η_t is

$$\eta_t = \left(\frac{P_i - P_{LC}}{P_i}\right) \times 100\ ,\ \% \qquad (4\text{-}20)$$

where

P_i = power input to transmission, hp

Another satisfactory method for determining power loss is based upon convection cooling and requires the assumption that gearbox efficiency does not change with slight changes in viscosity within the range of lubricant temperature used. The exterior surface of the gearbox is gridded into approximately equal areas with centrally located temperature sensing points. The individual areas should not exceed 50 in.^{2} An oil cooler or heat exchanger with a controllable cooling rate is employed and air flow conditions about the gearbox are maintained as constant as possible. The test procedure requires stabilized operation at two discrete oil cooler heat extraction levels, preferably with temperature levels of the oil out of the transmission at least 50 deg F apart. During each of these runs the power loss to the cooler (Eq. 4-19) is measured and the temperatures of the designated case monitoring points are recorded, along with the ambient air temperature. The increase in oil cooler heat rejection at the lower stabilized temperature condition is assumed equal to the decrease in convection heat rejection from the housings into the ambient air, allowing the solution of the following simple set of equations (the primed symbols indicate cold condition):

$$\text{Hot:} \sum P_L = P_{LC} + C_C(\bar{T}_S - T_A) \qquad (4\text{-}21)$$

$$\text{Cold:} \sum P_L = P'_{LC} + C_C(\bar{T}'_S - T'_A) \qquad (4\text{-}22)$$

$\sum P_L$ = total power loss, hp
P_{LC} = power loss to oil cooler (Eq. 4-19), hp
C_C = case convection cooling coefficient, hp/°F
$\bar{T}_S$ = average of external surface temperature readings, °F
T_A = ambient air temperature, °F

Because $P'_{LC} > P_{LC}$, $\bar{T}_S > \bar{T}'_S$, and ΣP_L and C_C are constant by definition, we have the immediate solution:

$$C_C = \frac{(P'_{LC} - P_{LC})}{(\bar{T}_S - \bar{T}'_S - T_A + T'_A)}\ ,\ \text{hp/°F} \qquad (4\text{-}23)$$

Substituting C_C into either of the initial hot or cold loss equations (Eq. 4-21 or Eq. 4-22) will yield the total power loss P_L. If the necessary temperatures are measured for each test condition, the individually calculated values of C_C may be averaged and a probable error computed by standard statistical methods.

4-1.4.2.7 Thermal Mapping Tests

Time and instrumentation capability permitting, final design modifications of the proportioning of lubrication distribution, along with necessary adjustment of bearing parameters such as clearance and internal preload, may be accomplished by thermal mapping. Thermocouples embedded in contact with bearing inner and outer rings and with gear blank rims or tooth fillets, for example, should be used to construct a thermal map of the transmission. Measurement of rotating component temperatures requires the use of slip rings or similar devices. The use of infrared photographs of operating gearboxes also has been very effective in thermal mapping. Hot spots or excessive thermal gradients are cause for corrective design measures.

4-1.4.3 Overpower Testing

Overpower testing, sometimes referred to as weak point testing or modified stress probe testing, is intended to yield rapid results to enable the designer to make timely changes. The purpose of this testing is to produce failures and define failure modes and fail-safe features, not to demonstrate reliable extended operation. However, a 100-hr failure-free overpower test at from 100 to 125% of maximum continuous power on two samples certainly would indicate that the gearbox was ready for life substantiation or qualification testing.

The maximum recommended overpower test level is 120-130% of normal red-line power, although in some instances 110% is used. For valid test results, the following conditions described in the paragraphs that follow should be satisfied:

1. Lubrication states should remain unchanged for the main power path components (Fig. 4-3). EHD film thickness as predicted by the Dowson equation (Eq. 4-5) is relatively insensitive to load (125% power should reduce h values by about 4% from their 100% power levels for an isothermal condition); however, because the temperature of the conjunction may increase as the 3/4 power of load, which in turn will reduce the viscosity of the typical MIL-L-7808 oil by 28%, and of the h value by 22%, a cautious evaluation is demanded.

2. Excessive deflection must not occur. If developed bevel gear patterns degenerate excessively, their reduced area, coupled with the increased tooth load, could result in doubling unit stresses at the

overpower levels. The "small-cutter" and other types of spiral bevel gears tend to resist pattern shift with increasing power and are good candidates for successful overpower testing. In well designed planetary gear reductions, it is not uncommon to find a 50% increase in unit stress for a 125% overpower test at constant speed.

3. The mechanical limitation of ball bearing load path constraints must not be exceeded. There should be sufficient race shoulder height and bearing mounting rigidity to retain the ball path fully at the overpower test condition.

4. Cylindrical roller bearings should have sufficient roller crown (or race crown) to preclude severe end loading due to increased shaft misalignment or simple Hertzian deflection.

5. The increased thermal gradients present during overpower testing must not result in excessive bearing preloading or gear misalignment due to housing distortions.

Design criteria for successful overpower testing must preclude gear tooth bending fatigue failure, case crushings, or scuffing (scoring) failure modes. Accelerated wear without compromise of the design function of the gearbox for the specified test interval is the criterion of success.

4-1.4.4 Other Life and Reliability Substantiation Testing

A 200-hr qualification test is required by AMCP 706-203, and follows the tests in the preceding paragraphs. Also required are a 50-hr preflight assurance test (PFAT) and a 150-hr "must pass" qualification test in a ground test vehicle (GTV). Beyond these tests, it is frequently desirable to conduct extended bench or GTV tests to assist in the determination of initial TBO levels and to uncover failure modes not detected in previous tests. All testing in these categories is based upon spectrum loading conditions. The selected spectrum should have an *RMC* power level in excess of the anticipated flight spectrum. Because most lubrication system elements (including shaft seals) exhibit failure modes that are insensitive to power level, no meaningful accelerated test programs exist for the lubrication system, and its evaluation requires the accumulation of many test hours. Although the majority of lubrication system components will have undergone some degree of evaluation in early tests (par. 4-1.4.1), evaluation of their performance in the total system environment must await these extended time or endurance tests.

4-2 TRANSMISSIONS

4-2.1 FAILURE MODES

Many competing failure modes exist simultaneously in any mechanical transmission device. The modes recognized as dominant are often representative of the life-cycle phase in which the observation is made. Recognition, classification, and definition of safe operating limits are fundamental to successful design. Failure modes may be identified as primary and secondary for ease of analysis. In one study based on component replacement at overhaul for the UH-1 and CH-47 gearbox, secondary failures were shown to exceed primary failures by at least an order of magnitude (Ref. 46). Although the majority of design effort is directed toward preventing primary failures, the cost of drive subsystem maintenance and overhaul reflects the total of both categories. Therefore, reduction in secondary failure modes is an important objective for future design.

4-2.1.1 Primary Failure Modes

Primary failure modes are identified as those that render a component unserviceable because of some self-generated conditional occurrence other than normal wear. Cracked, broken, pitted, or spalled elements that fail while operating at normal loads, speeds, and environmental conditions are representative of this failure category.

There is a reasonable statistical level of occurrence for primary failures, perhaps on the order of 0.5%/1000 hr, that typifies the normal dispersion associated with acceptable and cost-effective design practices. Failure rates in excess of this level are considered a result of design or manufacturing deficiency. Identification and elimination of components with excessive failure rates are the objectives of the qualification assurance testing outlined in AMCP 706-203.

Properly designed and manufactured drive systems must not exhibit catastrophic primary failure modes. It is not unreasonable to expect primary modes to be exclusively noncatastrophic. This criterion may be satisfied by inherent redundancy in load paths or load sharing, or by failure progression rates that are commensurate with available built-in failure detection and diagnostic devices.

Conscientious application of classical structural analysis methods as modified by relevant test and service experience, coupled with adequate quality assurance methods, effectively will eliminate static and bending fatigue failures. However, the surface durability of loaded members such as gear teeth and rolling element bearings is by no means thoroughly understood or easily predicted. The interaction of the effects of friction, lubrication, and wear (the modern discipline of Tribology) is the subject of intensive research (Ref. 47).

Drive design is influenced by variables such as metals (hardness, microstructure, chemistry, cleanliness, residual stress), finish (roughness, lay, texture), surface treatments or coatings, lubricants (base oil, viscosity, additive package), moisture and other contaminants, speed, slip, Hertzian stress, contact geometry, friction, and temperature. These variables, separately or in combination, may vary observed life at constant stress by a factor of 500 in conventional helicopter applications. Their combined effects also exhibit slope variations from −5 to −12 of log-log S-N curves. Because it is impossible to consider the quantitative effects of all permutations of the pertinent parameters described in current literature, the significance of relevant test experience cannot be overemphasized. The classical stress-life equations or published S-N data must be viewed only as starting points. Table 4-4 presents useful qualitative influences of some of the variables affecting S-N characteristics. There are many combination effects among these variables, but virtually none that result in contradiction of the indicated trends.

The presence of relatively high slide/roll ratios and thin lubricant films is necessary for the surface pitting life to be sensitive to the additional factors shown in Table 4-4. Pitting or spalling generally is considered to be the result of metal fatigue due to cyclic contact stress. Under idealized conditions, the initiation of pitting occurs at a considerable distance below the

TABLE 4-4. LIFE MODIFICATION FACTORS — SURFACE DURABILITY

VARIABLE	INCREASED LIFE	REDUCED LIFE	QUALIFICATIONS
METALS			
HARDNESS	Rc 60 – 63	< Rc 60	CARBURIZED AMS 6260
	Rc 60 – 64	< Rc 60	AISI 52100 M-50
RETAINED AUSTENITE	< 10%	≥ 15%	CARBURIZED AMS 6260
	< 5%	> 5%	AISI 52100
WHITE LAYER	REMOVED	PRESENT	AMS 6475
CLEANLINESS	CEVM	AIR MELT	INCLUSIONS & TRACE ELEMENTS
RESIDUAL STRESS	COMPRESSIVE	TENSILE	SURFACE TO MAXIMUM SHEAR DEPTH
SURFACE FINISH			
TYPE	HONED, POLISHED	GROUND	VERY IMPORTANT AT LOW VISCOSITY
LAY	⊥ TO SLIDING	∥ TO SLIDING	VERY IMPORTANT GROUND SURFACE
SURFACE TREATMENT	BLACK OXIDE	BARE	VERY IMPORTANT THIN LUB. FILM
	LIGHT ETCH	AS MACHINED	RAPID SURFACE BREAK IN
LUBRICANT	HIGH VISCOSITY	LOW VISCOSITY	$V_1 + V_2 < 2000$ ft/min
	MINERAL BASE	SYNTHETIC BASE	TRUE AT NORMAL STRESSES
	ADDITIVE & SYNTH	ADDITIVE & MINERAL	VERY TRUE AT LOW SPEED
	HIGH COEFFICIENT	LOW COEFFICIENT	PRESSURE VISCOSITY COEFFICIENT, α
	LOW ACIDITY	HIGH ACIDITY	DEGRADATION - TIME AND USE
WATER CONTENT	LOW	HIGH	WATCH DEGRADED SYNTH.
SPEED	HIGH	LOW	EXCEPT ROUGH SURFACES
SLIP	LOW	HIGH	
	POSITIVE	NEGATIVE	LOWER REL. SPEED - NEG.
FRICTION	LOW	HIGH	
TEMPERATURE	LOW	HIGH	SURFACE CONJUNCTION
GEOMETRY OF e	HIGH	LOW	$e = b/a$ FOR ELLIPSE b = AXIS ∥ TO ROLLING V a = AXIS ⊥ TO ROLLING V

surface at the level of maximum orthogonal shear stress, and classical theory has been developed about these conditions. However, recent studies establish that the shear stresses tend to be located nearer to the surface, even in the presence of very small magnitudes of slip (Ref.48). The traction stresses imposed by sliding can raise the surface shear stresses to within 40% of the maximum Hertzian stress (Ref. 49). These conditions lead to surface initiation of pitting or cracking that ultimately results in the gross pitting or spalling failures observed in most failed mechanical components. For example, it has been established that the vast majority of pitting (spalling) failures in UH-1 and CH-47 helicopter gearboxes are surface-initiated (Ref. 46).

4-2.1.2 Secondary Failure Modes

Secondary failure modes are all those modes that are not classified as primary. By definition secondary failure modes do not contribute directly to component MTBR; however, they contribute greatly to the cost of overhaul, and in some instances they limit severely the safe operating time after occurrence of primary failure.

Secondary failure modes are grouped into three categories, each with a different design avoidance technique.

4-2.1.2.1 Overload Failures

Components that are overloaded due to the failure of a parallel or series connected load carrying member frequently result in secondary failure in a short time. Tandem thrust bearings or multiple planet or epicyclic gear trains are typical parallel load-path configurations. Such components limit the progression rate of a primary failure by an automatic load reduction resulting from increased deflection or wear material removal of the failing primary component. In such designs, the secondary load-carrying members should be analyzed under full power to insure adequate life for safe continued operation. Such analyses should show a minimum life of 100 hr.

Series-connected secondary failures are typified by transfer of damage from one gear member to another in a train arrangement or by the upstream overload of a component due to an advanced downstream failure such as a "jammed" rolling element bearing with advanced retainer or ball fractures. The static yield strength of the primary power path components (gears, shafts, bearings, couplings, etc.) must be sufficient to withstand the maximum red-line power plus the incremental transient load required to fracture and break clear the relatively frangible primary failed component.

4-2.1.2.2 Debris-caused Failure

Debris from a spalled tooth or bearing often enters another gear mesh or bearing and results in sufficient denting, embossing, or brinelling to initiate a secondary failure after relatively few cyclic stressings. While the damage incurred in gears and cylindrical roller bearings is less severe than in ball bearings (due to the preferential debris entrapment of conformal contact bodies), the rate of replacement of secondarily damaged parts at overhaul has considerable cost impact (Ref. 46). Much potential damage can be avoided by compartmentalized designs or by use of shields or baffles to protect dynamic components by deflecting and re-routing debris to catch-trap or sump areas. The objective is localization of the damage to the primary failure component.

4-2.1.2.3 Environmentally Induced Failures

Oxidation, stress corrosion, galvanic corrosion, and aging or embrittlement fractures are examples of failures that could become significant to future MTBR data for helicopters with increased TBO intervals. These failures result from inadequate attention during design or production quality control to materials, protective plating, or finish coatings. Concentric anodes always should be used during plating of tubular shaft members to secure adequate protection of internal surfaces. Special attention must be given to providing drains to eliminate trapped water at gearbox stud, boss, and mast seal locations. Additional protective practices are recommended in par. 4-2.3.1.

4-2.2 DYNAMIC COMPONENTS

The dynamic components common to all drive subsystems are:

1. Toothed power transmission wheels (gears) that operate over a wide range of rolling or total velocities with moderate to relatively high sliding or slip velocities, under Hertzian stresses rarely exceeding 250,000 psi
2. Rolling element support members (bearings) that operate at similar total velocities, lower slip velocities, but generally higher Hertzian stresses
3. Interconnecting members (shafts and couplings) that are splined, bolted, or welded together and to gears, with external forces and moments imposed while rotating
4. Other miscellaneous elements such as shaft seals, nuts, and locking devices.

4-2.2.1. Gears

Helicopter gear design will be viewed from three aspects: limitations, analysis, and the drawing or specification. The primary emphasis is upon power transmission gearing rather than torque gearing (high load/low speed, as in actuator or hoist applications) or accessory gearing. The latter is discussed briefly in par. 4-5.

4-2.2.1.1 Gear Limitations

Successful helicopter gear designs usually have employed counter-formal involute spur, helical, and spiral bevel configurations.

Some applications, notably the Westland WG-13, have used conformal circular arc helical tooth forms. Although somewhat superior in performance with respect to surface durability, conformal gears have numerous configurational limitations, e.g., operating center distance is very critical, and frequently exhibit reduced tooth bending fatigue strength. Analytical techniques used for involute tooth strength analysis are not well-suited for circular arc teeth, although an excellent analytical finite element approach to bending stress calculation may be found in Ref. 50. The need for three-dimensional analysis is described in Ref. 51. This gear form is particularly sensitive to center distance variation unless considerable mismatch of tooth curvature is used between pinion and gear. The immediate result of this practice is a considerable increase in the maximum Hertzian stress with an attendant reduction in the theoretical surface durability of the pinion. The deflection inherent in the elastic reaction of the loaded tooth introduces a small degree of slip that further reduces the theoretical pitting endurance (par. 4-2.1.1). Conformal gears are of considerable interest from a research and development viewpoint, but at present, design knowledge and experience are not sufficient to permit a meaningful design discussion of this configuration.

The relative efficiencies of the various spur, helical, and spiral bevel tooth forms were discussed in par. 4-1.2.1.1. The high sliding and resultant power loss and limited load-carrying capacity of crossed helical and hypoid gears operating in synthetic lubricants eliminate their usefulness except in accessory drive applications.

4-2.2.1.2 Gear Analysis

Successful detail involute gear design analysis requires an understanding of the relative risks of the three failure modes (par. 4-2.1.1) under the particular operating conditions for each specific gear application. Initial insight may be obtained by reviewing the regions of dominant distress as shown in Fig. 4-20, based upon Ref. 42. This graphic relationship reflects the characteristics of case-hardened precision helicopter gears operating in synthetic turbine lubricants. The relative positions of each zonal demarcation will vary as a function of the diametral pitch P_d, pressure angle ϕ, contact ratio, root fillet form, surface finish, and material-processing characteristics of each individual design. Fig. 4-20 represents reasonably accurate estimates for a standard proportion, $P_d = 8.5$, 35 × 61 tooth set of full fillet form, ground flank, carburized AMS 6260 involute spur gears.

The variation in the three failure mode relationships when all factors are constant except for diametral pitch may be seen in Fig. 4-21 (Ref. 53).

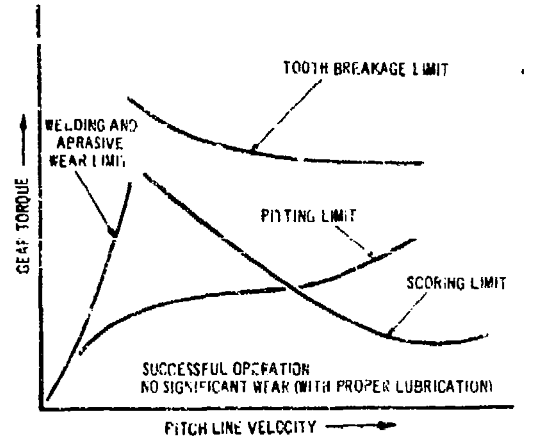

Figure 4-20. Graphic Relationship — Failure Modes — Load vs Velocity

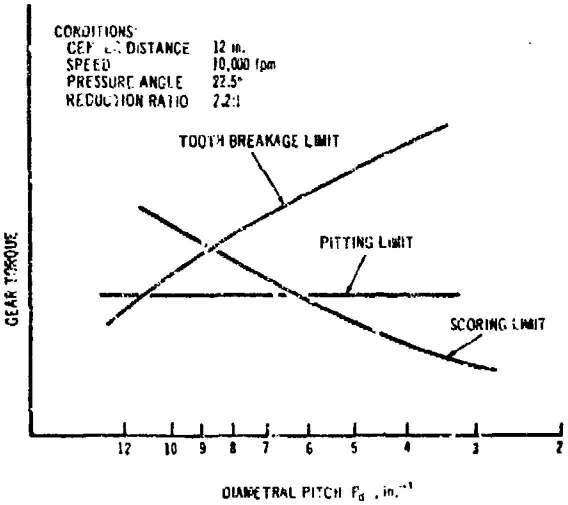

Figure 4-21. Graphic Relationship — Failure Modes — Load vs Tooth Size

4-2.2.1.2.1 Bending Fatigue Strength

The determination of bending fatigue risk involves three steps: load determination, stress evaluation technique, and definition of the properties of materials used.

The basic bending stress equations, generally in accord with AGMA practice, may be found in Refs. 54 (spur gears), 55 (helical gears), and 56 (bevel gears), together with computer solutions written in FORTRAN symbology. Because of the increased precision with which the basic Lewis equation geometry factors are treated, the use of these references is highly recommended. However, for a better understanding of the relative significance of the factors affecting tooth bending fatigue, the general AGMA equation will be rearranged and discussed term by term.

The basic equation for tensile stress is S_t due to tooth bending has been defined (Ref. 57) as

$$S_t = \left(\frac{W_t K_o}{K_v}\right)\left(\frac{P_d}{F}\right)\left(\frac{K_s K_m}{J}\right), \text{ psi} \qquad (4\text{-}24)$$

where

F = face width of gear tooth, in.
J = geometric shape factor, dimensionless
K_m = misalignment factor, dimensionless
K_o = overload factor, dimensionless
K_s = size factor, dimensionless
K_v = dynamic load factor, dimensionless
P_d = diametral pitch, in.$^{-1}$
W_t = gear tooth load, lb

Both test and analysis have confirmed that the dynamic load W_d is more correctly expressed as an independent variable rather than as being accounted for by applying the factor K_v to the gear tooth load W_t. Therefore, Eq. 4-24 should be replaced by

$$S_t = (K_o W_t + W_d) K_m \left(\frac{P_d K_s}{FJ}\right), \text{ psi} \qquad (4\text{-}25)$$

where

W_d = dynamic load, lb

Ref. 58 shows that the dynamic load W_d exists as an incremental load due to tooth errors and gear drive dynamics at operating speed, and, therefore, is relatively unaffected by transmitted power. An acceptable engineering approach to the evaluation of W_d also is given by Ref. 58. This evaluation is based upon the tooth spring rate calculation taken from Ref. 59, which in turn presents an analysis of the experimental data of Ref. 60, taken during tests of solid blank type gears. Inasmuch as helicopter gears have lighter back-up rim and web configurations than those used in the referenced tests, the dynamic load analyses based upon these methods will be conservative. Greater precision can be obtained by considering factors other than tooth deflections by the methods advanced in Refs. 61 and 62.

Although dynamic load factors and increments have been studied by numerous investigators for the past century, the results have led to only slight agreement. Therefore, well correlated test experience is of great importance. The interaction of profile modifications, tooth errors, deflection modes, spring constants, and inertias are of such complexity that generalized solutions are unlikely to be satisfactory. However, certain qualitative observations that find general acceptance are:

1. The dynamic load W_d increases essentially linearly with speed and spacing error. Hence, the importance of increasing precision with speed is underscored.

2. There is a limiting value for W_d that probably occurs when the duration of the velocity pulse is less than one quarter the period of natural vibration of the pinion-gear spring-mass system.

3. The value of W_d is proportional to both spring rate and gear effective mass. Hence, it is important to consider total deflection and mass of tooth and rim for helicopter gear designs. Expressions for approximate values of the deflection at point of mesh impact for solid rim and hub configurations are given by Refs. 61 and 62. For thin-rim helicopter gears (back-up rim thickness of approximately one tooth whole depth) with approximately 40 or more teeth, the deflection can be as much as twice the value for a solid rim. The limiting deflection of a spur tooth at the moment of mesh impact is approximately 10% that of a helical tooth, reflecting a spring rate an order of magnitude higher. As a result, the limiting dynamic load on the spur gear will be about three times greater than that on the helical gear. This analogy also holds for comparing straight and spiral bevel gears.

4. In most cases the torque capacity of a hardened precision helicopter gear set will be limited by surface durability rather than by tooth breakage resulting from dynamic loads. Many failures formerly attributed to dynamic loads have been recognized recently as resulting from resonant or vibratory conditions that may occur at high speeds (par. 4-2.4.1).

Determination of the gear tooth load W_t is relatively straightforward. For a first approximation it is

$$W_t = \frac{2Q}{D_p N}\text{, lb} \qquad (4\text{-}26)$$

where

Q = torque, lb-in.
D_p = pitch diameter, in.
N = number of driven gears in mesh with the driver, dimensionless

For precise calculations the pitch radius $D_p/2$ should be replaced by the radius to the highest point of single tooth loading (HPSTL) as determined by the profile contact ratio for spur gears or by other considerations for helical and bevel gears. These considerations are discussed further in a subsequent paragraph in conjunction with discussion of the geometric shape factor J.

The overload factor K_o is included in Eqs. 4-24 and 4-25 to account for the torque pulsation waveform or for roughness of the transmitted power, and hence is a difficult parameter to assess. This roughness can result from the power source, the driven member, or the response of the elastic drive subsystem itself. Because the bending stress S_t must be within the fatigue endurance limit of the material, K_o should not be used to account for occasional overloads of low cumulative cyclic duration unless the design life itself is relatively low; i.e., less than 10^6 cycles. Some examples of measured K_o values are

K_o = 2.0 — first gear drive from six-cylinder, reciprocating, four-stroke cycle, horizontally opposed aircraft engine
K_o = 1.25 — gear set adjacent to high angle Hooke's joint installation
K_o = 1.15 — gear set adjacent to typical tail rotor drive-shaft
K_o = 1.2 — third-stage gearing in six-cylinder reciprocating engine application
K_o = 1.0 — turbine engine speed reduction gear drive.

The misalignment factor K_m in the tooth bending stress equations takes into account the lengthwise or axial load distribution on the face of the loaded gear mesh. Three primary sources, which generally are additive, contribute to misalignment:

1. Initial misalignment due to manufacturing inaccuracy or deflected axes of rotation due to gearset load, external load, or thermal gradient
2. Tooth lead slope deviations due to inaccuracy in gear manufacture
3. Elastic deflections of shafts, gear webs and rims, and support bearings.

The generally accepted relationships for the misaligned factor K_m for spur and helical gears are

$$K_m = \frac{2F}{F_m}\text{, d'less, for } F_m \le F \qquad (4\text{-}27)$$

$$K_m = \frac{F_m}{F_m - (F/2)}\text{, d'less, for } F_m > F \qquad (4\text{-}28)$$

where

F = face width of gear tooth, in.
F_m = average value of effective face width F_e for given loading condition, in.

The average face width F_m in these equations is that width which can be considered to remain in contact under an effective tooth load W_t where, from Eq. 4-25

$$W_t' = K_o W_t + W_d\text{, lb} \qquad (4\text{-}29)$$

Empirical expressions for F_m are

$$F_m = \left[\frac{2W_t'}{eG}\right]^{1/2}\text{, in.} \qquad (4\text{-}30)$$

for spur gears, and for helical gears

$$F_m = 2\left[\frac{W_t' P_b}{eGkZ}\right]^{1/2}\text{, in.} \qquad (4\text{-}31)$$

where

e = pitch plane misalignment (net), in./in.
G = lengthwise tooth stiffness constant, psi
k = contact line inclination factor (Ref. 63), dimensionless
P_b = base pitch, in.
Z = total transverse length of line of action, in.

The value of the tooth stiffness constant G in these equations usually is $10^6 \le G \le 2.5 \times 10^6$.

The correct value for K_m for "point-contacts" such as those in spiral bevel gears may be considerably less than the values for spur or helical gears due to the conformal axial curvature and curvature mismatch used to localize the contact pattern within the tooth boundaries. It is common practice to use a value of K_m = 1.1 for aircraft spiral bevel gears. This low value is particularly justified for straddle mountings used in conjunction with so-called "small-cutter tooth developments". Use of cutter (and grind-wheel) diameters equal to the mean cone distance at low helix angles and equal to less than twice the cone distance at high helix angles is an example that would meet this criterion. In general, the contact pattern shifts toward the heel of the tooth as the load increases when too large a cutter is used, while a shift toward the toe results from use of an excessively small cutter diameter. The correct diameter will result in approximately equal pattern spreading toward both toe and heel.

The stress evaluation portion of Eq. 4-25 consists of the expression $(P_d/F)(K_s/J)$.

The diametral pitch divided by face width (P_d/F) defines the physical size and hence the basic strength of the gear tooth. K_s is a size factor to account for the phenomenon that larger components may not exhibit fatigue endurance stress levels equal to those for smaller components. It is grouped with other terms in the equation for gear tooth stress S_t so that ready comparison may be made with the basic material allowable stress.

For spiral bevel gear applications Ref. 56 recommends for $P_d \leq 16$ the use of

$$K_s = 2P_d^{-0.25}, \text{ dimensionless} \tag{4-32}$$

and for $P_d > 16$

$$K_s = 1.0, \text{ dimensionless} \tag{4-33}$$

However, for spur and helical gears for which $6 \leq P_d \leq 12$, Ref. 54 suggests $K_s = 1$ because test data have produced only a slight strength difference attributable to size.

It should be emphasized that the values for K_s given by Eqs. 4-32 and 4-33 are based upon the assumptions that the ratio of case depth to tooth thickness remains essentially constant for case-hardened gear teeth, and that the characteristics of case and core material and residual stress fields are unaffected by size. Although the latter condition can be achieved in the practical sense over a fairly wide range for case-carburized materials, it cannot be satisfied for nitrided materials. For materials such as AISI 4340, AMS 6470, and AMS 6475 that are to be case-hardened by nitriding, a correction becomes necessary as the diametral pitch increases. While there are few published data in this area, reasonable corrections for nitrided AISI 4340 or AMS 6470 for various values of pitch diameter D_p are

$$\begin{aligned} K_s &= 1 && \text{for } D_p \geq 12 \\ K_s &= 12/D_p && \text{for } 12 > D_p > 3 \\ K_s &= 4 && \text{for } D_p \leq 3 \end{aligned} \tag{4-34}$$

while for nitrided AMS 6475 it is appropriate to use

$$\begin{aligned} K_s &= 1 && \text{for } D_p \geq 9 \\ K_s &= 0.5 + 4.5/D_p && \text{for } 9 > D_p > 3 \\ K_s &= 2 && \text{for } D_p \leq 3 \end{aligned} \tag{4-35}$$

The geometric shape factor J is used to account for the shape of the cantilevered gear-tooth beam; and it includes the influences of stress concentration, load sharing, and the modified Lewis form factor. The recommended equations for J are:

for spur gear;

$$J = \frac{Y}{K_f m_n}, \text{ dimensionless} \tag{4-36}$$

where

Y = form factor, dimensionless

K_f = stress concentration factor (Ref. 63), dimensionless

m_n = contact ratio factor, dimensionless

for helical gear;

$$J = \frac{Y_c C_h \cos^2\psi}{K_f m_n}, \text{ d'less} \tag{4-37}$$

where

Y_c = form factor, dimensionless

C_h = load inclination factor, dimensionless

ψ = helix angle, deg

and for bevel gear;

$$J = \frac{Y_k R_t F_e P_d}{m_n K_i R P_m F}, \text{ d'less} \tag{4-38}$$

where

Y_k = form factor, dimensionless

R_t = distance from pitch circle to point of load application, in.

F_e = effective face width, in.

K_i = inertia factor, dimensionless

R = mean transverse pitch radius, in.

P_d = transverse diametral pitch (measured at large end of bevel gear), in.$^{-1}$

P_m = mean transverse diametral pitch, in.$^{-1}$

Eq. 4-38 is taken from a Gleason standard (Ref. 64) and is presented primarily for discussion of the pertinent parameters. As mentioned previously, a more thorough and accurate evaluation can be obtained with the computer program of Ref. 56.

The modified Lewis form factor appears as Y, Y_c, and Y_k in Eqs. 4-36, 4-37, and 4-38, respectively. This factor is based upon bending stress calculation for a parabolic cantilever beam inscribed within the involute tooth form with the point of tangency between the parabola and the involute and the point of load application being the significant factors governing the stress. This form factor reduces the extreme fiber tensile stress with a compression component of the tooth normal load. The load application point for the spur gear factor Y always should be taken at the calculated HPSTL and at the tip of the involute gear profile for the helical gear factor Y_c. The determination of the load point for the bevel gear factor Y_k is based upon certain assumptions concerning the load contact pattern geometry of the bevel tooth. The great difference between the assumptions of Ref. 64 and the updated version of Ref. 56 accounts for a considerable change in the calculated stress. Use of the modified Lewis form factors in conjunction with a stress concentration factor K_f has proven to be as accurate as any method known for involute gear tooth stress calculation within the range of pressure angles 14.5 deg < ϕ < 25 deg. However, a significant degree of inaccuracy may occur outside this range as well as for internal gear tooth forms and for maximum fillet radius configurations.

The influence of the stress configuration due to the relative fillet radius and load point location is accounted for in K_f, which is derived from the photostress work of Dolan and Broghamer. It should be noted that an effective K_f is included in the values for Y_k in Eq. 4-38.

The effective load apportionment due to load-sharing among the meshing teeth is accounted for in the contact ratio factor m_N. For spur gears for which the profile contact ratio m < 2.0, $m_N = 1$ because all calculations are based upon the HPSTL. For 2.0 ≤ m ≤ 3.0, $m_N = 2.0$ when the highest point of double tooth contact is used for determination of the geometric shape factor J. For helical gears the axial or face contact ratio is additionally accounted for by using

$$m_N = p_N (0.95 Z), \text{ dimensionless} \qquad (4\text{-}39)$$

where

p_N = normal circular pitch (helical gear), in.
Z = transverse length of line of action, in.

For spiral bevel gears, the modified contact ratio m_o, which is a root-mean-square (rms) summation of the effective profile and face contact ratio, is used. When $m_o \leq 2.0$, $m_N = 1$, but for $m_o > 2.0$

$$m_N = \frac{m_o^3}{m_o^3 + 2\sqrt{(m_o^2 - 4)^3}}, \text{ d'less} \qquad (4\text{-}40)$$

The C_h factor used in Eq. 4-37 accounts for the inclination of the load contact line and is derived from cantilever plate bending theory as presented in Ref. 65.

$$C_h = \frac{1}{1 - \sqrt{\frac{\nu}{100}\left(1 - \frac{\nu}{100}\right)}}, \text{ d'less} \qquad (4\text{-}41)$$

where

ν = helical tooth load line inclination angle = $\text{Tan}^{-1}(\sin\phi_n \tan\psi)$, deg
ϕ_n = normal pressure angle (helical gear), deg
ψ = gear tooth spiral angle, deg.

The inertia factor K_i in Eq. 4-38 accounts for a reduced contact ratio. For $m_o > 2.0$, $K_i = 1.0$ and for $m_o < 2.0$, $K_i = 2.0/m_o$.

Eqs. 4-36 through 4-38 are of assistance in evaluating the stress at the location assumed to be the weak point of gear teeth. Test results often indicate, however, that failures originate not in the fillet near the involute flank, but rather deeper toward the root or higher on the tooth flank.

In the former case, the crack propagation is frequently downward through the rim rather than in an arc across the tooth. This type of failure is not an acceptable failure mode because a large section of the gear rather than a single tooth tends to break off.

Breakage high on the flank may result from the use of too thin a tooth on a rigid base with less-than-optimum blend between the fillet and the flank. A low break frequently arises when the back-up rim is too thin. In this case the rim bending stress (due to tooth load moment about the rim neutral axis) can be greater than the assumed cantilever tooth bending stress. Because rim curvature and web resistance enter into this analysis, it is incorrect to consider only the rim thickness. However, for spur and low-helix-angle helical gears of about 40 teeth, a rim thickness equal to the tooth depth is generally accepted as adequate. Slightly lesser values may be used for gears with fewer teeth while greater values may be needed for gears with more than 40 teeth. The existence of high thrust loads in helical or bevel gears will complicate the analysis. Considerable web and rim reinforcement is necessary to develop full tooth strength potential in high-thrust-component spiral bevel pinions.

Because the preceding analyses are two-dimensional, they are not suitable for definition of maxima for a triaxial stress field. Computerized methods of finite element analysis are expected eventually to afford accurate solutions for these conditions

Additional precision of high-speed gearing analyses may be obtained by inclusion of the hoop stress S_h in the gear rim due to the centrifugal acceleration. A conservative value for this steady stress (at a constant speed) may be taken as

$$S_h = 7.095 \times 10^{-6} \rho (nD_r)^2 \text{ , psi} \qquad (4\text{-}42)$$

where

7.095×10^{-6} = dimensional constant
ρ = material density, lb/in.^{3}
n = gear speed, rpm
D_r = gear root diameter, in.

The oscillatory stress due to bending S_t may be combined with the hoop stress S_h by use of a modified Goodman diagram constructed on the basis of the material properties of the specific gear. The modified Goodman diagram can be used to account for the reduced allowable bending limit for an idler gear application in which the calculated stress is fully reversing. Use of the diagram is discussed in detail in Ref. 54.

A maximum safe working value of tooth stress $S_{t_{max}}$ due to bending can be determined as

$$S_{t_{max}} = \frac{S_{at}K_l}{K_tK_r} \text{ , psi} \qquad (4\text{-}43)$$

where

S_{at} = allowable endurance limit stress, psi
K_l = life factor, dimensionless
K_t = temperature factor, dimensionless
K_r = reliability factor, dimensionless

The life factor K_l is assumed as unity for all applications designed for infinite life, i.e., greater than 10^7 cycles. All Army helicopter power gearing designs must meet this criterion. The temperature factor K_t is taken as unity provided the gear blank operating temperature is below the hardness draw temperature for the material in use; this criterion must be satisfied by all Army helicopter power gearing designs. The reliability factor K_r effectively is a factor of safety that is used when the statistical confidence and reliability (test data scatter) are unknown for a given mean value of the endurance limit stress. In such cases, a value of 3.0 is recommended for K_r. When the allowable endurance limit S_{at} is known for the specified reliability level, $K_r = 1.0$

S_{at} always should be chosen to reflect the desired reliability for the design application. A generally recognized safe design practice for helicopter gearing is to select S_{at} as the value 3 standard deviations (3σ) below the mean endurance limit demonstrated by test. The value of the standard deviation σ, as well as the mean endurance limit, varies greatly with material, heat treatment practices, manufacturing variability, and the quality control level exercised in final component acceptance inspection and nondestructive test and evaluation methods. Endurance limit data and an applicable value for σ, may be obtained from an R. R. Moore rotating beam specimen test. Extreme care must be taken to duplicate every metallurgical and manufacturing characteristic of a gear itself or the data are useless. However, the value of σ from these tests invariably will be smaller than that for the more complex production gear.

Refs. 66, 67, and 68 present the results of independent test programs conducted to determine an accurate mean endurance limit for carburized AMS 6265 gears. The resultant S_a values vary from 160,000 to 210,000 psi. Relative strength data for many materials and processes as treated in single tooth (pulsor) machines over a 15-yr period may be found in Ref. 69. The variability of σ and S_{at} with material and process is shown in Fig. 4-22, extracted from Ref. 41. None of the test gears were shot peened since this process could have masked the inherent differences in the materials and processes.

The endurance limit is, of course, merely one of several factors that must be assessed by the designer in making his gear material selection. Aside from the obvious criteria of familiarity, confidence level in manufacturing and process control, cost, temperature environment, and size, the crack propagation characteristics must be satisfactory. Although the safe design stress for one candidate material may be far higher than that of another, it also must be assumed that ballistic damage or secondary damage will occur in any critical gear mesh of an Army helicopter. Such damage could result in an impact-type overload that could cause a through-hardened, high-tensile-strength material to shatter and fail instantly. The degree of ductility provided by a core structure of lower hardness (commonly core hardness 20 points Rockwell C lower than the case) is often sufficient to provide a safe failure mode with a relatively low rate of crack propagation. When this technique cannot be employed, the use of lower hardness gradients together with geometric crack stoppers may suffice. Ref. 70 is a useful primer on the fracture mechanics for the gear designer.

The use of properly controlled shot peening in the gear fillet area often can reduce scatter (smaller σ) and in some instances can increase the design allowable S_{at} by 15 to 25% in carburized AMS 6265 gears. Practices vary widely with regard to peening of gear tooth faces. Some specifications require masking the faces during peening, others require removal of the effects of peening by flank grinding or honing, and still others allow peening of the running surfaces. Peening requirements are tailored to the gear application, but certain generalities may be stated:

1. Peen in accordance with MIL-S-13165.

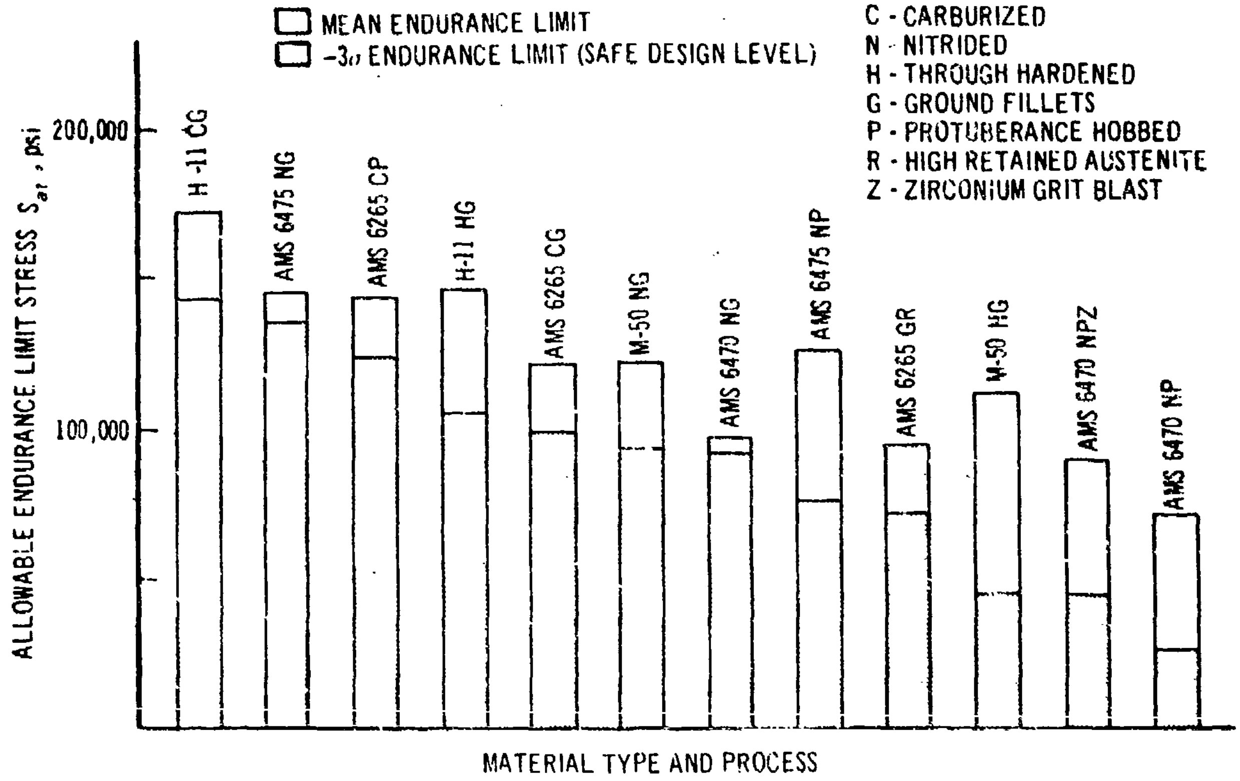

Figure 4-22. Single Tooth Pulsor Gear Fatigue Test Results

2. Use cast iron, steel, or cut steel wire shot of diameters no larger than one-half the smallest fillet radius.

3. Use several hundred percent coverage.

4. Determine intensity with Almen test strips positioned to simulate the exact surface location to be peened.

5. Never exceed 0.016A intensities.

6. Do not peen very hard (> R_c 64) or brittle surfaces such as nitrided AMS 6470, AMS 6475, or similar materials. However, cleaning and abrasive grit or glass shot at intensities up to 0.010N is permissible.

4-2.2.1.2.2 Scoring Failure

When two gear teeth slide together under load, there is considerable heat generation in the localized conjunction even in the presence of a lubricant. When the rate of heat buildup exceeds the rate of heat transfer away from the conjunction, the resultant temperature rise acts to reduce the lubricant viscosity, thus reducing the thickness of the separating oil film. In the full EHD lubrication zone, the temperature rise may serve to reduce the friction; however, as the film thins and conditions change to the transitional or boundary lubrication regime, the friction generally increases. Consequently, the unstable condition may be created that eventually will result in harsh metal-to-metal contact and a surface energy density sufficient to "melt" and smear a thin layer of the gear tooth surface. This smearing condition is referred to as scoring or scuffing. Although the physics of the phenomenon remains the source of much debate and intensive research, certain factors are understood sufficiently well to permit engineering design that will minimize scoring risk for a given set of operating conditions. There are five fundamental approaches the designer must consider:

1. Selection of the proper diametral pitch, taking care to balance bending strength against scoring (Fig. 4-21). Higher speeds call for the use of finer pitches. It always should be possible to select a pitch range in which putting endurance is the life-limiting failure mode.

2. Use of a contact ratio sufficient to insure load sharing by at least two pairs of mating teeth in the higher sliding velocity ranges of the tooth contact. The profile contact ratio for straight spur gears

operating in the velocity ranges of scoring sensitivity never should be less than 1.65, a value that permits two-tooth load sharing in the first and last thirds of meshing contact. The profile contact ratio may be reduced for helical and spiral bevel gears when the face contact ratio is sufficient to assure a total developed contact ratio of 2.0 or greater. Reliable achievement of these contact ratios requires accuracy of tooth spacing, profile slope, and lead slope.

3. Selection of tooth numbers to insure hunting tooth action. In general, this requires that the number of teeth in any two meshing gears be relatively prime; i.e., that there be no common factors. An indication of the significance of this requirement is given by Fig. 4-23, which illustrates the difference in scoring load limit between gear-synchronized and separate motor driven 4.0 in. diameter test discs operating in MIL-L-7808 oil (Ref. 71).

4. Modification of the involute profile to compensate for deflection of the teeth under load-deflected load so that the loading on the entering and leaving tooth pair contacts varies smoothly rather than in a step-function. This not only minimizes the transmitted load carried at the sliding velocity extremes, but reduces the dynamic load increment as well.

5. Provision of adequate, uniformly distributed oil flow to the entire tooth face width. The dual function of cooling and lubrication is best served by use of both in-mesh and out-of-mesh oil jets. When both jets cannot be employed (either to minimize windage losses or because of marginal pump supply), it usually is best to retain the in-mesh jet for high-speed gearing and the out-of-mesh jet for low-speed gearing.

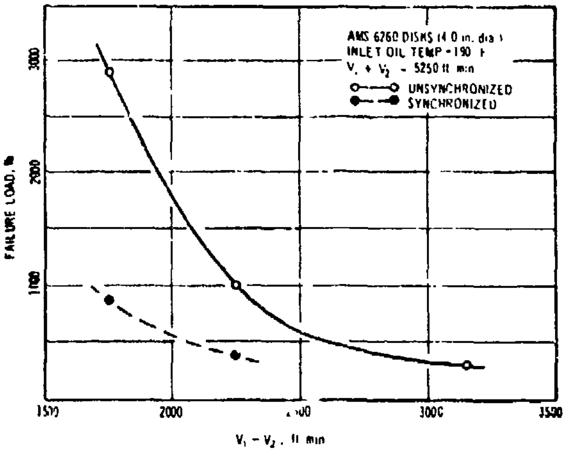

Figure 4-23. Scuffing Load vs Sliding Velocity — Synchronized and Unsynchronized Discs

The most satisfactory technique currently available for calculating gear scoring risk is based upon the critical temperature hypothesis (Ref. 72). This hypothesis suggests that for every oil-metal combination, there exists a critical constant conjunction temperature at or above which surface scoring occurs. Application of the concept requires determination of appropriate values of the critical temperature T_c and the temperature of the conjunction $T_i + \Delta T$ with T_i the initial temperature of the oil-mesh interface as it enters mesh and ΔT being the temperature rise during the meshing cycle. The critical temperature hypothesis implies that the limiting or failure load W_f is related to certain design variables. The specific relationship has been reduced to standard gear terminology and published by AGMA (Ref. 73). Average values for thermal conductivity, specific heat, density, and an assumed constant coefficient of friction of $f \approx 0.06$ have been incorporated into the empirical questions for the temperature rise ΔT and the scoring geometry factor Z_t

$$\Delta T = \left(\frac{W_t'}{F_e}\right)^{3/4} \left(\frac{50}{50 - S}\right) \left(\frac{Z_t n_p}{P_d^{1/4}}\right)^{1/2}, \quad {}^\circ\mathrm{F} \tag{4-44}$$

where

W_t' = total effective gear tooth load, lb
F_e = effective face width, in.
S = rms surface finish, μin.
n_p = pinion speed, rpm
Z_t = scoring geometry factor, dimensionless

$$Z_t = 0.0175 \frac{\left(\sqrt{r_p} - \sqrt{\left(\frac{N_p}{N_g}\right) r_g}\right) \left(P_d\right)^{1/4}}{\left(\cos\phi_t\right)^{1/4} \left(\frac{r_p r_g}{r_p + r_g}\right)^{1/4}} \tag{4-45}$$

where

N_p = number of teeth in pinion
N_g = number of teeth in gear
P_d = diametral pitch, in.$^{-1}$
r_g = radius of curvature of gear tooth, in.
r_p = radius of curvature of pinion tooth, in.
ϕ_t = transverse operating pressure angle, deg

In using Eqs. 4-44 and 4-45, T_i is assumed as equal to the oil inlet temperature. However, considerable error can be introduced through the actual values of f and T_i because the friction coefficient is often lower than 0.06 and the pinion-gear surface temperatures often are higher than those at the oil inlet. However, because these two errors are opposite in effect, sufficient cancellation occurs to render the equations acceptable for estimation purposes.

Ref. 73 defines the critical temperature T_c with respect to scoring risk, rating T_c = 500°F as a high risk, 300°F as a medium risk, and lesser temperatures as low risks. Therefore, the value of the conjunction temperature $T_i + \Delta T$ under design load conditions should be less than the value of T_c associated with an acceptable level of risk.

There is no accepted or inherently accurate method for calculating T_i, although measurements of typical helicopter pinions have shown values 100°F greater than oil inlet temperatures. Although it is well known (Refs. 74 and 75) that above a certain critical speed the scoring load of a given gear set will increase, the AGMA equation does not reflect this consideration since no speed term other than sliding velocity was used in the development of Eqs. 4-44 and 4-45.

An improved calculation method uses speed-dependent friction coefficients (Ref. 76) combined with the effects of tooth load sharing. The method for digital computer use follows:

1. Subdivide the active tooth profile into at least 20 equally spaced points.
2. Calculate r_p and r_g at each point.
3. Calculate Z_t at each point.
4. Calculate f at each point by method shown in Ref. 76 or using suitable empirical data.
5. Replace Z_t in Eq. 4-44 with Z_t', a modified scoring geometry factor,

$$Z_t' = \left(\frac{f}{0.06}\right)^2 Z_t \text{ , d'less} \quad (4\text{-}46)$$

6. Calculate a value of W_t' adjusted to account for load sharing at multiple tooth contact points, including effects of profile modifications.
7. Calculate ΔT and conjunction temperature $T_i + \Delta T$ at each point.

Proper load sharing distribution in the two-tooth contact zone must be provided for by involute profile modifications. There are many techniques in use for calculation of these modifications, most of them based upon the practices recommended by Ref. 77. Even though the calculated values are slightly low for thin-rim helicopter gearing, the tooth deflections obtained by the methods of Ref. 78 should be used for the profile modification technique. The first point of contact (pinion dedendum with gear tip) is the most critical with respect to the overload effects of tooth spacing errors, and produces the higher absolute values of the friction coefficient (Figs. 4-6 and 4-7) Therefore, to achieve the best profile modification for scoring risk reduction, the preceding calculations should be slightly biased to increase the material removal at first point of contact while decreasing the removal at the last point of contact. A 20-30% bias shift is generally satisfactory. If practicable reductions in scoring risk are to be obtained through involute modification, profile slope tolerances must be held between ± 0.0001 and ± 0.0002 in. for the modified zones and tooth-to-tooth spacing accuracies of 0.0002-0.0003 in. must be achieved. Adequacy of the calculated design values must be confirmed during initial gearbox bench testing. Proper profile modifications for helicopter applications must reveal full visual profile contact throughout the range 50-75% of the red-line power; if less than full contact is achieved, the resultant loss of contact ratio at normal cruise power may cause excessively rough and noisy operation with an attendant reduction in pitting life. When the level of sophistication described is used in the calculation of $T_i + \Delta T$, together with precision in manufacture, the risk evaluations of T_c should be modified; a value of T_c = 500°F remains high risk, but T_c = 400°F as a medium risk thus would be a suitable classification for carburized AMS 6265 gears operating in MIL-L-7808 or MIL-L-23699 lubricants.

The preceding analyses do not adequately account for certain factors that are known to influence the calculated temperature rise ΔT and the true T_c for synthetic lubricants. Among these factors are:

1. The differences in friction and wear additive efficiency between MIL-L-7808 and MIL-L-23699 oils
2. The influences of surface topography, lay, and texture
3. The influence of EHD behavior as a function of temperature and velocity.

When tests are conducted under closely controlled conditions wherein the friction coefficient f, the initial temperature T_i, and the EHD parameters are known with accuracy, it has been reported that the assumption that T_c is constant actually is invalid. Ref. 20 shows a semi-log correlation between T_c and a dimensionless EHD parameter ξ_f, which depends upon the initial viscosity μ_o; the sliding and total velocities V_s and V_T, respectively; pitch radius R; and compressive (hertz) stress at failure S_{cf}. In this correlation the value of T_c for well heat treated, low retained austenite, case carburized AMS 6265 operating in MIL-L-7808 drops from about 600°F to about 430°F when the value of ξ_f increases by a factor of 10^2 (from $\xi_f = 10^{-19}$ to $\xi_f = 10^{-17}$). Values of T_c are approximately 100 deg F less for lower quality (with high retained austenite) case carburized AMS 6265 operating in the same lubricant (Ref. 71).

The reduction in T_c with an increase in ξ_f is due to the complex interaction of V_s and V_T. For constant V_T, T_c falls sharply and then levels off as V_s is increased, while T_c increases exponentially with V_T when V_s is held constant. Because the ratio V_s/V_T is constant for a given gear design, these opposite effects tend to cancel each other over common ranges of gear operating speeds and loads, producing a relatively constant value of T_c.

When actual friction data for a given lubricant-metal combination are not available, the trends shown in Fig. 4-24 (Ref. 71) are helpful in design review and evaluation.

From a practical viewpoint, when overpower tests show that scoring risk is marginal, the problem may be eliminated by such relatively minor remedial actions as:

1. Improving the run-in cycle by using longer runs at increased load and reduced speed to refine the operating surface finishes
2. Reducing the manufactured surface roughness through better grinding practice or the use of gear tooth honing where possible
3. Reducing the value of T_i through increased lubricant flow or cooler lubricant supply.

When such measures prove inadequate, the lubricant and the metallurgical microstructure should be evaluated. If neither can be improved, it may be possible to improve the involute modification or profile and the tooth spacing error.

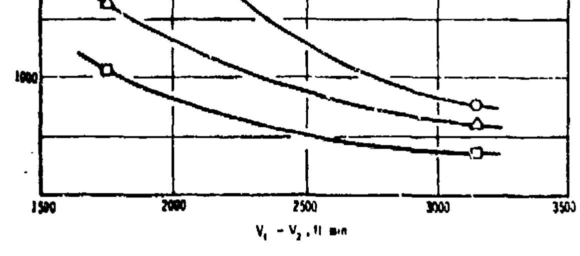

Figure 4-24. Scuffing Load vs Lubricant — Unsynchronized Discs

4-2.2.1.2.3 Pitting Failure

There are several pitting failure modes that in their advanced states produce the same end result: extensive spalling and tooth fracture. Only three of these modes are relevant to the type of gearing used in modern helicopter drive subsystems. They may be classified as case failure, classic or pitch-line fatigue, and wear-initiated failure.

4-2.2.1.2.3.1 Case Failure

This mode results simply from inadequate depth of case to support the operating load. It may be avoided by adjustment of either unit load or case depth to obtain a ratio of subsurface shear stress to shear yield strength in excess of a particular critical value. Ref. 79 recommends that a value of 0.55 for this ratio not be exceeded; however, for high-quality helicopter gearing transient operations at values between 0.55 and 1.0 should not result in failure. Extended operation above the critical ratio will cause subsurface cracks to occur near or in the case-core transition area as a result of the repetitive subsurface shear stressing. These subsurface cracks soon spread to the tooth profile surface and generally result in numerous brittle longitudinal fractures in the general area of the single-tooth contact zone. Total mutilation of the tooth profile then results from only a few additional cycles of load application.

The variation of subsurface shear stress with depth may be calculated in a straightforward manner. The magnitude of the subsurface shear for a given depth is a function of S_c and the Hertzian contact band semiwidth b. The calculation should be made for the lowest point of single tooth contact (LPSTC) on the pinion involute surface (unless the gear member is considerably weaker) because this produces the maximum value of S_c. S_c may be calculated in accordance with the methods shown in par. 4-2.2.1.2.3.2. The effective tooth load W_t' and the radii of curvature should be adjusted for the LPSTC. The Hertzian semiwidth b is related to S_c in the following manner:

$$b = (2.50 \times 10^{-7}) \left(\frac{r_p r_g}{r_p + r_g} \right) S_c \text{ , in.} \quad (4\text{-}47)$$

where r_p and r_g are, as defined previously, the radius of curvature of the pinion and gear tooth, respectively.

Table 4-5 next should be used to calculate values of shear stress S_s at 12 depths. These values then may be plotted along with the allowable stress as shown in Fig. 4-25. The allowable values shown are 55% of the

TABLE 4-5. SHEAR STRESS VS DEPTH

DEPTH, in. = $C_1 \times b$	SHEAR STRESS = $C_2 \times S_c$
VALUE OF C_1	VALUE OF C_2
0.05	0.090
0.10	0.160
0.25	0.276
0.33 [1]	0.314
0.50	0.293
0.60	0.278
0.75	0.252
1.00	0.211
1.25	0.179
1.50	0.154
2.25	0.107
3.00	0.082

[1] MAXIMUM VALUE OF ORTHOGONAL SHEAR STRESS OCCURS AT DEPTH ≈ 0.33 b FOR CYLINDRICAL HERTZIAN CONTACT

shear yield strength, as a function of hardness at the given depth. An approximate relationship between hardness to shear yield stress is shown in Fig. 4-26. The allowable values of shear stress near the surface are omitted because of the large residual compressive stress field normally in existence there. Because this residual field will reduce the effects of the imposed subsurface shear stresses, this region is not critical to the analysis; the occurrence of failure in this region is limited to the high hardness gradient transitional depths.

4-2.2.1.2.3.2 Classic or Pitch Line Fatigue

Classic or pitch-line pitting has been treated extensively in the literature and is related closely to classic bearing fatigue. Pitting life may be calculated as a function of Hertzian stress S_c; it is a phenomenon associated with rolling contact, and the theory is not applicable if surface traction or shear stresses are of considerable magnitude. Consequently, valid analyses are limited to full EHD lubricant film

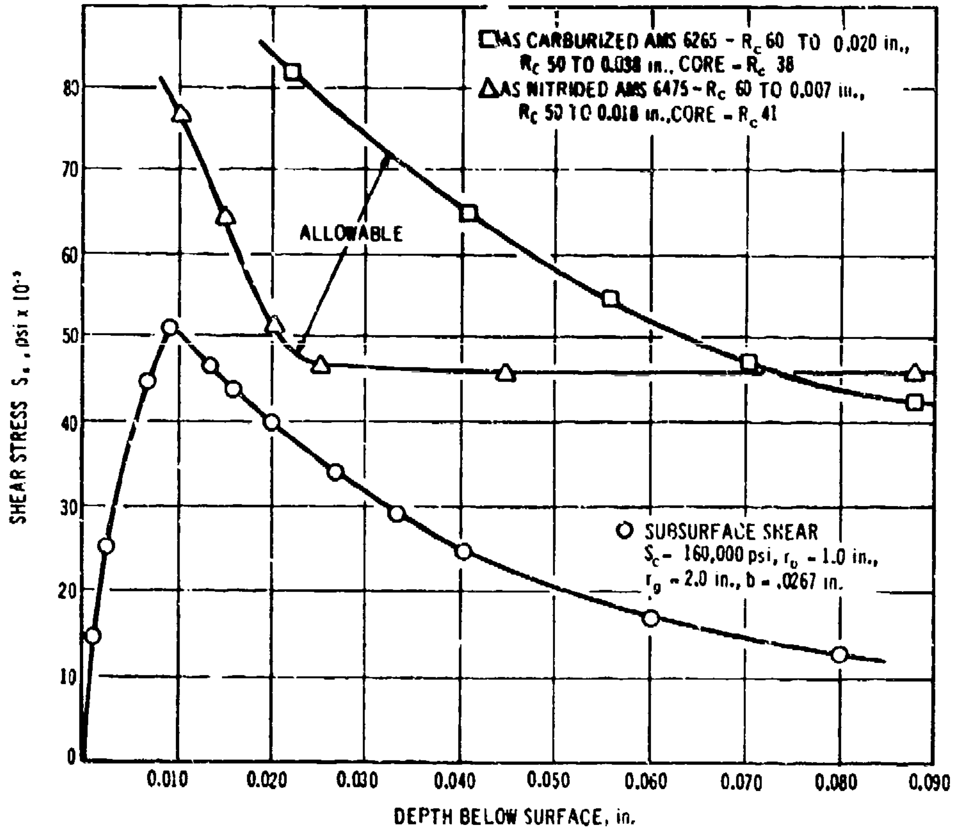

Figure 4-25. Case Depth Allowable vs Subsurface Shear

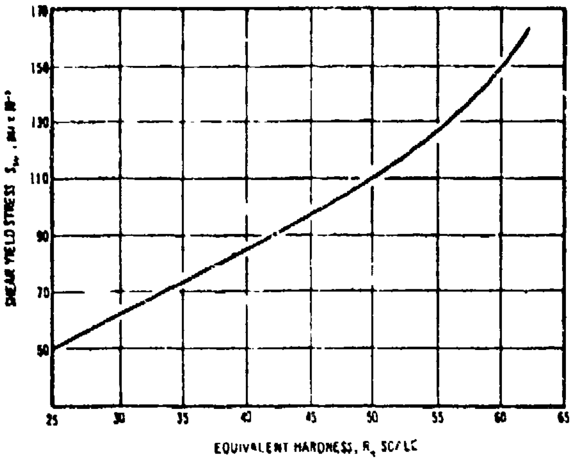

Figure 4-26. Shear Yield vs Hardness

separation as depicted in Regime II of Fig. 4-3. Operating in this region is not observed often in helicopter drive subsystems where low-viscosity synthetic lubricants are used, except in the very high speed gearing stages. Total velocities $V_T = V_1 + V_2$ of the order of 15,000 ft/min or greater are required to achieve Regime II conditions.

The fundamental AGMA approach to the calculation of pitting life is given in Ref. 80. For helicopter use, the following adaptation is suggested for calculation of the Hertz stress S_c:

$$S_c = C_p \sqrt{\frac{K}{I}}, \text{ psi} \qquad (4\text{-}48)$$

where

C_p = elastic coefficient, $(\text{psi})^{1/2}$
K = stress factor, psi
I = stress index modifier, dimensionless

In Eq. 4-48 the elastic coefficient C_p is given by

$$C_p = \sqrt{\frac{k}{\pi E'}}, \ (\text{psi})^{1/2} \qquad (4\text{-}49)$$

where

k = geometry factor, dimensionless (for cylindrical contact $k = 2.0$; for elliptical contact $k = 3.0$)
E' = combined modulus of elasticity, psi given as

$$E' = \frac{2}{\left(\dfrac{1+\mu_1^2}{E_1} + \dfrac{1+\mu_2^2}{E_2}\right)}$$

where μ_i are the Poisson's ratios.

The stress factor K is defined as

$$K = \frac{W_t'}{dF}\left(\frac{m_g \pm 1}{m_g}\right), \text{ psi} \qquad (4\text{-}50)$$

where

W_t' = total effective tooth load (Eq. 4-29), lb
d = pinion pitch diameter, in.
F = face width, in.
m_g = gear ratio N_g/N_p, a number $m_g \geq 1.0$

In Eq. 4-50 the term $(m_g + 1)$ is used with counterformal teeth and $(m_g - 1)$ is used with conformal teeth. The effective tooth load W_t' varies from that used in Eq. 4-29 in that W_t is taken at the pitch line, and for most applications $W_d = 0$ because the pitch line load is governing while an incremental dynamic load usually is limited to the initial mesh contact.

The stress index modified I in Eq. 4-48 is defined as

$$I = \frac{C_c'}{m_n}, \text{ d'less} \qquad (4\text{-}51)$$

where

$$C_c' = \frac{\cos\phi_t \sin\phi_t}{2}, \text{ d'less} \qquad (4\text{-}52)$$

ϕ_t = transverse operating pressure angle, deg

and the value for the contact ratio factor m_n also is calculated as in par. 4-2.2.1.2.1.

The value of S_c calculated using Eq. 4-48 should be used with the S-N curve shown in Fig. 4-12 to predict pitting life.

4-2.2.1.2.3.3 Wear Initiated Failure

This is the most frequently encountered failure mode, predominating throughout the transitional lubrication states between pure boundary layer and full EHD conditions (Fig. 4-3). Causes and corrective action are discussed in detail in pars. 4-1.2.1.2, 4-1.3.1, and 4-2.1.1.

In the absence of relevant test data or extensive experience, the best procedure for analyzing this failure mode is to calculate S_c by Eq. 4-48 and to apply this value to the applicable S-N curve of Fig. 4-18. More suitable life equations that take into account many of the significant variables other than S_c may soon become available from the many research programs now under way. One such program, entitled "Relationship of Lubrication and Fatigue in Concentrated Contact", is being conducted by the Research Committee on Lubrication of the ASME.

4-2.2.1.3 Gear Drawing and Specification

Without a drawing or specification adequate to insure control of the critical variables, little confidence can be placed in the value of gear analyses relative to expected service performance; reliability goals cannot be guaranteed and the results of any specific Airworthiness Qualification Specification (AQS) test become relatively meaningless. To achieve a workable logistical, maintenance, safety of flight, and otherwise cost-effective helicopter program, consistency of product must become a paramount consideration. Consistency or reduction of variability is of far greater importance at the operational level than is the achievement of any other criterion of performance such as power-weight ratio, strength, or efficiency.

The gear drawing must be amplified by numerous supporting specifications. However, the decision as to what class of data falls into each category is a matter of individual preference provided the result is a workable system for procurement, quality control, and necessary engineering review and change. The drawing is the document governing definition of the component, and it must clarify any ambiguities in or between supporting documents.

The following review list is intended as a minimum guide for assuring completeness of data, but no stipulation is made whether it be provided by drawing or by specification:

1. Raw material:
   a. Chemistry
   b. Certification condition
   c. Grain orientation
   d. Processing requirement
   e. Shape and size reduction from case ingot
   f. Finish
   g. Decarb limits.
2. Heat treatment requirements:
   a. Process controls
   b. Certification
   c. Properties, including microstructure
   d. Case hardness, surface and gradient, case depth and tolerance, and core hardness.
   e. Quenching and tempering limitations including time, temperature, and interval regulations
   f. Limits on reprocessing.
3. Serialization:
   a. Proper identification and traceability
   b. Location of codes and numbers
   c. System for transfer during processing
   d. Control of marking methods, size, and point(s) during processing for application.
4. Drawing technique:
   a. Specifications and Standards (MIL-D-1000), MIL-STD-10 with dimensioning practices to ANSI Y14.5
   b. Gear reference axis definition with location tolerance for inspection set-up
   c. Specified taper, waviness, roundness, concentricity, and finish requirements, assuring compatibility for journals
   d. Boundaries to cased areas.
5. Finishing requirements:
   a. Specified methods and limitations on use
   b. Specified peening techniques including set-up, shot, gaging, coverage, and certification frequency
   c. Means to avoid embrittlement and stress corrosion in all electrolytic, acid, or caustic processes.
6. Stock removal:
   a. Limits on stock removal (minimum and maximum if required) during grinding on all cased areas within tolerances compatible with Item 2 and with design stress analysis
   b. Specified methods of control.
7. Nondestructive testing:
   a. Specified requirements and methods for magnetic particle, penetrant, and etchant tests
   b. Specified frequency and sequence
   c. Specified frequency of certification of processes
   d. Specified equipment and precise location of identification for necessary hardness measurements on critical areas.
8. Balance requirements:
   a. Planes of measurement, limits, and speeds established and located, and permissible techniques specified when dynamic balancing is required
   b. Specified location, limits, and material removal methods for meeting balance requirements.
9. Tooth form:
   a. Provide clear enlarged detail of tooth form, graphically specifying tooth thickness, flank and root finish, over pin (or ball) dimensions, OD root diameter, and minimum fillet radius or equivalent
   b. Applicable data listed; i.e., N, P_d, ϕ, D_p, circular pitch p_c, involute base circle diameter D_b, and ψ.
10. Involute data:
    a. Slope and modification zones specified, e.g., by use of degrees rolled off base circle
    b. Critical diameters such as start of true involute, and edge break limits defined.
11. Lead data:
    a. If applicable, slope and crown defined
    b. End break limits and blend specified.
12. Allowable errors:
    a. Specified limits of manufacturing deviation

from the desired

b. As necessary, equipment or certifiable equipment capabilities required to measure such errors specified

c. Repeatability and standardization of proof check methods and frequency for inspection equipment check specified

d. Gear mounting within location limits given in Item 4 or equivalent specified for inspection

e. Tolerances on adjacent and accumulated tooth spacing, profile slope, lead slope, hollow or fullness of profile and lead, waviness of profile and lead, and undercut or cusp specified where applicable.

13. Chart format:

a. Inspection chart for involute profile and lead required to conform to a predetermined standard for proper interpretation and consistency

b. A sample chart with explanation of interpretive technique specifying magnification and paper travel speed provided.

14. Pattern limitations:

a. For spiral bevel gears bearing pattern checks required in lieu of profile and lead checks

b. Methods and machines by which bearing or contact checks are performed on production components run against "working masters", checked in turn against "grand masters" specified).

c. Data defining gaging dimensions, pattern size, shape, and location; and boundary tangencies through specified V and H and profile settings specified (See Ref. 81 for further definitions).

4-2.2.2 Bearings

The discussion of bearing application design, life analysis, and drawing controls that follows is limited to radial ball, angular contact ball, and radial cylindrical roller configurations. However, the basic principles introduced are sufficiently general to serve well in application design of any rolling element type bearing. Efficiency, reliability, survivability characteristics, and standards recommendations were treated previously in pars. 4-1.2.1 and 4-1.3.3.

Army helicopter transmission bearings have exhibited a primary failure rate two times greater than that for gears and four times higher than that for all remaining transmission components (Ref. 46). Also, their replacement rate at overhaul was three times that of gears and 15 times that of the remaining components. The majority of these replacements were due to secondary failure such as debris ingestion and corrosion. Also of importance to the designer is the finding that ball bearings (predominantly thrust applications) exhibited ten times the primary failure rate of cylindrical roller bearings.

In order to achieve the failure rate reductions required by modern MTBF goals, it is advisable that, as a minimum, the designer:

1. Come to agreement with the bearing supplier with respect to specific application needs.
2. Clearly specify application requirements by pertinent drawing or specification.
3. Evaluate the effectiveness of potential gains available with amended specifications in order to understand what changes in price are justifiable.
4. Inspect bearings for compliance with specification.

AMCP 706-201 describes the elements of bearing type selection and gives many examples of typical helicopter configurations. The primary function of helicopter bearings is to provide accurate positioning of gear and shaft components under wide ranges of speed while also exhibiting satisfactory life. Means of achieving this goal are described in the paragraphs that follow.

4-2.2.2.1 Application Design

Four general areas appear to create the major difficulties in bearing application design. They are:

1. Mounting practices
2. Lubrication techniques
3. Internal characteristics
4. Skidding control.

4-2.2.2.1.1 Mounting Practices

In most helicopter applications of the rolling element bearing, the loads are relatively large in relation to the physical dimensions and weight of the bearing. Good design requires consideration of the elastic behavior of such a system; adequate support for both the rotating and nonrotating rings is necessary, and the supporting shafts and housings must have greater rigidity than the bearing rings. This criterion may be satisfied by use of shaft wall sections that are at least as large as the bearing inner ring thickness, and of total housing-liner-quill cross sections equal to the total bearing cross section. Use of thinner sections should be avoided unless careful stress and deflection analyses prove that they are feasible.

Fretting wear, creep, and spinning are undesirable phenomena generally associated with the bearing inner ring-shaft interface (inner ring rotating with respect to load vector). Proper inner ring interference fit is the most important parameter for control of these conditions.

Fretting wear is the result of localized rubbing of very small amplitude at the interface, and is difficult to control unless the ring cross-sectional thickness is large enough for the loading conditions involved.

Ring creep is the slow relative (lagging) motion of the ring with respect to the shaft and occurs in some instances as a result of sufficient fretting wear to reduce the interference fit. A relative rotational speed of as little as $10^{-6} \times$ shaft speed may result in sufficient wear over several hundred hours to reduce the design interference. Often, creep occurs initially because of insufficient design interference.

Spinning is a term used to denote an advanced state of creep that occurs in cases with loose fit up or no interference between the shaft and the inner ring of the bearing. With hardened and ground precision interface surfaces, polishing and advanced wear rates often result when the relative rotational speeds approach 10 to 20% of shaft speed under operating conditions not unlike those in a simple sleeve bearing. Fig. 4-27 (from Ref. 82) shows a wear vs time function for a cylindrical roller bearing application.

There are a number of design practices that may be used to counteract these phenomena. They are described here in a descending order of preference.

1. The extent of the interference fit necessary to provide sufficient radial force to prevent creep may be calculated and employed in the design. Two factors that must be considered are:

a. Circumferential stretch of the bearing ring under applied rolling element loads which effectively increases the inside diameter (ID) of the ring.

b. The influence of the temperature gradient from shaft to the bearing ring upon relative thermal expansion. This gradient is a function of the cooling paths and of heat generation or friction loss.

Ref. 6 presents a calculation technique suitable for interference fit determination. However, the calculations cited assume a solid shaft and generous ring sections such as are found in 200 or 300 series bearings. Therefore, when dealing with application of this formula to lighter section bearings combined with hollow shafts, it is necessary to compensate for the reduced radial pressure per unit interference by the method of elastic ring theory as defined in Ref. 83. Use of interference fits that produce surface tensile stresses in the circumferential direction above 10,000 psi should be approached with caution because the fatigue life of the race may be reduced.

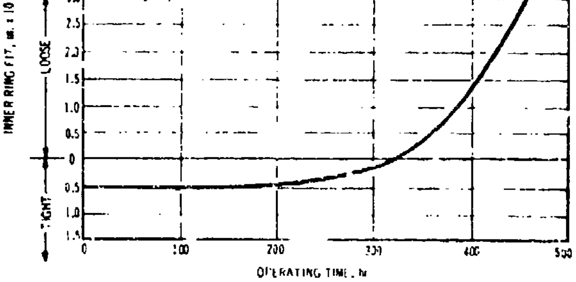

Figure 4-27. Creep Wear — Inner Ring Fit vs Operating Time

2. Excessive thermal gradients should be avoided. Because the circulated lubrication oil acts to modulate these gradients through forced convective cooling, an increase of oil flow to the shaft and bearing often can be used to alleviate thermal problems.

3. Very thin ring sections as used in AFBMA sizes below series "0" for ball-type bearings and series "1" for cylindrical roller-type bearings should be avoided for high-load applications. Elimination of the inner ring by use of integral shaft raceways for cylindrical roller applications is an effective means of avoiding the problem altogether.

4. Hardened (R_c60), ground, or honed journals (roughness ≤AA10) with dimensional tolerances equal to or closer than the rings of the bearing will assure attainment of desired calculated pressures and will withstand frequent assembly and disassembly and long service with minimal loss of interference.

5. It is desirable to use 2/3-lip-depth shoulders, spacers, and clamp nuts that are square with the journal surface and provide rigid axial clamping. Positive nut lock devices always should be used because ring creep under high axial loading may rotate the nut. Because bearing ring creep results from a lag of the ring behind the shaft rotational speed, it is simple to determine whether such conditions will serve to tighten or loosen the nut.

6. Copper or silver plating on the shaft interface has been used with some beneficial results in reducing or virtually eliminating fretting corrosion and thus prolonging component service life.

7. Positive ring-shaft interlocking with notched rings and keys have been employed to prevent creep. However, satisfactory installations are difficult to achieve because sufficient fretting corrosion may occur between the shaft-key-bearing ring surfaces to precipitate bending fatigue failure in one or more of the members.

Outer ring (nonrotating load) diametral clearances and clamping require somewhat less diligent attention than do the inner rings of bearings. However, heavily loaded angular contact thrust bearings must be well clamped and their outside diameters must be well supported to prevent excessive coning under the action of the angled rolling element load vector. When only radial loads or light thrust loads are involved, retaining rings or similar devices are satisfactory for axial retention. The diametral fit generally should be nominally line-to-line to approximately 0.0003 in. tight at operating temperature to reduce ring rotation. Outer ring interference fits often are limited by requirements for ease of assembly and disassembly. Higher speeds call for tighter fits.

Outer ring rotation is normally opposite in direction to shaft rotation due to the traction forces exerted by the loaded rolling elements. However, in the case of lightly loaded, outer-land-guided, cage-type bearings, the viscous drag may be sufficient to reverse the norm.

It is customary to use thermally fitted and pinned steel liners in aluminum or magnesium housings to reduce wear and the rate of increase in outer ring mounting clearance due to rising temperature. The increased clearance at operating temperature should be compensated for when the fit is specified at room temperature. In the absence of thermal gradients, the fit in a massive steel liner having the same coefficient of thermal expansion as the bearing will remain unchanged at operating temperature, but the fit in a light alloy housing without liner will loosen in proportion with the product of bearing outside diameter (OD) temperature rise, and the difference in the thermal expansion coefficients of the alloy and steel. The change in the outer ring fit in the presence of a steel liner installed in a housing with considerable diametral interference will lie between these two boundary conditions, and is calculated easily. On the assumption that liner is fitted to a minimum of 250 deg F interference (line-to-line contact when the temperature differential is 250 deg F) at room temperature (par. 4-2.3.2), there is an appreciable, uniform pressure at the liner-housing interface. This pressure results in an elastic reduction in the diameter of the installed liner bore. As the temperature is increased, the pressure is reduced and the bore expands. The initial pressure, and, hence, the expansion rate, is dependent upon relative section thicknesses and material properties that may be evaluated by application of the elastic cylinder theory (Ref. 83). Fig. 4-28 presents a graphic solution to the room temperature fit correction factor. To illustrate two practicable extremes, conditions are represented both for a 140-mm-OD bearing installed in a 0.045-in.-wall liner that is in turn fitted (250 deg F shrink fit) into a 0.05-in.-wall thickness aluminum housing; and for a 60-mm-OD bearing, with 0.090-in. wall liner (250 deg F shrink fit) and 0.040-in.-wall aluminum housing. If the operating temperature is 220°F, the corrections are 0.0013 in. for the 60-mm bearing and 0.0038 in. for the 140-mm bearing in this illustration. However, when, as is often the case, the bearing is the principal heat source and the housing provides appreciable heat conduction, these values should be reduced to compensate for the temperature gradient. A reasonable correction for most design applications is 60%. Consequently, the room temperature bearing outer ring fit-ups should be tightened by approximately 0.0008 in. for the 60-mm OD bearing and approximately 0.002 in. for the 140-mm OD bearing.

Occasionally, due to space limitations or a desire to eliminate unnecessary detail components, integral external flanges are used on bearing outer rings for axial retention and prevention of rotation. In this case, care must be exercised to avoid radial restraint at the flange holes so as to preclude race distortions due to thermal or load-induced deflections. Under the influence of high radial loads, a bearing of this design always will exhibit greater stiffness at the flange end, thus causing a shift in load intensity toward the flange. This characteristic can be used to compensate for bending deflections of the shaft or housing by appropriate location of the flange side of the bearing.

4-2.2.2.1.2 Lubrication Techniques

While theoretically it is true that only a slight amount of oil is needed to lubricate retainer rubbing surfaces and to supply fluid to rolling element-raceway conjunctions, helicopter applications often require the use of substantially greater quantities of oil. For example, circulating oil may be used to remove retainer wear particles, water condensation and sludge, and to transport spalling failure debris to chip detectors or similar diagnostic aids. Increased oil flows also act to modulate temperature gradients by forced convective cooling and help to reduce differential thermal expansion distortions that otherwise could reduce component life. As speeds and loads increase, thermal stability can be attained only through the rapid rates of cooling provided by high oil flows. In addition, centrifugal accelerations and windage barriers at high speeds make it very difficult to get oil to inner raceway and cage lands, leading to a requirement for forced pressure lubrication. Finally, critical bearings often must be lubricated by redundant systems so as to increase operational reliability and permit safe operation should the primary system fail.

Generally, lubricant applications may be grouped

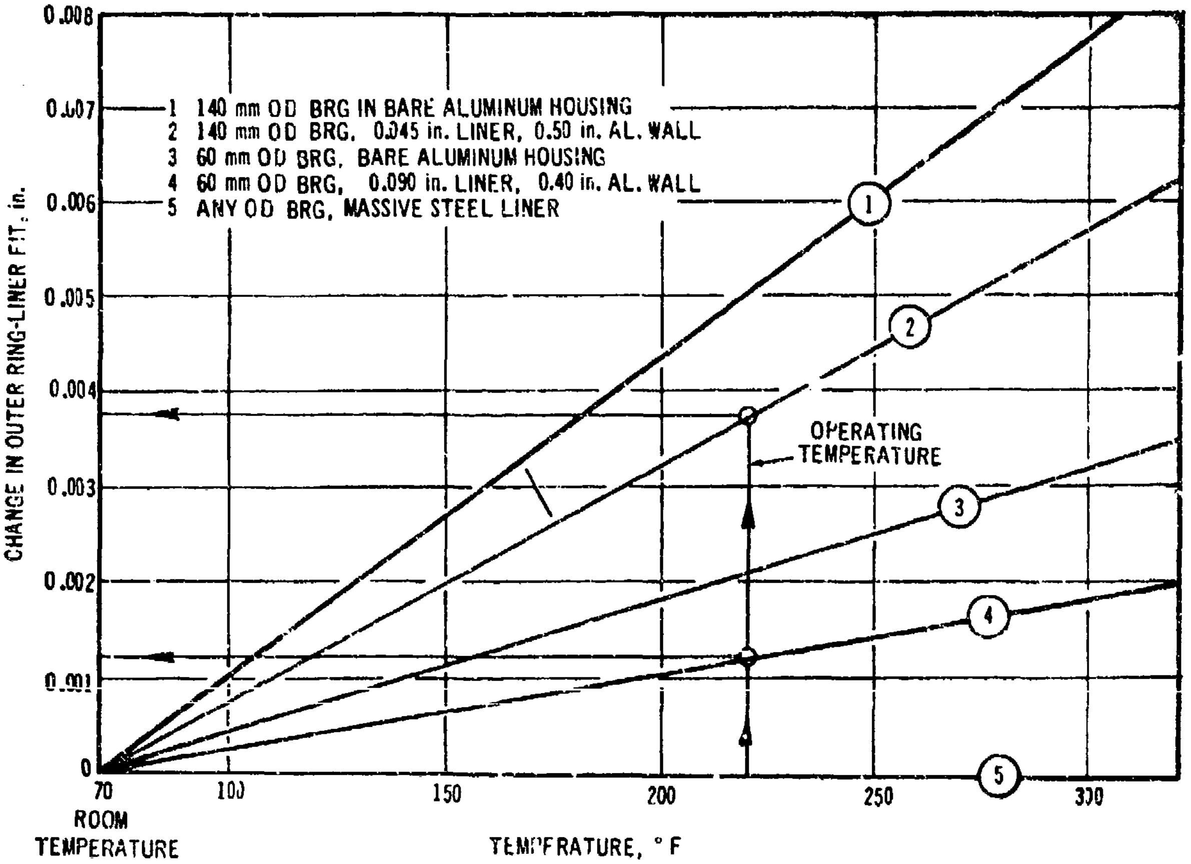

Figure 4-28. Temperature vs Outer Ring — Liner Fit Reduction

by approximate *Dn*-values (diameter *D*, mm; multiplied by speed *n*, rpm) according to the method of lubricant supply. The groupings that follow are arbitrary and may overlap in specific applications:

1. Low speeds ($Dn < 0.3 \times 10^6$):

a. Oil mist by either natural or forced flow of oil-laden atmosphere

b. Wick feed to either ring or retainer

c. Splash or dipping by dammed oil level (at least to center of lower rolling element)

d. Gravity feed through drilled or cast passage from trap located to catch sump return oil

e. Surface tension and/or centrifugal feed from rotating hollow shaft with oil acquisition from pressure jet or other mechanical means in combination with preceding methods

f. Pressure jet stream impinging on retainer-ring gap.

2. Moderate speeds ($0.3 \times 10^6 < Dn < 1.0 \times 10^6$):

a. Lightly loaded bearings may respond well to the methods of Item 1

b. Heavily loaded bearings require pressure jet impingement or integral pressure feed through ring face slots or feed holes to retainer lands and/or unloaded raceway surfaces.

3. High speeds ($Dn > 1.0 \times 10^6$):

a. Lightly loaded bearings with relatively open faces can be lubricated by high-velocity jet impingement

b. Heavily loaded or restricted configurations require internal pressure feed as described in Item 2. Oil egress must be considered. Outer ring counterbored ball-type bearings and lipless outer ring cylindricals frequently are employed.

4-2.2.2.1.3 Internal Characteristics

Of all of the internal geometric properties of a bearing the most important with respect to operating characteristics is diametral clearance. Such factors as control of initial shaft displacement to reduce gear misalignment, load sharing of the rolling elements in radial load applications, elimination of thermally induced radial preload, reduction of externally induced deformation loads (such as the pinch effect of planet idler gears), and determination of ball bearing contact angle are basically dependent upon diametral clearance.

A specific operating diametral clearance must be maintained under all conditions. While radial load deflection contributes to needed clearances, it is generally insufficient to compensate for bearing installation, or fit-up, practices or thermal expansion effects.

Changes in race diameter due to fit-up and to temperature differential between inner and outer rings can be calculated directly from elastic cylinder theory as presented in Ref. 83. Whether or not an increase in inner ring temperature will tend to reduce the raceway enlargement due to initial fit-up will depend upon shaft temperature and heat flow conditions. It is not uncommon to find highly loaded angular contact bearings operating at moderate to high speeds with the inner ring temperature 50-100 deg F above the outer ring temperature. Tolerances selected for shafts, housings, and bearings will have a direct influence upon the success of clearance compensation. The range of variations of bearing deflections and lives in a given application and, therefore, the life scatter within a lot of ostensibly identical gearboxes, can be reduced greatly by use of bearings of the higher precision ABEC and RBEC classification. When high interference fit-up and clearance compensation are required, there is a limiting practicable value for the ratio of ball diameter to radial cross-sectional thickness of the bearing. Ratios greater than 0.63 should be approached with caution unless there is considerable experience from which to draw.

The successful design of angular contact ball bearings for use in stacked sets of two or more requires a knowledge of their elastic behavior. DB (back-to-back; i.e., inner ring thrust faces opposed) and DF (face-to-face; i.e., inner ring thrust faces adjacent) configurations often are used to provide a combination of thrust and radial load capability, while DT (tandem) bearings generally are reserved for conditions where the thrust load is less than 40-50% of the radial load.

Fig. 4-29 represents a single-row, angular contact bearing. When operating speeds are such that the centrifugal force on the balls is not significant and the ball-outer race loads are essentially equal to ball-inner race loads, the line of contact is established by the centers of the race curvatures. In Fig. 4-29 the radii of the races are denoted f_i and f_o (inner and outer, respectively). When a radial displacement of the inner race $\Delta(D_p/2)$ occurs due to fit-up or thermal growth, the radial clearance and contact angle are reduced. If the inner ring is allowed to displace axially until ball contact exists with no load, a ring thrust face protrusion δ_H results. If a clamped-ring DB or DF mounting with a tight housing fit is employed, the resulting compression produces an internal preload and a compensating increase in contact angle. This may be eliminated by manufacturing each bearing with a thrust face intrusion equal to δ_H. The initial contact angle β_o also should be reduced by approximately $(\beta_o - \beta_1)$.

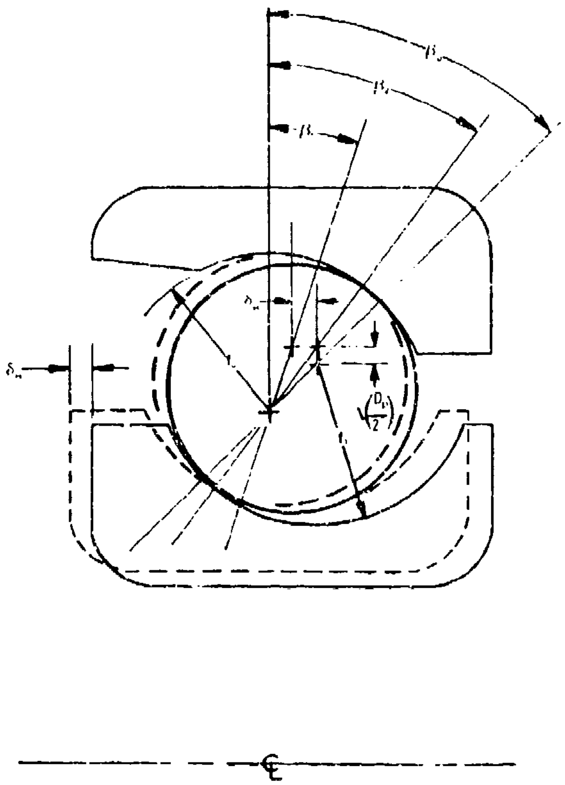

Figure 4-29. Bearing Geometry Change With Inner Ring Expansion

If there is a possibility for sizable thermal gradients and resultant preload, the DB mounting is preferred to the DF mounting because the load per unit of thermal expansion is considerably less with the DB mounting. Fig. 4-30 presents a graphic explanation of this condition. The thermal growth of x compensates for that of y in DB applications while the reverse is true for DF mountings. The resultant preload is an exponential function of the relative Hertzian compression, δ_N.

Load-sharing equalization of DT installation may

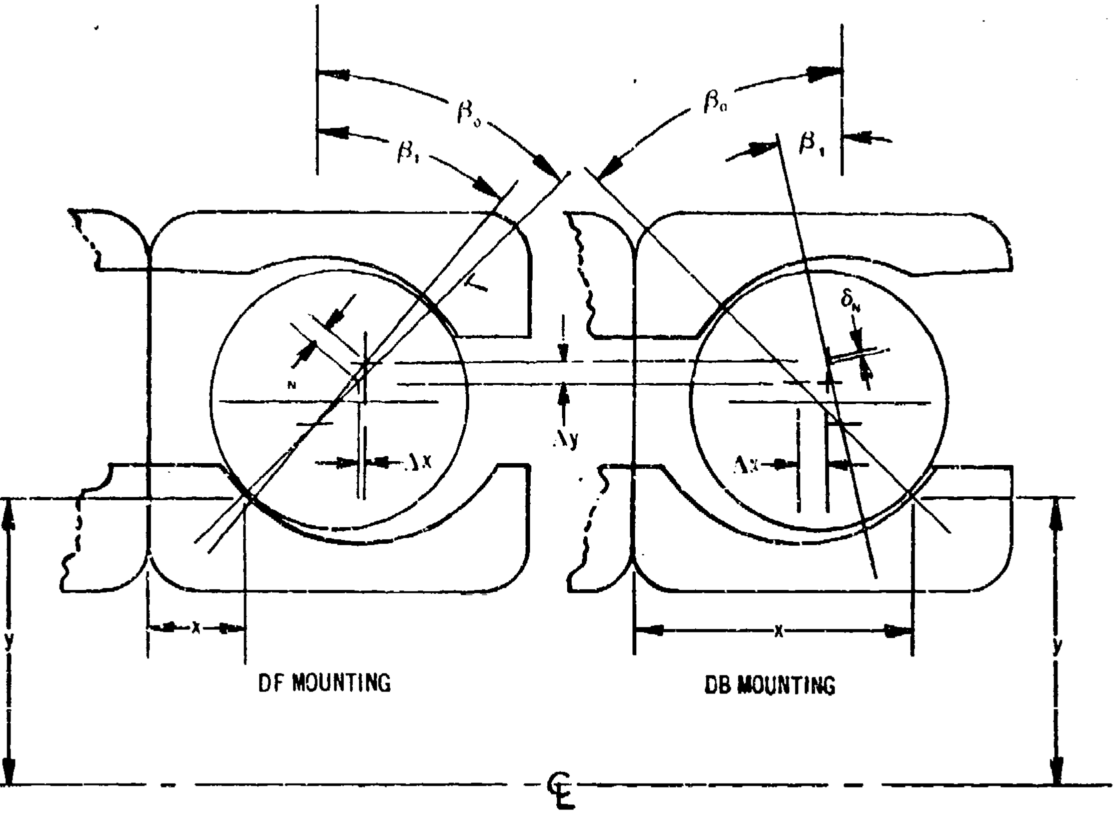

Figure 4-30. Relative Thermal Preload — DF vs DB

be enhanced significantly by requiring that the faces be flush (or equally offset) under a relatively heavy axial gaging load. This load should be at least 25% of the operating thrust load for maximum benefit.

Raceway groove shoulder heights must be adequate to support the elliptical contact area of the ball-race. Shaft misalignment and combination radial-thrust loads can affect a skewed ball path that contributes to this requirement. Most heavily loaded helicopter angular contact bearings require a shoulder height-to-ball diameter ratio of 0.25. Excessively skewed paths may require higher shoulders and increased radii of curvature for at least the nonrotating race groove. Such conditions also increase the requirement for retainer pocket-ball clearance to prevent excessive retainer wear or fracture. The balls in the loaded zone of operation will position themselves as the race-ball traction conditions dictate, including high retainer loads if clearances are insufficient.

Single-row angular contact bearings may be fabricated with two-piece inner rings (J-type bearing), and, in special instances, with two-piece outer rings. This eliminates the need for counter-boring and permits the bearing to resist thrust in either direction. The inner groove generally is ground with shims between the split halves so that, upon removal of the shim and assembly of the bearing, a Gothic arch shape results in the raceway groove. This configuration reduces the axial play of the bearing for a given contact angle. This fabrication technique generally enables one additional ball to be assembled into the complement, thus increasing load-carrying capacity for the same external envelope. Thrust-to-radial load ratios must be approximately 2.0 to prevent degenerative three-point contact. Misalignment and the resultant ball path skew also must be considered in avoiding three-point contact. The shim thickness that may be used in grinding the ring halves is limited by the need to maintain race clearance in the absence of a thrust load. Because the loaded inner ring is only half the width of that used in a conventional counterbored ring bearing, total radial pressure between the loaded ring and the shaft is considerably less for the same interference fit-up. Consequently, fretting and ring creep also are more difficult to control.

4-2.2.2.1.4 Skidding Control

Lightly loaded high-speed bearings may operate with gross sliding between the rolling element complement and the rotating inner race. Such operation can produce smearing or race surface failures not unlike those caused by gear tooth scuffing. The centrifugal acceleration present at high speed creates a considerable rolling element/outer race load, with braking traction forces exceeding driving traction forces at the rolling element/inner race contact. Retainer drag forces and lubricant viscosity also play an important part in determining load-speed-slip conditions. Historically, this distress mode has been a greater problem with cylindrical roller bearings than with ball bearings. Calculation of slip-critical conditions is relatively uncertain, but some useful insight may be gained from Ref. 8.

One method for preventing gross slip is to maintain at the inner race/rolling element contact the load required to obtain sufficient driving traction. This may be accomplished on DB or DF angular contact bearings with internal preload. A preload spring may be required with single-row ball angular contact bearings.

There are several methods for controlling gross slip with cylindrical roller bearings:

1. Out-of-round outer raceways may be employed to produce a pinch effect upon installation. Installation orientation is required such that the external gear loads are orthogonal to the pinch load plane (Ref. 84).
2. A small number of oversized radially tight, hollow rollers may be dispersed at even intervals throughout the complement. This method requires careful sizing of the hollow rollers to preclude bending fatigue failures (Ref. 85).
3. When the configuration permits, the roller bearing may be mounted very slightly off-center with respect to the shaft axis. If the shaft is positioned by an additional pair of bearings, sufficient radial preload may be effected to provide the necessary traction load.
4. Out-of-round liners have been employed to produce the same result as in Item 1. The primary disadvantage to this technique is the increased difficulty in assembly and disassembly.

To minimize skidding tendencies in high-speed bearings, the smallest acceptable diameters should be used both for the rolling element and for the pitch circle of the complement of rolling elements. Retainers may require balancing to obtain satisfactory operation. As in element skidding, a critical speed exists; the centrifugal acceleration at this speed will displace an out-of balance retainer off center until all land contact occurs in a single local zone of the retainer. This may, in turn, cause rapid retainer wear at the pockets as well as the guiding rails. Once started, the wear rapidly accelerates until failure occurs — often in 20 hr or less.

However, a dichotomy often exists with respect to clearance requirements. While controlled reduction of internal clearance to minimal values tends to reduce the skidding tendency of lightly loaded bearings, it comprises their ability to operate without lubrication, i.e., fail-safe operation (see par. 4-4.3, Emergency Lubrication). Normal heat distribution within a bearing with inner ring rotation results in a negative temperature gradient from inner ring through rolling elements and outer ring to the housing with the inner raceway operating broadly from 50 deg to 100 deg F hotter than the outer raceway. The shaft and inner ring heat flow paths offer less rejection capability than the outer ring and housing paths. This, coupled with the customarily higher heat generation rate attendant with inner race sliding velocities and counterformal contacts, results in the higher inner race operating temperatures. Under normal operating conditions, the lubricant removes the bulk of the heat and maintains thermal stabilization within this gradient. However, when the cooling effect and the friction reducing characteristics of the lubricant are absent, temperature stabilization can only occur at the higher gradients dictated by the increased friction and reduced heat rejection. If internal clearances are sufficient to accommodate the expansion attendant with the new gradient and increased overall temperature, then stable fail-safe operation is theoretically attainable. However, if inadequate internal clearance exists, a radially tight condition results. This in turn leads to a divergent increase in temperature until bearing seizure or shaft failure occurs.

As described previously, the mechanical means of providing positive rotation for the rolling elements in order to reduce skidding tendency can be applied in conjunction with greater internal clearance to affect a design without skidding and with fail-safe operating capability. Since the skidding tendency is highest in lightly loaded high speed bearings, it is possible to install nonload-carrying hollow rollers in cylindrical roller bearings without loss of needed capacity. This offers the driving feature required to defeat skidding while providing adequate radial clearance to accommodate thermal growth during fail-safe operation. Bearings with lower speed and higher loads exhibit progressively less skidding tendency and are designed with adequate radial clearance for fail-safe operation without need for auxiliary positive driving features. Thermal growth due to fail-safe operation in angular contact duplex ball bearins can be accommodated by providing adequate internal clearance initially (minimum contact angle of say 30 deg) or, if initially preloaded, by mounting the bearings back-to-back (DB). Back-to-back mounting allows the inner rings to grow radially and axially without generating additional preload, i.e., radial growth tends to increase preload while axial growth relaxes preload.

4-2.2.2.2 Life Analysis

Modern techniques for calculation of fatigue life of bearings are based upon the pioneering theoretical engineering and statistical analyses of Refs. 86 and 87. Certain empirical constants in these analyses were determined by evaluation of experimental data; hence, it may be argued that the effects of certain physical phenomena not specifically addressed in the theory are in fact represented in the final equations. If this is a valid argument, it follows that the bearing life in an application that differs substantially from the laboratory conditions could vary significantly from the calculated value. Fortunately, the statistical model used (a modification of the function originally presented in Ref. 88) is sufficiently general to permit meaningful interpretation of failure modes as diverse as human mortality, light bulb filament burnout, or wear-initiated gear tooth spalling. Consequently, valid test and field service experience can be used satisfactorily to add life modification factors with corrected dispersions to the Weibull distribution for bearing life prognosis.

4-2.2.2.2.1 Assumptions and Limitations

The basic AFBMA life calculations commonly used in the U.S. are based upon Refs. 86 and 87 and hence contain certain key assumptions and limitations:

1. The failure mode is subsurface-initiated pitting or spalling. Cracks begin at microscopic weak points, most probably at the depth of maximum subsurface orthogonal shear beneath the Hertzian contact. The developed solution, therefore, is based upon stressed volume theory. However, it has been indicated (par. 4-2.1.1) that a preponderance of the failures in the analysis of helicopter bearings at overhaul were surface initiated.
2. Hertzian stress theory is based upon the local compressive deformation of contacting bodies and hence ignores any possible effects of gross elastic changes of shape in these bodies.
3. Empirical coefficients used in the AFBMA formulas reflect the characteristics of air-melt AISI 52100 steel of R_c60 nominal hardness operating in medium-viscosity mineral oils at relatively low temperatures and moderate speeds and loads.
4. Any effects upon life caused by speed of rotation are omitted.
5. The calculated lives are based upon the number of cyclic stressings to produce failure in 10% of the population of a statistically significant sample size.
6. Bearings manufactured by different sources are assumed to belong to the same statistical population.

4-2.2.2.2.2 Modification Factor Approach to Life Prediction

A useful method has been advanced (Ref. 89) to account for many variables common in modern design applications. An adjusted life L_A is calculated as the product of adjustment, environmental and/or design factors, and the AFBMA calculated life L_{10}.

$$L_A = DEFGHL_{10}, \text{hr} \tag{4-53}$$

where

D = material factor (reflecting actual steel chemistry and purity), dimensionless

E = processing factor (accounting for CEVM and other melting practices, thermomechanical metal working, forging grain flow orientation, and absolute and element differential hardness), dimensionless

F = lubrication factor (considering lubricant EHD film formation and relative surface roughnesses), dimensionless

G = speed effects (considering centrifugal acceleration and slip conditions), dimensionless

H = misalignment factor (applicable to crowned and cylindrical roller bearings), dimensionless

It is not uncommon in helicopter bearing design for the value of the multiplicative group of factors to vary between 0.3 and 18 due to the range of conditions and requirements encountered. Digital computer programs often are used to define factors F, G, and H; while factors D and E are assigned values whether the life calculation is by simple AFBMA equation solution or by computer analysis.

4-2.2.2.2.3 Complete Elastic and Dynamic Solutions

Dynamic forces associated with high-speed operation not only change bearing operating characteristics greatly from those assumed for the static design, but also impose limiting speeds based upon failures due to sliding or contact slip heat generation. Fig. 4-31 shows that centrifugal acceleration at high speeds not only increases the outer ring/ball load for an angular contact bearing, but results in different contact angles at each race. The definitive axis for ball rotation is dependent upon the contact that has the greater "grip" on the ball. At high speeds this may be the outer race, which then forces the inner race contact into gross sliding. Also, because the ball rotation is not coincident with the bearing axis of revolution, a gyroscopic precession moment is induced. For balls of large size and high contact angle, this moment may induce complete precession slip, often with immediate overheating failure. Analyses of the governing forces are treated in Ref. 90. General computer solutions employing the equations of this reference also may consider the elastic deformation of shaft and housings in combination with the Hertzian deflections between the race and the rolling element as they influence the load distribution among a number of individual bearings on a common shaft.

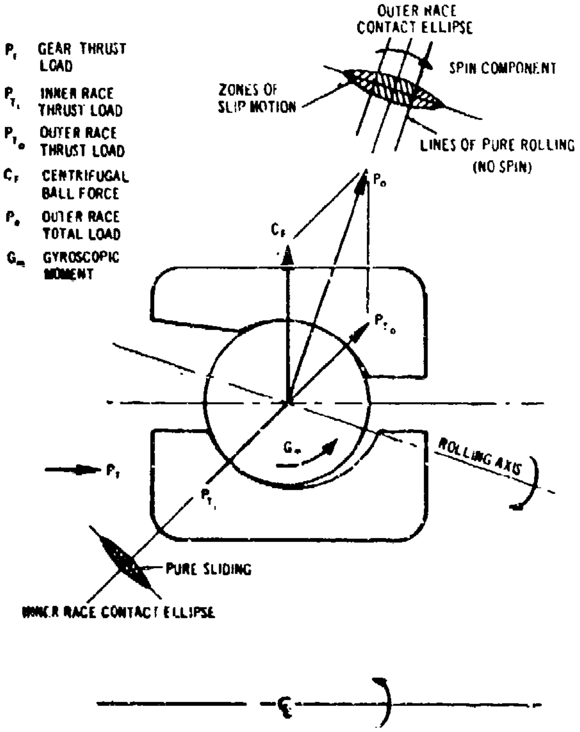
P_t GEAR THRUST LOAD
P_{T_i} INNER RACE THRUST LOAD
P_{T_o} OUTER RACE THRUST LOAD
C_F CENTRIFUGAL BALL FORCE
P_o OUTER RACE TOTAL LOAD
G_m GYROSCOPIC MOMENT
OUTER RACE CONTACT ELLIPSE
SPIN COMPONENT
ZONES OF SLIP MOTION
LINES OF PURE ROLLING (NO SPIN)
P_o
C_F
P_{T_o}
G_m
ROLLING AXIS
P_t
P_{T_i}
PURE SLIDING
INNER RACE CONTACT ELLIPSE
℄

Figure 4-31. High Speed Angular Contact Ball Thrust Bearing Forces

A special case of elastic deflection influence upon calculated life occurs in planetary idler bearings whose outer races are integral with the idler gears. As a result of the squeeze effect of the sun and ring gear radial load components, and of the moment upon the gear centroid due to tangential tooth loads, considerable deformation occurs and may create additional bearing loads of sufficient magnitude to reduce bearing life significantly. Rim section properties and internal clearances also have strong effects upon resultant life. Typical functions are shown in Fig. 4-32 (Ref. 91).

4-2.2.2.3 Drawing Controls

Confidence cannot be placed in the reliability or performance of a drive transmission bearing without a thorough evaluation of the important characteristics of the bearing as defined for the specific application. Bearing characteristics may be controlled by drawing, secondary specification, or manufacturers' source documents — depending upon individual preference. The following are minimal guidelines for such control:

1. Raw material:
    a. Chemistry
    b. Method of melt
    c. Certification limits
    d. Size reduction from ingot
    e. Grain orientation
    f. Thermomechanical processing limits if applicable
    g. Decarburization.
2. Heat treatment requirements:
    a. Process controls
    b. Certification
    c. Properties including microstructure, hardness
    d. Case and core properties where applicable
    e. Limits on reprocessing
    f. Retained austenite, where applicable, or time-temperature stability requirements.
3. Serialization and identification:
    a. Traceability
    b. Location of codes and numbers
    c. Process step for application
    d. Match marks for high points of eccentricity on precision sets
    e. OD code marking for verification of proper stacking of matched sets.
4. Dimensioning technique:
    a. Applicable ABEC and RBEC grades
    b. Pitch diameter; rolling element dimensions; race curvature; contact angle (unmounted); radial clearance; shoulder heights; flushness and gaging

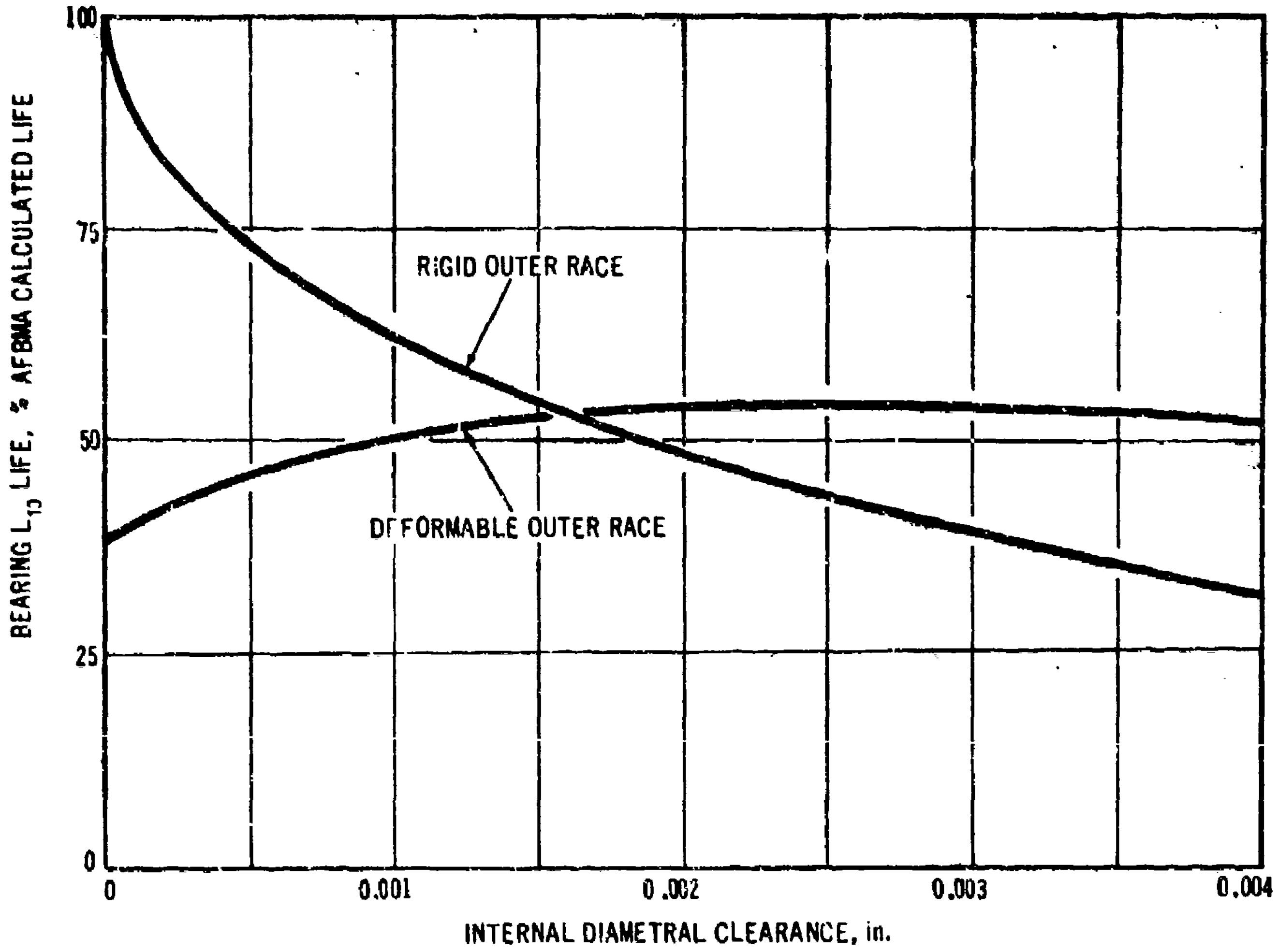

Figure 4-32. Clearance vs L_{10} — Elastic and Rigid Solutions

loads; corner breakout limits; retainer dimensions including pocket clearances, surface finishes; and applicable special dimensions which differ from ABEC/RBEC standards.

5. Finishing requirements:

a. Methods and limitations on plating, peening, honing, polishing, and stock removal, when applicable

b. Protection against embrittlement and stress corrosion.

6. Nondestructive testing:

a. Requirements for magnetic particle, penetrant, and etchant techniques

b. Control frequency and sequence of test or inspection

c. Frequency of certification processes.

The use of life modification factors (Eq. 4-53) cannot be warranted or substantiated unless specifically controlled by the fabrication and/or procurement document.

4-2.2.3 Splines

Basic introductory and classification information concerning splines is contained in AMCP 706-201. Therefore, this discussion is limited to specific design applications of power-transmitting splines.

The primary failure mode for a properly designed spline is wear. When relative motion is slight, fretting corrosion often accelerates wear. Good design practice will insure a wear life in excess of the useful component life. Galling and pickup (welding) occur only under excessive compressive stress in the presence of slight motion. Tooth breakage and fatigue seldom occur unless shaft bending moments were neglected in the design analysis. Neglect of proper fillet radius control or advanced fretting corrosion often contributes to such failures. Bursting of the internally toothed member is rare, but results from an excessively thin tooth backup structure that is insufficient for the hoop or tensile stresses imposed by the tooth separating and centrifugal forces.

Two basic spline types are employed in drive system design: face splines and concentric splines.

4-2.2.3.1 Face Splines

Face splines are typically used to couple two shafts or a shaft and a gear. Flat-faced, tapered, V, or square form teeth may be milled, shaper cut, or ground — depending upon material hardness and accuracy requirements. The fabrication of high-quality, interchangeable, concentric joints with uniform tooth contact is relatively difficult and expensive; consequently, they are seldom used in drive systems.

The most common face spline is the Curvic* system (Ref. 92). Curvic* splines are easily fabricated with pressure angles between 10 deg and 30 deg, although the higher value is predominant. Most design deficiencies result from inadequate localization of tooth contact relative to the tooth center or from inadequate clamping means to resist the tooth separating forces and the bending moments on the joint.

4-2.2.3.2 Concentric or Longitudinal Splines

This type of joint has its load-bearing surfaces essentially parallel to the rotational axes of the coupled components that comprise the external and internal mating elements. This discussion is limited to the commonly used involute tooth form, although other types are occasionally used.

The involute spline may be manufactured by any involute gear production method, in addition to methods not well suited to full-depth, high-strength gear tooth forms. Depending upon the limitations imposed by production volume and precision, involute splines may be produced by milling, shaping, shear cutting, broaching, cylindrical thread rolling, rack cutting, shaving; by rolling, hobbing, and form or profile-generating grinding. Relatively trouble-free applications are limited to misalignments of 0.001 in.-/in. and are either clamped or floating. Operating misalignment of the axes of the mating parts of 0.25 deg or greater under load requires the use of flexible couplings (par. 4-3.2.1). It is very difficult to obtain satisfactory wear life with floating splines operating with misalignment.

4-2.2.3.3 Properties of Splines

Involute splines are designated as major-diameter or sidefit, depending upon the controlling dimensional features. Minor-diameter-fit splines should be avoided in all but special applications (such as with a weaker internal member) due to the excessive stress concentration caused by the sharp tooth-root fillet radius on the external member.

Splines are clamped to prevent relative motion. Light interference fit dimensioning of major-diameter or side fit splines may be employed to assist in attaining secure clamping. When radial loads dominate the joint design, adjacent mating cylindrical bores and shoulders with light press fits frequently are used to complement the spline; successful application requires close control of concentricity between joint elements.

Floating spline joints primarily are used to accommodate axial motion. Diametral looseness and backlash must be sufficient to provide clearance under operating conditions. The choice between major-diameter and side fit control normally is predicated upon concentricity and balance requirements. Major-diameter control is preferable for precision applications where rotational speeds or alignment are critical. While side fit splines of 20 deg or greater pressure angles provide self-centering under torque loads, their looseness may permit excessive component imbalance and eccentric operation under no-load, high-speed operation.

Conventional involute splines may offer appreciable resistance to axial motion while under load. A safe design value axial force A_f for oil-lubricated spline may be taken as:

$$A_f = 0.4\,Q/D_p, \text{lb} \qquad (4\text{-}54)$$

where

Q = torque, lb-in.

D_p = pitch diameter, in.

Reduction in slip force may be achieved by use of special lubricants, friction-reducing tooth coatings, platings (silver, etc.) and treatments; with ball splines; or by the introduction of considerable misalignment (par. 4-3.2.1). Nylon (Ref. 93) and epoxy-bonded molybdenum disulfide coatings are often effective.

Floating spline joints also are used to provide a slight accommodation for radial, axial, and angular misalignment. Under these operating conditions, fretting and galling wear modes may prove troublesome. Their occurrence is difficult to predict, and determination of secondary effects and solutions frequently must await design development testing. Depending upon the severity of the problem and the design restrictions, the following solutions have found widespread use individually or in combination:

1. Increased hardness and accuracy (generally a matter of gear tooth grinding precision)
2. Shot peening of one or both members (relatively high intensities and surface texture modification are desired)
3. Use of dissimilar materials, types of heat treatment, and hardnesses

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*Registered Trademark — Gleason Works

4. Crowning of the external member tooth flanks and major diameter

5. Increased oil flow or other lubrication improvements.

4-2.2.3.4 Spline Strength Analysis

AMCP 706-201 gives allowable bearing pressures S_{br} for various classifications of involute splines. These values reflect approximate current practice in the helicopter industry and are defined by

$$S_{br} = 2Q/\left(D_p^2 F\right), \text{ psi} \tag{4-55}$$

where

D_p = pitch diameter, in.
F = face width, in.
Q = torque, lb-in.

for standard SAE or ANSI B5.15 tooth proportions where tooth addenda are one-half those of AGMA standard 201.02 gears. However, these values do not represent true bearing pressures because the accuracies, stiffnesses, and geometric proportions of typical splines combine to reduce the true contact area to less than the 100% tacitly assumed in Eq. 4-55.

The accuracy of splines is determined by the tolerances as specified and the method of inspection employed rather than by the method of manufacture. Splines may be gaged (go-no-go systems), gaged and partially inspected analytically, or completely inspected analytically in the manner of gears. The following values of the fraction of theoretical contact γ achieved with splines of the various classifications are realistic:

Classification	Contact Fraction
Gaged USASI Cl.5 (commercial grade)	$0.25 \le \gamma < 0.45$
Gaged and measured ANSI B5.15-1950 Cl.5	$0.45 \le \gamma < 0.75$
Analytically measured SAE Cl.3 (about 50% tolerance of B5.15 Cl.5)	$0.75 \le \gamma \le 0.95$

The variation of γ within each classification is dependent upon stiffness, proportions, and, in some instances, the ductility when the design load approaches limit shear strength.

An external involute spline with face width $F = D_o/3$ on a solid shaft will exhibit greater shear strength than the shaft if D_o (outside diameter of the shaft) is only slightly smaller than the spline minor diameter D_{mi}. Therefore, excessive spline lengths can offer little benefit while increasing manufacturing difficulty and costs. Reasonable proportions for spline face width F and pitch diameter D_p are:

$0.4 \le F/D_p \le 1.0$ for torque-transmitting applications

$0.8 \le F/D_p \le 2.0$ for location and alignment applications.

Lengthwise tooth load uniformity can be enhanced further by adjusting the shaft diameters and wall thicknesses to secure matching torsional deflections and by avoiding excessive radial stiffness at either end of the spline joint. Stress concentration must be avoided by specifying minimum fillet radius values, chamfering or otherwise blending tooth ends into the shaft section, and achieving uniform loading.

For most helicopter applications, spline fatigue endurance is not the limiting criterion because oscillatory loading due to shaft bending or torque fluctuation is avoided by proper design. If bending fatigue is a design consideration, recourse to use of the modified Goodman diagram (par. 4-2.2.1.2) with appropriate stress concentration factors is required. Static stress analyses must demonstrate a positive margin of safety both for limit torque compared to material yield strength and for ultimate torque compared to material ultimate strength. Limit spline torque is defined as 1.5 × maximum drive system continuous torque and ultimate torque as 1.5 × limit torque.

The following spline static stresses should be calculated in addition to the bearing stress values S_{br}:

1. Spline shear stress S_s:

$$S_s = \frac{2Q}{\gamma D_i \text{ [illegible]}}, \text{ psi} \tag{4-56}$$

where

Q = spline torque, lb-in.
N = number of spline teeth
T_c = circular tooth thickness, in.
γ = fraction of theoretical contact, dimensionless

2. Torsional shear stress (external toothed members) S_{ts}:

$$S_{ts} = \frac{16 Q D_{mi}}{\pi\left(D_{mi}^4 - D_i^4\right)}, \text{ psi} \tag{4-57}$$

where

D_i = inside diameter of shaft, in.
D_{mi} = minor diameter of spline, in.

3. Bursting stress (internal toothed member) S_b:

$$S_b = 7.095 \times 10^{-6}\rho(nD_2)^2 + \frac{4Q\tan\phi}{\pi F D_p^2\left[\left(\frac{D_2}{D_1}\right)^2 - 1\right]}, \text{psi} \tag{4-58}$$

where

D_1 = major diameter of spline, in.

D_2 = outside diameter of spline tooth member = $D_1 + 2 \times$ (back-up rim thickness), in.

n = rotational speed, rpm

ρ = material density, lb/in.^{3}

ϕ = spline pressure angle, deg

Limit and ultimate margins of safety must be calculated using appropriate torque values in these equations and comparing the stresses calculated with Eqs. 4-56 and 4-57 with the allowable yield and ultimate shear stresses of the respective parts. The stresses calculated by Eq. 4-58 are compared with the allowable yield and ultimate tensile stresses of the internal toothed member.

4-2.2.3.5 Drawing Design and Control

Splines should be specified on the engineering drawing or other document in a manner similar to gear teeth. An enlarged, dimensioned sectional view and a data block should be included. An even number of teeth is preferred for over/under wire inspection purposes and manufacturing ease. To prevent involute undercutting and permit use of the maximum number of manufacturing technique options, tooth numbers must be no lower than those in Fig. 4-33.

The enlarged spline drawing ($D_p/2$ is a convenient scale) should present major diameter, pitch diameter, form diameter, minor diameter, minimum fillet radius, circular tooth thickness (external), circular space width (internal), tooth tip chamfer, dimension over (external) or under (internal) gage wires, and surface finish.

The data block should present number of teeth, diametral pitch fraction a/b (where a represents $P_d = D_p/N$ and b is the value of P_d when it is expressed as the reciprocal of the addendum length), pressure angle, base diameter, total or composite index error, maximum deviation of parallelism tooth-to-tooth for given length of engagement, and parallelism limits with respect to part reference axis or surfaces.

Involute profile tolerances should be presented as data, for gage inspection, or as a chart, for analytical inspection. For gage inspection techniques, major diameter and pitch diameter eccentricities must be absorbed within the limitations of the major diameter and effective tooth (or space) thickness tolerances. For externally gaged splines the maximum effective and minimum actual circular tooth thicknesses must be specified; while for internally gaged splines, the minimum effective and maximum actual circular space width must be determined. When inspection with gages is specified, the diameters of over-and-under wires called for are referenced data. With analytical inspection techniques, tooth thickness and space width are given as actual minimum and maximum limits, and tooth-to-tooth spacing tolerances also must be specified. In addition, allowable tooth lead error should be substituted for parallelism error.

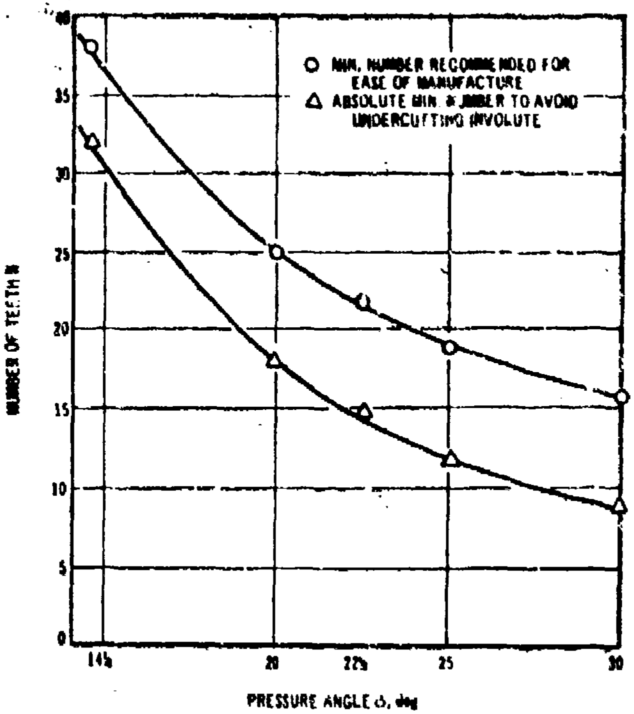

Figure 4-33. Involute Spline Data ϕ vs N

The manufacturing method must be considered when detailing the spline. Shaper cut splines should have a minimum chip and cutter overrun gap equal to their total depth. The minimum relief diameter for an internal spline should be equal to the major diameter plus one quarter of the whole depth, and for an external spline, the minor diameter minus one quarter of the whole depth. These diametral clearance values also may be used for broached splines. Hobbed or ground splines, of course, must provide overrun clearance for the wheel or hob radius.

4-2.2.4 Overrunning Clutches

Certain overrunning (free-wheeling) clutch requirements were described in par. 4-1.2.2. The lowest drive system weight will result from placing the clutch in the location of highest speed; i.e., between the engine and the first stage of reduction gearing. However, other considerations in multiengine configurations — such as vulnerability, safety, and reliability — may require locating the clutch between the first and second stage of reduction gearing.

Operational requirements for clutches vary with helicopter configuration, mission, and life cycle. However, current Army clutch requirements for twin-engine helicopters typically are.

1. The minimum ultimate torque capacity of the clutch *shall* be 2.0 × limit torque (limit torque = 1.5 × maximum continuous drive torque).

2. 2-hr, full-speed continuous overrunning *shall* be possible without operational capability impairment.

3. 30-min, full-speed continuous safe operation *shall* be possible after total loss of lubricant system.

These requirements apply to the entire clutch system, including support bearings and, often, seals.

In addition to these requirements, there exist other significant design considerations as evidenced by observed failure modes in existing helicopter clutch applications:

1. Brinelling due to the presence of oscillatory torque pulsations. Depending upon operating stresses and configuration, overrunning clutches will safely tolerate only 10 to 30% continuous oscillatory torque. External shaft bending and radial or moment loads must be eliminated from the clutches by use of relatively rigid support bearings that maintain concentricity at all times between the driving and overrunning members. When modeling a drive system for torsional analysis, it is important to consider the clutch as a relatively soft torsional spring. Stiffness values typically range from 35,000 to 350,000 in.-lb/rad (Ref. 94).

2. Excessive wear at intermediate overrunning speeds. Maximum wear conditions usually are encountered when the output member is operating at full speed and the engine is at idle speed. Most sprag- and roller-type clutches evidence their greatest wear rates when (input speed) / (output speed) $\approx$ 0.5 due to the product of centrifugally induced compressive stresses and sliding velocities.

3. Failure to engage at high speeds. In many instances reported, the second engine has failed to engage after the first has accelerated the system to ground idle speed. Both sprag and roller clutches require a critical friction coefficient of about 0.05-0.07 to engage. Hydrodynamic or elastohydrodynamic oil film formation and/or externally induced vibratory modes may lower friction coefficients below this level at the moment of speed synchronization, resulting in momentary or complete overspeeds. Subsequent adjustments of input or output speeds may lead to abrupt engagement with attendant shock loads sufficient to fail adjacent drive system components.

4. False brinelling of clutch elements or support bearings. Clutch support bearings operate in a static mode whenever the clutch is engaged because both inner and outer bearing rings rotate in unison. External vibration thus may cause fretting or false brinelling at the rolling element/race contacts. The entrapment of wear particles and sludge in the outer race often accelerates such wear. Therefore, it is important to maximize the static capacity of the support bearings for the available envelope and to provide good oil circulation without stagnation areas.

Design considerations peculiar to particular types of clutches are given in the paragraphs that follow.

4-2.2.4.1 Sprag Clutches

Sprag clutches are the most widely used type for helicopter drive systems. Two variations have been used with success. Both employ a complement of equally spaced, full-phasing sprag cams operating between concentric circular races. A detailed study of their geometric and operating characteristics is presented in Ref. 95. Race cross sections must be sufficiently large to prevent elastic deflection under load from increasing the sprag space by more than about 0.002 in. Race hardness and case depth must be adequate to support operating Hertzian stresses of 450,000 to 500,000 psi at the sprag/inner race contact. Successful applications of these designs are based upon between 3×10^6 and 10^7 cycles of full torque application without failure.

For moderate to high-speed operation it is preferable to use outer race power input with inner race overrunning to reduce the centrifugally induced sprag/race contact stress. In such usage the sprag complement should remain stationary with respect to the outer race and should slip at the inner race during overrunning. This arrangement also permits centrifugal-feed lubrication through the inner race and reduces the race/sprag sliding velocity for a given overrunning speed.

Both clutch types usually employ a degree of centrifugal self-energization by virtue of sprag center of gravity (CG) offset with respect to their contact engagement axes. This can cause some problems with high-speed applications because the drag torque (power loss) and wear may be excessive.

The two clutch types differ in some characteristics. For example, one uses two concentric cage elements to separate the sprags, while the other uses a single outer cage. The double-cage type uses an inner race drag spring to react the centrifugal self-energization during slip conditions, and may produce considerably lower overall drag forces at intermediate slip conditions. The single-cage design frequently employs an integral rib on the sprag which contacts the adjacent sprag and thus limits sprag overload rock angle.

The overload failure modes of the two types also differ. The double-cage design fails by sprag turnover (with resultant permanent loss of drive) while the design with ribbed sprag and a single cage fails by slipping. However, both failure modes generally exceed the 2 × limit torque requirement by comfortable margins. The single-cage clutch usually has a higher torsional spring rate than the double (for equal envelopes) along with a greater tolerance for oscillatory loading conditions.

Proper lubrication of either type requires complete oil immersion. This often is accomplished by use of full-depth circular dams on both sides of the sprag unit.

4-2.2.4.2 Ramp and Roller Clutches

Ramp and roller clutches also have found extensive successful application in existing helicopters. Such designs employ a cylindrical outer race as in sprag clutches, but use cylindrical (hollow or solid) rollers in lieu of sprags, plus a multiple-cam-surface inner race to provide a wedging action on the rollers upon engagement. Overrunning usually produces roller complement rolling contact with the outer race and sliding with the inner. Consequently, most moderate- to high-speed applications feature inner race input with outer race overrunning. This design may require forced feed (pressure) lubrication through the inner race to obtain satisfactory full-speed overrunning.

Spring-loaded cages or individual roller springs are used to force the rollers into the wedge to secure reliable and rapid engagement upon race speed synchronization. A thorough analysis of the design geometry and speed characteristics of these clutches is given in Ref. 96.

Due to the reduced radius of curvature in the roller as compared to the sprag cam, roller clutches have lower torque capacities than sprag clutches of comparable size. Overrunning drag torque also is greater at high speeds for the roller clutch. Failure mode of this clutch type at overtorque is slip, if the cam/race components are sufficiently strong to preclude their fracture, drive capability is not lost. In most installations, the roller clutch has shown a superior tolerance to oscillatory torque-induced wear.

4-2.2.4.3 Self-energizing Spring Clutches

Although there have been no applications of spring clutches in production helicopter systems, considerable interest has developed in them because they have the potential advantage of reduced weight and size for a given torque capacity. The principal deterrent to the use of spring clutches in helicopters has been their poor release characteristics in overrunning. Recent design improvements feature a tapered-width helical spring of rectangular cross section and cylindrical outside diameter (Ref. 97). The torque transmission is between a cylindrical outer race and the outside diameter of the spring. The device may be servo-actuated with an energizing pawl that contacts the small end of the spring or self-energized by friction forces between the spring end and the outer race. Recent development and test experience is reported in Ref. 98.

4-2.2.5 Rotor Brakes

AMCP 706-201 describes the basic requirements for rotor brakes, while AMCP 706-203 presents the minimum qualification test requirements. This discussion, therefore, is confined to typical detail requirements and limitations and to basic design and analysis procedures.

While this paragraph treats only hydraulically actuated disk- and puck-type brakes, the basic analytical techniques presented are sufficiently general to aid in the development of design criteria for other types of rotor brakes. The disk brake has become virtually the standard for helicopters due to its relative simplicity, ease of inspection and maintenance, and reliability.

A rotor brake differs significantly from a wheel brake both in failure modes and in functional requirements. The catastrophic failure mode for a wheel brake is failure to engage, or failure to stop the aircraft. Puck clearances are nil, contact speeds are moderate, significant cooling may occur during and after use with disk ventilation assisted by rotation, and repetitive use with short operating cycles and intervals is common. The catastrophic failure mode for a rotor brake, on the other hand, is unintentional operation. Puck clearances must be very large, contact speeds may be very high, the primary cooling is provided by the disk heat sink, and repetitive use in less than a 5-min time interval is virtually impossible.

4-2.2.5.1 Requirements and Limitations

Recent performance specifications for Army helicopter rotor brakes have included the following requirements:

1. *Shall* stop rotor from 100% speed in 30 sec
2. *Shall* allow 1000 stops without part replacement
3. Must hold rotor stopped against 45-kt wind while helicopter is not in use
4. Must hold rotor stopped while engines are at idle
5. Must not be located on a main driveshaft, or where lining debris could cause FOD to engines or APU
6. Activation and control *shall* be fail-safe. Safeguards are required to prevent inadvertent activation. Engine control interlock required with positive retention in lock and unlock modes.

Development and performance histories of helicopters also suggest the following guidelines:

1. The best (in simplicity, reliability, and safety) hydraulic system is a manual hydrostatic type. If operated from or boosted by pump/accumulator systems, these systems should be divorced completely from flight control or servo actuator systems.
2. Automatic self-adjustment is undesirable because it compromises reliability. Sufficient fluid should be provided to accommodate the useful wear life of the linings; manual hydrostatic, dual-level, mechanical advantage systems have been developed to accomplish this requirement (Ref. 99).
3. The disk should be stiffly coupled. A short, torsionally stiff takeoff drive on the main rotor transmission is often desirable. Soft mounted disks (such as on intermediate or tail rotor gearbox drive-shaft hangers) invariably become a vibration and antinode at some speed during engagement with resultant oscillatory loads on disk and/or puck attachments. These loads may cause intermittent brake chatter, dynamic system overloads, or crew annoyance.
4. The puck or caliper assembly should have a high spring rate mounting that is stiff in all loading vector component directions.

From a practical viewpoint, there may be limitations that place two or more of these requirements into conflict. Often, specification compromise or multimode brake activation systems are the result. For example:

1. Short stop time, high rotor inertia, and landing gear skid friction limits may combine to cause ground loop.

Braking torque Q_B is

$$Q_B = I\dot{\Omega}, \text{ lb-ft} \qquad (4\text{-}59)$$

where
I = rotor inertia, slug-ft^2
$\dot{\Omega}$ = rotor (angular) deceleration, rad/sec^2
and skid torque Q_S is

$$Q_S = Wfd, \text{ lb-ft} \qquad (4\text{-}60)$$

where
W = helicopter weight carried on each braked wheel, lb
f = effective coefficient of friction, gear to ground, dimensionless
d = track of braked wheels, ft

For a stop time of 15 sec, rotor inertia I = 7500 slug-ft^2 and Ω = 250 rpm rotor speed, Q_B = 13,090 lb-ft (ignoring aerodynamic rotor decay). If each of the braked wheels is loaded to 4000 lb, the wheel-track is 80 in., and f = 0.4 (rubber sliding on asphalt), the skid torque Q_S = 10,667 lb-ft, and therefore, a dangerous ground loop potential would exist. The minimum stopping time under such conditions would be 18.4 sec. A safe limit for the pilot-activated rotor brake mode would apply about 9800 lb-ft of torque to the main rotor mast (equivalent to a 20-sec stop).

2. Although the static breakaway friction for the disk/puck brake may be somewhat higher than that for dynamic conditions, the severe consequences of inadvertent rotor rotation during engine idle operation suggest the need for an additional safety margin. This can be provided by use of the lower value of friction coefficient in design calculations. Thus, a typical pair of engines might develop the equivalent main rotor torque of 12,000 lb-ft at 60% gas generator speeds. If the safe pilot-activated rotor stopping mode is limited to a main rotor torque of 9800 lb-ft in accordance with Item 1, a second brake mode with increased pressure (interlock-protected for engine start sequence only) would be indicated.

4-2.2.5.2 Design and Analysis

Two basic determinations are required for the calculation of safe brake performance: (1) limit energy rate per unit area to yield satisfactory wear life and preclude disk scuffing, and (2) disk heat sink capacity.

Surface energy rate varies with such lining and disk properties as thermal conductivity, diffusivity, convective cooling, and critical temperature. Solid steel disks in helicopter applications have been operated successfully at an energy dissipation rate E_D of 25 Btu/in.^{2}-min. The referenced area is the swept area under the puck. The energy to be dissipated is the kinetic energy of the rotor at time of brake application less any applicable rotor aerodynamic decay increment. Wear life for common brake puck materials is dependent upon surface temperature, pressure, and velocity. Existing rubber-asbestos lining materials have demonstrated wear rates of approximately 0.0004 in.^{3}/stop at pressures of 240 lb/in.^{2} for mean rubbing velocities of 6000 fpm over 20-sec stop periods.

The second determination involves the heat sink capacity of the disk. For stops on the order of 20 to 30 sec, steel disk thicknesses in excess of 0.5 in. offer little help in reducing peak surface temperatures at the end of a stop due to the limited thermal conductivity of steel. Current systems operate well with values of energy/pound- of-disk near 100,000 ft-lb/lb with peak disk rim temperatures of about 500°-600 °F.

Other materials such as beryllium and carbon graphite recently have been employed with relative success for brake applications. The greatest improvement seems to be available with a configuration which uses a proprietary low modulus structural graphite composition for both the disc and puck lining material. A beryllium heat shield is used between the graphite lining and the hydraulic slave cylinders. The saddle or caliper assembly is fabricated from aluminum. Such brakes have been successfully constructed and tested with disc thicknesses to 1/5 in., puck diameters of 5 in., and disc diameters of 18 in. Considerable increase in energy storage, allowable operating temperature, and wear life has been demonstrated. Safe disc temperatures of 3000°F (incandescent white light) and a thermal capacity of 300,000 ft-lb/lb of graphite are characteristic of the design. Approximately 20% of the total stopping energy is dissipated in ablative wear. The puck material also serves as a heat sink and may be considered in the thermal design capacity.

The advantages due to these characteristics seem to indicate that weight savings on the order of 50% and wear life increases of 800% relative to conventional steel/ rubber-asbestos systems are obtainable. These factors are probably sufficient to offset the initial high cost to the extent that a life cycle cost reduction can be achieved.

Disadvantages lie in initial costs and structural limitations of the graphite material. Although ballistic impact characteristics are satisfactory and handling damage susceptibility is relatively low, graphite cannot compare with steel. Through bolt or spline attachments cannot be used for the graphite disc — a high pressure squeeze plate or friction drive attachment is required. Similarly, the disc cannot be drilled locally to achieve dynamic balance requirements. However, a steel reinforcing ring may be used for this purpose. The structural graphite material costs about $400/lb as of this writing (1975) — future costs may be significantly less with adequate production volume.

4-2.3 STATIC COMPONENTS

Static or nonrotating components of the transmission and drive system include the gearbox housings, liners, quills, mounts, studs, and dowels that serve to enclose and support the dynamic components. This paragraph addresses only the most significant components; i.e., cases and housings, and quills.

4-2.3.1 Cases and Housings

Helicopter gearbox cases and housings are fabricated almost exclusively from lightweight aluminum alloys and castings and forgings or from magnesium sand castings. These materials exhibit excellent thermal conductivity and ultimate strength-to-weight ratios, are readily machinable, and in many instances may be salvaged by welding and stress relieving with little resultant loss of strength properties. Although the general approach to casting and forging design is well covered in available literature, certain aspects peculiar to Army helicopter applications are summarized in the paragraphs that follow.

4-2.3.1.1 Design and Analysis

Housings and cases may be classified as primary structural load paths (rotor mast support or control system reaction member) or simply as gear housings for which externally applied loads are not significant. This distinction is fundamental in the selection of the design and analysis methods employed. Criticality classifications of castings and forgings are defined in MIL-C-6021 and are interpreted in AMCP 706-203.

In most instances specified crash load factors and limit maneuver loads will require ultimate and yield strength levels in primary structural cases and housings of such magnitude as to permit design definition by static analysis as opposed to fatigue analysis. Fatigue analysis will be used to define only the rotor control reaction portions of the cases and, occasionally, the gearbox support or mounting lugs when rotor vibratory loads or dynamic reaction loads from ground resonance or landing conditions are sufficient to cause concern for low-cycle fatigue.

Static and fatigue test requirements are outlined in AMCP 706-203, which describes basic design load requirements of MIL-S-8698. The critical design criterion generally is the satisfaction of the static test requirements. Because the integrity of a casting or forging is governed by the type of quality control established by applicable drawings and specifications, it is imperative that required static tests be performed on the least acceptable specimens. The radiographic acceptance standard ASTM E-155, as well as other inspection criteria, then may be based upon these static test results.

Recent Army helicopter RFP requirements have emphasized increased crew safety through more crashworthy design in accord with the recommendations of Ref. 100. Limit load conditions are based upon +3.5 and −0.5 maneuver load factors at the helicopter CG and ultimate load conditions upon normal load factors of +20/−10, and lateral or longitudinal load factors of ±20. Combination loading also must be considered as the simultaneous occurrence of loadings in accordance with any of the three conditions that follow:

	Condition I	II	III
Longitudinal	±20	±10	±10
Vertical	+10/−5	+20/−10	+10/−5
Lateral	±10	±10	±20

Where a structural support case is of relatively simple configuration, forgings are preferred to castings because of the superior strength-to-weight ratio and the inherently lower variability in strength of the former.

The four design deficiencies found most frequently in current Army helicopter housing components are:

1. Insufficient attention to corrosion protection. Success in attainment of required component life is highly dependent upon avoiding corrosion. Housing designs must avoid traps for water from rain, washdown, or condensation. Open upward-facing traps, such as stud or bolt counterbores or seal enclosure cavities, must be provided with effective drains. External size of case joints should be mismatched, with the upper member being larger so as to eliminate standing water. Such joints may be sealed externally with relative ease by nonnardening, fillet-forming compounds. Sharp edges and rough surfaces must be eliminated by chamfering, polishing, or tumble (slurry) de-burring to avoid inadequate resin or paint coverage due to surface tension effects. Cathodic particles must be removed completely from casting surfaces by thorough cleaning prior to resin impregnation.

Galvanic corrosion protection in the form of resin, acrylic, or zinc chromate application should be used between all metals except those immediately adjacent in the activity series (MIL-STD-889). Excessive steady tensile stresses due to assembly clamping (such as use of bolted clevis lugs without spacers) should be avoided to reduce the susceptibility to stress corrosion.

2. Lack of attention to differential thermal expansion. Steel bearing clamp nuts and similar devices installed in magnesium or aluminum threaded bores often lose their entire axial clamping force at operating temperatures. Such applications either must have an initial deflection that is greater than the amount of thermal relaxation or else threaded steel liners must be inserted in the case bores. Static bearing and hoop stresses should be checked throughout the possible ambient temperature range (normally −65° to +300°F) when steel and light alloy cases are joined with piloted flanges. Thermally fit steel liners in alloy cases should have a nominal 300°F interference and the bore of the light alloy ring section surrounding the liner also should show a positive margin on limit stress at −65°F. Where steel bearings are installed in the liner, their line-to-line fit-up temperature and outer ring cross section must be considered in calculating limit stress. The hoop tension in the housing bore fiber at −65°F assuming the bearing fit is line-to-line at a temperature of +72°F may be taken as:

$$S_h = \frac{(\alpha_2 - \alpha_1)\,(c^2 + d^2)}{t\,(d^2 - c^2)} \times \left[\frac{365}{\dfrac{1}{E_1}\left\{\dfrac{\left(\dfrac{c}{b}\right)^2 + 1}{\left(\dfrac{c}{b}\right)^2 - 1} - \mu_1\right\}} + \frac{502}{\dfrac{1}{E_2}\left\{\dfrac{\left(\dfrac{d}{c}\right)^2 + 1}{\left(\dfrac{d}{c}\right)^2 - 1} + \mu_2\right\}} + \frac{137}{\dfrac{1}{E_1}\left\{\dfrac{\left(\dfrac{c}{a}\right)^2 + 1}{\left(\dfrac{c}{a}\right)^2 - 1} - \mu_1\right\}} \right] \quad (4\text{-}61)$$

where

a = bearing outer ring bore, in.

b = bearing OD, in.

c = liner OD (steel), in.

d = light alloy section OD, in.

E = Young's modulus, psi

α = linear coefficient of thermal expansion, in./in.-°F

μ = Poisson's ratio, dimensionless

Subscript 1 = steel properties

Subscript 2 = light alloy properties

3. Improper attention to joint and fastener requirements. Sufficient flange thickness must be provided to distribute loads uniformly among the preloaded tension fasteners (bolts or studs) used on case flange joints. Fastener preload must be sufficient to maintain tension at −65°F, preclude stress reversals during normal oscillatory loading, and maintain flange contact under tension loading. For properly designed flanges with compatible fastener spacing, a conservative value for fastener tension loading P_t for moment-loaded cylindrical joints is given by

$$P_t = \frac{8M}{3D_{BC}N}, \text{ lb} \qquad (4\text{-}62)$$

where

M = moment, in.-lb

D_{BC} = bolt circle diameter, in.

N = number of bolts or studs (equal spacing assumed)

The yield and ultimate strengths given in MIL-HDBK-5 for standard AN studs represent the minimum values obtained with proper installation. However, manufacturing characteristics often have a more deleterious effect upon the installed strengths of smaller AN studs than upon larger sizes. Even with maximum practicable perpendicularity for tapped holes, the combined effects of taper and squareness of joints may induce bending loads such that 3/8-in. series studs demonstrate tensile failure at 90% of handbook minimum values. The significance of this is probably better demonstrated by the fact that typical 7/16-in. diameter studs usually demonstrate double the installed tensile strength of comparable 3/8-in. diameter studs with approximate cost and weight penalties of 10% and 25%, respectively.

Blind tapped holes for stud installation must be vented.

When torque must be transferred through a flange joint, a torque capacity Q_f in the absence of external tension loads is given by

$$Q_f = 1.25\frac{Q_s N D_{BC}}{D_{PD}}, \text{ lb-in.} \qquad (4\text{-}63)$$

where

Q_s = stud torque, lb-in.

N = number of studs

D_{BC} = bolt circle diameter, in.

D_{PD} = stud pitch diameter, in.

Beyond this value, dowels, keys, or other mechanical locking devices should be used to prevent bending and shear loading of the studs. Although it is common to consider stud shear strength in determining ultimate joint strength, this factor should not be relied upon for normal design torque analysis.

The joint configuration theoretically necessary to meet the strength requirements having been determined, it becomes imperative to assure compatible detail design of the machined surfaces for the fasteners. Fracture resistance as well as fatigue considerations require careful attention to seemingly less significant detail. Generous fillet radii must be provided in spotfaces, counterbores, and keyways. All sharp corners must be chamfered, and feather edges must be removed to minimize stress concentration. Smooth blending of intersections on critical machined surfaces is also necessary to minimize stress concentration. Additionally the counterbore or spotface size must be adequate to provide wrench clearance for normal maintenance operations.

4. Failure to consider deflection and loading in contiguous structure. Many record instances of gearbox mounting lug failure are attributable to externally induced loads that were ignored in the design analysis. The cast gearbox structure, which often is stiffer than the airframe structure to which it is mounted, may provide a load path for bending and for torsion reactions present in the airframe structure due to landing gear, rotor thrust, or vibratory responses. Examples are the attachment of a four lug accessory gearbox to a deck structure that has vertical bending modes, or the similar mounting of an intermediate gearbox on tail boom structure that undergoes torsional deflections with large tail rotor pitch inputs. Because it is desirable to obtain the safety inherent with mounting redundancy, it is usually a good practice either to provide greatly stiffened airframe structure locally or to introduce additional compliance at one or more gearbox attachment points by use of elastomeric members. Beefing up the gearbox often merely modifies the failure mode.

4-2.3.1.2 Materials and Processes

Magnesium has fallen into disfavor compared to aluminum for cases and housings because shortening of service life due to corrosion has become a significant maintenance and spares replacement expense to the Army. Because magnesium is anodic with respect to all other metals, failure to employ adequate design, process, and preventive maintenance measures has caused an overemphasis of the deficiencies of the material. Bare magnesium actually is affected less by exposure to marine atmosphere than is unprotected mild steel (Ref. 101). However, corrosion of magnesium alloys can be avoided successfully only if the designer and fabricater follow the complete sequence of

1. Design
2. Cleaning
3. Chromating or anodic film application
4. Surface sealing or impregnation
5. Painting
6. Assembly
7. Routine preventive maintenance.

The most frequently occurring inadequacies in recent Army experience involve design and maintenance.

Aluminum alloys should be used in areas of high susceptibility to corrosion. Ref. 46 reports the replacement rate for AZ91 magnesium main transmission case at UH-1 overhaul as:

Top case — 16.0%
Main case — 1.7%
Support case — 2.3%
Sump case — 2.9%
Quills — 1.0%

In the case of replacements of the top case, 1/3 were attributed to improper protection of bare surfaces during shipment after removal of the main rotor mast, and 2/3 to in-service corrosion. The relative replacement rates suggest that the environment in which only the top case operates is particularly conducive to corrosion. Aluminum alloys with high silicon content (6 to 12%) have been found to be superior to other aluminum alloys and magnesium alloys with respect to wear resistance. Properly designed splines of these materials will exhibit negligible wear when operating with floating steel mating splines. Magnesium and aluminum alloys commonly used in helicopter housings and cases are listed in Table 4-6.

The specification of co-cast test bars in designated areas can insure the highest allowable strength properties. The use of MIL-A-21180 control specifications rather than those of QQ-A-601 generally will insure 25% higher allowable fatigue strength, although the cost may be 20 to 50% greater. The potential weight savings may be as high as 40% if sta. strength defines the design and provided that minimum wall section restrictions are not imposed. The tensile properties of many aluminum forging alloys may be improved by cold working or mechanical stress relief (Ref. 102).

Procurement and process specifications for castings and forgings are defined in MIL-C-6021. The detail design drawing must require the following processing and nondestructive test (NDT) inspection procedures as a minimum:

1. Impregnation. Thermosetting polyester resins per MIL-STD-276 are recommended for casting impregnation. Vacuum processing is essential to remove gas bubbles from casting pores and to permit good resin permeation. Leak checks may be aerostatic or hydrostatic, although the former is preferred for sensitivity and cleanliness.

2. Radiographic inspection. Radiographic inspection is required in accordance with MIL-STD-452. The detail drawing must call out x-ray views and should include a stress diagram to assist in determination of techniques and interpretation to be employed. The x-ray technique should be able to resolve 2% of the thickness being examined. Film interpretation is based upon discontinuity gradations as defined in ASTM E-155.

3. Surface crack inspection. This must be accomplished by fluorescent penetrant techniques as defined in MIL-I-6866 and MIL-I-25135. Inspection must be performed after forging, final heat treating or aging, cold working, stress relieving, grinding, welding, and machining, but before polishing, tumbling, shot peening, plating, resin impregnation, or painting.

4. Hardness inspection. All castings and forging should be checked for hardness by the standard 500 kg Brinell (or equivalent) method to ascertain that full heat treating and/or solution aging has been accomplished. This inspection must be performed prior to shot peening, plating, or painting.

4-2.3.2 Quills

External and internal quills frequently are used to house a gear/bearing subassembly to facilitate modular maintenance techniques and reduce the complexity of the primary gearbox housing. Problems typically encountered in helicopter applications are associated with excessive wear or high temperature creep of the housing bore that accepts the quill. The primary cause of wear is the use of material combinations that permit differential thermal expansion with looseness becoming excessive at operating temperatures. Combinations that tighten at elevated temperatures alleviate this problem while retaining ease of assembly at room temperatures. However, the thermal stress effects at −65°F also must be considered (par. 4-2.3.1.1) to assure that the material yield strength is not exceeded. Because steel liners frequently are used in light alloy quills, the solution must be extended to consider the effects of four concentric rings with respect to their individual tolerancing, material strengths, fit-up, and thermal expansion coefficients. Sealant compounds always should be used at external quill-housing joints to prevent water entrapment.

TABLE 4-6. HELICOPTER TRANSMISSION CASE MATERIALS AND APPLICATION DATA

MATERIAL DESIGNATION		PROPERTIES, APPLICATION, AND RESTRICTIONS
ALUMINUM ALLOYS	A356 CAST	MOST FREQUENTLY USED AL CASTING, EXCELLENT CASTABILITY, BEST CORROSION RESISTANCE, HIGHEST DUCTILITY. LOSES STRENGTH ABOVE 250° F, GOOD WEAR PROPERTIES, BEST CASTING FATIGUE STRENGTH
	A357 CAST	SAME AS 356 BUT + 11% S_{tu}, + 17% S_{ty}
	249 CAST	BEST S_{tu} AND S_{ty} ABOVE 350° F. POOR WEAR. S_{tu} 20% ABOVE 357. CORROSION AND FATIGUE PROPERTIES POORER THAN 357
	224 CAST (AMS 4226) 5083 FORGED	GOOD BALLISTIC PROPERTIES. RELATIVE TO A 357 S_{tu} UP 10%, S_{ty} DOWN 7%. FATIGUE STRENGTH LOWER, WEAR AND CORROSION SAME AS 249
	XA201.0 CAST (AMS 4229, KO-1)	BEST S_{tu} AND S_{ty} BELOW 350° F. CASTABILITY, WEAR, AND FATIGUE ALL POORER THAN A357
	2014 FORGED	MOST FREQUENTLY USED. FATIGUE, S_{tu}, S_{ty}, AND FORGEABILITY ALL GOOD. CORROSION GOOD, WEAR POOR
	4032 FORGED	BEST WEAR PROPERTIES, FATIGUE LESS THAN 2014
MAGNESIUM ALLOYS	AZ91 CAST	MOST FREQUENTLY USED. GOOD CASTABILITY. LOSES STRENGTH ABOVE 250° F
	AZ92 CAST	S_{ty} HIGHER THAN AZ91, OTHERWISE SIMILAR
	ZE41A CAST	EXCELLENT CASTABILITY. AVERAGE S_{tu} AND S_{ty} HIGHER THAN AZ92. EXCELLENT STRENGTH AT HIGH TEMPERATURES. RADIOGRAPHIC INSPECTION DIFFICULT
	QE22A CAST	BEST HIGH TEMPERATURE PROPERTIES RELATIVE TO AZ91. S_{tu} UP 9%m, S_{ty} UP 62%. EXCELLENT CASTABILITY. RADIOGRAPHIC INSPECTION DIFFICULT

S_{tu} = ULTIMATE TENSILE STRESS, psi

S_{ty} - YIELD TENSILE STRESS, psi

4-2.4 SPECIAL CONSIDERATIONS

Although many special disciplines affect detail design success, recent experience indicates that two of particular importance are vibration control and diagnostics.

4-2.4.1 Vibration Control

Drive system torsional dynamics are treated in par. 5-5, AMCP 706-201, while additional considerations of engine governing, overlap, and damping are detailed in par. 8-7. This paragraph addresses problems associated with component resonant vibration, particularly as it affects gear members. Recent Army helicopter RFP specifications have stated that gear resonant frequencies *shall* be a minimum of 30% away from the design continuous operating speed. While this may be impossible to achieve with some gears when all vibration modes are considered, the intent may be satisfied for the potentially dangerous modes. As an alternative, sufficient damping may be employed to render such resonant frequencies harmless; i.e., the vibratory stresses will be safely below the endurance limits for the structure.

As increasing design speeds for drives have resulted in pitch line velocities above 10,000 fpm, many fatigue failures occurred that initially were attributed to dynamic tooth loading. However, investigation revealed that the fatigue nucleations usually were located in the bottoms of the tooth roots or on the insides of the back-up rim. The crack propagation generally was radial rather than across the tooth base, resulting in the loss of a large segment of the gear rather than a single tooth. Such failures are typical of resonant conditions in which the tooth meshing frequency or one of its harmonics coincides with a particular natural vibration mode of the gear.

Lightweight gear designs for helicopter use often will exhibit various types of vibratory modes, such as with radial nodes or circular nodes, singly and in combinations. Typical vibration modes for a thin web spur gear with integral shaft are given in Ref. 103. Generally, the mode of concern is that involving radial nodes that put the gear rim into axial waveform vibration. The lower orders (say, up to the fifth diametral mode) are more likely to involve higher amplitudes and, hence, higher oscillatory bending stresses. However, relative resonant response amplitudes for constant forcing input intensity at various discrete resonant frequencies vary enormously with gear blank configurations. One gear blank may respond most to a third diametral mode frequency while another to the fifth. The flange, web, and hub design, all influence this relationship as well as the ratio of higher order resonant frequencies to the fundamental. The resonant frequencies are best determined by experimental bench test techniques using only the gear in question. Excitation can be by mechanical shaker, acoustical siren, or electromagnetically by an induction coil mounted very near the rim surface. Excellent visual determination of resonant response may be accomplished with sand pattern techniques if the gear web is of suitable configuration. In other instances, nodal and antinodal zones may be clearly detected by manual probing of the rim and web surfaces with a lightly hand held soft steel rod. Audible detection is also sufficiently precise for resonant frequency identification, although a microphone feedback coupled with a radially opposed input from the exciter into an oscilloscope is required to produce Lissajou patterns necessary to distinguish between fundamental and overtone responses. Magnetic sensors may be used for the feedback phasing signal with equal ease.

When resonance vibration amplitudes are sufficient to produce significant stress levels, the most practical quantitative evaluation can be made by attaching strain gages to appropriate antinode regions of the rim and web. The higher energy level resonance frequencies readily may be identified. In many instances, designs will exhibit very low vibration due to the favorable mass and stiffness configurations of the flange, web, and hub, and therefore will not produce a detectable strain gage output. Considerable input energy may be required for a realistic determination. Although actual operation in the transmission is the final arbiter the acoustical siren generally will produce sufficient input to get the job done.

Audible detection also is sufficiently precise, although a microphone feedback coupled with orthogonal axis input from the exciter into an oscilloscope is required to produce Lissajou patterns in order to distinguish between fundamental and overtone responses. The observed standing waves are the product of a forward and backward traveling (with respect to rotational velocity) waveset. If the gear is rotating at a given speed ω, two resonant frequencies are observed for each static fundamental radial vibration mode:

Forward wave natural frequency

$$f_f = f_o + n\omega/2, \text{ Hz} \tag{4-64}$$

Backward wave natural frequency

$$f_b = f_o - n\omega/2, \text{ Hz} \tag{4-65}$$

where

f_o = static resonant frequency, Hz
n = number of radial nodes
ω = rotational speed, Hz

A graphic presentation of the phenomenon is contained in Fig. 4-34. The fundamental radial node static resonant frequencies are designated on the ordinate by the number of their radial nodes. The abscissa is rotational speed n relative to normal operating speed n_{op}. The inclined line represents the gear tooth meshing frequency for a 41-tooth pinion driving at a normal speed of 20,000 rpm. Note that the forcing function represented by the pinion tooth mesh intercepts the forward traveling 8 node vibration near ground idle speed, the backward wave from the 10 node at about 83% speed, the forward 10 node wave at near normal operating speed, and the backward 12 node wave at overspeed. The 8, 10, and 12 node vibrations are all potentially hazardous.

Even if it were demonstrated that the cyclic stresses were below the material endurance limit, objectionable acoustical energy radiation would occur at these intercepts. Redesign of the shape and mass of the gear is not very practicable in this example because a 25% change in gear rim and web thicknesses affects a maximum change of 8% in resonant frequency for a specific design (Ref. 104).

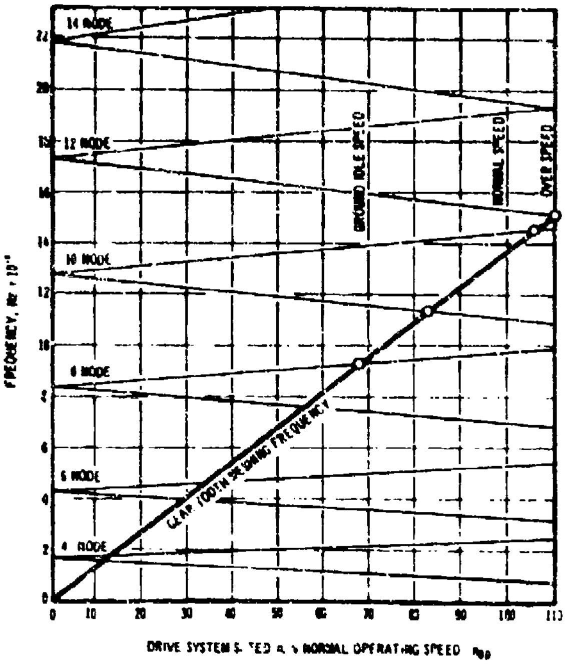

Figure 4-34. Radial Mode Resonance Frequency vs Gear Tooth Meshing Speed

The illustrated situation thus demands the introduction of sufficient damping to reduce the vibration significantly. The spiral damper ring or snap ring as shown in Fig. 4-35 has proven to be effective in many instances, producing damping ratios of 0.04 to 0.10 with a resultant reduction of cyclic resonant stress to 30-40% of the undamped magnitudes.

Although intended as noise reduction devices, other types of damping (viscoelastic) and torsional absorbers have been evaluated with partial success in helicopter transmissions. Certain viscoelastic damping treatments have been shown to reduce some gear vibration mode amplitudes by 50% (Ref. 105).

4-2.4.2 Diagnostics

Many cockpit indicators, warning lights, and gages fit the general definition of diagnostic aids. Recent Army RFP specifications include indications for oil pressure and temperature, low pressure warning, high temperature warning, oil quantity, and chip detection. Additional ground inspection techniques routinely include impending oil-filter bypass warning flags, oil leakage checks, visual and aural failure detection and location, and oil sampling for spectrographic analysis. However, these methods by themselves have proven inadequate to allow the widespread use of safe and cost-effective conditional maintenance on existing Army helicopter drive subsystems. To a large extent, the missing ingredient is thorough and comprehensive consideration of these techniques in the initial design and development test phases.

Safe and cost-effective implementation of conditional maintenance methods requires thorough definition and accurate use of early failure detection, diagnosis, and prognosis as defined in Ref. 106. Major efforts underway in Army-sponsored programs are aimed at reducing the time required for detection and at improving scheduling. This philosophy results in scheduling of unscheduled maintenance provided that experience-based judgments are available concerning the severity and rate of progression of the detected failure.

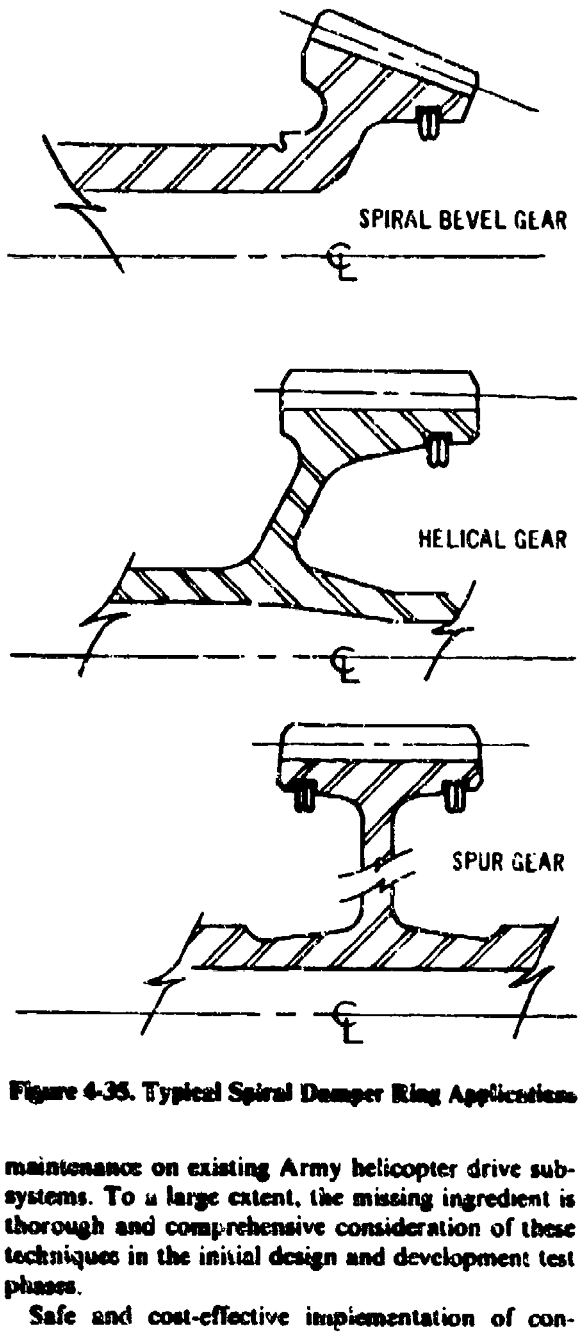

Figure 4-35. Typical Spiral Damper Ring Applications

Detection methods may be classified as related to the internal oil system or external to the oil system, and include:

1. Oil system dependent:
   a. Spectrographic oil analysis
   b. Electronic or electro-optical oil monitoring
   c. Oil filter differential pressure
   d. In-line electrical resistance filter grids
   e. Electric chip detectors
   f. Magnetic chip collector plugs.
2. Independent of oil system:
   a. Noise analysis
   b. Vibration monitoring
   c. Temperature measurement.

Spectrographic oil analysis has proven expensive and relatively unreliable because poor correlation exists between failure severity and detection. Little is actually known concerning failure, therefore, diagnosis and prognosis are impossible.

Oil monitoring remains in the evaluation stage; some techniques have proven totally useless while others show some promise.

Oil filter differential pressure can be correlated quite well with failure progression rate of bearings when the rate of change of differential pressure is examined. Usefulness of this system is compromised, however, by the accumulation of normal wear, dirt, and contamination particles. Visual inspection of the filter element debris by experienced technicians can determine the generic failure mode but not the location.

Filter grids are best employed in a fashion similar to use of oil monitoring. The basic difference is that the filter grid passes smaller contaminant particles, hopefully retaining only the larger metal failure flakes. The percentage of grid clogging is detected and may be converted electrically to a rate-of-change display.

Chip detections have been used extensively in Army helicopters for two decades. The debris particles are captured magnetically to bridge electrical contact points, which, in turn, energize a caution light. The usefulness of this method is dependent upon the location of the detector and the resulting crew action. Failure indication often results from the slow accumulation of normal wear debris, and all too often the crew prognosis results in a mission abort to investigate the cause.

Use of chip collector plugs emphasizes an attempt at better diagnosis in that generally a large number of plugs with greater debris storage capacity and no electrical caution panel light are used to aid in localization of a failure to a specific area or module. Inspection is accomplished at daily or periodic intervals and sufficient debris collection is usually present to permit an experienced technician to identify the failure mode. Good success has been claimed for such a system on commercial air carrier fleets, but success in helicopter adaptation is completely dependent upon well-designed installations that entrap and localize failure debris, convenient and accessible detector locations, and experienced technicians to correlate findings with other available diagnostics and to schedule maintenance accurately.

Vibration monitoring and noise analysis differ only in the sensing techniques. Accelerometers or other forms of contacting vibration sensors measure vibration, while microphone sensors measure noise. Substantial research and development efforts are being made on these systems. Rudimentary go-no-go systems with a warning threshold signal requiring engine shutdown have long been used in turbine engine installations. The real challenge lies in the electronic signal processing and its conversion into an identification and quantitative assessment of the failure mode. Signal analysis is being investigated by such techniques as auto-correlations, Fourier transform analysis, power spectral density examination, cross power spectral density, amplitude probability distribution, and real time classification of waveform by convolution of other techniques (Ref. 107). The operational success of any such system is dependent upon its ability to isolate the faulty component signature from the background created by other internal and external forced and resonant vibrations, side bands, and beat frequencies, at an earlier time than the chip detector, and to remain online to monitor the rate of progression of failure of the particular component. Only in this manner can the remaining useful safe life be predicted, and maintenance scheduled wisely. The ultimate goal of these programs is automatic identification and prognosis as well as detection. This goal requires comprehensive test data on all failure modes, both individually and in combination, along with improved sensor reliability and an analysis system having a minimum reliability an order of magnitude greater than that of the drive system being monitored.

Temperature measurement systems, limited at present to relatively integrated bulk temperature measurement, seldom can detect other than very advanced (virtually emergency condition) failures. Capability improvement requires a multitude of individual component sensors combined with rate-of-change analysis corrected for base line variation in bulk environment temperature.

Rapid improvements in diagnostics for helicopter drive subsystems are dependent largely upon:

1. Improved initial design:
   a. Failure-forgiving design techniques as defined in par. 4-1.2.1.2
   b. Integration of necessary diagnostic aids and sensor provisions into the original design.
2. Redirected development testing effort
   a. Seeking out of failure modes as discussed in par. 4-1.4.4
   b. Compilation of progression rate data and correlation with detection level.
3. Implementation of procedures for effective prognosis
   a. Determination of safe residual operating times for failed components, based on overpower operation to provide additional safety margins
   b. Determination of impact of secondary failure modes from excessive continued operation.

4-3 DRIVE SHAFTING AND INTERCONNECT SYSTEMS

In a helicopter transmission and drive subsystem, the drive-shafting is comprised of all the shaft components necessary for the transmission of power from the engine(s) to a gearbox and from a gearbox to a rotor or propeller. This includes drive shaft tubes, couplings, bearings, and bearing supports (hangers).

4-3.1 GENERAL REQUIREMENTS

Specific drive shaft requirements are determined by the particular helicopter design, including the engine(s) selected. Shaft size, span, coupling, hanger design, and other configuration details are defined primarily by the power requirements of the specific location, but are influenced by critical speeds, deflections to be encountered, vulnerability, maintainability, logistics, manufacturing cost, and weight.

General requirements for reliability, maintainability, and survivability are discussed in par. 4-1.2.1. In addition to these all-inclusive requirements there exist general characteristics and features peculiar to the shaft function, location, and operating rotational speed. The requirements applicable to specific shaft locations are discussed in the paragraphs that follow.

4-3.1.1 Engine-to-Transmission

Requirements peculiar to engine-to-transmission shafting depend upon relative locations of the engine and transmission and airframe mounting systems, engine PTO (power-takeoff) speed and configuration, and PTO loading limitations.

Airframe mounting systems generally determine the misalignments and dynamic axial motion that the shafting must accommodate. These mounting systems may be grouped conveniently into three categories:

1. Engine(s) mounted directly to main gearbox
2. Engine mounted to airframe with tubular struts, using rod-end bearings to relieve thermal and load deflections; with the gearbox bolted directly to the airframe
3. Engine mounted as in Item 2; gearbox mounted flexibly. The rotor pylon usually is integral with the gearbox, that is, airframe mounted with elastomeric springs and hinged links or struts.

System 1 generally uses an internal splined quill shaft lubricated with gearbox oil. The splines may be hardened and ground, slightly crowned or straight; or medium hard and hobbed, shaped, or roll-formed. Wear and fretting corrosion may be reduced by shot peening and/or soft plating or coatings such as silver, nickel, nylon, Teflon, or molybdenum disulfide dispersed in a bonding vehicle.

Installation requirements for System 2 can be met with external shafting and couplings with relatively low misalignment capability — on the order of 1 deg or less to accommodate airframe structural deflections and mounting tolerances. There seldom is need to accommodate dynamic axial deflection but variations in static length due to installation tolerances and thermal growth of the engine must be considered in the shaft design. Care must be taken to assure that the airframe structure provides support at the engine and gearbox attach points adequate to minimize deflection. It often is possible to comply with ultimate and crash load strength requirements, yet encounter relatively high deflections under normal flight load conditions. Special attention should be given to torque reactions and to airframe modes.

Angular deflections of 1 deg or less easily are accommodated by flexible disk (Thomas type), flexible diaphragm (Bendix type), elastomeric, tension element (Bossler type), or crowned tooth gear couplings. The first three types generally are preferred because they do not require grease or oil lubrication. (See coupling discussion, par. 4-3.2.1).

System 3 conditions offer the greatest challenge in coupling design characteristics. The misalignment requirements vary with pylon mount spring rates and the geometric arrangement of mounting links as well as with the proximity of the engine PTO to the gearbox input. The greater the distance between the two, the less is the angular misalignment for a given suspension system. For a pylon system with fixed planar focusing, or roll center, reduction of the distance from this center to the gearbox input location also reduces both misalignment and axial motion.

The amount of axial motion that must be accommodated usually will determine the type of drive shaft couplings used. All known drive shaft systems offer a resistance or damping force opposite to the direction of axial motion while transmitting torque. The maximum acceptable value for such forces usually is established by either the rotor vibration isolation system or the engine PTO design specification limits. Damping force characteristics for various couplings and spline combinations are discussed in par. 4-3.2.1.

Other input driveshaft design criteria are governed by maintainability, vulnerability, and reliability requirements and by additional engine PTO design and specification limits.

Maintainability considerations require that the engine-to-gearbox shaft contain "quick-disconnect" features. Ease of accessibility also is required to facilitate drive shaft inspection and servicing, and engine or gearbox replacement. Since these tasks must be performed at the direct support level, the absolute minimum of special tools, fixtures, and skills should be required.

Although vulnerability and reliability have been discussed previously, it is important to consider the consequence of drive shaft failure. The large kinetic energy of the input drive shaft categorizes it as a potentially hazardous or lethal object should it separate at either or both of the engines and gearbox adapters. The use of antiflail devices, i.e., secondary components or structure capable of capturing a failed drive shaft, is highly desirable.

In addition to limits on axial force specification at the engine PTO pad, allowable moments and steady and oscillatory radial loads are usually specified. Since engine-to-gearbox shaft rotation speeds are in the range 6000-20,000 rpm, compliance with the oscillatory load limits generally require kinematic and dynamic balancing of the individual elements of the drive shaft assembly. When positioning or locating tolerances between mating surfaces of elements such as couplings, adapters, and shafts cannot be controlled adequately, it may become necessary to balance the complete assembly to eliminate excessive vibration.

4-3.1.2 Interconnect Shafting

An interconnect shaft system for multiple main (or lifting) rotor helicopters transmits power between the engine gearbox (or the collector gearbox in multi-engine helicopters) and the main rotor gearbox(es) while also maintaining phase relationship between rotors. The primary considerations for such an interconnect shaft system are reliability and survivability, as functional failure of the shafting becomes a catastrophic failure almost immediately, with a collision of the intermeshing rotor blades. Achievement of the necessary level of reliability requires detailed consideration of operating stresses and margins of safety, critical speed margins, number and type of dynamic components (such as bearings, hangers, dampers, couplings, and splines), redundancy in mounting and support structure, and ease of inspection. Criticality of the interconnect system allows little latitude for reliability trade-offs and compromise with weight, cost, and maintainability goals. Optimization of design, then, must be in the direction of minimum number of parts, low stress (high margin of safety), and high tolerance to ballistic damage. Therefore, the drive shaft tubes will be relatively large diameter, thin wall, and long (within a safe buckling) length/diameter (L/D) ratio and critical speed limit. Intermediate bearing hanger design must permit relubrication, with ready access to the whole hanger for visual inspection. The selection of drive shaft tube material can necessitate further considerations of axial motion due to differential expansion between the airframe (generally aluminum) and the driveshaft (steel, aluminum, titanium, or composite). Airframe deflections due to flight maneuvers or load distribution also can contribute to the axial deflections of the drive shaft. These deflections will necessitate couplings capable of absorbing the anticipated motion. If axial deflections are small, then flexible disk couplings frequently are the choice; for larger axial deflections, the geared coupling or ball-spline disk combinations are better suited. Under any specific set of requirements, the primary design emphasis must be reliability and ready-access for service and inspection.

4-3.1.3 Tail Rotor or Propeller Shafting

The drive shaft system used on a single main rotor helicopter to power the tail or antitorque rotor, spans between the main gearbox and the tail rotor gearbox. This system must provide power to the tail rotor from the main rotor in the event of loss of drive from the engine(s). In normal operation, the engine(s) drive through a freewheeling clutch to the main gearbox. During autorotation, when the free-wheeling unit is overrunning, tail rotor power is extracted from the main rotor autorotational, or kinetic flywheel, inertia.

Tail rotor drive shafting will be subjected to severe transient loads and cyclic torsional oscillations as well as normal steady torque inputs. Torque requirements for most flight conditions are moderate in nature, with maximum steady torque required during hover at high gross weight. The total power required to hover is main rotor power plus tail rotor power required to offset the main rotor torque, plus losses. The tail rotor also must counteract the main rotor cyclic thrust vector and an aerodynamic drag couple from the tailboom. Conventional rotor or propeller theory, including an efficiency factor applicable to the specific tail rotor can be used to calculate the steady tail rotor torque. However, experience has shown that the transient torque requirements can be from 200-400% of the steady-state design torque. High levels of transient torque result from sideward flight in an adverse quartering wind, from yaw accelerations, and from unusual inflow conditions resulting from combinations of main rotor downwash, tail rotor blanking from aircraft structure, and adverse winds at hover or low flight speed.

Transient torque inputs also can be introduced to the tail rotor drive system by engine compressor stall, violent flight maneuvers, rapid throttle movements (chops), or abrupt engine power loss. Under such conditions the abrupt relief of the windup of the tail rotor drive-shaft combines with the flywheel inertia of the tail rotor and with secondary effects of main rotor inertia to cause several cycles of extremely high amplitude torque oscillations in the drive shafting. Although occurring infrequently, this low cycle-high stress phenomenon can cause fatigue damage to the tail rotor drive system unless the components of this system are designed for torsional loads well in excess of the normal steady power requirements. Transient design criteria for the tail rotor drive stipulated in MIL-T-5955 and AMCP 706-203 are 300% of the power required to hover at design gross weight and density altitude or 150% of the maximum power required in the most severe maneuver within the flight envelope, whichever is higher.

Such requirements are rather straightforward with respect to fatigue design of the gear teeth and the cantilever rotor shaft in that the need for infinite life criteria due to the high rate of cyclic accumulation (rotation speed) is evident. However, with respect to the remainder of the drive system, where start-stop cycles, throttle chops (T.R. inertia overruns), airborne engine restarts, and yaw control pedal excursions account for the bulk of the high stress cycles, a far lower frequency of accumulation exists. In such instances, past experience with the fitting of theoretical spectrum analysis to subsequent flight strain survey results is rather essential in efficient design work. When fatigue spectra are unknown, a reasonable approach has been to design static yield strength levels to a minimum of 3 times the transient fatigue stress used for the gear teeth and rotor shaft infinite life critria. The throttle chop transient response is often the greatest oscillatory torque felt in the T.R. drive system. The system can be modeled for the computer using the engine-main rotor decay curves, the appropriate lumped mass and spring rate analogues, and the coupling discontinuities, with reasonable accuracy. A preproduction flight strain survey will provide sufficient information on the torsional characteristics of the tail rotor drivesystem to enable substantiation of the integrity or revelation of the unanticipated weak points.

4-3.1.4 Subcritical Shafting

Analytical methods for determining critical speeds of a drive shaft are covered in Chapter 7, AMCP 706-201. As defined there, the critical speed is that rotational speed at which the elastic forces are overcome by the unbalanced centrifugal forces and the "bow" of the shaft increases divergently. Theoretically, the critical speed of a perfect shaft, i.e., a shaft that is perfectly balanced, homogeneous, and equally displaced about the rotating axis, will occur as predicted by [illegible] analysis. The behavior of such a shaft is shown [illegible] dashed line in Fig 4-36. No vibration occurs [illegible] the rotational speed η approaches the critical speed η_{cr} where divergence occurs almost without warning.

Practically any real shaft has some initial unbalance that provides a centrifugal driving force which increases with increasing rotational speed η. Such a shaft exhibits vibration/rotation characteristics such as are shown by the solid line in Fig. 4-36. While vibration levels at normal operating speeds

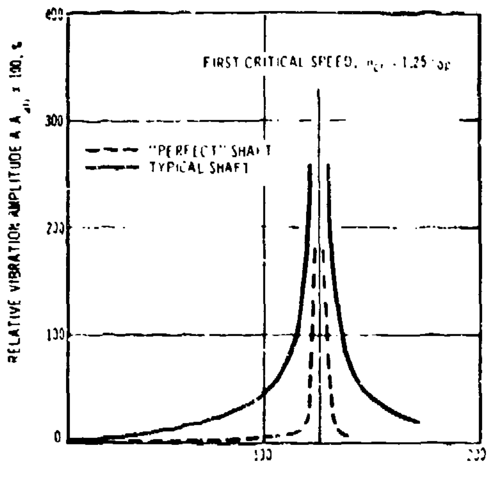

Figure 4-36. Relative Shaft Speed vs Relative Vibration Amplitude

may be acceptable, the unbalanced forces can increase rapidly as η increases above η_{op} with the possibility of resultant damage. This situation can effectively reduce the critical speed margin to an unacceptably low level. The inference is that simply by balancing the drive shaft an acceptable critical speed margin easily can be realized. However, the cost of dynamically balancing the shaft and/or shaft assembly must be included in the trade-off, together with a careful assessment of the contributions of the end conditions and/or mounting compliance to the vibration/rotation characteristic.

The majority of existing drive system applications use subcritical shafting, for which the lowest value of $\eta_{cr} > \eta_{op}$. Requirements for balancing can be met with ordinary balancing techniques and equipment; relatively short shafts minimize production and logistic problems; and ballistic tolerance design parameters are known. On the other hand the cost of a subcritical shaft installation with several separate spans may be higher than that of a comparable supercritical installation. The manufacturing cost for the short shafts may not be much different than the cost of a single long shaft, while the number, and hence cost, of machined parts probably will be higher for the subcritical installation.

A single span of the subcritical system consists of a drive shaft tube with end fittings, drive adapter, hanger assembly with bearing, splined adapter, and coupling. A typical example is shown in Fig. 4-37. Design of the drive shaft requires a determination of the shaft cross section necessary to accept safely the steady and transient loads stipulated in the pertinent design specification, and of a shaft length that will operate safely within the critical speed limitations. An efficient design generally consists of the least number of spans with acceptable critical speed margins and torsional buckling strength. Large diameter thin-walled tubes, generally of nonferrous metals; a grease lubricated bearing sealed on one side

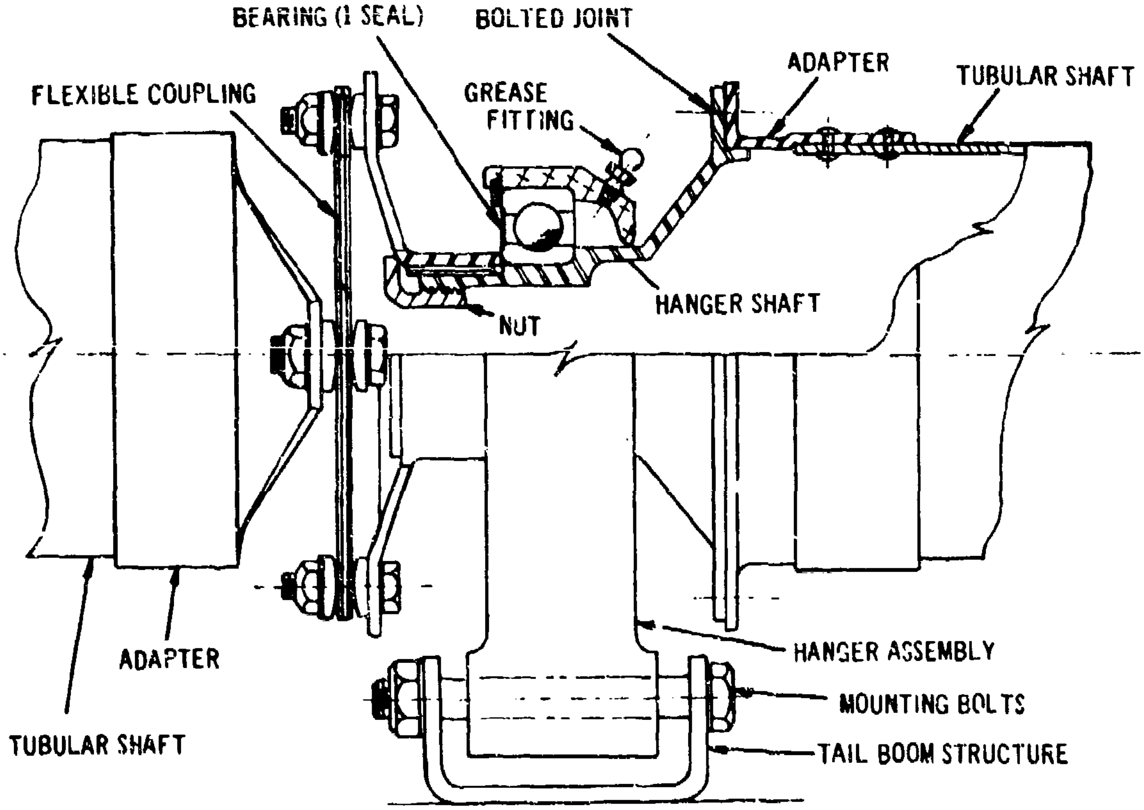

Figure 4-37. Typical Bearing Hanger Assembly — Subcritical Shaft Assembly

with a fitting for a periodic relubrication; and a flexible disk coupling are typical of current design practice. The end fittings and their attachment represent a considerable portion of the cost of the manufacturing of such drive shafts. The fittings may be attached by adhesive bonding, riveting, bolting, electron beam welding, or brazing. Tolerance of mating parts must be closely maintained to ensure good parallelism of end fittings and low vibration characteristics.

The type of couplings selected, their mass, location, and friction characteristics, influence critical whirling modes as well as torsional response modes of the shafting. A recent investigation of coupling induced whirl phenomena on turboshaft powered helicopters is given in Ref. 108.

4-3.1.5 Supercritical Shafting

Supercritical shafting usually operates at a speed between the first and second critical speed of rotation, although even high orders are possible. The main rotor and tail rotor shafts, or masts, often pass through the first critical speed before reaching operating speed. However, the critical speed is relatively low, the dwell time is momentary, and aerodynamic damping forces are quite large. On the other hand, interconnect drive shafting and tail rotor drive shafting generally operate at relatively high speed with very little inherent damping.

The advantages of a supercritical shaft design are the smaller number of detail parts and bearing hanger assemblies. The disadvantages are the need for dampers, which for reliability should be redundant, and the physical length of the shafts, which may impact on logistics. Also, other than normal design factors may determine shaft diameter and wall thickness. Shaft sizes larger than those required by the power requirements may be necessary to maintain a *L/D* ratio sufficient to avoid critical torsional buckling, or to counter a specific ballistic threat. Once size has been determined, the design requirements for shafts in the supercritical speed range center primarily on damping and dynamic balancing. As shown in Ref. 109, the necessity for balancing to a very close tolerance over the entire span of the supercritical shaft is paramount for successful operation.

It is incorrect to assume that a supercritical shafting system will automatically weigh less than a subcritical system. Directly comparable designs for a given helicopter application have to be made and the total installed weights determined accurately and compared. The weight saving apparently achieved by eliminating the hangers necessary for the subcritical shafting will be partially or totally offset by the addition of a damper or dampers. The elimination of hangers and the attendant maintenance requirements also may be offset by the addition of maintenance requirements for the dampers.

Acceptable tolerance to ballistic strikes requires hardware testing under simulated service conditions. Parameters for ballistic-tolerant designs for supercritical shafting have not been defined and dependence on individual tests is almost complete.

A method for calculation of critical speeds and bending modes for high-speed shafting is well presented and explained in Ref. 110.

4-3.2 COMPONENT DESIGN

The basic drive shaft system components, couplings, bearings, and shafts are discussed separately in the paragraphs that follow.

4-3.2.1 Couplings

The primary purpose of the shaft coupling is to provide freedom for angular misalignment and axial motion between various shafting elements and the engines and gearboxes. This relieves stresses in the shafting, bearing, gearbox, and engine components induced by bending moments and axial forces. The relative motions between these components may be due to airframe structural deflections, thermal expansion, or pylon excursions required by rotor vibration isolation schemes. There are six major types of couplings that have been used in helicopters, and the selection of one among them for a given application depends greatly upon the required displacements and the loads that can be tolerated in their supporting elements. The six are:

1. Laminated flexible disk couplings (Thomas type). This type of coupling, shown in Fig. 4-37, is probably the simplest design for angular misalignments ≤1 deg. It has been used on the CH-47 synchronizing shaft and on the OH-58 tail rotor drive system. Each driving spider may have two or three attaching points (four or six equally spaced holes in the disk complement). The larger number is preferred from the viewpoints of vulnerability and survivability. The laminated disks are generally circular rings, although square and hexagonal shapes have been used. One problem that has been encountered is disk fretting at the bolt attachment.

This coupling features high torque capacity, lightweight, simplicity, and constant angular velocity. The torque capacity can be varied easily by the addition or deletion of laminates. However, increasing the number of laminates reduces the angular misalignment capability of the particular design. An additional relationship exists between the number of attachment points and the torque capacity, and misalignment capability. A four-point attachment (two-bolt shaft adapter) provides the maximum misalignment capability and also is the least expensive to manufacture. The flexible disk is capable of small axial deflections, and where predicted axial motions are low, this coupling serves well. No lubrication is required.

Thomas couplings often are used in series with a sliding involute spline to accommodate variations in initial shaft assembly length due to accumulation of manufacturing tolerances. However, when axial deflection occurs under operating torque, the slip resistance of the spline is so great that appreciable axial force will be applied to the disk laminates. For splines of this type the breakaway slip force F_{bs} rarely is less than:

$$F_{bs} = 0.4\, Q/D_p, \text{ lb} \qquad (4\text{-}66)$$

where

Q = torque, lb-in.

D_p = pitch diameter, in.

In some special cases where certain dry film lubrications are applied to the splines (Ref. 93) breakaway forces of half this value may be realized.

2. Flexible diaphragm (Bendix type) couplings. This coupling (see Fig. 4-38) has been used on the OH-6 helicopter. This type of coupling is generally capable of a maximum angular misalignment of about 1 deg per diaphragm pair, but very little axial deflection capability and therefore must be used in series with a sliding spline. Recirculating ball splines generally are preferred for minimum breakaway slip force F_{bs}. Also, the diaphragm stack is very sensitive to fatigue failure due to oscillatory axial deflections since the weakest member of the stack will often oil-can and provide nearly all of the axial deflection. Under such conditions it may be necessary to provide a self-aligning monoball (or similar type) bearing at the intended flexure center of the stack to transfer axial loads into the supporting adapter and to force the ball spline to move. Although theoretically it is possible to obtain very low breakaway slip forces with the ball spline, practical considerations with respect to minimum length of the ball track grooves, numbers of balls, and seal provisions usually limit these forces to a minimum of

$$F_{bs} = 0.15\, Q/D_p, \text{ lb} \qquad (4\text{-}67)$$

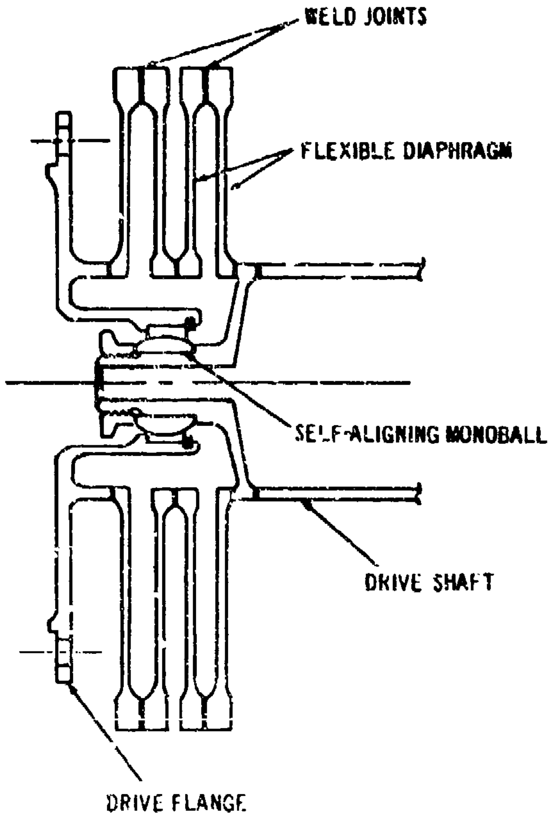

Figure 4-38. Flexible Diaphragm Coupling

All too often, the vibratory forces due to imbalance and rotor vibrations lead to false brinelling of the ball track grooves, which results in turn in much higher axial forces with increasing service time.

Although the flexible diaphragm elements need no lubrication, the splines and the monoball occasionally require lubrication.

3. Axially loaded straight element flexible coupling (Bossler coupling). This coupling (Fig. 4-39) requires no lubrication and has the ability to accommodate combined axial motion, misalignment, and torque. A series stack of warped rectangular plates with a centrally located rectangular cutout leaving slender sides characterizes the coupling. Opposed corners of these plates are bolted to adjacent elements and to end fittings or adapters. Design characteristics of this coupling are defined in Ref. 111.

Applications to date have been largely experimental with some flight time accumulated on the HH-2 helicopter (Ref. 112) and UH-1 helicopter. The angular misalignment capability appears to be about 0.5 deg per plate element. However, an increase in the number of elements used results in a reduction in first whirling critical speed. A satisfactory lightweight design for an engine-to-gearbox shaft for moderate angles (of the order 2.5 deg) and 0.25 in. oscillatory axial motion probably would be required to operate in the super-critical range if the engine output speed were above 6000 rpm.

4. Elastomeric couplings. Considerable development work culminating with experimental flight testing on helicopters such as the YH-51 have been accomplished with this type of coupling (Fig. 4-40). However, all successful applications have had low angular misalignment and axial deflection requirements. Efforts to develop higher capabilities (up to 2.5 deg steady misalignment and ±0.25 in. oscillatory axial displacement) have met with failure. The low angle configurations have used simple rubber elements in shear or compression, while for higher angles very thin, multiple layer, rubber-metal-rubber combinations, such as are now common in certain rotor system bearings have been used. The principal development problems have been elastomer fatigue due to reversed loading (alternating tension/compression) at high angle/low torque conditions.

The basic advantages of elastomeric couplings are high compliance (low shock and noise transmission), no lubrication required, no susceptibility to fretting corrosion, and potential savings of cost and maintenance. Inherent disadvantages are deterioration in an oily environment and aging, and reduced critical speed due to high compliance.

5. Hooke's joint. The Hooke's or Cardan type of universal joint coupling (Fig. 4-41) is capable of relatively high angular misalignment, of the order of 30 deg at moderate speeds and 15 deg at high speeds. However, unlike all other couplings discussed in this paragraph, the output is not a constant angular velocity, and significant bending moments are induced in the attaching adapters and supporting structures. Consequently, this type of coupling generally is employed as phase-matched pairs to cancel the oscillatory angular velocity or singularly with systems that are very soft torsionally and hence can absorb the angular velocity oscillation. The H-13 tail rotor drive system is an example of the latter type of application.

These couplings have no axial motion capability and normally are used in series with either a sliding (involute or square tooth) or a recirculating ball spline. The input and output yokes of the coupling normally are attached to the cross with cupped needle bearings. When these components are sized properly for a given torque and angular velocity oscillation, the breakaway sliding force of the adjacent spline usually is well within the axial load capacity of the coupling. Common failure modes are spalling of the cup and needle bearing, and fatigue fracture of the yokes. The needle bearings require lubrication.

6. Gear couplings. Gear couplings (Fig. 4-42) with highly crowned external involute gear teeth mating with straight toothed internal gear teeth have been used on helicopters far more extensively than all other coupling types combined. These couplings are capable of providing moderately high angular misalignment and axial motion during operation at high speed and torque for very low weight. Common operating conditions are 3 deg continuous and 6 deg

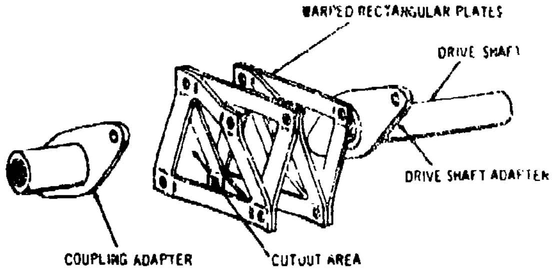

Figure 4-39. Bossler Coupling

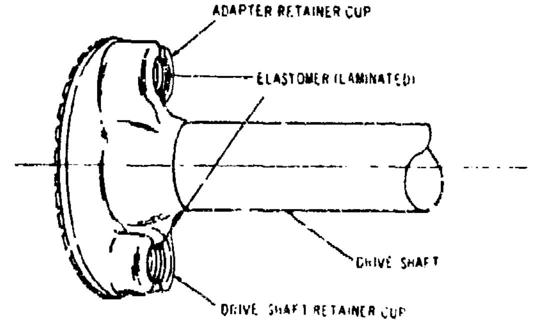

Figure 4-40. Elastomeric Coupling

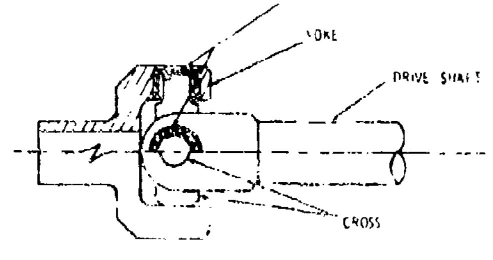

Figure 4-41. Hooke's Joint (Universal)

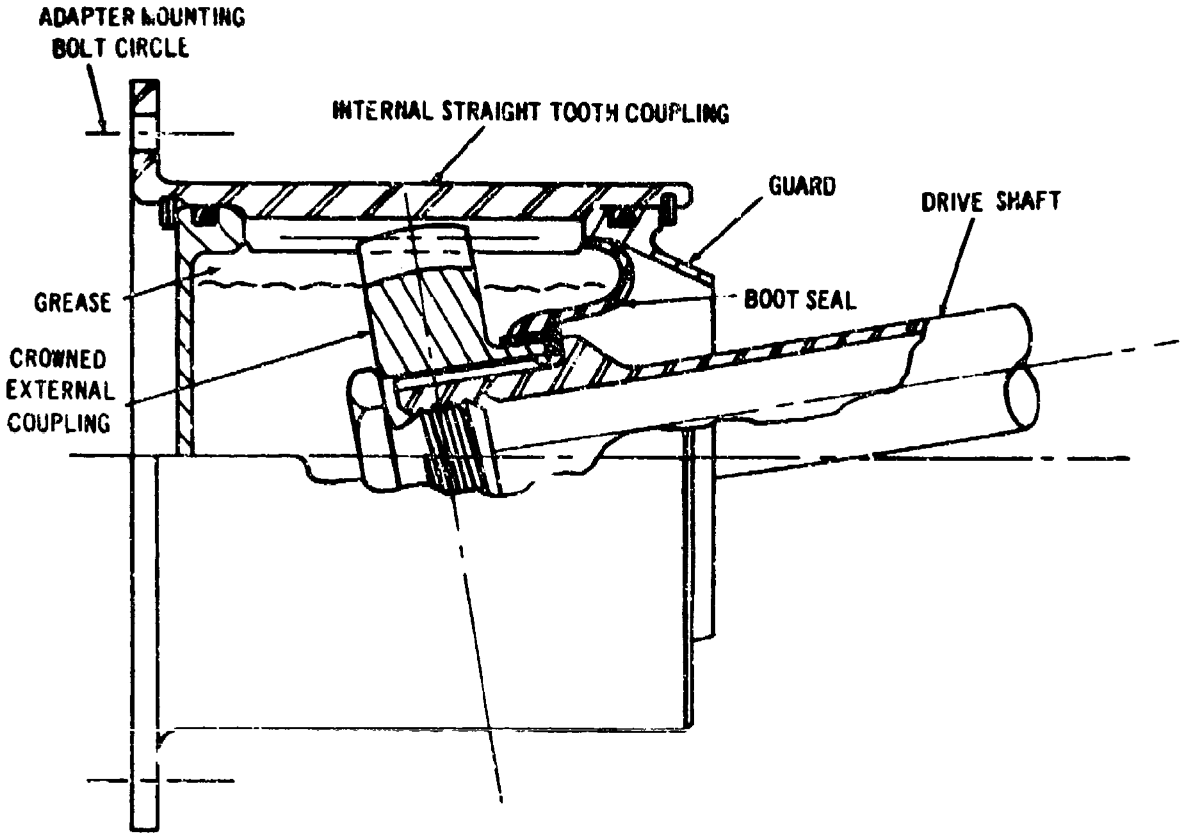

Figure 4-42. Gear Coupling

transient and ±0.75 in. oscillatory axial motion at frequencies in the range of 10 Hz. Unlike straight splines or recirculating ball splines, the breakaway sliding force F_{bs} can be very low; actually F_{bs} reduces as the misalignment angle increases. A comparison of this force for a typical gear coupling is shown in Fig. 4-43. This phenomenon is due to the fact that the contact between loaded teeth is at a high sliding velocity due to the angular misalignments. Consequently, a superimposed axial motion is resisted only by the relatively low dynamic friction coefficient rather than a static value.

Gear coupling operating limits are thermal rather than fatigue, which is the limiting consideration for the five coupling types previously discussed. Specially developed grease lubricants have provided the best load carrying (least friction) capability for gear couplings. However, the operating environment (high centrifugal field, high mechanical stroking frequency, and elevated temperature) combine to make the vast majority of greases unsuitable for this application.

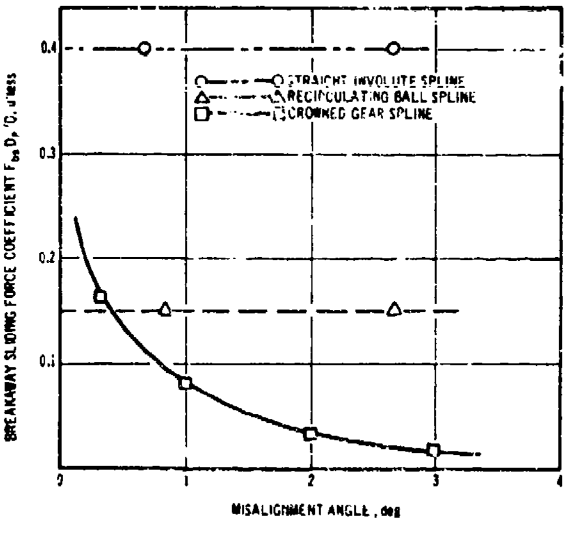

Figure 4-43. Breakaway Sliding Force vs Misalignment for Various Spline Devices

Aside from the need for periodic relubrication (600-hr intervals are common) the greatest difficulty with gear couplings is providing adequate sealing for the grease. Guillotine slider seals and elastomeric boots are most often found in high angle applications while modified lip-type shafts seals can be used for low angle (<1 deg) operation. Shaft speeds of 20,000 rpm, which are now common, offer a distinct challenge to the designer since few seal designs can tolerate the high centrifugal field. Overheating due to loss of lubricant, followed by plastic shear of the hot teeth, is the predominant failure mode for this type of coupling.

Fully hardened, ground, and properly modified gear coupling teeth operate well with tooth loads in the range of 50,000-70,000 psi at sliding velocities well over 100 in./sec. Operation out-side of these boundaries, use of improper tooth materials, and use of inferior lubricants can result in contact melting, smearing, and welding of the teeth.

A method of determining the tooth load distributions for varying combinations of tooth crown curvature, profile modification, misalignment angle, and torque is given in Ref. 113. These loads may then be used to calculate root fillet bending stress, Hertzian contact stress, and flash temperature indices as shown in par. 4-2.2.1.2.

4-3.2.2 Bearings

The criteria for design of hanger bearings for drive shafting differ considerably from the normal power loaded bearings used in gearboxes. The loads P to which the hanger bearings are subjected are very light ($C/P << 10$ where C is the capacity of the bearing for a life of 10^6 cycles with 90% probability survival) and sizes are determined by the torque requirement of the shaft through the bearing. With high tensile strength heavy wall shafts used to reduce shaft outside diameter, a relatively small bore (light) series bearing can be used in the hanger.

Bearing mounting on the shaft should be closely controlled to assure true running and that internal clearances are adequate to prevent radial preloading under operating temperature differentials. Grease lubrication normally is used, and sealed nonrelubricatable as well as relubricatable bearings may be used. The lack of adequate internal clearance is a common design error found in many existing hanger bearing designs.

Considerable effort has been expended, as described in Ref. 114, to evaluate greases for hanger bearings. The grease most commonly used is MIL-G-81322.

After loss of the lubricating oil in the grease by evaporation or migration, the common failure mode exhibited by grease lubricated hanger bearings is overheating, failure and expulsion of the cage, and finally, expulsion of balls. Severe shaft vibration, due to loss of centering provided by the bearing, or shaft failure may follow loss of balls.

Degradation of the lubricant also is caused by entry of water or debris into the bearing. The means of sealing bearings provided by bearing manufacturers are generally inadequate to preclude a significant failure rate in the Army environment unless additional protection is provided. One such means is to enclose the drive shaft with a cover to exclude the bearing areas from the contaminating environment. Another simpler, but less effective, means is to install rotating slingers on each side of the bearings with closely controlled clearances at the slinger OD. This provides shield against the entry of water, debris, or cleaning fluids during helicopter washdown. Although an effective seal may be designed that will reliably assure reasonable bearing life (1000-2000 hr), a hanger that is designed to permit relubrication can greatly reduce hanger bearing replacement. Frequent introduction of a fresh charge of lubricant can revitalize and/or purge the old charge of contaminated and thickened grease. However, relubrication adds to the maintenance burden and the risk of servicing with an incorrect and unsuitable lubricant is everpresent, but most lubricants will provide satisfactory operation of the bearing for at least a short period. The selection of nonserviceable replaceable bearings or relubricatable designs is a trade-off involving many factors such as bearing cost, maintenance man-hours, reliability, and survivability.

The design of the hanger assembly must be such as to prevent inadvertent bearing overloads. Nominal bearing loads are limited to shaft weight and rotating unbalanced loads, neither of which should be detrimental. However, misinstallation of the hanger can introduce static angular misalignment between inner and outer rings (shaft to housing) causing a moment load to be imposed on the bearing. Although system compliance (hanger, shaft, and airframe) may preclude loads of sufficient magnitude to cause spalling fatigue, the bearing balls will skid as a result of contact angle reversal due to these moment loads. Such operation will cause cage distress and overheating with abbreviated service life. Adequate relief from angular misalignment, in the form of proper internal clearances and/or self-aligning outer ring mounting, must be provided in the hanger design.

4-3.2.3 Shafting

Design of the drive shaft itself is concerned primarily with material, size, and end fitting selections. For high torque applications, where tube wall thickness permits, a spline or similar drive mechanism may be used to adapt the shaft to couplings or other drive components. With thin wall tubes, an adapter with a thicker section must be attached to the tube to permit use of a bolted or splined attachment to the coupling.

Adapters may be adhesively bonded to thin wall tubes. The adapter joint must be proportioned properly to avoid excessive stress concentration at the bond interface. This can be done by machining the end fitting bore and shaft OD in a tapered or parabolic shape so that the angle of twist is constant. If the section modulus is constant over the length of the bonded joint, the distribution of shear stress in the bond material will be even. If the joint were not so designed and an abrupt change in section modulus were encountered at the end of the fitting, a differential angle of twist would occur causing a severe shear stress concentration in the bond material. The strength of such a joint would be considerably lower than intended.

The fittings can be riveted effectively to the larger diameter drive shaft tubes with adequate margins of safety. The stress concentration effects normally associated with riveted joints must be taken into account in the design of this type of assembly. Bolted joint designs are similar to the riveted joints.

Welded joints can be made effectively when ferrous materials are used both for tube and adapter. Normal efficiency factors for welds must be used when sizing the joint for steady torsional load and the effects of a metallurgical "notch" or stress concentration must be included in the fatigue analysis. Brazed joints also are effective for some designs. Induction brazing is developed easily and is a cost-effective method. The heat affected zone in the brazed joint normally is tempered, and the torsional strength of the joint must be based on the minimum allowable strength of the tube or adapter in the tempered zones.

Machining may be necessary subsequent to the attachment of the end fitting to provide parallel and concentric mounting surfaces so that the drive shafting runs true.

Materials used for drive shafting include steel, aluminum, titanium, and nonmetallic composite structures. Steel shafting is used for engine-to-transmission applications and other areas defined as fire-zones. Aluminum, titanium, and composite shafting are suitable for interconnect shafting and tail rotor drive shafting. Mill run tube stock can be used in low-speed applications where balance requirements are not stringent. Tube stock and bar stock, bored and completely machined, are used for higher speed application where straightness and true running are necessary to meet close tolerance balancing requirements. Composite materials usually are fabricated by laminating epoxy preimpregnated carbon or boron filament at zero, 45 deg and 90 deg lay to the shaft axis and curing in an autoclave. The composite shaft has a very high strength to weight ratio but the cost is considerably higher than for other materials. Balancing requirements are less stringent for the composite shaft due to the lower specific weight material, but machinable material should be added at approximately one third span positions to facilitate dynamic balancing when required. Large diameter (3.0 in.OD) thin-walled aluminum tubes have demonstrated excellent ballistic tolerance to low and high velocity, and tumbled 7.62-mm bullets. Torque transmitting capability is somewhat reduced following a hit by this type of projectile, but the vibration characteristics are not affected adversely for subcritical shafting. Composite shafting exhibits ballistic tolerance to 7.62-mm bullets similar to that of aluminum shafting although the tolerance to lower velocity projectiles, impact of a dropped tool, or handling damage is considerably reduced.

4-4 LUBRICATION SYSTEMS

A helicopter gearbox can be designed to meet load and speed requirements but the useful life of the gearbox is a direct function of the lubrication and cooling system. The amount of power loss as heat is governed by the design of the heat generating elements in the gearbox. The lubrication system assures attainment and maintenance of a minimum value of heat loss as well as minimum wear. The concurrent function of the lubrication system is to carry away heat.

Heat transfer occurs between bearing outer rings and housings by conduction and from housings to atmosphere by convection. This mode of heat transfer is minimal compared to the heat transferred directly to the inside walls of the housings by the cascading oil, with convection again taking place. The third means of heat rejection is by direct transfer to forced air in an air/oil or, in rare cases, fuel/oil heat exchangers (oil coolers).

During stabilized operation a balance is maintained between heat transfer by conduction/convection/radiation from the gearbox cases and the heat exchanger, if one is provided. Some gearboxes are designed for continuous operation without an external heat exchanger. In this case the surface area (external wetted area) provides adequate cooling margin, especially if forced air is directed across the gearbox.

A somewhat different mode of heat transfer occurs in gearboxes that are grease lubricated. Gearboxes that are grease lubricated depend almost entirely upon the transfer of heat from the gears along the shaft to the bearings, through the bearings, and to the housings. A secondary flow of heat is provided by slowly migrating grease as agitation occurs but this is minimal compared to the direct conduction of heat to external gearbox walls through the shafts and bearings. Tests conducted on grease lubricated gearboxes using USAF MCG 68-83 grease (Refs. 37 and 115) indicate that grease migration is not significant. The lack of migration can be an advantage in meeting fail-safe operational requirements since little or no grease loss would be anticipated in the event of a ballistic strike in the housing.

4-4.1 OIL MANAGEMENT

The delivery of oil from pump to filter to manifold and then to load points must be systematic and deliberate to assure proper lubrication and cooling. Placement of the oil must be specific to prevent surging, foaming, and cavitation. As the used oil leaves the gear mesh and/or bearings, a natural gravitational flow path must be provided. Traps around rotating components can cause excessive churning and heat buildup, thus adding to the cooling burden. High speed gears can create vortices that will suspend large amounts of oil against the housing around the gear. Excessive oil flow to gears and bearings can cause heat generation and buildup greater than the amount of heat coming from the loaded conjunctions. Therefore, controlled movement of the oil after egress from the rotating elements and heat generating points must be provided to allow the oil to find its way uninterrupted back to the sump. Close fitting shrouds around gears, and return lines from cavities between bearings and shaft seals provide effective means of preventing oil entrapment and excessive churning. Judicious placement of ribs and webs in the gearbox housings and ample provision for oil flow beneath or around the structure will help assure proper oil return.

The pump inlet placement and arrangement must be considered carefully in the design of the pump, housing, and sump. Maintenance of a sufficient oil supply at altitude is directly affected by the volume and depth of oil at the oil pump inlet and the effect of flow constrictions into the inlet. If the return oil is hampered in getting to or through the oil inlet, cavitation and loss of oil pressure can ensue.

In splash lubricated gearboxes oil flow is more difficult to attain. However, because the primary function of the lubricant in this type of gearbox is to lubricate the gears and bearings sufficiently to minimize the heat generation, the amount of oil required at the friction points is minimal. Nevertheless, management of the oil is still critical to the adequacy of lubricating and cooling; provision must be made for oil to be delivered to each bearing, gear, and seal. Natural laws are employed to accomplish this; centrifugal head, gravity feed, and dynamic pressure differentials can impart sufficient impetus to the oil to attain directed flows. Oil splashed to the inside of a rotating shaft can be caused to flow continuously through the shaft by tapering the bore from the oil "inlet" end to the outlet. The outlet can be at the end of the shaft where return is accomplished by gravity flow through bearings or it can be through radial holes in the shaft, with centrifugal head forcing the oil into the bearings. Cooling (though minimal) also is provided by this flow by ultimate impingement of the warm oil onto gearbox interior walls. Agitation of this oil is primarily by gear members dipping into the oil sump and splashing the oil to the housing walls, bearings and gears, or to the inside of shafts. Auxiliary splashing can be accomplished by providing rotating dippers or slingers. Maximum cooling of the oil can be accomplished by the agitation and slinging action, but care must be exercised to determine the maximum oil level that can be tolerated before churning losses override the cooling effect of the agitated oil.

Grease lubricated gearboxes have a different set of operating characteristics. Although the high viscosity of grease provides good lubricating qualities, this high viscosity also prevents free migration inside the gearbox. As a result, the grease must be forced to remain in the bearing and gear cavities, usually by means of shrouds and baffles. The grease is thereby "captured" around each bearing, and the grease quantity must be such as to assure an adequate supply around gears. The percent "fill" in the gearbox is critical, as it is with the oil lubricated gearbox, especially the minimum level inasmuch as successful lubrication of the gears is predicated on grease quantity as well as location.

4-4.1.1 Function

The satisfactory fulfillment of the dual functions of cooling and lubricating requires that the design be approached systematically. Oil flow requirements should be determined and followed from pump outlet through the system and back to the pump outlet.

Oil must be provided for lubrication of all heat sources and supplied in such a way as to be most effective by pressure jets, oil mist, and/or internal pressure passages to bearings and gears. The amount of oil required to lubricate a conjunction, i.e., to prevent metal to metal contact, is very small. However, if the frictional heat generated in the conjunction is high, a much greater amount of oil must be supplied to absorb the heat and carry it away from the conjunction. Local heat buildup can cause scoring in gear teeth with subsequent welding and tooth breakage possible, and also loss of internal clearance in bearings that can cause spalling failure or seizure. An adequate supply of oil serves to reduce the friction losses to a minimum level and to maintain stable operating temperature consistent with the power and speed. The cooling oil requirement in a loaded contact is a direct function of the load intensity and the speed. A lightly loaded gear mesh operating with thick film lubrication will generate very little frictional heat and will require very little cooling oil. Such a mesh can be lubricated adequately by air/oil mist or by splash. Conversely, a highly loaded gear mesh will generate considerable heat and require a copious flow of directed oil for cooling. The same is true for both lightly loaded and heavily loaded bearings.

4-4.1.2 Components and Arrangement

An oil system, as a minimum, will consist of a supply of oil in a gearbox and a means of gaging oil quantity, e.g., sight glass, dip stick. The sump must be so located that oil circulation will be accomplished by a gear or rotating element dipping into the sump and splashing the oil to the gear and bearing elements. This arrangement for splash lubrication can be used effectively in gearboxes with single meshes normally operating at light load where the wetted gearbox area provides adequate convective cooling. Tail rotor drive and accessory drive gearboxes fall in this category. Although the power transmitted by the tail rotor gearbox occasionally is high, the condition is transient and bulk heat buildup is generally negligible. Gearbox heat loss at hover power can be transferred effectively from the gearbox housings to the airflow caused by the rotor downwash. Less power is required during cruise conditions, and more airflow is available. Large wetted areas are common for accessory drive gearboxes, and power requirements are predictable and constant. The splash system is an inherently wet sump system.

On the other end of the spectrum are oil systems consisting of oil pump, oil lines and passages, relief valve, filter, manifold, regulator, oil cooler, thermal bypass, flow bypass, temperature sensor and gage, pressure sensor and gage, sight glass, and multiple oil jets. A schematic of such a system is shown in Fig. 4-44. Power for the oil pump can be provided by an accessory drive gear with power takeoff from the main drive train. Oil is pumped from the sump through the filter, the cooler, and the manifold to the internal passages and jets and finally to bearings and gear meshes.

This is typical of both wet sump and dry sump pressure lubrication systems. The dry sump system normally will have a scavenge pump to remove oil from the "free" sump area and to feed the pressure pump inlet cavity with a continuous supply of lubrication oil. The dry sump system finds wide use where the likelihood of oil starvation exists, as in violent flight maneuvers, or in gearboxes that are subject to different functional requirements in a single flight, as with a compound helicopter or convertiplane. Another use for the dry sump system is with a system of gearboxes utilizing a single lubrication system. The oil must be scavenged from all remote gearboxes and deposited in the pressure pump sump area.

The pressure pump in general will be of the constant displacement type, sized for the pressure and flow rate determined by cooling requirements and system pressure losses. Pumps can be designed to meet widely varying flow requirements with single element pumps possible with flows in the range of 70 gpm and speeds up to 12,000 rpm. The pump drive may be required to have a shear section to satisfy the requirement that no catastrophic damage be done to the main drive train in the event of accessory failure.

The filter system should consist of a pump inlet screen to prevent ingestion of large particles and, downstream of the pump, finer filtration. A primary disposable filter element of required fineness in a housing with full flow bypass capacity, and possibly with bypass indicator, should be provided. Replacement of this filter element will be part of the periodic maintenance requirements. Additional filters may be required in the system to meet filtration requirements of full bypass flow, if stipulated by the RFP or PIDS. In this case a secondary filter sytem will be installed in the bypass sytem to assure continued clean oil delivery to the gears and bearings subsequent to complete clogging of the primary filter. The secondary filtration requirements generally are less stringent than primary filtration. Filter elements of the order of 40 microns suitable for secondary filtration can be of porous bronze, steel mesh, or paper element types. The bronze and steel filters are cleanable and reusable while the paper element generally is

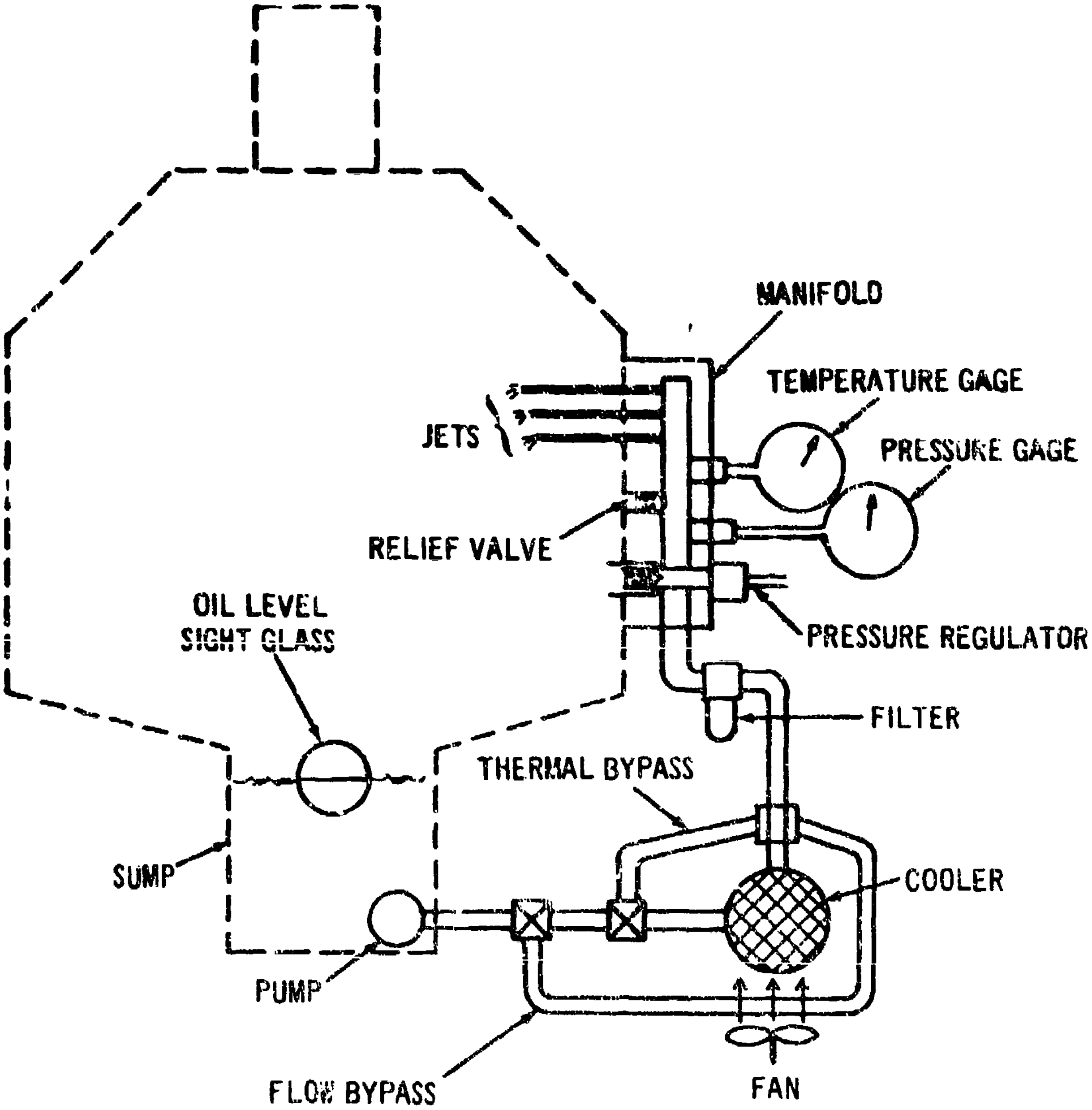
MANIFOLD
TEMPERATURE GAGE
PRESSURE GAGE
JETS
RELIEF VALVE
OIL LEVEL SIGHT GLASS
PRESSURE REGULATOR
FILTER
THERMAL BYPASS
COOLER
SUMP
PUMP
FLOW BYPASS
FAN

Figure 4-44. Oil System Schematic

disposable. The paper elements are used more frequently in the primary system where absolute filtration down to 3-5 microns can be attained and maintained with less than 15 psi pressure drop across the filter (Ref. 116). However, absolute filtration of 15-micron particles with 99% efficiency of filtering 5-micron size particles or larger has been shown to be cost-effective and adequate (Ref. 117), while finer filtration presented filter clogging problems that affected service intervals and reliability.

The oil cooler system normally will consist of an air/oil heat exchanger with thermal bypass for cold weather starting, and forced air cooling. Oil cooler design requirements and procedures are presented in Chapter 8, AMCP 706-201, and are amplified in par. 4-6.2. Forced air can be from shaft driven blowers or bleed air turbines. Oil cooler fan design procedure also is described in Chapter 8, AMCP 706-201.

Cooler location should be consistent with applicable ballistic threat survivability requirements.

Coolers integral with the gearbox or, if separate, surrounded by protective components or structure are possibilities. The use of either auxiliary systems or armor plating should be considered only as a last resort. The integral oil cooler has been shown to be effective (Ref. 116). It virtually eliminates the need for external plumbing and minimizes the ballistic threat to the gearbox oil system, and the inherent protection of the surrounding airframe structure is enhanced by proximity of cooler to gearbox. A pressure bypass has been used to divert full oil flow to the gearbox oil system to circumvent oil flow to the cooler in case of ballistic strike on the cooler.

The manifold is an oil distribution mechanism that normally houses the oil pressure regulator, temperature sensor, pressure sensor, and distribution passages. Oil is carried from the manifold through gearcase internal passages to oil jets for pressure lubrication of gears and bearings with direct impinging streams of high velocity oil. Internal passages also can be provided to direct oil to bearings encapsulated in housings and liners.

Externally mounted oil system components such as pumps and filter housings often present sealing problems and service problems associated with the seals. Gaskets and O-rings normally are used for sealing between the mating parts. Components requiring frequent removal can more effectively be sealed with O-rings then gaskets. The compressed gasket material adheres to both the sealed surfaces, and mechanical removal of the gasket residue often is required. This becomes more difficult around studs. Each gasket application generally is unique and hence maintenance support requires stocking of unique parts, while O-rings are stocked for multiple applications and are supplied from a common stock. Gaskets possibly have a cost advantage by virtue of the elimination of the O-ring groove. Provisions for O-rings also can result in slightly higher weight than for gaskets.

Some system protection is provided by gaskets by their inherent ability to "blow out" in case of overpressure surges. Where close tolerances must be held between locations within the mating parts, the use of a gasket becomes impractical. The gasket material can compress and generally is not consistent from one gasket to another. In this case an O-ring should be used.

4-4.1.3 Special Considerations

High flow oil systems may require multiple element pumps. Constriction free inlet design, high rotational speed, and high flow rate may not be attainable with a single element pump. Multiple element pumps (or more than one pump) also may be necessary in a dry sump design, when a scavenge pump is required. The scavenge pump extracts oil from the sump area and feeds it directly to the pressure pump or to an oil inlet sump for the pressure pump. Oil return requirements, sump capacity, and turn-around time must be compatible. A 20-gpm flow requirement with an 8-qt sump capacity results in complete turn-around of the oil ten times a minute. If the height of the gearbox is appreciable, with extensive baffling, there is a danger of pump cavitation and interrupted lubrication. Even without the danger of interrupted lubrication the oil has insufficient dwell time for deaeration. Therefore, excessive foaming and inadequate lubrication or cooling are possible. Turnaround frequencies greater than 3.5-4 times per minute become questionable with respect to proper deaeration and attendant cooling characteristics. Adequate film thickness is difficult to achieve relative to surface finish in loaded contacts when low viscosity synthetic oils are used for gear and bearing lubrication. Boundary lubrication states, often characteristic with low viscosity oil, can still provide adequate wear life in gear teeth and bearings but the surface roughness must be low enough to prevent progressive metal-to-metal contact (see par. 4-1.2.1).

Synthetic oil, especially MIL-L-23699, has a moisture absorption capability and its lubricating ability is diminished by moisture content. Hence, extreme care should be exercised in the design and location of gearbox vents to prevent water ingestion. Areas where atmospheric air can impinge directly on shaft seals also should be avoided. Positioning rotating shields in front of shaft seals is a very effective means of preventing dust, dirt, and moisture-laden air from being ingested into the gearbox.

Secondary effects of moisture absorption are internal corrosion. Synthetic lubricant that is contaminated with moisture becomes highly corrosive to the bare steel parts inside the transmission, with the lower roughness surface finishes being particularly susceptible. It should be noted that once contaminated with water, MIL-L-23699 does not release that water when heated to normal operating temperature (>212°F). Therefore, both the poor lubricating quality and adverse corrosive tendency are present, and every effort should be made to prevent moisture absorption.

The most consistent problem facing the designer of oil lubricated gearboxes is proper sealing. Leaking seals represent the single largest replacement item or cause for removal of gearboxes in the military helicopter (Ref. 33). Although it is infrequent that a seal leak rate is sufficient for depletion of the gearbox lubricating oil to occur in a single mission, that appearance is presented nevertheless. The oil residue from a leaking shaft seal accumulated on the surrounding components is so extensive that a minor leak manifests itself as a major problem. Certainly effective seal designs are laboriously, if ever, achieved. Carbon face and circumferential seals required for high-speed and high-temperature applications generally require an extensive test and development program. Elastomeric shaft seals for lower speed applications are designed more easily but successful sealing often is equally difficult to attain. Investigations are being conducted continuously by seal manufacturers and users to develop a universally acceptable and effective seal design. Based on the premise that no seal is completely effective, one design approach that can be taken to minimize the leakage problem is multiple seals. A shaft seal of conventional design, either elastomeric lip seal or carbon face seal, can be used in conjunction with other types of seals to affect seal staging. One suitable method is to use an inner lip seal with an outer labyrinth seal. The oil lubricates the lip seal, which assures adequate seal life, while the labyrinth provides secondary sealing from both directions. The shielding effect of the labyrinth precludes atmospheric debris that would normally accelerate elastomer and shaft wear from collecting on the lip seal. A rotating slinger in close proximity to the housing on the outside will produce further baffling and increase the sealing effectiveness and seal life. Oil that weeps past the lip seal in normal operation can be removed through an overboard drain.

Research with various lip contact configurations for rotating shaft lip seals has shown promise during testing but no striking improvement has been observed in service. A ribbed lip was observed to produce a pumping action that prevented oil flow from the oil side of the test gearbox. Another lip design, a waved contact lip, produces a similar wiping action and retains some lubricant on the seal-shaft contact that provides good sealing and coincident lubrication. A radially segmented carbon seal has been extensively tested and evaluated at NASA for high-speed shaft sealing. This seal consists of several semicircular segments fitted together and spring-loaded to contact the shaft. In operation the seal lifts off slightly and virtually frictionless contact results.

4-4.2 COOLING REQUIREMENTS

Determination of the power loss in bearings and gears as described in pars. 4-2.2.1 and 4-2.2.2 provides the basis for determination of minimum heat rejection requirements. The gearbox frictional losses and windage losses, having been determined, an estimate of the oil flow requirements can be made.

4-4.2.1 Heat Exchanger Sizing

The maximum size required for a heat exchanger, or oil cooler, would be that size necessary to reject all the heat losses from the transmission. On the other end of the spectrum, considering forced air convection around the gearbox, no oil cooler may be required. This would occur if the surface area were sufficiently large, heat generation low, and internal oil flow distribution such that transfer of heat to the housing inside walls were adequate. Characteristic heat transfer rates from aluminum and magnesium gearbox housings are in the range of 0.001 Btu/in.^{2}-min-°F (Ref. 4). Hence, when the friction and windage loss has been determined and the surface area has been established, it easily can be decided whether an oil cooler will be necessary. Tail rotor drive gearboxes and accessory gearboxes generally fall into this category. However, in the interest of compact design it is rare that no cooler is required for a main rotor gearbox. A general design requirement for the cooler is to reject 67% of the heat generated from the gearbox during critical operating conditions. Critical operation generally occurs during hover at design gross weight in hot-day conditions (35°C, 4000 ft), when maximum main rotor power is required and forced air convection is minimal. As the gearbox power-to-weight ratio increases with improvements in design and material technology, larger size coolers will be required to reject the increased amount of heat that will result from higher specific gear and bearing loads and decreased wetted areas of housings.

The physical size and configuration of the oil cooler, together with oil and air flow rates and pressure drops, can be determined with the help of the cooler manufacturer. The cooler core size and density are determined by the heat rejection requirements and the available airflow. The procedural approach to cooler size determination consists of the following:

1. Determine heat generated in the gearbox in gear teeth, bearings, and by windage loss (par. 4-2.2.2).
2. Determine effective external wetted area of gearbox. Effective area is exclusive of appendages.
3. Apply heat transfer factor, 0.001 Btu/in.^{2}-min-°F for hot-day performance and power condition, and take algebraic difference between heat generated and heat transferred.
4. If heat generated exceeds heat transferred, then a cooler will be required to reject the excess generated heat.

5. Determine location for cooling fan and, based on required heat rejection rate of the cooler, choose a fan that will meet the airflow requirements of the cooler. Cooler specifications to be met are:

*	Rated oil flow	gpm
	Rated air flow	lb/min
*	Rated heat rejection	Btu/min
*	Oil inlet temperature	°F
*	Oil outlet temperature	°F
*	Oil inlet pressure	psig
*	Oil pressure drop	psi
	Air static pressure drop	in. H_2O

(*These values will be established by the system design and the heat rejection requirements.)

Any number of actual combinations of airflow and cooler sizes can meet the established heat rejection requirements. However, the final choice will be based upon the system interfaces, i.e., available fan drive power; location of fan and ducting required; fan size limitations imposed by surrounding airframe or other hardware; and resulting limitations on airflow cooler location, and size limitations. The necessary calculations for a cooler and fan design are presented fully in Ref. 148.

4-4.2.2 Cooling Fan Sizing

Sizing of the cooling fan can be accomplished by the method described in Chapter 8, AMCP 706-201. The airflow requirements (volume, pressure, and velocity) will be determined by the heat rejection required of the cooler, the oil flow rate, and cooler core parameters. Based on the required airflow, a fan that will interface with the available drive and space can be designed to meet the requirements. Both axial and centrifugal flow fans are used in cooler blowers. Choice of the type of fan is dependent upon airflow volume and pressure requirements. The axial fan generally is used where higher shaft speeds are available, and pressure head at the cooler is high. The centrifugal flow fan generally is used where high volume flow at lower pressures is required. An adverse side-effect possible with the axial flow fan is a high pitch noise.

4-4.3 EMERGENCY LUBRICATION

Design of a power transmission system to meet specific emergency operation requirements entails a comprehensive evaluation of each and every dynamic component that can influence the loss of drive continuity as a result of interruption or loss of lubrication. Redundancy of power paths, dormant auxiliary lubricants, secondary cooling systems, and specific design tolerances are considerations directly pertinent to emergency lubrication or operation without benefit of primary lubrication and cooling systems. Much work has been accomplished in establishing and evaluating design parameters associated with emergency operation, i.e., fail-safe design (Refs. 35, 37, and 38). The basic criterion that has been established is safe continuation of flight for a minimum of 30 min subsequent to total loss of lubricant. As a minimum, the continuing flight *shall* be at the power level required to maintain the speed for maximum range at sea level standard conditions.

Loss of lubricant initially is synonymous with loss of cooling and is followed immediately by an increase in the coefficient of friction with attendant increase in heat generation, due to the change to dry operation. As the primary heat transfer medium of oil is lost, an immediate heat buildup occurs at the heat-generating points; and the secondary heat transfer paths become paramount. If unstabilized heating is to be averted, the heat generating element must maintain a balance of heat generated to heat transferred through the secondary medium at a maximum temperature that is safe. Heat sources (gears and bearings) must be designed to minimize dry friction losses, and the generated heat must be transferred away by the most efficient means available.

Gears designed for fail-safe operation must have sufficient clearance to prevent interference at the higher stabilized temperature. The clearance necessary is determined by calculating the differential expansion between steel gear centers and the same distance in the housing, which is usually nonferrous material. For instance, a gear set consisting of straight spurs operating at a center distance L_{CD} of 6.0 in. and a normal operating temperature of 200 °F may attain a temperature of 900°F while the aluminum housing containing the gears (and bearings) only rises to 400°F. The rate of expansion of the steel S_{fe} is 6.5 × 10^{-6} in./in.-°F and the aluminum expansion rate S_{fe} is 12 × 10^{-6} in./in.-°F. The differential amount of expansion would then be

$$
\begin{aligned}
\Delta L_{CD} &= (\Delta T_{Fe}\,\delta_{Fe} - \Delta T_{Al}\,\delta_{Al}) \times L_{CD} \\
&= [(T_2 - T_1)_{Fe}\,\delta_{Fe} - (T_2 - T_1)_{Al}\,\delta_{Al}] \\
&\quad \times L_{CD} \\
&= [(900 - 200)(6.5)10^{-6} - (400 - 200) \\
&\quad (12)10^{-6}] \times 6.0 \\
&= [(700)(6.5 \times 10^{-6}) - (200)(12)(10^{-6})] \\
&\quad \times 6.0 \\
&= [(45.5)10^{-4} - (24)10^{-4}] \times 6.0 \qquad (4\text{-}69) \\
&= 0.013 \text{ in.}
\end{aligned}
$$

The significance of the preceding calculation is that the gears expand in a radial direction toward each other to reduce running clearance more than the housing expands to separate the gears. In the example, to allow for operation at the assumed conditions without interference the teeth would have to be cut deeper (smaller root diameters) and/or the outside diameters decreased, equivalent to spreading the gear centers by a total of 0.013 in.

The same type of calculation can be made for ball and roller bearings on a radial clearance basis and for duplex ball bearings, considering contact angles, on a radial and axial basis. Radial growth characteristics in bearings and gears and the effects of dry running are presented in Ref. 38. Optimum design for minimum friction loss in gears and bearings is covered in par. 4-2.2.

To operate a gear or bearing at temperatures of 900°F and above, it is necessary that the component be fabricated from material(s) that exhibit a reasonable tolerance to high temperature. Materials such as AMS 6475 (Nitralloy-N) and AMS 6490 (M-50) are well suited to the purpose. AMS 6490 exhibits excellent hot hardness characteristics and has proven to be one of the most fatigue resistant bearing materials available. AMS 6475 is a precipitation hardening material widely used in helicopter gearing which also exhibits high hot hardness characteristics. The more common gear and bearing materials, AMS 6265 and AMS 6444, respectively, do not have high hot hardness characteristics, but they can withstand moderate loads for a short period. Fail-safe operation for 30 min can be obtained using AMS 6265 and AMS 6444 gears and bearings, but the applications are limited to moderate power levels and speeds.

Bearing cages are also critical to the design of fail-safe bearings. Bronze and plastic materials are not acceptable for fail-safe operation. The characteristic failure mode for a bearing with a bronze cage is mechanical plating of the bronze onto the rolling elements with immediate loss of running clearance and temperature instability, followed by seizure. Plastics such as nylon, Teflon (tetrafluoroethylene) and fiberglass offer little resistance to failure at elevated temperatures. Carbon graphite is an excellent cage material for dry operation, but its tensile strength is too low for normal use. Manufacturing problems and scrap rates are significant; carbon cages can be armored with steel reinforcing rings and side plates but the cost is quite high. The most adaptable cage material at the present time appears to be mild steel with silver-plated pockets. Dry friction between the rolling elements and the silver-plated cage is moderate, and the steel retains adequate strength for this application. Clearances are necessarily a very important part of the cage design. Outer-land-riding cages offer the risk of entrapping slag-type debris between the cage and the outer land, with fracture or seizure possible, while inner-land-riding cages risk loss of clearance due to thermal differentials between the cage and the inner ring of the bearing. The most difficult cage design is seen in high speed bearings. It is desirable, especially in roller bearings, to provide inner-land-riding cages with inner ring through-lubrication to make maximum advantage of the traction force vectors. Also, it is desirable to minimize guiding-land-to-cage clearance from a dynamic balance standpoint. Therefore, if normal high speed design parameters are followed, risk of seizure or "burn-out" of the bearing increases for dry operation.

Several means of augmenting lubrication or supplying lubricant after loss of the primary oil system that may be developed are:

1. Inclusion inside rotating shafts of high melting point lubricant that melts and flows into bearings and onto gears after dry running commences
2. Providing oil traps with metering holes
3. Wicking oil into bearings from oil absorbing materials
4. Encapsulating lubricant in containers with heat activated drain plugs
5. Providing auxiliary (idler) gears of oil absorbing or dry lubricant material to mesh with power gears.

4-5 ACCESSORIES

4-5.1 PAD LOCATION AND DESIGN CRITERIA

On small helicopters the accessories may consist only of an oil pump, hydraulic pump, tach generator, and cooler fan. A simple co-axial arrangement of oil pump, tach generator, and hydraulic pump as on the OH-58A may be the most effective means of arranging an accessory drive. The accessories are driven by a concentrated contact spiral bevel pinion powered by the input bevel gear.

On medium weight helicopters, where system redundancy may be required, multiple accessory pads usually can be provided on the main gearbox. Hydraulic pumps for primary control actuation must be located at widely displaced locations to thwart loss of both systems to a single small arms bullet. The size of the main gearbox normally will be adequate to allow such displacement while still providing pads for generators, tach drive, etc. Accessibility for maintenance must still be a prime criterion for location.

On large helicopters the most effective means of providing accessory drives normally is from a gearbox remote from the main rotor gearbox. Multiple systems with redundancy become imperative, and the complexity and power required for ground checkout establishes the need for an auxiliary power unit (APU). With multiple drive pads and high continuous power requirement the remote accessory drive gearbox must have a recirculating oil system, complete with oil pump and filter. For emergency lubrication considerations, the gearbox must be self-contained to prevent oil depletion from the main transmission in the event of the accessory gearbox being hit by small arms. The location for the accessory gearbox must not introduce unacceptable noise levels in crew compartments.

4-6.2 ACCESSORY DRIVE DESIGN REQUIREMENTS

Accessory gearbox design and configuration factors must be compatible with the main gearbox power takeoff, airframe and cowling, work platform provisions, CG, and minimization of gear-induced noise. Particular drive pad power requirements are determined by the accessory (hydraulic pump, generator, alternator, etc.), and the proper MS, AND, or QAD pad must be provided to meet the continuous power rating and seizure torque level. The accessory drive shaft normally is provided with a shear section that must fail in the event of seizure of the accessory rather than permit damage to the accessory gearbox. Coincidentally the accessory drive gearbox must be provided with a connecting drive shaft system that will isolate effects of accessory gearbox seizure from the main rotor gearbox. Multiple clutch arrangements are required to provide isolation of the APU during normal helicopter operation and overrunning isolation of the gearbox from the main rotor gearbox during APU drive (ground checkout, etc.). APU shaft mounted centrifugal clutches are well suited to the former application and one-way sprag clutches are well suited to the latter. Functionally, the APU must power the accessory gearbox by driving through the APU input clutch while the main gearbox drive is disengaged by means of the one-way sprag clutch in the accessory gearbox. Conversely, as the main rotor(s) become operative, the APU is shut down and disengaged while the main rotor gearbox drives into the accessory gearbox through the one-way clutch. Additional clutches may be required to limit the number of accessory drives that operate during ground operation.

4-6.3 SPECIAL REQUIREMENTS

As mentioned previously, the noise generated by the accessories must be considered in the choice of gearbox location. The hydraulic pumps are especially severe noise generators and close proximity to a crew compartment can cause intolerable high-pitch sound levels. Elastomeric mounts can be an effective noise isolation means. APU exhaust ducting must be adequate to prevent noxious gas and heat from invading the personnel compartments. As with other gearboxes accessibility must be provided to oil level indicators for preflight maintenance. One man should be able to change accessories without assistance.

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CHAPTER 5
ROTOR AND PROPELLER SUBSYSTEM DESIGN

5-0 LIST OF SYMBOLS

A = propeller inflow angle, deg
a = speed of sound, fps
a_n = coefficient which is dependent upon mass and stiffness distribution and has a different value for each mode of vibration, dimensionless
B = tip loss factor, dimensionless
BL = blade loading, lb/ft^2
b = number of blades
b_s = blade semichord, ft
C = empirical constant, dimensionless
$\bar{C}_D$ = mean rotor blade profile drag coefficient, dimensionless
C_L = coefficient of lift, dimensionless
$\bar{C}_L$ = mean rotor blade lift coefficient, dimensionless
C_M = coefficient of pitching moment, dimensionless
C_P = power coefficient, dimensionless
C_T = thrust coefficient, dimensionless
c = distance from beam neutral axis to outer fiber, in.
c_d = airfoil section drag coefficient, dimensionless
c_l = airfoil section lift coefficient, dimensionless
$c_{l_{max}}$ = maximum section lift coefficient, dimensionless
D = propeller diameter, ft
E = modulus of elasticity, psi
EF = excitation factor, dimensionless
EI = stiffness, lb-in.^{2}
e = location of flapping hinge from the center of rotation, in.
F = force, lb
g = acceleration due to gravity, ft/sec^2
HP_0 = profile power required, hp
I = mass moment of inertia, slug-ft^2
I' = moment of inertia, in.^{4}
I_p = polar moment of inertia (per blade for a tail rotor), slug-ft^2
I_p = propeller mass moment of inertia, slug-ft^2
I_R = mass moment of inertia of the rotor, slug-ft^2
I_{zz} = helicopter yaw mass moment of inertia, slug-ft^2
K = ratio of total tail rotor thrust to net tail rotor thrust, dimensionless
K_T = notch factor, dimensionless
K_g = gust alleviation factor, dimensionless
K_n = coefficient dependent upon mass distribution and the mode of vibration, dimensionless
KE = kinetic energy, ft-lb
KE_R = rotational kinetic energy, ft-lb
L_w = wing lift, lb
l = length, in.
M = bending moment, in.-lb or ft-lb
M = Mach number, dimensionless
$M_{adv\,tip}$ = advancing tip Mach number, dimensionless
m = mass per unit length of the beam, slug/in.
m_R = mass of spanwise increment at outboard end of blade (R), slug
m_e = mass of spanwise increment at inboard end of blade (e), slug
n_z = load factor, dimensionless
n = the number of vibratory stress cycles accumulated at a particular stress level or at a particular operating condition
n = tail rotor rotational speed, rev/sec
n_g = gust load factor, dimensionless
P = actual power required, hp
P = pressure, psi
Q_E = engine torque, units as required
Q_P = propeller torque, lb-ft
Q_{mr} = main rotor torque, lb-ft
q = dynamic pressure, lb/ft^2
R = propeller tip radius, ft
R = rotor radius, units as required
R_{mr} = main rotor blade radius, ft
R_{tr} = tail rotor radius, ft
R'' = outside blade radius, in.
r = radius, ft
r = radius of curvature, in.
S = Laplace operator, sec^{-1}
SFP. = stall flutter parameter, dimensionless
S/A = ratio of blocked disk area to total disk area, dimensionless
T = thrust, lb
ΔT = change in thrust, lb
T/A = tail rotor disk or thrust loading, psf
T_Q = tail rotor thrust required to compensate for main rotor torque, lb
T_{tr} = total tail rotor thrust required, lb
T_{net} = total tail rotor thrust minus the fin force, lb
t = propeller axis downtilt from wing zero-lift-line, deg
V = true airspeed, kt

V = average velocity of contacting surfaces, fpm
V_i = indicated airspeed, kt
V_i = induced velocity, fps
V_v = vertical airspeed, fps
W = gross weight, lb
W_a = aft adjustable weight, lb
W_f = forward adjustable weight, lb
W_p = propeller weight, lb
$W_{s_{max}}$ = maximum allowable weight for abrasion strip, lb
$W_{s_{min}}$ = minimum allowable weight for abrasion strip, lb
w = disk loading, lb/ft^2
X = distance between center of main rotor and tail rotor-antitorque moment arm, ft
$\overline{X}$ = dynamic axis, in.
ΔX = clearance between main rotor and tail rotor blade tips, ft
x = chordwise distance from blade leading edge to centroid of mass increment, in.
Y = spanwise distance from flapping hinge to centroid of mass increment, in.
α = rotor angle of attack, deg
β = rotor blade coning angle, deg
β = propeller blade angle, deg
γ = $\tan^{-1}(C_D/C_L)$
δ_3 = pitch-flap coupling angle, positive if pitch is decreased when the blade flaps up, deg
θ = rotor blade angle, deg
μ = advance ratio, dimensionless
μ = coefficient of friction, dimensionless
μ_R = rotor mass ratio
ρ = air density, slug/ft^3
σ = standard deviation, defined as the root-mean-square value of the deviations between individual data points and the mean
σ = rotor solidity, dimensionless
σ_B = blade bending stress, psi
σ_b = rotor blade solidity, dimensionless
ψ = propeller inflow angle, deg
ψ = yaw rate, rad/sec
ψ = yaw acceleration, rad/sec^2
Ω = rotor angular velocity, rad/sec
Ω_{tr} = tail rotor angular velocity, rad/sec
Ω_R = rotor tip speed, fps
Ω_1 = precession velocity, rad/sec
ω = propeller speed, rad/sec
ω_t = natural torsional frequency, rad/sec
ω_{R_n} = natural frequency of a rotating beam, rad/sec

5-1 INTRODUCTION

In general, all rotors and propellers are mechanical devices used to produce thrust by accelerating a fluid mass. They range in sophistication from simple two-bladed, fixed-pitch configurations to coaxial counterrotation systems with individual rotor collective and cyclic pitch control. The analytical techniques for all types are very similar. However, there are minor variations in the definition of rotor-propeller nondimensional parameters which prove to be unimportant once it is realized that data can be transposed readily from one format to another.

The overall performance of a rotor or propeller may be described by its tip speed, airfoil characteristics, solidity ratio, and disk loading. Rotational inertia also is important to rotor design because it affects helicopter autorotational performance. Based upon selected values for these parameters, the detail design of the rotor is largely a task of optimizing the configuration in terms of the number of blades, flapping and inplane freedoms, dynamic response to externally applied cyclic forces, and the assurance that the hardware can be built with a fatigue or service life compatible with the design requirements.

The paragraph addressing design parameters reviews those preliminary design factors which will be converted to useful hardware in the design of the rotor system.

The paragraph on rotor system kinematics discusses the blade motions to be accommodated in the detail design; in particular, the flapping, leadlag, and blade-feathering motions. Typical rotor systems accommodate these motions by means of teetering, fully articulated, or hingeless hubs. The paragraph also describes a number of methods that provide for both cyclic and collective feathering of individual blades.

The paragraph on rotor system dynamics addresses the internal stiffness and mass distributions of the rotor blades, and the relative effects of these factors on aeroelastic stability, vibration response, flutter, ground resonance, and other phenomena related to system damping and periodic forcing functions. Also covered in this paragraph are rotor responses to such transient excitations as gusts and acoustic loadings.

The discussion of blade retentions includes the various means of attaching the blades to the rotor hub. Among these are elastomeric bearings, tension-torsion straps, and antifriction bearings. Also described are auxiliary devices used at the hub to alleviate blade forces associated with blade pitch, and the lag hinge dampers used to dissipate the excess energy of the inplane motion of the blades. Blade-folding provisions, both manual and powered, are discussed as well.

The paragraph on rotor blades discusses trade-offs in blade geometry, such as airfoil section and root-to-tip taper and twist, and their relationship to the corresponding parametric analyses discussed in AMCP 706-201. Design considerations that provide for manufacturing simplicity, inservice adjustments of blade balance and track, and the blade materials and joining techniques needed to position masses and stiffnesses properly, are addressed. Also discussed are rotor system fatigue lives.

The paragraph on propellers deals generally with the design requirements for propellers and develops design considerations in the same manner as do prior paragraphs for rotors.

The paragraph on antitorque rotors reviews the knowledge gained in recent years concerning the desirable direction of rotation, the flapping freedom required, the merits of pusher versus tractor configurations, etc. The advent of "flat-rated" engine-transmission systems with high-altitude capability has placed additional demands on tail rotor control power. Additionally, the airspeeds encountered in normal operation have increased markedly, creating adverse environmental conditions for tail rotors. These, and other problems, are discussed in light of the lastest knowledge.

5-2 DESIGN PARAMETERS

The selection of rotor parameters is quite complex, as each major variable interrelates with all other variables. The basic analytical procedure for determining rotor performance are outlined in Chapter 3, AMCP 706-201. Included is a discussion of the type of parametric analysis required to optimize a rotor for a given group of performance requirements. The discussion herein supplements that description of preliminary design procedures, with emphasis upon the considerations pertinent to the detail design phase.

The parameters that are considered in connection with rotor performance include:

1. Disk loading
2. Blade loading
3. Blade tip Mach number and advance ratio
4. Number of blades
5. Blade twist
6. Airfoil section(s).

For an Army helicopter that will be required to operate in the nap-of-the-earth and in combat, compliance only with specified performance requirements will not produce an acceptable design. Additional design criteria that may or may not be defined quantitatively for a particular helicopter rotor include:

1. Maneuverability
2. Noise
3. Radar cross section
4. Damage tolerance against
   a. Striking a solid object such as tree limb
   b. Being struck by weapon fire, either solid or HE
5. Repairability
6. Fatigue life
7. Weight
8. Cost

Specific values of the performance parameters probably will have been selected during preliminary design. Compliance with the operational criteria is dependent largely upon the materials and method of manufacture, which will be selected during detail design.

The design problem initially is broken down into the requirements for hover, high-speed level flight, and high-speed maneuvering and each is discussed independently. The total problem then is considered and some approaches are offered.

5-2.1 HOVER

Selection of the optimum hovering rotor involves all the performance related parameters listed previously, with the exception of advance ratio. Hover power is divided into "induced power" (that chargeable to providing lift) and "profile power" (that chargeable to blade profile drag).

5-2.1.1 Disk Loading and Induced Power

The relationship between induced power and disk loading is described in Chapter 3, AMCP 706-201. A more extensive discussion can be found in Ref. 1.

Disk loading frequently is determined by factors other than performance. For example, a requirement for air transportability may dictate a fuselage length limitation that, in turn, will limit the rotor diameter. Rotor downwash and wake effects also are involved because induced velocity is proportional to the square root of the disk loading. Thus, the higher the disk loading, the higher the induced — or hovering downwash — velocity, which will result in increased ground erosion and greater difficulty for personnel and cargo operations in rotor wake areas.

Another effect of disk loading on performance concerns vertical drag, or download. Vertical drag results from the impingement of the wake upon the fuselage, horizontal tail, and wings (if any). The effect of vertical drag appears as an increment of rotor thrust required over and above the vehicle weight. However, evaluation of vertical drag is not precise. One of the methods described in par. 3-2.1.1.9, AMCP 706-201, employs wake velocity distributions, such as those given in Ref. 2, to obtain dynamic pressure distributions. Drag coefficients are established consistent with the body shapes in the wake, and the vertical drag is calculated by a strip analysis. One weakness of this method is the relative inaccuracy of the wake geometry described in Ref. 2. Improved accuracy of vertical drag calculations is desirable although this may require extensive development of more refined wake analyses. Model tests can be performed with scaled rotor and airframe models. However, Reynolds number effects on data from these tests can be significant. For conventional helicopter shapes (without wings) and values of disk loading, download is normally about 4-6% of the vehicle gross weight.

Hovering induced power also is affected by blade twist. This effect is due primarily to alterations in spanwise load distribution as a result of twist. Ref. 1 details twist effects for the "ideal" rotor. Twist selection for the actual rotor is covered in par. 5-2.1.5.

The "swirl", or inplane component of induced velocity is another factor that affects induced power. This inplane component frequently is omitted in the determination of the induced power of the rotor in hover or axial flight. Fig. 5-1, based on work reported in Ref. 3, shows that the swirl velocity effectively reduces the magnitude of the rotational velocity seen by the blade element. For lightly loaded rotors, this swirl component can be considered insignificant, but it can be substantial in the more heavily loaded rotors used today. In general, swirl effects should be included in hovering-power-required computations unless disk loading $w < 3.5$.

5-2.1.2 Blade Loading

The thrust produced by a rotor per unit of blade area is the blade loading BL. This parameter can be defined most simply in terms of the disk loading w and the rotor solidity σ.

$$BL = \frac{w}{\sigma}\ , \text{lb/ft}^2 \qquad (5\text{-}1)$$

where

w = disk loading, lb/ft²

σ = rotor solidity, blade area/disk area, dimensionless

More meaningful than this parameter is the mean blade lift coefficient $\overline{C}_L$. This coefficient can be used to define the aerodynamic operating point for the rotor blade airfoil sections and, therefore, to determine the drag coefficient. The profile power required HP_0 is proportional to the mean drag coefficient $\overline{C}_D$, and can be expressed as

$$HP_0 = \frac{\overline{C}_D \sigma \pi R^2 \rho (\Omega R)^3}{4400}\ , \text{hp} \qquad (5\text{-}2)$$

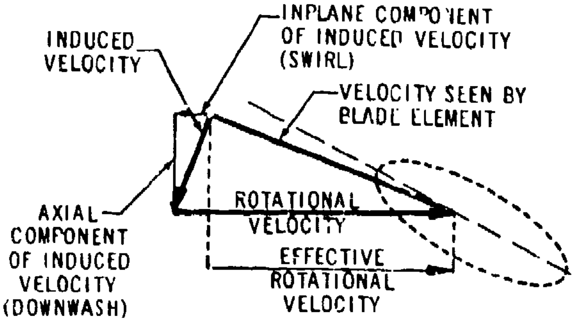

INPLANE COMPONENT OF INDUCED VELOCITY (SWIRL)
INDUCED VELOCITY
VELOCITY SEEN BY BLADE ELEMENT
AXIAL COMPONENT OF INDUCED VELOCITY (DOWNWASH)
ROTATIONAL VELOCITY
EFFECTIVE ROTATIONAL VELOCITY

Figure 5-1. Vector Diagram of Swirl in Hover

where

$\overline{C}_D$ = mean rotor profile drag coefficient, dimensionless

R = rotor radius, ft

ρ = air density, slug/ft³

Ω = rotor angular velocity, rad/sec

The mean rotor profile drag coefficient $\overline{C}_D$ is a function of the mean blade lift coefficient $\overline{C}_L$. In the "ideal" case (Ref. 1) $\overline{C}_L = 6C_T/\sigma$

$$\frac{C_T}{\sigma} = \frac{T}{\sigma \pi R^2 \rho (\Omega R)^2} \qquad (5\text{-}3)$$

and

$$\frac{C_T}{\sigma} = \frac{w}{\sigma \rho (\Omega R)^2} \qquad (5\text{-}4)$$

where

C_T = thrust coefficient, dimensionless

T = thrust, lb

ΩR = rotor tip speed, fps

For the more realistic case, where tip losses and other effects are considered, $\overline{C}_L$ can be described more accurately as $7C_T/\sigma$ (see par. 3-2, AMCP 706-201). Also, a single curve of airfoil section lift and drag coefficients c_l and c_d characteristic of the section is not representative of the actual rotor case, where Reynolds number and compressibility effects are significant. When there are spanwise variations in blade planform and/or airfoil shape, the actual values of these characteristic coefficients deviate even further from the ideal.

The relationship between $\overline{C}_L$ and $\overline{C}_D$ can be developed from flight tests of rotor configurations similar to the one being designed (i.e., similar in Mach number, twist, and airfoil sections); or it may be developed from detailed power-required calculations that include the spanwise variation of all parameters.

The optimum value of mean blade lift coefficient

$\bar{C}_L$ generally is that value corresponding to $(\bar{C}_L/\bar{C}_D)_{max}$ (Ref. 4). Further, it is preferable to obtain a blade configuration (planform, twist, and airfoil section(s)) such that the ratio of section lift and drag coefficients c_l/c_d is maximum simultaneously all along the blade span.

5-2.1.3 Blade Tip Mach Number

Performance and weight considerations generally are in conflict when efforts are made to optimize rotor tip speed. High tip Mach numbers (greater than 0.65) can be attractive from the points of view of both transmission and blade weight, but they have detrimental effects upon both power required and noise propagation. If higher tip Mach numbers are employed, tip airfoil selection becomes more critical to hover performance; thin airfoils (thickness less than 10% of chord length) are desirable, and the twist must be selected so as to maintain relatively low tip lift coefficients.

5-2.1.4 Number of Blades

Upon selection of values for disk loading, mean blade lift coefficient, and blade tip Mach number, rotor solidity has been defined uniquely. With any significant variation from a rectangular planform for the rotor blades, the effective rotor solidity σ_e should be evaluated using the method of Ref. 1.

Blade area is defined by the product of rotor solidity and disk area and can be divided among any number of blades. Propeller design experience indicates that efficiency increases with increasing numbers of blades. However, recent analytical advances, confirmed by flight and whirl test data, show that this is not true necessarily for the hovering rotor. Apparently, interblade interference can reduce the hovering efficiency of multibladed rotors (Ref. 5). The selection of the number of blades, therefore, is dependent more upon considerations of overall rotor system weight than upon aerodynamic efficiency (see par. 3-4.1, AMCP 706-201).

5-2.1.5 Twist

Selection of blade twist for the "ideal" rotor is covered in Ref. 1 In current helicopters, twist generally is linear in order to simplify manufacturing. If stretch-formed spars are used, nonlinear twist is obtained quite easily. In any event, twist selection is a function of disk loading and blade tip Mach number. The higher the disk loading, the greater the optimum twist; and the higher the tip Mach number, the greater the required twist. Twist optimization is achieved by systematic variations using detailed analytical methods.

5-2.1.6 Airfoil Sections

Rotor blade airfoil sections preferred for their aerodynamic characteristics frequently are incompatible with structural design requirements, and a compromise must be made. In general, for the hovering rotor the inboard airfoil should be of a low-drag type (at least with extensive lower-surface laminar flow). Outboard of 70% radius, compressibility effects must be considered, and the lift-to-drag ratio L/D for the airfoil section should occur at the local Mach number and angle of attack. These conditions suggest a spanwise variation in airfoil contour. If a constant airfoil is employed, its selection should be weighted toward complying with the angle of attack and Mach number conditions at or near the blade tip (outboard of 80% radius).

5-2.1.7 Hovering Thrust Capability

The capability of a hovering rotor to produce thrust can be expressed by a simple relationship. However, the agreement between the calculated and measured values of thrust produced for a given amount of power applied to rotors of practical configuration is not good. Several improvements are available and are reviewed in par. 3-2.1.1, AMCP 706-201. The method most appropriate for calculating the capability of a new rotor possibly is dependent upon the similarity to rotors for which analytical and experimental results are available. The limitations of the available methods for prediction of the performance of hovering rotors also is discussed in Ref. 5.

5-2.1.8 Guidelines

Evaluation of the "ideal" hovering rotor parameters requires systematic parametric variation involving all of the major variables given previously. This analysis is discussed in detail in par. 3-4.1, AMCP 706-201. Generalized results are given in the paragraphs that follow.

In current helicopter designs, disk loading generally does not exceed 10 lb/ft^2. Light helicopters (less than 5000 lb gross weight), tend to have disk loadings of 3-5 lb/ft^2. The medium-size helicopter, 5000-15,000 lb tends to be in the 6-8 lb/ft^2 class, and for larger helicopters disk loading is of the order of 10 lb/ft^2. Size and weight effects bias the disk loadings higher as gross weight increases.

The current emphasis on high-altitude, high-temperature design conditions results in values of mean blade lift coefficient $\bar{C}_L$ values of the order of 0.44 to 0.54 for sea level standard day conditions at primary mission gross weight.

Current helicopters have hovering blade tip Mach numbers ranging from 0.50 to 0.75. Weight and structural considerations suggest higher minimum values, and noise considerations suggest lower maximum values — resulting in a compromise design range between Mach 0.6 and 0.7.

5-2.2 HIGH-SPEED LEVEL FLIGHT

To maximize high-speed level flight performance, the same parameters are considered as in optimizing hover performance. In addition, the ratio of flight speed to rotational tip speed, or advance ratio μ, is introduced.

In high-speed design, the basic compromise is between advancing blade tip Mach number and advance ratio. The advancing tip Mach number $M_{adv\,tip}$ can be defined as

$$M_{adv\,tip} = \frac{V + \Omega R}{a} \tag{5-5}$$

where

a = speed of sound, fps
V = true airspeed, fps

At a given forward speed, decreasing tip speed decreases the amount of blade that is providing useful lift and propulsive force, because more and more of the disk is in reversed flow. This effect is accompanied, necessarily, by increased lift coefficients over the "working" part of the disk, which eventually can lead to significant amounts of stall.

The alternative approach is to increase rotor tip speed. This leads to increasingly higher advancing blade Mach numbers. Eventually, drag divergence is attained over a significant portion of the advancing blade, with increased power requirements as a result.

Increasing blade area with a given value of rotor tip speed will lower the mean blade lift coefficient and, therefore, allow operation at higher advance ratios.

Increasing twist tends to alleviate the retreating blade stall problem up to a point, but also can result in negative lift on the advancing blade tip. The latter is disadvantageous because higher lift coefficients must be achieved over the positive-lift portions of the disk in order to compensate for the negative lift on the advancing blade tip. Also, with large amounts of blade twist, drag divergence — with an accompanying increase in power required — may occur due to high negative angles of attack on the advancing blade.

It is necessary to determine the combination of tip speed, solidity, and twist that results in the minimum power required for a given speed, or the maximum speed for a given amount of power available.

Normally, the high-speed performance problem cannot be divorced from the hover and maneuver requirements. However, it is discussed as a separate problem here, where for a given amount of power available, airspeed is to be maximized. Initially, a source such as Ref. 6 can be used to determine an initial set of values for twist, solidity, and tip speed. This source requires that values for gross weight and vehicle parasite drag area first be assumed. Ref. 6 also assumes a particular airfoil section and a linear twist distribution. From this starting point, modifications of blade tip airfoil section, planform shape, and twist can be made in order to achieve speed increases up to the limits of the power available.

To increase the advancing blade tip Mach number at which drag divergence becomes critical, airfoil thickness can be reduced. For symmetrical airfoils, reduction of thickness to values of less than 12% normally results in a reduction of maximum lift coefficient. This is detrimental for the lifting capability of the retreating blade. This effect can be altered by introducing camber into the airfoil section of reduced thickness in order to maintain an acceptable value for $C_{L_{max}}$ while also attaining an increased drag divergence Mach number. However, excessive amounts of camber will result in undesirable blade pitching moments at high level-flight Mach numbers.

Sweep of the blade tip can be employed to decrease the effective Mach number, thus allowing higher values of actual advancing tip Mach number $(V + \Omega R)/a$ before the drag rise due to compressibility becomes unacceptably high. However, care must be exercised to avoid the loss of effective area and, therefore, of retreating-blade lift capability.

Nonlinear twist distributions may assist in optimizing speed for a given amount of power available. The effects must be investigated in a detailed rotor analysis by consideration of radial and azimuthal variations of angle-of-attack and Mach number. No rules can be offered; trial-and-error is the only approach currently available.

5-2.3 HIGH-SPEED MANEUVERING FLIGHT

Achievement of the desired maneuver capability at a given airspeed also may affect the selection of final values for the basic rotor design parameters. Because of increasing amounts of retreating blade stall, the higher the forward speed (for a given tip speed) the more difficult it is to achieve high maneuvering load factors.

To begin with, a static analysis is not satisfactory for determination of maneuvering flight capability. As discussed in Ref. 7, rotor pitch and roll rates are involved in both symmetrical and turning maneuvers, and can affect load factor capability significantly. These maneuver rates alter the angle-of-attack distribution obtained during steady-state flight at a given speed and rotor thrust level.

In general, to achieve high maneuver capability, blade loading in trimmed steady-state flight, i.e., normal load factor $n_Z = 1.0$, must be low. Load factor, or maneuver, capability can be related to $(C_T/\sigma)_{max}/(C_T/\sigma)n_Z = 1.0$. Thus, for a given rotor design with known $(C_T/\sigma)_{max}$, the lower the trim thrust coefficient (or blade loading), the greater the load factor capability. The other term under design control is $(C_T/\sigma)_{max}$. The major variables for maneuver capability are advance ratio, airfoil section, and twist for a given solidity ratio.

As advance ratio increases, $(C_T/\sigma)_{max}$ decreases (Ref. 8). Therefore, for a given flight speed, an increase in rotor tip speed increases $(C_T/\sigma)_{max}$. However, as for level flight, a maximum value of advancing blade tip Mach number must not be exceeded. Advancing blade shock stall can be encountered if the Mach number is too high.

The magnitude of $(C_T/\sigma)_{max}$ for a given advance ratio is a strong function of the maximum section lift coefficient near the blade tip. This is not necessarily a direct function of the section, or two-dimensional, maximum section lift coefficient $c_{l_{max}}$, because of complicating factors such as spanwise flow and oscillating airfoil effects. However, it is a good general rule that an increase in $c_{l_{max}}$ of the tip section will improve the rotor maneuvering thrust capability. Because retreating blade stall generally occurs first, the magnitude of $c_{l_{max}}$ at the retreating blade tip Mach number also is quite important. New airfoil design developments (Ref. 4) allow a tailoring of the section profile to obtain the peak value of $c_{l_{max}}$ at a desired Mach number.

Blade twist also affects maximum thrust capability by controlling the lift distribution at the retreating blade tip. Optimum twist is determined only by detailed analyses of the maneuvers including major effects such as pitch and roll rates. However, optimization of twist for the maneuver case usually is detrimental to level-flight performance, so a compromise often is required. Computation of helicopter maneuvering flight performance is discussed in more detail in par. 3-5.2, AMCP 706-201.

5-2.4 INERTIA

Rotor inertia is a major parameter in autorotative landing characteristics. Rotor angular velocity and inertia uniquely define the rotational kinetic energy of the rotor that can be used in the development of a decelerating force to arrest descent velocity in a zero-power or partial-power landing. The amount of rotational kinetic energy KE_R is defined as

$$KE_R = \frac{I_R(\Omega^2_{initial} - \Omega^2_{final})}{2}, \text{ ft-lb} \quad (5\text{-}6)$$

where

I_R = mass moment of inertia of the rotor, slug-ft²

The symbols $\Omega_{initial}$ and Ω_{final} represent the rotor angular velocities at the beginning and end of the flare maneuver, respectively. However, the determination of an acceptable value for Ω_{final} for a new rotor is largely judgmental, with little more than the designer's experience available to assure that the rotor remains controllable throughout the flare. Computation of helicopter autorotative performance is discussed in further detail in par. 3-5.1, AMCP 706-201. In par. 3-5.3, AMCP 706-201 an autorotative index AI is developed. Acceptable values of this index, and hence of the rotor inertia, also are discussed.

5-3 ROTOR SYSTEM KINEMATICS

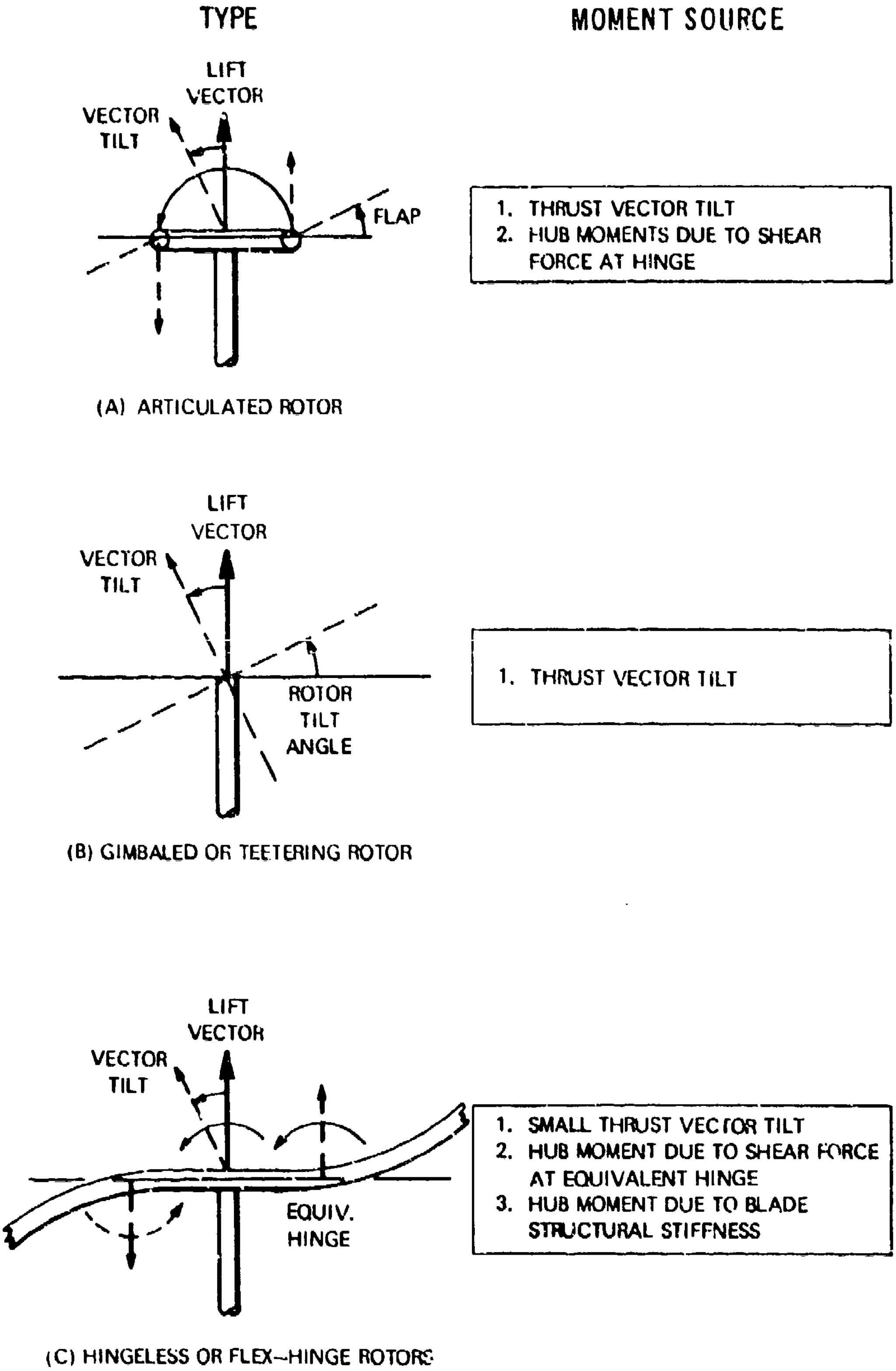
5-3.1 GENERAL

Rotor systems can be described as articulated, gimbaled (or teetering), hingeless (sometimes referred to as "rigid"), and flex-hinge.

The blades of an articulated rotor system are attached to the hub with mechanical hinges, allowing the blade freedom to flap up and down, and swing back and forth (lead and lag) in the disk plane. The blades of the hingeless rotor are attached to the hub without mechanical hinges for flapping or lead-lag motion. The flex-hinge, or strap-hinge, rotor employs a flexible structural attachment of the blade to the hub and thereby achieves a compromise between the high stiffness of the hingeless rotor and the low stiffness of the articulated or gimbaled system.

Generally, the type of rotor system will have been selected during preliminary design. In par. 3-3.3, AMCP 706-201, each of the types of rotor system is described, together with the methods by which each is controlled and in turn is used to provide control of the helicopter. The discussion includes a simplified summary of the flapping motions of a flapping (fully articulated) rotor, while the dynamics of rotor systems are described in detail in Chapter 5, AMCP 706-201. The descriptions of the several types of rotors in par. 3-3.3, AMCP 706-201, include discussions of the advantages and disadvantages of each, together with a review of the helicopter sizes for which each may be most appropriate.

The discussion of rotor systems kinematics and control which follows supplements the introductory


TYPE
MOMENT SOURCE
LIFT VECTOR
VECTOR TILT
FLAP
1. THRUST VECTOR TILT
2. HUB MOMENTS DUE TO SHEAR FORCE AT HINGE
LIFT VECTOR
VECTOR TILT
ROTOR TILT ANGLE
1. THRUST VECTOR TILT
LIFT VECTOR
VECTOR TILT
EQUIV. HINGE
1. SMALL THRUST VECTOR TILT
2. HUB MOMENT DUE TO SHEAR FORCE AT EQUIVALENT HINGE
3. HUB MOMENT DUE TO BLADE STRUCTURAL STIFFNESS

(A) ARTICULATED ROTOR

(B) GIMBALED OR TEETERING ROTOR

(C) HINGELESS OR FLEX-HINGE ROTORS

Figure 5-2. Control Moment for Basic Rotor Types

description in par. 3-3.3, AMCP 706-201, and the theoretically oriented presentation of Chapter 5, AMCP 706-201.

5-3.2 HELICOPTER CONTROL

Inflight control of the helicopter, using the rotor types cited, is provided by:

1. Moments acting upon the rotor hub
2. Tilting the resultant rotor lift vector
3. A combination of these.

The control moment source for each type is illustrated in Fig. 5-2. For the gimbaled rotor, a given rotor tilt produces a corresponding tilt of the lift vector, which, in turn, produces a control moment about the helicopter CG. An additional control moment exists in an articulated rotor as a result of the hub shear force acting at the flap hinge to produce a moment at the hub. In the case of the hingeless and flex-hinge rotors, the structural spring at the equivalent hinge provides an additional component of control moment at the hub.

The conventional method of achieving rotor control is through collective and cyclic pitch changes at the blade roots. These changes are accomplished through control linkage between the rotating blades and a swashplate (a structural element that constitutes a fixed plane that defines the blade pitch as a function of azimuth). Individual blades are mounted on spindles that provide feathering freedom for control. Collective pitch of the blades is introduced by a scissor mechanism or by raising or lowering the swashplate; cyclic pitch, required to produce a tilt of the rotor disk plane, is accomplished by tilting the swashplate.

Blade pitch changes also are made in some rotor systems by connecting the swashplate to a servo tab on each blade, or by connecting the swashplate to a servo rotor or gyro bar that in turn acts as a swashplate for the main rotor.

5-3.3 ARTICULATED ROTOR

The kinematics of an articulated rotor with an outboard lag hinge are illustrated in Fig. 5-3. Vertical motion of the pitch link in response to swashplate tilt as the blade travels around the azimuth produces pitching rotation at the pitch bearings corresponding to the cyclic pitch of the rotor. The position of the pitch bearings with respect to the blade lag freedom varies with the rotor system design. In the example (Fig. 5-3), the pitch bearing of the rotor system is inboard of the lag axis, whereas that of the CH-46 rotor is outboard of the lag hinge.

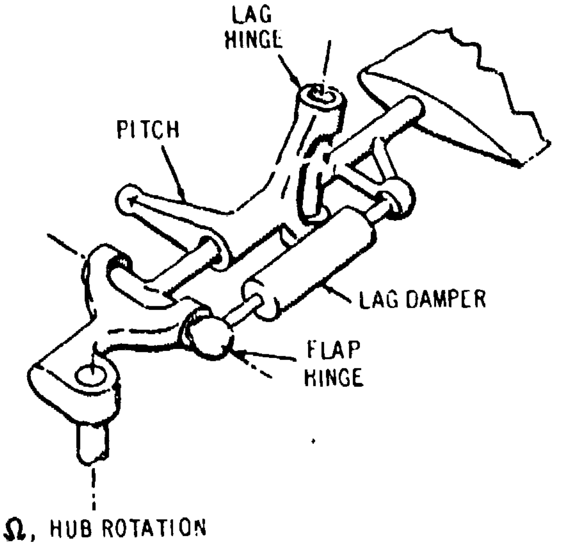


Figure 5-3. Articulated Rotor Schematic

As shown in Fig. 5-3, the lag hinge allows the blade to move, leading and lagging, in the disk plane. Lag freedom of the blade is necessary in this particular design so that the steady chordwise bending moment at the blade root is reduced. In the equilibrium lag position of the blade, the chordwise moment due to the drag loads on the blade is balanced at the lag hinge by an opposite moment due to the centrifugal force and the lag displacement of the blade. Individual blade lag dampers are required to provide energy dissipation adequate to control the mechanical instability associated with the coupled rotor/airframe system as described in par. 5-4.3 (also see Chapter 5, AMCP 706-201). The rigid-body lag natural frequency of articulated rotors usually is between 0.20 and 0.40 times the rotor speed.

Blade flapping freedom in the articulated rotor is provided by a horizontal hinge, which is located close to the rotor centerline in order to minimize the flap bending of the rotor hub (Fig. 5-3). The steady moment about the flap hinge from the centrifugal force acting through the moment arm of the blade — vertically displaced by the blade coning above the disk plane — is balanced by a moment of the same magnitude, but in the opposite direction, due to the steady lift on the blade. The natural flapping frequency of an articulated design is near resonance with the rotor shaft speed. However, aerodynamic damping in the rigid flap mode approaches 50% of critical damping, with the result that the near-resonant condition provides an acceptable design.

Coupling between flap and pitch motions is an important design consideration for a rotor control system. Generally, the rotor should be designed so that, as the blade flaps upward, the mechanical pitch angle of the blade remains the same or decreases. The kinematic coupling that varies the feathering, or pitch, angle of the blade with flapping is defined as δ_3, and the standard notation is that an increase of pitch with an increase of flapping angle is positive. Flap-pitch coupling can be introduced mechanically by a skewed flap hinge, or by radial location of the attachment of the pitch link to the pitch arm inboard or outboard of the flap hinge. Negative δ_3 generally is required to improve stability of the rotor (see Chapter 5, AMCP 706-201).

Pitch bearings outboard of the lag hinge produce a kinematic coupling that changes the blade mechanical pitch angle with blade lag motion. This configuration has the potential for unstable pitch-lag blade motion.

Fig. 5-4 illustrates the general arrangement for a rotor with coincident flap and lag hinges. This rotor has a compact arrangement of flap and lag hinges exactly like a universal joint. Flap hinges located further outboard provide greater control power, but also increase the flap bending moment at the hub. The location of the lag hinge closer inboard results in a lower lag natural frequency, with increased damping being required to prevent ground resonance.

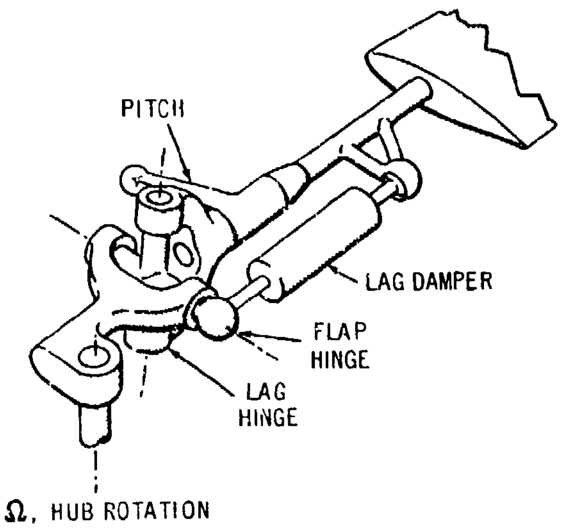


Figure 5-4. Coincident Flap and Lag Hinge Rotor

5-3.4 GIMBALED (TEETERING) ROTOR

Fig. 5-5 provides a schematic of a gimbaled rotor system; only two blades of a four-bladed rotor are shown, although any number of blades may be used. Each blade is mounted on a spindle attached to a yoke that interconnects the blades. The yoke, which defines the rotor disk plane, is gimbal-mounted to the helicopter mast (the top of the rotating shaft). In a gimbaled rotor, no cyclic pitch motion of the blades occurs relative to the spindles for any steady hovering condition, regardless of CG location or flapping relative to the mast.

The phase relationship of the one-per-rev excitation of the primary inplane bending mode is such that the blade root moments are reacted internally in the yoke, leaving the rotor hub undisturbed. The yoke structure must be stiff enough that the natural frequency of the blade cantilever mode is sufficiently greater than the rotor speed to avoid excessive amplification of one-per-rev loads.

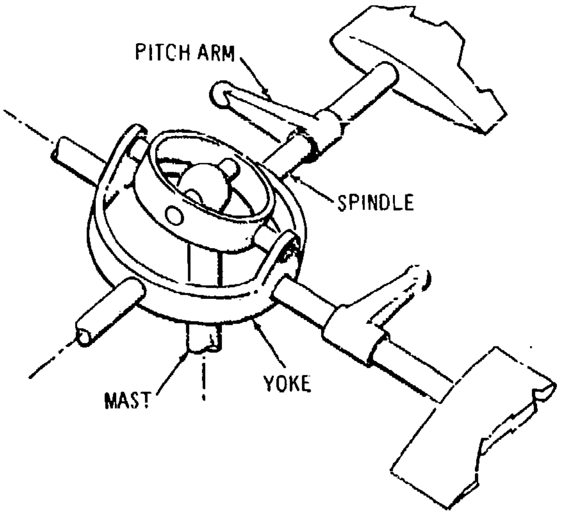


Figure 5-5. Gimbaled Rotor Schematic

For two-bladed, or teetering, rotors (Fig. 5-6), the gimbal mounting of the blades may be replaced by a single teetering hinge that allows only seesaw or flapping motion of the blades. Cyclic and collective blade pitch occurs about the yoke spindles. For rotor tilt relative to the shaft, the blades are forced by the trunnion out of their ideal position in the cone of the rotor twice each revolution. This results in a bending moment on the mast in the direction of rotor tilt that varies at a frequency of two-per-rev.

A sketch of a teetering rotor system is shown in Fig. 5-7. This rotor is connected to the shaft by a hinge, the axis of which passes approximately through the CG of the rotor in order to minimize vibratory hub and control loads. The stabilizer bar provides stability by increasing the lag time between shaft tilt and rotor tip path plane tilt. The stabilizer bar is connected to the blades through mixing levers between the sides of the bar. The inner ends of the mixing levers are pinned to the bar, the outer ends are

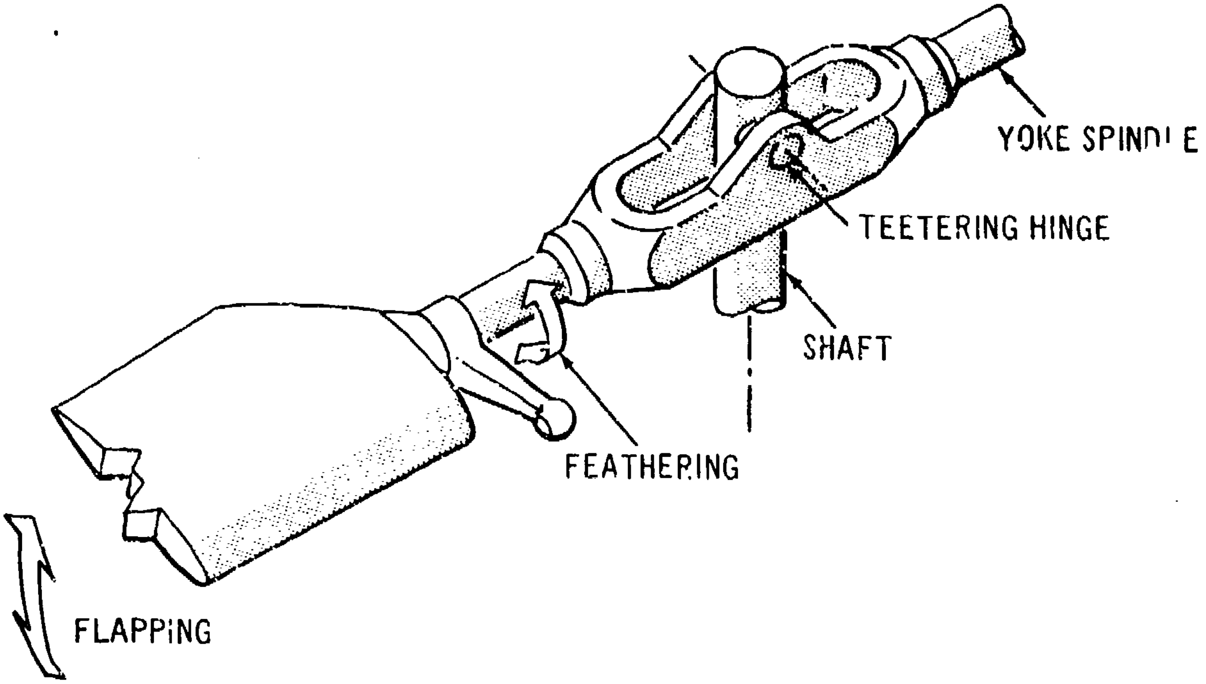


Figure 5-6. Teetering Rotor Schematic

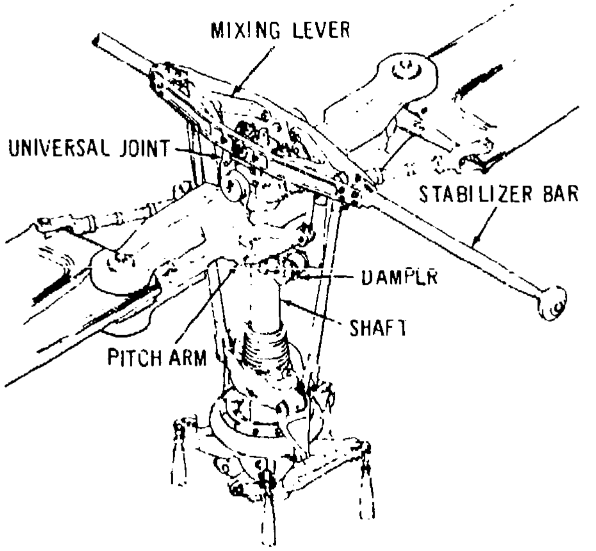


Figure 5-7. Teetering Rotor

connected to the swashplate, and the middle is connected to the pitch arms. The damper regulates the rate at which the stabilizer bar follows the tilt of the rotor shaft. An increase in damping quickens the following rate and improves the maneuverability, but also degrades the stability.

5-3.5 HINGELESS ROTOR

One blade of a hingeless rotor is illustrated schematically in Fig. 5-8. In this type of system, no mechanical means are provided to allow chordwise or flapwise displacement of the blades. The blades are cantilevered from the rotor hub, which is attached rigidly to the rotating shaft. Collective and cyclic pitch inputs for variation of thrust and control moment are made through the pitch links in response to pilot input to the swashplate. The pitch angle is changed by rotation of the blade about the feathering axis just as is an articulated rotor. Following a cyclic pitch input, the hingeless rotor responds as shown in Fig. 5-2(C), providing a control moment about the helicopter CG as a result of both tilting of the resultant lift vector and a moment acting at the hub.

The natural frequency of the first flapwise bending mode fixes the offset of the equivalent flap hinge. The dynamic characteristics, control power, and pitch and roll damping for a hingeless rotor are identical to those of an articulated rotor whose mechanical flap hinge is located at the equivalent hinge point. A hingeless rotor with a fundamental flap frequency of between 1.10 and 1.15 times rotor speed would have

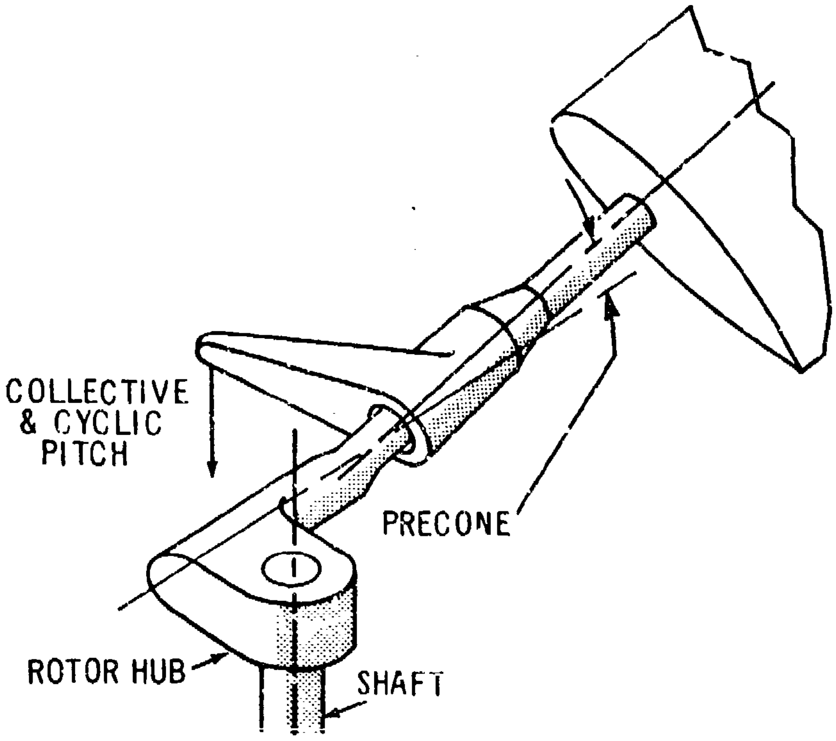


Figure 5-8. Hingeless Rotor Schematic

characteristics similar to those of an articulated rotor with a flap hinge located at 20% of the blade radius.

Precone of the blades, typical of the gimbaled and hingeless designs, permits cancellation of the steady lift moment by the moment due to the centrifugal force of the rotating blade acting through the vertical displacement of the blade above the disk plane. Kinematic coupling in a hingeless rotor is influenced by the location of the feathering axis with respect to the precone angle and to the equivalent flapping hinge location.

5-3.5.1 XH-51 Rotor System

A schematic of the XH-51 hingeless rotor, which has cantileverd blades with only a feathering degree of freedom, is shown in Fig. 5-9. A mechanical stabilizing gyro is connected by one set of links to the blade pitch arms and to the rotating swashplate by another set. Blade cyclic pitch is controlled by the control gyro, which, in turn, is controlled by the swashplate input. The blades of this rotor have high chordwise stiffness, but are provided with flap flexibility by a flat spring section inboard of the feathering axis. The blades are swept forward about 3 deg ahead of the feathering axis to locate the CG of the blade ahead of the feathering axis. Thus, the inertia forces acting through the CG produce moments about the feathering axis, producing, in turn, feedback forces at the gyro. Analysis has shown that this displacement of the CG forward of the feathering axis permits the gyro effectively and simply to provide stabilizing pitch inputs to the rotor.

In this system, upward flapping of the blade puts an up force on the pitch link, causing the control gyro to precess. The tilting of the gyro then puts cyclic pitch back into the blade at the proper phase to minimize blade flapping motion. The control gyro also provides the blade pitch angle changes necessary for stability of the helicopter. The AH-56A rotor also is of this type, but uses a door-hinge arrangement of the pitch change (feathering) bearings in order to obtain the desired high chordwise stiffness of the hub without excessive drag.

5-3.5.2 OH-6A Rotor

The OH-6A rotor system, shown in Fig. 5-10, is another type of hingeless rotor. Multiple straps transmit the centrifugal force from one blade across the hub to the opposite blade, and are flexible enough to allow both flapping and feathering of the blades.

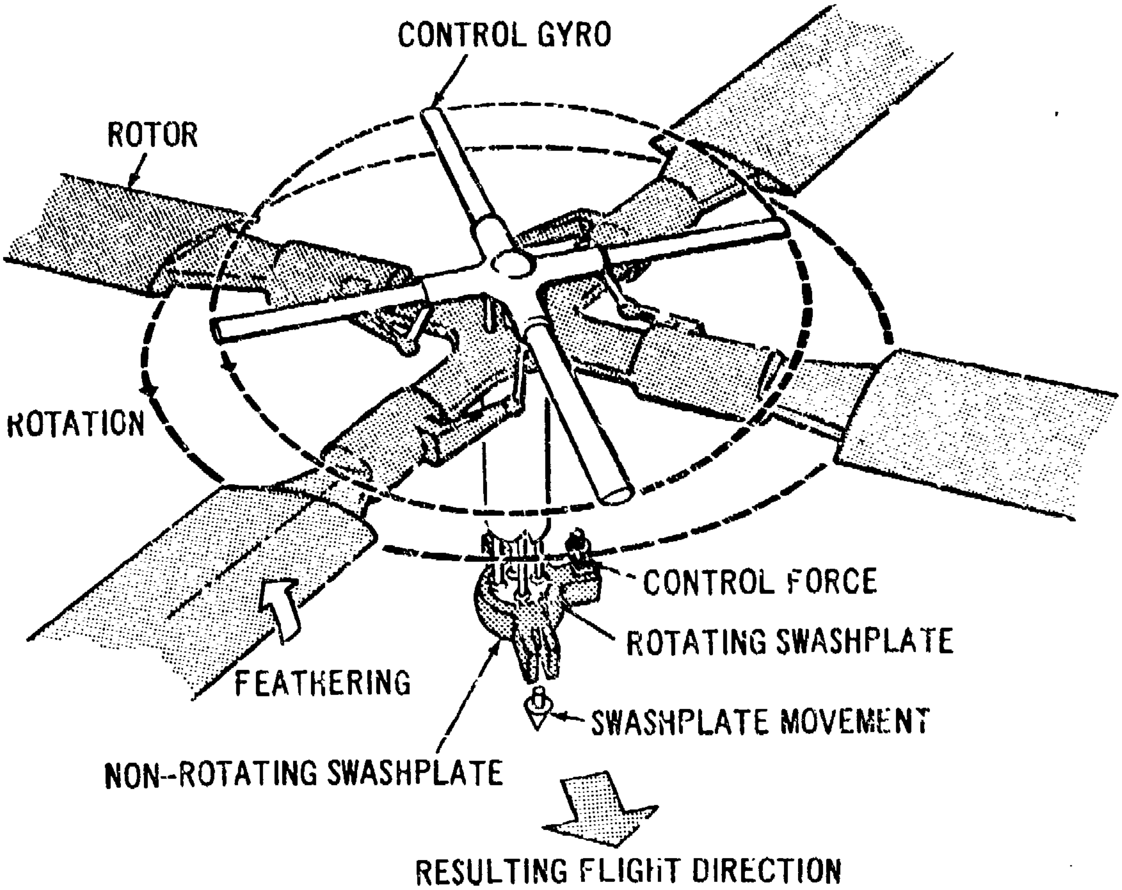


Figure 5-9. XH-51 Rotor System

Curved shoes installed at the points at which the straps are clamped to the hub prevent excessive bending of the straps at any one point. In this otherwise hingeless rotor, lag hinges are located at the outer ends of the tension-flap-torsion straps. Excessive static droop of the blades is prevented by stiff cuffs that are attached to the blades and cover the straps. Contact between the inboard ends of the cuffs and the hub limits the downward and upward flapping excursions of the blades.

5-3.6 ROTOR SYSTEM KINEMATIC COUPLING

Adverse kinematic coupling can result in various types of instability in a particular rotor system. This paragraph reviews the subject independently of the rotor type, but considers the blade and its retention system. The mechanism of rotor instability resulting from blade kinematics is examined separately under the categories of pitch-lag, pitch-flap, and flap-lag. The analytical investigation of these instabilities is outlined in Chapter 5, AMCP 706-201.

5-3.6.1 Pitch-lag Instability

Rotor blades with substantial chordwise displacement have a potential "pitch-lag" instability. The critical degrees of freedom involved are flap and lag. However, the critical design parameter is a kinematic coupling that causes a blade pitch angle change in response to lag motion, or chordwise displacement.

The mechanism of this instability is depicted in Fig. 5-11. As the blade lags (A), and if the pitch-lag coupling causes the blade pitch to decrease (B), there is a loss of lift. Downward flap of the blade occurs due to lift loss (C), and produces a Coriolis force in the lag direction (D), causing additional blade lag. Further discussion of this phenomenon may be found in Refs. 9 and 10.

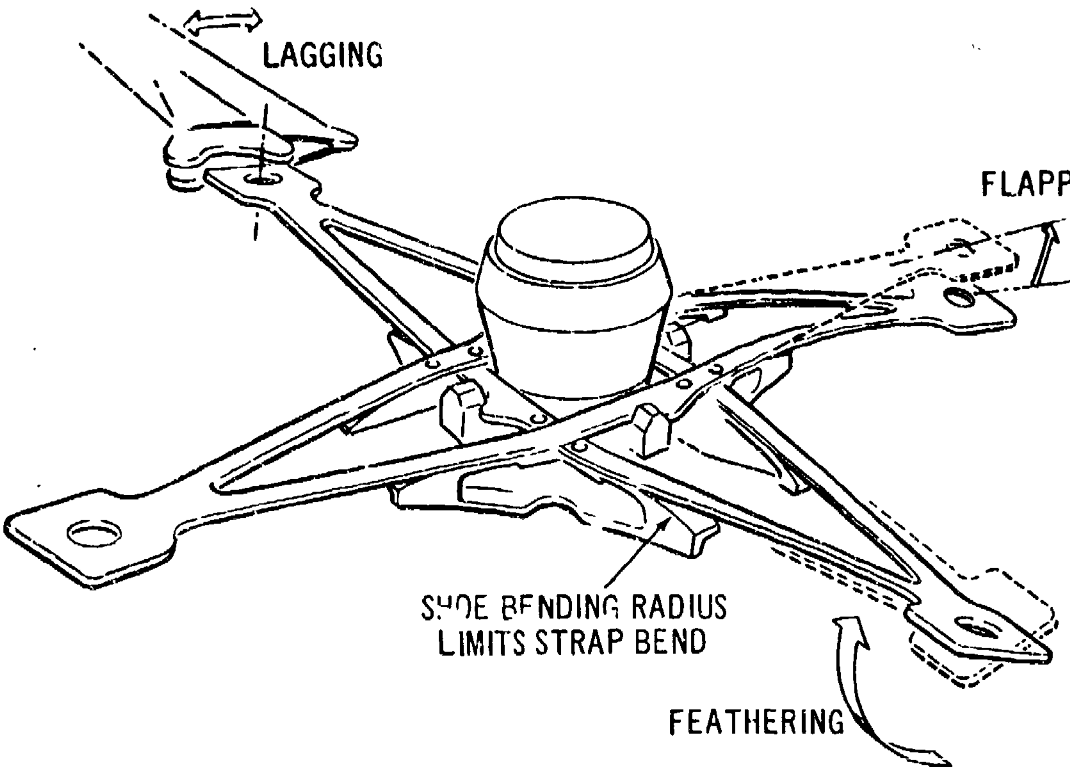


Figure 5-10. OH-6A Hingeless Rotor System

5-3.6.2 Pitch-flap Instability

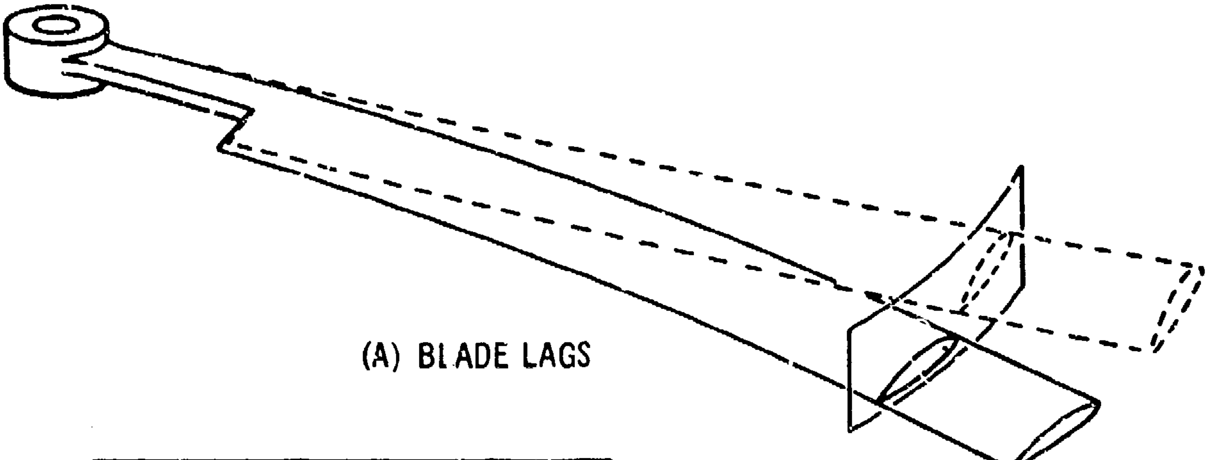
Rotor blades are subject to the same sort of dynamic instabilities as are fixed wings. For example, they are susceptible to the classical bending-torsional flutter discussed in Ref. 11. For hover or vertical flight, the major difference between the rotating and the fixed wing is the velocity variation spanwise along the blade due to rotation. The principal parameters influencing this mode in both systems are the chordwise distance between the CG of the airfoil section and the aerodynamic center, and the torsional stiffness.

In addition to torsional deflections, either flapwise or chordwise displacements of the blade deflections also may interact in such a way that a pitch-flap instability can occur. The case of a blade with flapping deflections above the feathering axis (flapping hinge outboard of the pitch bearing) is illustrated in Fig. 5-12 (A). As the blade flaps with respect to its steady-state position, the resulting Coriolis force produces a pitching moment about the feathering axis. If there is flexibility in the pitch control system, this pitching moment causes the pitch of the blade to change. Therefore, stiffness of the control system also is a significant factor in pitch-flap stability.

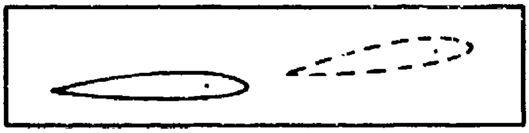
As shown in Fig. 5-12 (B), the same blade section initially is at distance $r \sin\beta_o$ from the feathering axis. As the blade rotates noseup about the feathering axis, the blade section and the lift force acting upon it are displaced backward $\theta r \sin\beta_o$, producing a nosedown pitching moment. This moment, together with those caused by variations in the inplane aerodynamic force and the centrifugal force acting through the moment arm $r \sin\beta_o$, results in coupling between the pitch and flap degrees of freedom and, consequently, affects the stability characteristics. The net pitching moment about the feathering axis changes the blade pitch angle, hence the angle of attack, by an amount that is inversely proportional to the control system stiffness. Completing the cycle for pitch-flap motion, which may be unstable, the blade section lift varies as a result of the angle-of-attack change. The lift variation causes blade flapping, which, in turn, produces additional Coriolis forces. Steady inplane bending deflection or blade sweep also can introduce pitch moments as a result of lift variations.

5-3.6.3 Flap-lag Instability

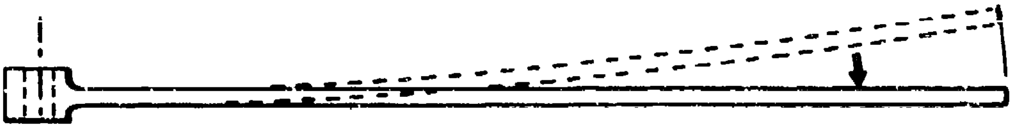
Ref. 12 describes flap-lag instabilities as a result of finite blade deflections. The conclusion from this work is that lifting rotors that have no lag hinges may, under certain conditions, be subject to limit-



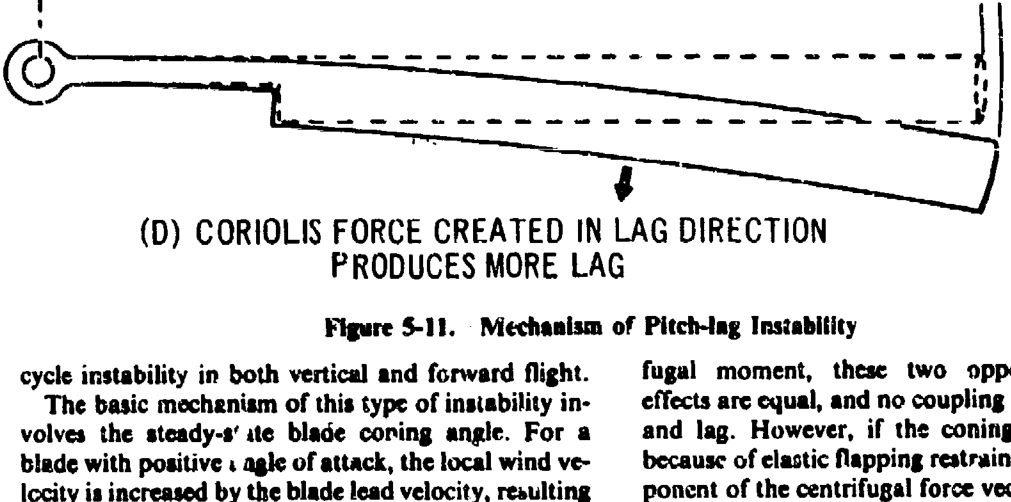
(A) BLADE LAGS



(B) PITCH ANGLE DECREASES



(C) BLADE FLAPS DOWN DUE TO LIFT LOSS



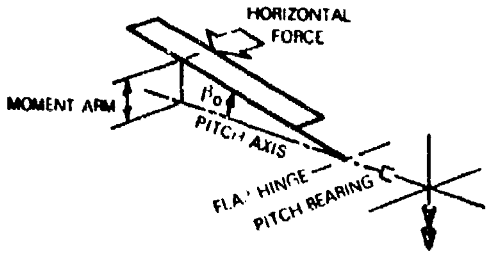
(D) CORIOLIS FORCE CREATED IN LAG DIRECTION PRODUCES MORE LAG

Figure 5-11. Mechanism of Pitch-lag Instability

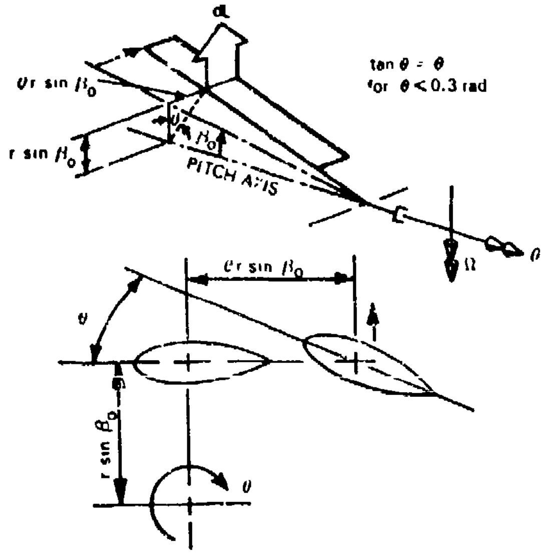
cycle instability in both vertical and forward flight.

The basic mechanism of this type of instability involves the steady-state blade coning angle. For a blade with positive angle of attack, the local wind velocity is increased by the blade lead velocity, resulting in an additional lift, as shown in Fig. 5-13 (A). The resulting incremental flap-up aerodynamic moment is counteracted by an equal flap-down centrifugal force moment. The incremental centrifugal force also results from the lead velocity or incremental rotational velocity (Fig. 5-13 (B)). If the steady coning angle is obtained by the balance of thrust and centrifugal moment, these two opposing incremental effects are equal, and no coupling exists between flap and lag. However, if the coning angle is reduced because of elastic flapping restraint, the vertical component of the centrifugal force vector is reduced and the incremental aerodynamic flapping moment exceeds the centrifugal restoring moment. In this case, the flap-up moment produced by a lead velocity of the blade can result in unstable blade motion.

In forward flight, an additional destabilizing term occurs. This is an aerodynamic flap moment proportional to the product of mean lift, lead velocity, ad-


HORIZONTAL FORCE
MOMENT ARM
β_0
PITCH AXIS
FLAP HINGE
PITCH BEARING

(A) CORIOLIS MOMENT FROM BLADE FLAP


dL
$\dot{\theta}r \sin \beta_0$
tan θ = θ
for θ < 0.3 rad
r sin β_0
β_0
PITCH AXIS
Ω
$\dot{\theta}r \sin \beta_0$
θ
r sin β_0
θ

(B) PITCH MOMENT FROM LIFT OFFSET

Figure 5-12. Pitch-flap Coupling of Rotors

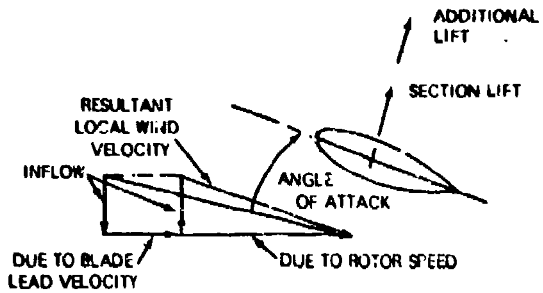
vance ratio, and the sine of the azimuth angle. A subharmonic instability of blade motion can occur at moderate advance ratio, e.g., $\mu = 0.4$. This type of instability can be suppressed by an adequate amount of inplane damping. Ref. 13 suggests that, if the blades are sufficiently flexible in torsion, pitch-lag coupling can be used as an additional means of suppressing lag motion instability.

5-4 ROTOR SYSTEM DYNAMICS

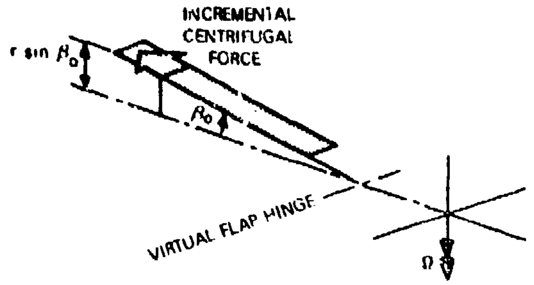
5-4.1 OSCILLATORY LOADING OF ROTOR BLADES

The oscillatory loading of rotor blades is important for both vibration and fatigue.

Rotors usually are designed by extrapolation from previously successful designs. This can be expensive because many rotors have been designed, built, and flight tested without significantly extending the useful envelope of existing rotors in relation to loads and vibrations. Rotor design considerations — such as fatigue spectra (based on mission profiles and maneuver frequencies), details of bearing stress, peak stress, and stiffness of complex shapes — remain difficult. However, other considerations, e.g., static deflection and gross vibratory response, are becoming fairly well defined and calculation procedures are available.


ADDITIONAL LIFT
SECTION LIFT
RESULTANT LOCAL WIND VELOCITY
INFLOW
ANGLE OF ATTACK
DUE TO BLADE LEAD VELOCITY
DUE TO ROTOR SPEED

(A) ADDITIONAL LIFT DUE TO BLADE LEAD VELOCITY


INCREMENTAL CENTRIFUGAL FORCE
r sin β_0
β_0
VIRTUAL FLAP HINGE
Ω

(B) INCREMENTAL CENTRIFUGAL FORCE DUE TO BLADE LEAD VELOCITY

Figure 5-13. Mechanism of Flap-lag Instability

5-4.1.1 Hypothetical Consideration of Rotor Vibratory Loads

The procedure for development of preliminary rotor design into a detail design requires continuous iterations by aerodynamics, dynamics, stress, and fatigue specialists. The integration of computed vibratory rotor loads into the detail design is described best by a generalized method toward which the industry is moving.

Ideally, the following program, which hinges on the existence of a complete aeroelastic flight-vehicle analysis, could be used to link the preliminary and detail designs:

1. A maneuver spectrum should be defined, based upon previous helicopter experience and the mission requirements of the new helicopter.

2. The maneuver spectrum should be "flown" analytically with the preliminary design helicopter, and loads and vibrations should be computed at the

locations where flight test instrumentation will be placed. These calculations should cover the complete ground-air-ground cycle, including landing, taxiing, and towing. In such a sequence, all of the static, transient, and oscillatory stresses and vibrations should be computed.

3. The stress and vibration calculations should be reviewed to determine where design optimization should start. If the performance, stresses, vibrations, handling qualities, and other rotor characteristics meet but do not exceed the specification requirements, then the preliminary design may represent an optimum configuration. If the helicopter greatly exceeds requirements in one or more of these areas, a weight penalty generally results. Design changes should be made and at least a portion of the analysis based upon the maneuver spectrum should be recomputed. If the design fails to meet the specifications in certain areas, the design must be improved and a portion of the maneuver spectrum analysis should be recomputed to demonstrate compliance.

At the end of the iteration, the detail design should start with a set of loads for static and dynamic stress calculation, and a reasonable assessment of vibration.

The detail design of the rotor system should be initiated by making dimensional drawings of all parts. The designer should follow the preliminary design as closely as possible. However, compromises often may be required, some of which can alter the dynamic characteristics significantly. As the design progresses, section properties, weights, inertias, and stresses should be computed for the detail parts. As soon as the first design iteration is completed, the rotor natural frequencies, loads, and stresses should be recomputed. These stresses should be used as the basis for a fatigue analysis, which should be guided additionally by experience from the test histories of similar parts and idealized material samples. Depending upon the outcome of this first design iteration, the design should be accepted or another iteration started.

Significant advances have been made with helicopter flight simulations. For example, Ref. 14 describes a recently developed analytical tool for computer "flight testing" of VTOL designs. Steady-state flight, maneuvers, and gust response effects are included so that required configuration changes can be made readily during the preliminary and detail design stages. This objective is achieved by detailed representation of the aircraft, including rotors, wings, auxiliary propulsion, and control systems. Complete blade element analysis of the main rotor(s), tail rotor, and propeller(s), as applicable, are performed through a maneuver, and are based upon the instantaneous aerodynamic and dynamic environments. Time histories of rotor blade loads and bending moments are calculated. The basic equations and programming procedures are presented and discussed in Ref. 14. The representation of airframe and rotor parameters, the types of maneuver inputs, and the available output formats also are discussed in detail. Typical case studies are given. This analysis is capable of evolving into a generalized procedure with the addition of details such as elastic pylon and fuselage, and a fully aeroelastic rotor. Complete documentation of this particular method can be found in Ref. 15. Comparable methods have been developed by other contractors.

5-4.1.2 Oscillatory Load Design Considerations

Oscillatory loads are a major factor in rotor design; but the calculation of oscillatory loads is not yet sufficiently accurate for life prediction and design assurance. Therefore, rotor design is guided by calculated natural frequencies, static loads, and factored oscillatory loads. The final demonstration of design adequacy comes from flight and fatigue testing.

5-4.1.2.1 Rotor Oscillatory Load Calculation

Most current procedures for computing rotor natural frequencies and loads are based on Myklestad's development of the dynamics of a rotating beam (Ref. 16) and on simplified, two-dimensional aerodynamics (Ref. 1). Typically, such analyses can be used to compute natural frequencies and airloads separately; then the two analyses are combined to compute the forced steady-state response. Many versions of this procedure have been developed. A detailed description of one adaptation, which has been used for designing two-bladed rotors for nearly a decade, is given in Ref. 17.

5-4.1.2.2 Drawing Board Phase

As noted previously, the drawing board phase of the rotor detail design is an iterative procedure and typically involves several groups of engineering specialists. The procedure is best explained by briefly describing the functions of several elements:

1. The aerodynamics group sizes the rotor, develops the blade contours, and helps determine the static load spectrum during the preliminary design phase. Often this work carries over, with little change, into the first step of detail design.

2. The rotor design group starts the design iteration by laying out the preliminary rotor design and developing dimensional drawings for the detail parts. The designers must be cognizant of the stresses and

vibrations resulting from rotor oscillatory loading. Other considerations requiring attention include dynamic stability, weight control, bearing applications, mechanical function, value analysis, materials, bonding, and manufacturing processes, and the designer must rely upon specialists in many of these fields.

3. The rotor stress group begins preliminary design with the development of section properties. These properties, in turn, are used for determination of the nominal stresses that result from the highest combinations of centrifugal force (CF) and maneuver thrust conditions. The bending moments for these conditions are computed in the steady-load portion of the rotor-load analysis (par. 5-4.1.2.1).

Fatigue may be considered by an empirical method that provides a simple and reliable account of load spectrum shape and severity in relation to component fatigue strength in the initial stages of rotor component design (Ref. 18). This method uses the flight loads calculated for maximum level flight speed, the condition that usually produces the highest continuous (nontransient) alternating loading. A factor is applied to the calculated loads to obtain design loads that will produce a satisfactory fatigue-life structure. This factor is a function of the material S-N curve, the maneuver spectrum severity, the loading frequency, and the fatigue life required by the helicopter system specification. The design curves for this factor are based on an analysis of several flight load surveys.

Natural frequencies also are calculated for the rotor as defined at this stage of the design. Changes in section properties and/or concentrated weights are used to produce a frequency distribution that avoids principal resonances.

During the detail design phase, proper rotor natural frequency placement insures the lowest possible oscillatory rotor loads; and rotor and fuselage frequency placement, isolation, and superposition insure the lowest possible fuselage vibration. The rotor characteristics must be such that low-frequency vibrations of the fuselage are avoided. High-frequency vibrations generally are not so critical and may be corrected during the flight test phase.

The aim of this continuous iteration is a design optimized, or balanced, with regard to performance, function, strength, life, weight, and vibration. When an acceptable balance has been achieved, the design is considered to be adequate, the drawings are completed, and rotor components are manufactured. However, further changes usually result from the flight and fatigue tests.

5-4.1.2.3 Flight Tests

The design calculations made during the drawing board phase insure that a rotor meets the static strength criteria and has an infinite life for the low-cycle high-stress variations associated with the ground-air-ground cycle. However, because of the superimposed high-cycle, low-stress oscillations, many rotor and control system parts will have finite fatigue lives. Design changes often are required to insure that the adequate component fatigue lives of all components are adequate. Flight and fatigue testing is required to determine these lives.

Prior to the flight test phase, the strain gage instrumentation, the flight load survey tests, and the data reduction format *shall* be specified. The measured loads are used, along with the approved maneuver and frequency-of-ocurrence spectra, to determine the fatigue lives of all fatigue-critical components. The required flight load survey tests are outlined in Chapter 8, AMCP 706-203.

The calculations made during the drawing board phase are effective in guiding the design so as to avoid excessive loads and vibrations at the lower frequencies; however, additional tailoring of the rotor and fuselage usually is required during flight test to minimize high-frequency loads and vibrations. Measures taken include optimizing the amount and location of the concentrated blade weights; changing blade stiffness, such as through use of trailing edge stiffeners; optimizing the pylon suspension parameters, and detuning the fuselage by varying stiffness and the location of certain concentrated weights, e.g., the battery.

5-4.1.2.4 Fatigue Tests

The determination of fatigue lives of components is discussed in Chapter 4, AMCP 706-201. The requirements for fatigue testing of critical components are given in Chapter 7, AMCP 706-203. The parts to be tested, the number of samples, and the method of loading *shall* be specified. The samples are cycled to failure, or for a prescribed number of cycles at several stress levels. The results usually are plotted as an S-N diagram. Basic information on the need, methods, and interpretation of fatigue testing in the helicopter industry is presented in Refs. 18, 19, and 20. By use of the fatigue test data and the frequency-of-ocurrence spectrum, the fatigue lives of the critical parts *shall* be determined by a method acceptable to the procuring activity (Chapter 4, AMCP 706-201). If one or more parts have lives shorter than required, the flight envelope may be restricted or the parts may have to be redesigned and requalified. Failure to comply with the fatigue-life requirement of the helicopter system specification usually will result in a penalty being applied to the contractor.

5-4.2 AMPLIFICATION AND NATURAL FREQUENCIES

A summary of the field of rotor vibrations is contained in Ref. 21. Although highly mathematical, this is a valuable reference for those seeking both general and specific information on rotor loads and vibration. The paragraphs that follow discuss specific amplification and natural frequency information.

Fig. 5-14 presents natural frequency information for a two-bladed, teetering rotor, but the method of presentation and the design information are general enough to warrant a detailed discussion. Two graphs are made — one labeled collective mode and the other cyclic mode. These names stem from the types of forces and motions caused by the collective and cyclic controls. For the two-bladed rotor, collective modes are excited by even harmonic airloads while cyclic modes are excited by odd harmonics.

The ordinates are natural frequencies; the abscissas are rotor speeds; the vertical lines mark the normal rotor speed range, and the radial lines are harmonic excitation lines. Coupled and uncoupled natural frequencies are indicated by curves with and without coding symbols. The applicable mode shapes are shown schematically. Flapwise is normal to the plane of rotation; inplane is in the plane of rotation. Uncoupled means without blade torsion and feathering; coupled modes include these degrees of freedom. The collective plot contains even harmonic excitation lines, while the cyclic plot contains odd harmonic lines.

The circled numbers identify three collective and four cyclic modes. All natural frequencies increase with rpm, but the flapwise frequencies increase much faster than the inplane frequencies (such as the third cyclic mode) because the centrifugal stiffness is a larger percentage of the total stiffness in the flapwise direction. The collective modes show very little intermodal coupling with twist and pitch, while the cyclic modes show strong coupling.

These fan plots are the primary design guide for rotor dynamics. Myklestad's development (Ref. 16) made it possible to compute the variation of the uncoupled frequencies with rpm. Finally, the method of Ref. 17 made it possible to compute the intermodal coupling due to twist and collective pitch. Accurate computation is necessary because of the fine tuning required to avoid resonance through the eighth harmonic of excitation, especially for compound and composite aircraft rotors having wide ranges of operating rpm.

Transient resonances such as Points A, B, and C in the collective mode plot (Fig. 5-14) cannot be avoided, but they cause no significant load or vibration problems. Such points are used for test verification of the calculated natural frequencies. Without such verification to confirm or adjust the fan plots during the first ground runs of a new rotor, several variations of tuning weights might be required to establish correct trends at operating rpm, resulting in considerable cost and loss of time.

The main objective of calculating rotor natural frequencies is to prevent coincidences (resonances) such as Point A from occurring in or near the normal operating rotor speed band. Such steady-state resonances are not catastrophic, but often produce stresses high enough to reduce fatigue life significantly, as well as fuselage vibrations that may require restriction of the flight envelope for comfortable operation.

Generally, resonant amplification factors cannot be computed with sufficient accuracy for design purposes. Once amplification factors have been determined for an existing rotor, fairly accurate predictions of loads can be made for variations of parameters; however, extrapolating these values to a new rotor involves considerable risk. The damping factors used in the typical rotor-load computing program are empirical and the value of the predicted loads necessarily is low.

Progress in evaluating damping mechanisms has had to wait for improvements in both dynamic and aerodynamic computing methods because an observed level of response at resonance may be due either to a low level of force or to a high level of damping. Some damping concepts associated with the fan plots of Fig. 5-14 are:

1. All modes contain structural damping on the order of 0.50-1.00% of critical, which is relatively insignificant.

2. Flapwise modes such as the ones shown in the collective mode plot are strongly damped aerodynamically, while inplane modes are not.

3. Modes with strong intermodal coupling, such as those shown in the cyclic mode plot, have frequencies that vary significantly with blade pitch; thus, they benefit from an effective damping mechanism referred to as cyclic detuning. The cyclic variation in pitch limits the resonance to a few degrees of azimuth, preventing steady-state resonant amplification. This damping source is at least as significant as the flapwise aerodynamic damping.

5-4.3 GROUND RESONANCE

The helicopter *shall* be free of mechanical instability at all rotor speeds and operating conditions (takeoff, flight, landing, and taxi) regardless of the type of landing gear; under the entire range of gross

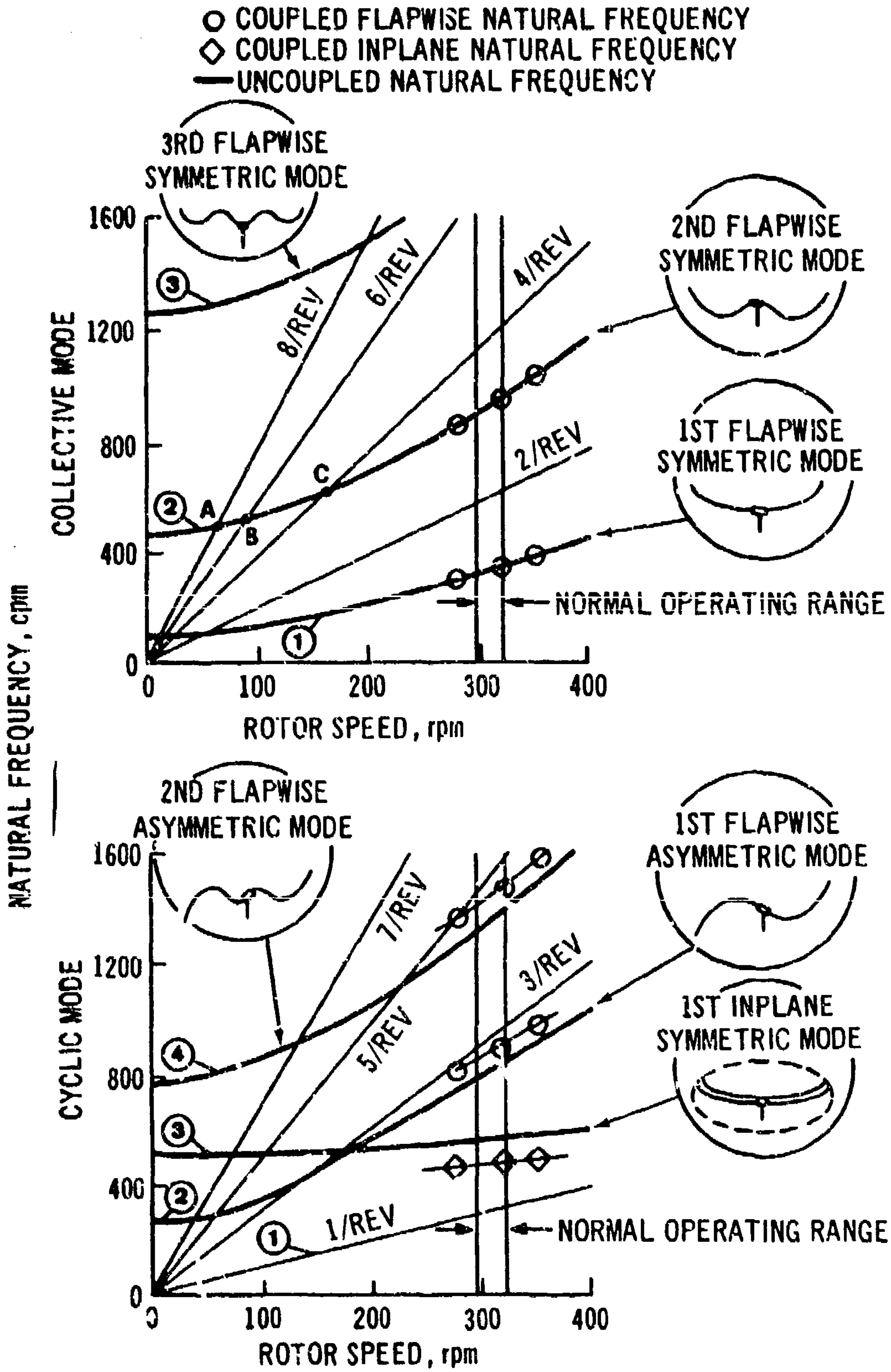


Figure 5-14. Typical Plots of Rotor Natural Frequency vs Operating Speed

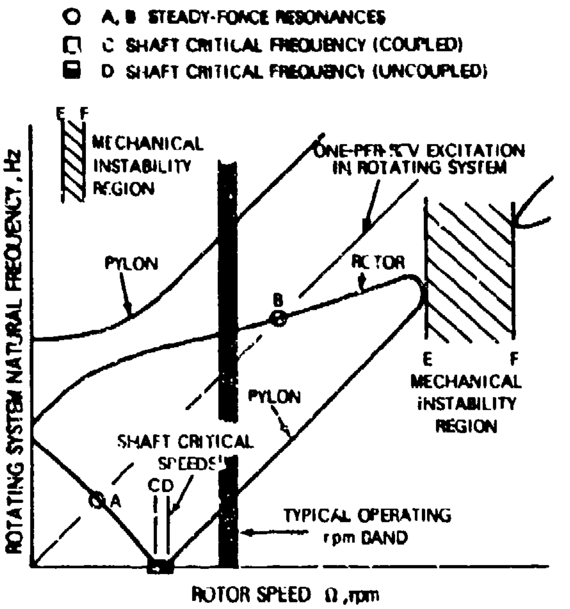


Figure 5-15. Single-degree-of-freedom Coleman Plot

weight conditions, and throughout the extreme temperature range specified by the contract. An extensive series of tests is specified by Chapter 9, AMCP 706-203. These tests can be relaxed, or eliminated, provided that it can be demonstrated to the satisfaction of the procuring activity that the helicopter is free of mechanical instability.

Occurrence of the instability problem usually is a result of a primary failure in the system. Additional information is contained in Refs. 22, 23, and 24. The discussion that follows supplements the analytical review given in Chapter 5, AMCP 706-201.

5-4.3.1 Two-bladed Rotor With Hinged Blades

The analysis of the mechanical instability of two-bladed, hinged rotors differs from that of rotors with three or more blades. In the original analysis (Ref. 22) it was assumed that the support system below the rotor — including the fuselage, pylon, and landing gear — could be resolved into a single-degree-of-freedom support in each of two perpendicular (fore-and-aft and lateral) directions. This necessitates the computation of the effective mass and effective lateral and fore-and-aft hub springs in the rotor plane for each mode of vibration of the support. It was further assumed that each support system mode can be treated independently.

The analysis is used to compute the coupled natural frequencies of the model, which has elements of both fixed and rotating systems; thus, the frequencies can be calculated and plotted in either

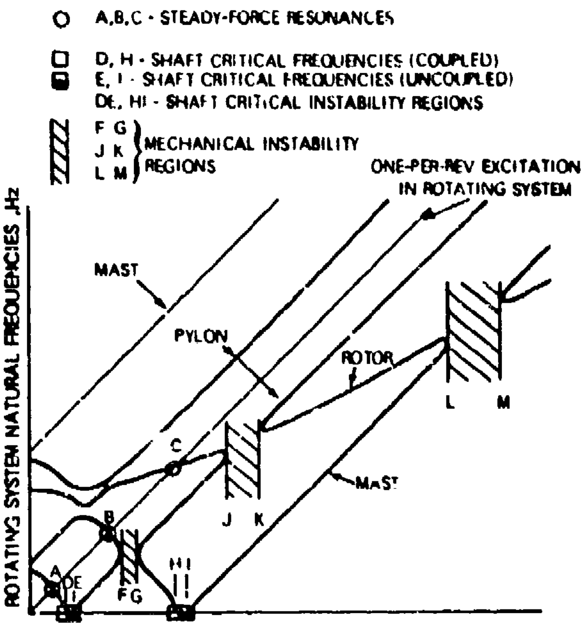


Figure 5-16. Two-degree-of-freedom Coleman Plot

system. Such graphs (Coleman plots) present both frequency and stability information according to conventions detailed in Ref. 22.

Potential instabilities — A through F — indicated in the frequency plot in Fig. 5-15 are classified in three catagories in accordance with the mode of excitation:

1. Points A and B: ordinary resonances (resonance with steady force) that occur at one-half the support system coupled natural frequency for any support mode (Point A) and at the coupled rotor-pylon natural frequency (Point B)
2. Points C and D: shaft-critical speeds (excitation by an out-of-balance force) that occur at the coupled (Point C) and uncoupled (Point D) support system frequencies
3. Points E and F: coalescence of pylon and blade frequencies, marking boundaries of the mechanical instability range (self-excited modes due to negative damping forces). These points always occur at rotor speeds greater than the first inplane rotor natural frequency (Point E).

The analysis of Ref. 23 includes two degrees of freedom of the support system in each direction. The rotational system frequency plot generated by this analysis is given in Fig. 5-16. It shows an additional steady force resonance B, an additional shaft critical range H-I, and two additional instability ranges F-G and L-M.

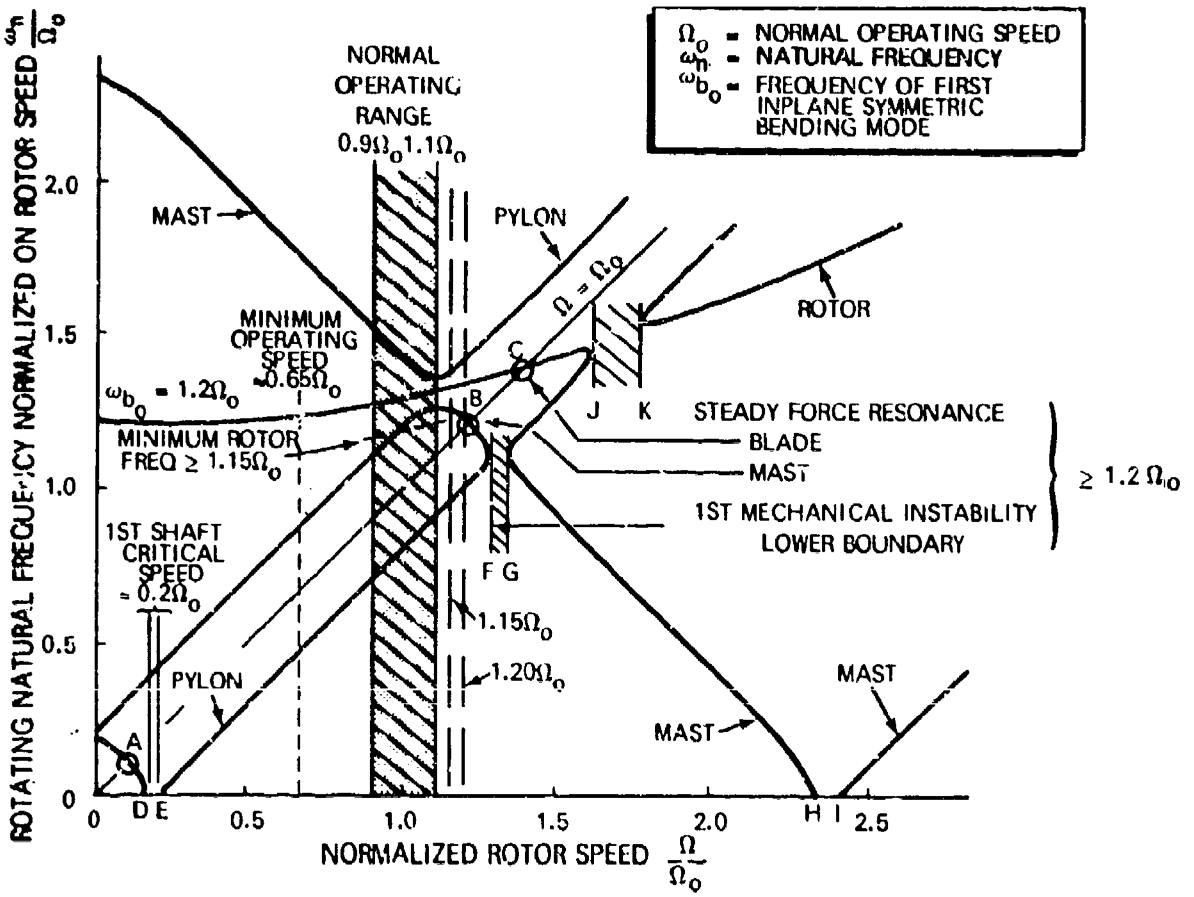


Figure 5-17. Two-degree-of-freedom Coleman Plot Showing Satisfaction of Minimum Frequency Criteria for Two-blade Hingeless Rotor

5-4.3.2 Two-bladed Rotors Without Hinges

It can be shown that two-bladed hingeless rotors can be approximated by rotors with effective hinges; thus, the analyses discussed in par. 5-4.3.1 also can be applied to this case. Because of the location of the rotor natural frequencies, the ground resonance requirements can be met by satisfying the following criteria (see Fig. 5-17):

1. Sufficient damping must be provided by the landing gear and undercarriage structure of the airframe to remove the instabilities (such as D-E) associated with the rigid-body degrees of freedom of the airframe on its landing gear. The exact amount of damping required to stabilize the region D-E is not known, but it is probably less than 3% critical. For configurations with skid gear and pylon isolation, the inherent damping of the system always is sufficient to prevent instability.

2. The mast frequency in the rotating system (Point B) *shall* be no less than 1.2 per rev. This guarantees that the mechanical instability associated with the coalescence of the two support system frequencies will be sufficiently removed from the operating range.

3. The first inplane natural frequency of the rotor at high collective pitch *shall not* be less than 1.15 times the rotor speed. This assures that the mechanical instability associated with the coalescence of the rotor natural frequency and the lowest support system mode (Point C) will be above the overspeed operating range (110% rotor speed), and prevents excessive response to steady-state one-per-rev excitation ($\Omega = \Omega_o$ line).

5-4.3.3 Multibladed Rotors

The analyses discussed in pars. 5-4.3.1 and 5-4.3.2 are special cases of the classical analysis given by Coleman (Ref. 25) for multibladed rotors. For rotors with three or more blades, there is no preferred direction of motion associated with out-of-balance forces; the stability ranges given in Fig. 5-16 by D-E and H-I are reduced to simple resonances of the support modes with the operating speed of the rotor. Thus, the number of degrees of freedom is reduced, simplifying the equations of motion. However, the analyses of the mechanical instability ranges J-K and L-M in Fig. 5-16 are identical to the two-bladed case and must be treated with the same considerations.

5-4.4 FLUTTER ASSESSMENT

The other types of potential rotor instability are divergence and flutter. The discussion that follows supplements the analytical review given in Chapter 5, AMCP 706-201.

When the developing aerodynamic forces simply overpower the elastic constraints and the motion exceeds some preselected bounds, i.e., goes unstable, divergence has been reached. Flutter usually involves a change in and coalescence of two or more system natural frequencies because of dynamic or aerodynamic effects, and the coupling of oscillatory motion of the lifting surface with the airstream in such a way as to derive energy from the airstream to increase the motion.

The first formulation of the flutter problem was published in 1934 and subsequently republished as Ref. 26.

Each potential flutter problem is related to modal couplings that are configuration-oriented, and the number of such couplings is very large. Further, in each specific configuration, the stability equations involve a large number of parameters whose meaning and measure are only made clear by a rather precise analytical diagram or model. Thus, a specific, rather than general, method has evolved. No method has yet been devised for writing meaningful specifications and simple instructions for designers for the prevention or avoidance of these instabilities. Notable attempts toward simplification are given in Ref. 27 for fixed-wing aircraft and Ref. 28 for helicopters, and a recent attempt at ordering and classifying is given in Ref. 29.

5-4.4.1 Current Criteria

The most comprehensive list of static and oscillatory aeroelastic instabilities compiled to date for helicopters is presented in Ref. 29. Several specific configuration-oriented problems are named, and formulas are given for determining static and oscillatory stability boundaries. This list could be extended to form the basis for a usable specification dealing with the aeroelastic phenomena basic to helicopters.

5-4.4.2 Design Considerations

To discuss design considerations, it is convenient to classify the applicable aircraft configurations as helicopter and compound; and under each to discuss fixed and rotating system divergence and flutter, or static and oscillatory aeromechanical instabilities.

5-4.4.2.1 Helicopter

5-4.4.2.1.1 Fixed System

Historically, relatively little attention has been given to fixed-system divergence and flutter for helicopters, as other design and operational requirements such as static strength, fatigue life, and operating speed have precluded divergence and flutter.

5-4.4.2.1.2 Rotating System

A number of rotor aeromechanical instability problems were encountered in the period before 1960. Solutions usually were worked out by trial and error long before they were understood mathematically. Included were problems such as weaving, pitch-flap, pitch-lag, and pitch-cone instabilities, and stall and binary flutter. Of these only one of recent occurrence is stall flutter, which has been actively researched (Refs. 30 and 31). This work has provided an understanding of the problem and the ability to predict the stall flutter boundary with reasonable accuracy.

The basic design changes that solved most of these problems were overbalanced blades, torsionally stiff blades, and, with the advent of hydraulic boost, increased control-system stiffness.

During the design of conventional rotors, the current practice is to forego elaborate calculations. The only mandatory check for main rotors is that the chordwise location of the effective CG of the blade be forward of 25% chord, and preferably forward of 24% chord. For unconventional designs, serious consideration should be given to detailed quantitative analysis. The list of known problems should be checked to see if an analytical solution is available, preferably a method that has been checked against experimental results.

5-4.4.2.2 Compound

5-4.4.2.2.1 Fixed System

In the development of compound helicopters, considerable attention has been given to divergence and flutter due to the extension of the speed range beyond that of conventional helicopters. Conventional practices, such as those outlined in Chapter 5, AMCP 706-201, are adequate. In one known case, significant buffeting of the vertical fin was encountered due to the impingement of disturbed air from the hub and pylon. The problem was solved by cleaning up the flow.

5-4.4.2.2.2 Rotating System

A great deal of research has been accomplished during the extension of the helicopter speed range by compounding. Ref. 32 is a good summary of the early work in this area. This work showed that increasing advance ratios and blade tip Mach numbers require progressive unloading of the rotor and reduction of rotational speed — even down to zero. The associated dynamic phenomena are continuous and trackable until zero rotor speed is approached.

At high advance ratios, thrust and flapping control are difficult because of high sensitivity to gusts. The principal dynamic problem is limit-cycle flapping instability. This instability produces both harmonic and nonharmonic flapping, the latter being visible as a weaving of the tip path plane. The history of this problem is sketched and the picture clarified in Ref. 33, which shows two azimuthal regions of instability, one on the advancing side due to negative spring rate, and one on the retreating side due to negative damping. Measures for stabilizing both are discussed.

5-4.5 ACOUSTIC LOADING

Conventional rotors, because of their relatively low disk loadings, experience negligible acoustic loading. Airflow over the blade surfaces can reach sonic velocity locally and momentarily produce a shock wave. Aerodynamic loading variations occur, primarily at the blade tip, which generate the characteristic acoustical signature referred to as blade slap. Rotors have operated in such an environment for years with no evidence of structural fatigue caused by either aerodynamic or acoustic loading. Blade slap has been shown to be drag-related, hence, the acoustic loading occurs inplane along the axes of maximum strength of the blade skin.

Proprotors and propellers located in close proximity to either a fuselage or a wing can experience some degree of acoustic loading. However, the rotor air pressure impingement on the fuselage or wing structure is an order of magnitude more significant than acoustic loading of the rotating system.

Tail rotors located close to a tail boom experience aerodynamic loading caused by the partial blockage of airflow and the interference by the fin with the blade surface pressure field. As with conventional main rotors, there is no evidence of measurable acoustic loading. Ducted fans used as antitorque devices, because of their relatively high disk loadings, may produce greater acoustic loading than conventional tail rotors. However, the duct surrounding the fan experiences higher loadings than do the fan blades.

5-4.6 GUST LOADINGS

The need for meaningful gust loading specifications has increased with the development of high-performance helicopters and compounds. The first definitive work was Ref. 33, in which the nature of the problem was elucidated and some solutions were offered. A more precise treatment required more sophisticated analytical methods. One such method (an extension of the rotorcraft flight simulation method discussed in par. 5-4.1.1) was developed and used for an extensive study of the problem (Ref. 34). This reference also reviews the state of the art. The problem of helicopter response to gusts and some of the design considerations are discussed in pars. 5-4.6.1 and 5-4.6.2 that follow.

5-4.6.1 Discussion of the Gust Problem

Rotary-wing aircraft experience milder reactions to gusts than do most fixed-wing aircraft. One of the earliest reports of this difference presents qualitative reactions of two pilots on a dual flight, one in a helicopter with side-by-side rotors and the other in a fixed-wing airplane. A similar test was conducted later by NACA with instrumentation to measure normal forces in both types of aircraft flying through turbulent air.

The relatively mild reaction of the rotary-wing aircraft is not substantiated by the simple theoretical expressions currently in use, particularly those that evolved from fixed-wing experience. Fig. 5-18 shows an example of gust load factors resulting from sharp-edged gusts, computed by the procedure in Ref. 35. This procedure is conservative in that it neglects stall and compressibility effects and assumes instantaneous changes in rotor angle of attack, induced velocity, and blade flapping.

MIL-S-8698 provides a gust alleviation factor by which theoretical gust load factor may be reduced.

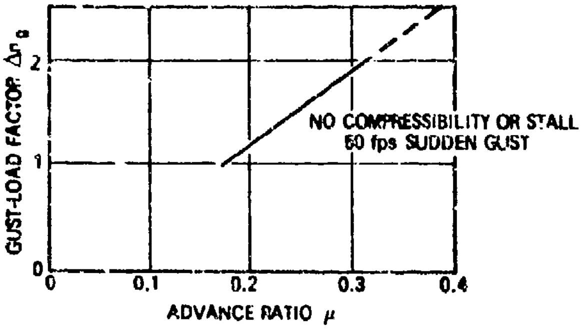


Figure 5-18. Gust Load Factor Computed for the UH-1B Helicopter Using Linear Theory

This gust alleviation factor K_g is given as a function of rotor disk loading, and is equal to unity for disk loadings greater than 6 lb/ft², as shown in Fig. 5-19.

At high speeds and for disk loadings greater than 5, gust load factors computed in accordance with MIL-S-8698, including the gust alleviation factor, are very high. When maneuver loads are superimposed on these gust loads, an unrealistic situation results. On the other hand, studies indicate that the thrust capability of a rotor actually decreases with increasing advance ratio. The aerodynamic limit shown in Fig. 5-20 is calculated by a digital method based on Ref. 36 which includes the effects of stall and compressibility. Also shown in Fig. 5-20 is a practical limit for the same rotor based on flight test data. The practical limit is a result of oscillatory rotor loads and stall flutter effects, and is the controlling limit on rotor thrust capability at high advance ratios. This conclusion is supported by Ref. 30.

Unloading the rotor by adding a wing would give the rotor a greater margin to accept gusts. The advantage, however, is not as great as might be expected, because the rotor usually will assume the larger share of the lift increase resulting from gusts, as shown by tests with an AVLABS-Bell high-performance helicopter. The gust alleviation factor given by MIL-S-8698 was found to be unrealistically low for a rotor unloaded in this manner.

An early attempt at treatment of gust effects on rotary-wing aircraft is reported in Ref. 33. Sine-squared gust shapes were considered instead of sharp-edged gusts, and a mass ratio comparable to that used in fixed-wing analysis replaced disk loading in the determination of the gust alleviation factor. Fig. 5-21 shows the results of that study. It suggests a gust alleviation factor considerably smaller than that given in MIL-S-8698. The scope of the study, however, was insufficient to define requirements for all types of rotary-wing aircraft. Furthermore, gradual penetration into the gusts, nonsteady aerodynamics, and aeroelastic feedback were not considered. Subsequent studies (Refs. 34 and 37) attempted to remedy these deficiencies and to include unsteady aerodynamics as well as additional variables such as gust shape and intensity, forward speed, disk loading, thrust coefficient-solidity ratio, and advancing tip Mach number. The design considerations derived from these studies are reviewed in par. 5-4.6.2.

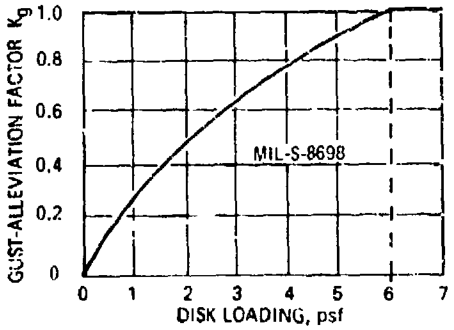


Figure 5-19. Gust-alleviation Factor (MIL-S-8698)

5-4.6.2 Gust Design Considerations

From the studies in Refs. 34 and 37 several principal conclusions can be drawn. The gust alleviation factors K_g given in MIL-S-8698 are too conservative. The use of a rotor mass ratio μ_g to determine K_g by analogy with the fixed-wing approach, as suggested in Ref. 33, also does not give satisfactory results. Nevertheless, gust loads cannot be ignored in rotor design. Pending preparation of criteria to replace the gust load requirements of MIL-S-8698, alternative methods for determination of gust load factors may be used, subject to the prior approval of the procuring activity.

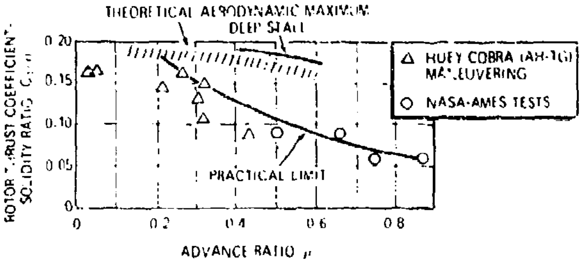


Figure 5-20. Rotor Limits as a Function of Advance Ratio

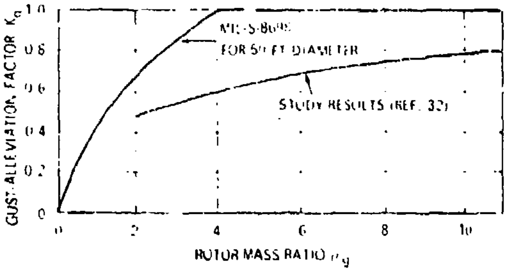


Figure 5-21. Results of a Load Gust Study Compared With Military Specification Requirements

For all the helicopters and compounds investigated in Ref. 34, the rotor gust-load ratio $\Delta T/T_{hover}$ can be expressed by the empirical expression

$$\frac{\Delta T}{T_{hover}} = \frac{0.057}{\left(\frac{C_T}{\sigma}\right)_{hover}} + \frac{0.85L_w}{T_{hover}} - C \qquad (5\text{-}7)$$

where

C = empirical constant, dimensionless

L_w = wing lift, lb

L_w/T_{hover} = wing lift ratio prior to gust encounter, dimensionless

This relationship gives reasonable accuracy with appropriate conservatism with $C \approx 0.2$ for semirigid (teetering) rotors and $C \approx 0.1$ for rigid and articulated rotors. For a compound helicopter, the wing gust load should be determined by conventional fixed-wing methods. However, an additional alleviation of the wing gust load, owing to the interaction with the rotor, was found to be related to the rotor thrust coefficient-solidity ratio C_T/σ. With further refinement, this approach may provide an acceptable basis for gust design requirements.

The relative effects of various parameters on gust response are summarized in Table 5-1.

TABLE 5-1. THE RELATIVE EFFECTS OF VARIOUS PARAMETERS ON GUST RESPONSE

PARAMETER	EFFECT
DISK LOADING	LITTLE INFLUENCE
ROTOR THRUST COEFFICIENT-SOLIDITY RATIO, C_T/σ	MAJOR EFFECT (SEE EQUATION 5-7)
COMPOUNDING	CONSIDERABLE EFFECT AT HIGH VALUES OF C_T/σ DUE TO LIFT SHARING WITH A WING
ROTOR TYPE	SOME EFFECT, DEPENDS ON DYNAMICS
NUMBER OF BLADES	LITTLE EFFECT
NUMBER OF ROTORS	INCREASED EFFECT FOR TANDEM CONFIGURATION
FORWARD VELOCITY AND ADVANCING-TIP MACH NUMBER	LITTLE INFLUENCE
LOCK NUMBER	SLIGHT REDUCTION OF GUST LOAD WITH INCREASED LOCK NUMBER
PITCH-FLAP COUPLING	LITTLE EFFECT
PITCH-CONE COUPLING	APPRECIABLE EFFECT
BOBWEIGHT IN COLLECTIVE SYSTEM	APPRECIABLE EFFECT

5-4.7 TORSIONAL STABILITY

Gas turbine engines first developed for aircraft included fixed-turbine engines for both turbojet and turboshaft configurations, and free-turbine engines for driving rigid loads or loads with natural frequencies far above the maximum response range of the engine governor. Sophisticated hydromechanical governors were optimized for these systems; however, these governors proved unsatisfactory for helicopter applications. A serious drive system instability was predicted for the XH-40, the first helicopter using the free-turbine engine (this analysis was later published in Ref. 38). The predicted instability occurred as predicted, but the consequences were not serious. Means to stabilize the system had been provided and the test program was able to proceed, although with a rather sluggish governor.

Extensive analog computer studies have shown that serious penalties would be incurred if the helicopter drive system were modified to solve the problem, but that modification of the governor resulted in only a slight penalty. As a result, Ref. 39 was formulated. This publication assigns responsibilities to both the engine and helicopter manufacturers in order to insure early recognition and solution of the drive system stability problem in future applications. The discussion that follows supplements the reviews of drive system torsional stability in Chapter 5, and of the engine/airframe integrated control system in Chapter 8, both in AMCP 706-201.

5-4.7.1 Discussion of Problem

The dynamic characteristics of three systems are involved: the gas producer and its governor, the power turbine and its governor, and the helicopter drive system with one or more low natural frequencies. The system equations of motion conveniently can be put in transfer function form and arrayed as a block diagram (Refs. 38 and 40). During throttle movements, the gas producer governor controls the engine. At steady-state throttle conditions, the power-turbine governor controls power-turbine speed by modulating fuel flow. When this speed drops below the selected value, additional fuel flow is called for to nullify the error. Accordingly, when the power turbine overspeeds, fuel flow is decreased. This cycle of events can be stable or unstable depending upon system parameters.

The drive system for rotors without drag hinges usually can be simplified to a system consisting of the main-rotor and power-turbine inertias connected by the effective shaft stiffness between them. This is a single-degree-of-freedom system in which the main rotor can be considered as nodalized because its inertia is much larger than that of the power turbine. This mode usually has a natural frequency below 5 Hz, is lightly damped, and is continuously excited at a low level by rotor control motions and external transients. Resulting oscillations in the power turbine speed are sensed by the flyball governor, which modulates the fuel flow accordingly. With a governor optimized for a system with a high natural frequency, the oscillation of the torque at the turbine wheel that follows will be so phased that it reinforces the original, low-frequency, drive system oscillation. Unstable torsional motion results.

It is not feasible to stabilize the system either by including mechanical dampers in the drive shaft or by stiffening the shafting sufficiently to move the natural frequency out of the response range of the governor. Furthermore, the gains and time constants of the engine and governor can be varied only within narrow limits. One effective solution is to use a small amount of valve overlap, which allows the flyball governor to oscillate the fuel valve a small amount without modulating fuel flow.

5-4.7.2 Design Considerations

Ref. 39 establishes an effective, three-phase procedure that adequately deals with the problem. Briefly, the steps include:

1. The engine designer provides as much flexibility as possible in the engine governor parameters.
2. Early in the preliminary design, the engine and helicopter designers exchange system parameters and each designer conducts an analysis of the system.
3. The helicopter designer selects the optimum solution determined by the analyses.

Several efficient computer methods of analysis now exist, including analog, digital, and hybrid. The analysis can be complete, including the nonlinear parameters for the full range of engine operation, or a perturbation analysis can be performed in which an operating point is selected about which oscillatory stability is determined. The complete analysis is much more complex, but it determines transient response and droop as well as stability. The perturbation analysis usually is adequate for determining stability alone.

5-5 BLADE RETENTION

5-5.1 RETENTION SYSTEM DESIGN CONSIDERATIONS

The fully articulated, gimbaled (teetering), and rigid (hingeless) rotors are described in par. 5-3. The blade retention requirements for each are different, and are discussed in the paragraphs that follow.

5-5.1.1 Articulated Rotors

The fully-articulated rotor system provides freedom of blade movement about flap and lag hinges in response to aerodynamic forces resulting from pitch change and/or flight conditions. This flap and lag freedom reduces the flap and lag moments to zero at the respective hinges. The lag dampers (shock absorbers) prevent unstable blade oscillations about the lag hinges. Dampers usually are not required for flap hinges because of the amount of aerodynamic damping provided by flapwise blade motion.

The design considerations for a typical articulated rotor, the motions and loads for each hinge, and their effect on the helicopter are discussed in par. 5-5.1.1.1. This rotor has the flap hinge inboard, then the lag hinge, and then the pitch axis hinge outboard. The effects of reversing the hinge arrangement are considered in par. 5-5.1.1.2.

5-5.1.1.1 Typical Articulated Rotor Considerations

The Boeing-Vertol Model 107 rotor is shown in Fig. 5-22. The hinge arrangement of this rotor also is typical of the Sikorsky S-51, the Bristol Model 171, the Alouette II, and the Russian Mil 6, 8, and 10 aircraft, among others.

The flap, lag, and pitch hinges on the Model 107 rotor have oil-lubricated, cylindrical roller bearings. The three flap hinges have a common, centrally located reservoir, while each of the other hinges has its own reservoir. Each reservoir has one or more sight glasses to indicate oil level. Radial, positive-contact seals retain the lubricant in the bearings.

The flap hinge is offset both radially and in the direction of rotation. The radial offset of the flap hinge axis is small, approximately 1.7% of the blade radius, and is as close to the rotor center as shaft and hinge sizes permit. Because of the small radial offset of the flap hinge, the control forces generated by this rotor come primarily from thrust vector tilt, with a small contribution from the vertical component of the flap hinge forces (see Fig. 5-2 (B)).

The offset of the flap hinge in the direction of rotation (also known as the torque offset, and shown as dimension "a" in the plan view, Fig. 5-22) is chosen to satisfy two requirements:

1. To equalize loads on the flap-hinge bearings when the blade is in the lag position, corresponding to normal flight torque
2. To avoid reversing axial motion of the flap hinge bearings due to blade lead-lag motion in normal flight.

The flap hinge cylindrical roller bearings withstand blade centrifugal force, alternating loads due to blade lag oscillation, and blade vertical shear forces, and experience one-per-rev flapping oscillations of ±4-6 deg. Thrust loads are carried by a bronze thrust bearing.

Permanent stops prevent excessive blade droop or flap motion due to winds while the helicopter is parked or during rotor shutdown. These stops are set so that no contact occurs in normal flight. The flap stops prevent possible overturning of the blade; droop stops prevent blade/fuselage contact.

Blade angular displacement about the lag hinge — stop to stop — varies inversely with the radial position of the hinge. However, large angular lag displacement adversely affects:

1. Blade-to-blade clearance in a tandem helicopter
2. Pitch arm kinematic error
3. Lag damper stroke.

The Model 107 lag hinge is located as far inboard as possible, consistent with the lag displacement considerations listed, in order to keep the mass of the pitch hinge and the blade retention joint inboard and thus reduce the centrifugal loads.

The lag damper, in addition to meeting requirements for stability of the lag motion, functions as the lead and lag stops of the blade. Design loads for the stops are the rotor starting condition and the rotor braking loads at shutdown. The damper is positioned so that the centrifugal load on the damper will be along the piston-rod axis and will not result in piston-rod bending or internal-bearing wear. Lag damper end bearings are lined with Teflon fabric for maintenance-free operation.

The rolling element bearings of the lag hinge are subject to centrifugal loads plus reactions from blade alternating and steady bending moments. Vertical shear loads are carried by bronze thrust bearings. Blade lag oscillation is in the order of ±1 deg.

The cylindrical roller bearings of the pitch (feather) hinge react blade steady and alternating bending moments and shears at the root of the blade. Typical oscillation angles in forward flight are ±4-6 deg. Tension-torsion straps, consisting of many slotted stainless steel elements, react the blade centrifugal loads. These straps twist easily, providing freedom for blade pitch change with negligible effect on the pitch control linkage forces. The pitch arm connects to the upper end of the pitch link nearly in line with the flap hinge axis; the locations of this attachment and the lower attachment of the pitch link to the swashplate were chosen to obtain favorable coupling of blade pitch with flap and lag displacements.

Two taper pins in a multiple clevis joint attach the blade to the outboard end of the pitch housing. The clevis provides freedom for manual folding of the blades about one or the other taper pin, depending upon the required direction of fold.

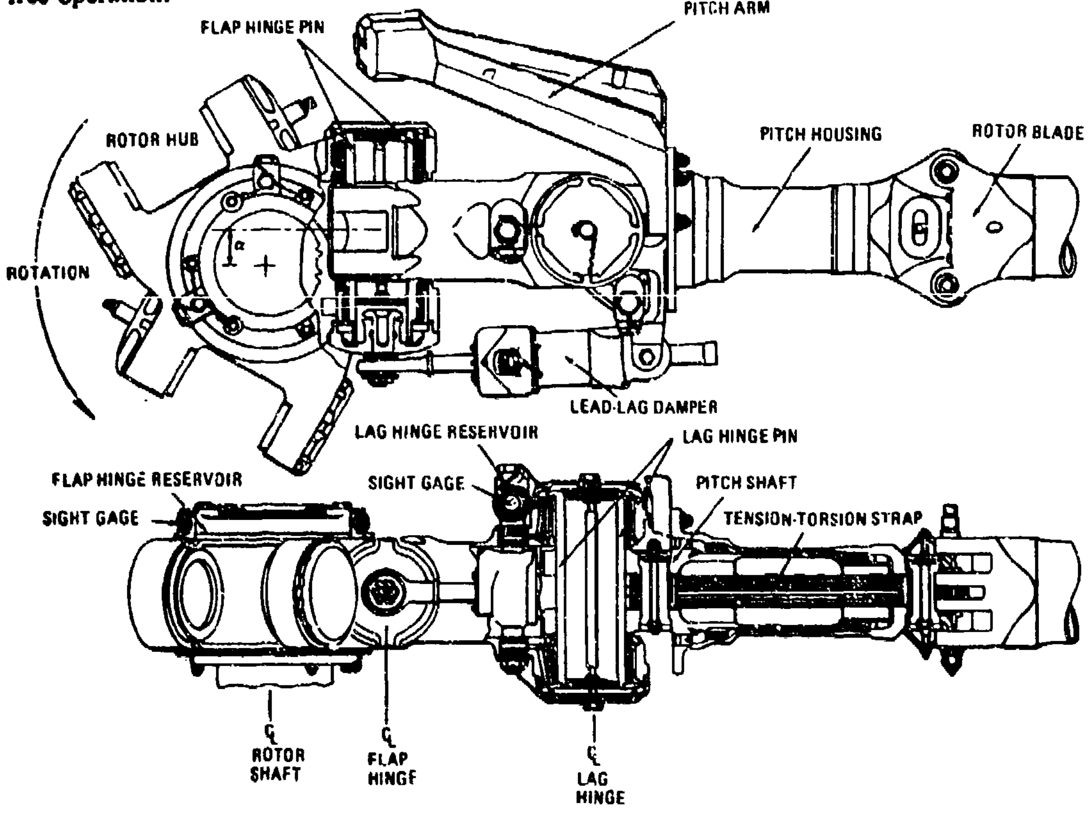


Figure 5-22. Articulated Rotor (Boeing Model 107)

5-5.1.1.2 Reversed Hinge Articulation

The coincident lag and flap hinge arrangement shown schematically in Fig. 5-4 is used on all production Sikorsky helicopters from the S-55 through the S-65. The radial location of these coincident hinges is roughly 5% of the blade radius. Good control power results, permitting liberal CG travel in these single-rotor helicopters. The loads and motions resulting from this hinge arrangement and the retention methods used in a typical coincident-hinge rotor (Sikorsky S-61) are discussed in the paragraphs that follow.

As the S-61 lag hinge bearings are mounted in the star-shaped hub, normal blade coning and cyclic flapping result in sizable vertical thrust loads along the lag hinge axis.

A pair of conical roller bearings transfers both radial and vertical thrust loads of the lag hinge to the upper plate of the hub, while a cylindrical roller bearing transfers radial loads to the lower plate.

With this coincident-hinge arrangement, the flap hinge leads and lags with the blade. Loads on the lead and lag bearings of the flap hinge are equalized by a symmetrical location about the lag hinge axis. Axial thrust along the flap hinge axis is due to blade chordwise shear forces. These forces are low and are carried by thrust faces.

The pitch (feather) hinge is just outboard of the coincident hinge, and blade moments and shears are of low magnitude, with the primary reaction for this hinge being the centrifugal force on the blade. A stack of angular contact ball bearings arranged in tandem carries the centrifugal forces in a very compact arrangement. A radial bearing pre-loads the angular contact set and assists in carrying moments and shears.

At least one helicopter (CH-47) has the pitch hinge inboard of the lag hinge. In this configuration the lag axis rotates with blade pitch changes and is perpendicular to the flap hinge axis at only one blade pitch. The weight of the rotor head is thus reduced, as the blade can be removed or folded at the lag hinge and an additional attachment joint is not required. Hinge loads and motions for this configuration are similar to those of the typical rotor arrangement first discussed, with differences resulting mainly from the moments and shears at the different radial locations of the pitch and lag hinges. Control system loads do not differ significantly.

This rotor configuration, with the pitch axis hinge inboard of the lag hinge, results in two major characteristics:

1. There is no kinematic coupling of pitch with lag.
2. The "parked" helicopter has reduced blade/fuselage clearances in that the weight of the blade in a lead or lag position results in moment about the pitch hinge, causing the controls to "drift" and the blade to droop below the normal position.

5-5.1.2 Gimbaled and Teetering Rotors

The gimbal-mounted rotor and the two-bladed teetering, or see-saw, rotor have blade pitch-change hinges rigidly mounted to the central hub. This hub assembly in turn is free to pivot with respect to the mounting structure in response to one-per-rev blade forces, thus minimizing loads in the blade root and the hub due to first harmonic flapping. Coriolis forces in the lag direction similarly are reduced. Alleviation of these two types of load has a significant effect on rotor-head strength requirements and therefore on the weight of the components. Elimination of the lag hinge and lag damper reduces maintenance requirements but at the expense of providing strength for lead-lag moments that do not go to zero. Also, controllability is somewhat lower with these hubs because it results from thrust vector tilt alone.

5-5.1.2.1 Gimbal-mounted Hubs

Two-bladed hubs, fully gimbal-mounted on the rotor shaft, see-saw about one axis for cyclic flapping and are tilted about an orthogonal axis for cyclic pitch control. Collective pitch is input by individual links to each blade. The OH-13 and the OH-23 are examples of helicopters that use this hub.

The gimbal pivot bearings react the rotor lift forces and also transmit the drive system torque. When the rotor plane is tilted, the Cardan joint characteristics of the gimbal cause oscillating speed-torque characteristics in the drive system that must be considered in the design of the retention system. Axial load capability in the gimbal hinges is required to react the in-plane rotor forces.

The pivot bearings on the gimbal axis parallel to the blade span oscillate with cyclic feathering of the blades while the bearings on the axis normal to the blade span oscillate with flapping of the rotor. Thus, the bearings on the pitch axis are required to accommodate only the collective pitch motions of the blades.

The pitch axes are preconed to reduce the steady blade flap bending moments on these hinges and on the hub structure. The hub structure containing the hinges is underslung below the gimbal pivot so that the vertical location of the CG of the blade assembly in the normal flight position is close to that of the gimbal pivot point. This reduces the chordwise oscillation of the blade CG when pitch changes are made without flapping, as discussed in Ref. 41.

Moments and shear forces for the pitch bearings are higher than those of an equivalent fully articulated rotor. Motions are of the same order of magnitude but, as noted previously, do not include the oscillations due to cyclic pitch. Retention methods are the same as those previously described for the pitch axis of the fully articulated hub.

5-5.1.2.2 Teetering Hubs

Another common two-bladed rotor hub configuration, used on the OH-58A and the Fairchild-Hiller FH-1100, is a hub free to teeter about an axis normal to both the blade span and the rotor shaft for cyclic flapping. Control linkages to each blade change collective and cyclic pitch. This type of hub has preconed pitch axes underslung below the teetering hinge, and, in general, the loads and motions are similar to those of the two-bladed, gimbal-mounted hub. However, larger chordwise moments are caused by the lack of full gimbal provisions, and the pitch axis bearings must accommodate the oscillatory motions of cyclic pitch as well as collective motions.

5-5.1.3 Rigid Rotor

The "rigid," or hingeless, rotor blade retention configuration attaches the rotor blades firmly to the hub, which, in turn, is attached rigidly to the rotor mast. The blade retention system must be capable of transferring forces and moments in both flapwise and lead-lag directions. The blade centrifugal force may be reacted by any of the conventional methods (a tension-torsion strap, an elastomeric bearing, or a stack of antifriction bearings). The retention must provide a pitch change capability (both cyclic and collective), and the centrifugal force must be reacted across the pitch change, or feathering, hinge.

No hinges are provided for flapwise or lead-lag motion and the only flap or lag movement of the blades relative to the fixed support structure is due to structural deflection. The specific layout of the hub and the retention determines the manner and extent to which these deflections couple with and affect the pitch motion of the blades.

As with other systems that do not incorporate a flapping hinge, a precone angle usually is built into the blade retention for a hingeless rotor. This built-in angle helps to alleviate the flapping moment that the retention must react. The geometry of the retention also may include sweep and/or droop of the spanwise axis of the blade relative to the feathering hinge axis. The direction and amount of these alignments, together with the specific geometry of the blade pitch control input, define the feedback coupling between blade motion and pitch control input. The necessity for and significance of these coupling effects upon the stability of the rotor system are discussed in detail in Chapter 5, AMCP 706-201.

The advantages of the hingeless rotor system include the high level of control provided by the transfer of moments. The system may be physically simple, but the strength required to transmit the forces and moments across the blade retention system can cause the retention to be heavier than is true for other types of rotors.

5-5.2 COMPONENT DESIGN CONSIDERATIONS

The design considerations applicable to the use of rolling element bearings in rotor blade retention systems are reviewed in this paragraph. Additional discussion of bearings, for blade retentions and for other applications, is found in par. 16-3.

Other components associated with the blade retention system that also are discussed in this paragraph are: lag dampers, lead and lag stops, droop and flap stops, and droop and flap retainers.

5-5.2.1 Rolling Element Bearings

Rolling element bearings are widely used in rotor hinges. Experience with them has been good and the technology, which is based upon both analytical methods and empirical data, has been verified by extensive service experience. Generally, these bearings are compact. Bearing friction is low and has a negligible effect on hub loads. However, the effect of pitch axis bearing friction on control system loads should be evaluated; on large helicopters this friction usually is low compared with the aerodynamic and dynamic loads, but on smaller helicopters the effect may be significant.

The failure mode of oscillating rolling element bearings most generally encountered is gradually progressive spalling that results in looseness, heat generation, and aircraft vibration. These factors have some incipient failure warning characteristics.

Both grease- and oil-lubricated bearings have been used in rotor hinges. The use of oil is favored for the majority of current helicopters. Some characteristics of the two systems are:

1. Oil Lubrication:

a. Oil is satisfactory over a broad temperature range and is changed easily in response to environmental changes.

b. Oil sight gages provide positive indication of lubricant presence.

c. Oil permits "on condition" maintenance.

d. Reservoirs should be located so that centrifugal force drives oil into the bearings.

e. Reservoirs and lubrication cavities should be refillable without the necessity for venting to avoid air pockets.

f. Oil requires elaborate seals and the maintenance of seal integrity. Radial-positive-pressure seals are commonly used for dynamic sealing. These seals should be installed so that contaminants do not enter them through centrifugal force.

2. Grease:

a. Shields or simple seals are adequate when grease bearings are relubricated at regular intervals. Grease retention is fairly good with a failed seal.

b. Purged grease tends to exclude external contaminants from the bearing.

c. Grease in oscillating bearings tends to channel, and the soap base may harden. Regreasing may be ineffective because the hard soap may prevent proper distribution of new grease. Premature failure may result. Yet, if channeling occurs, debris detection will not be an effective means of failure detection.

d. Changing greases (as for extreme low-temperature use) can be accomplished effectively only by disassembly.

5-5.2.1.1 Cylindrical Roller Bearings

Cylindrical roller bearings combine high radial capacity with small space requirement. Design considerations mainly are empirical, and many of the factors are discussed in detail in Ref. 42. One method of computing the basic load capacity with oscillatory motion is given in Ref. 43. Other factors must be determined by endurance testing and service experience.

Some of the factors that influence the life of cylindrical roller bearings in rotor blade retentions are:

1. Angular deflection of the hinge pin, which can cause concentrated load on one end of the rollers
2. Crowning of the rollers, which can provide a better stress distribution and tend to minimize the effect of hinge pin deflections
3. Roller guidance (e.g., use of a cage), which can reduce roller misalignment
4. Large angles of oscillation, which can increase the number of stress cycles on the bearing rollers and races
5. Mounting fits, which must be as specified for particular application in order to obtain rated capacity
6. Type of lubricant.

5-5.2.1.2 Tapered Roller Bearings

Tapered roller bearings have very high radial and thrust capacities. Design factors are similar to those of cylindrical roller bearings and are discussed in Ref. 42. These bearings can be mounted in pairs to provide a capability to react moments as well as forces. The roller taper direction of paired bearings can be reversed to increase the tolerance to misalignment.

5-5.2.1.3 Angular Contact Ball Bearings

An angular contact ball bearing carries a high, one-directional thrust load in combination with radial loads. The end faces of these bearings can be ground so that two or more bearings in tandem will share a thrust load. A common method of pitch (feather) hinge construction is to use stacked angular contact bearings to react the blade centrifugal force, with a reversed bearing to preload the set and to react thrust reversal. This configuration has the capability of reacting moments and radial loads as well as thrust.

Design factors for these bearings are discussed in Ref. 44.

5-5.2.2 Teflon Fabric Bearings

Teflon fabric bearings are used in the main hinges of several operational helicopters, e.g., in the OH-6 lag hinge and AH-1G pitch bearing. These bearings are even more widely used in rotor control systems where they have demonstrated their ability to withstand high loads in an adverse environment.

The Teflon fabric liner varies in thickness from approximately 0.01 to 0.02 in. depending upon the manufacturer. Strands of a material such as cotton, Dacron, or Fiberglas are interwoven in the back surface of the fabric. The fabric then is bonded to the outer housing, with the non-Teflon strands providing good bond adherence. Various bonding agents, bonding procedures, and fabric weaves are used; these can result in different characteristics of the finished bearing.

Design considerations for these bearings are:

1. Loads and motions. The effect of loads and motions on Teflon bearing lives is based upon empirically determined factors. It is general design practice to compare bearing lives as a function of *PV*, where *P* = pressure, psi, based on the projected area of the bearing surface in the direction of load, and *V* = average velocity of contacting surfaces, fpm.

The acceptable value of *PV* for a given life varies with the pressure. Also, load reversals may reduce the acceptable value by half. Large-diameter bearings appear to withstand a higher *PV*-level than do small bearings. It is clear that *PV* is only a convenient index for comparison of bearings in similar applications, and real design values will depend upon endurance test data for full scale bearings.

2. Friction. Measured values of the coefficient of friction of Teflon-fabric bearings under loads comparable to those of rotor hinges generally are in the

range of μ = 0.1 to 0.2. Contamination of the bearings in service has resulted in higher values. High friction may add significantly to rotor system loads and should be considered in the design. Radial bearing forces due to differential expansion also should be considered as a possible source of damaging frictional loads.

3. Wear characteristics. Teflon fabric bearings operating at a given load level will wear at an essentially constant rate, up to approximately half the liner thickness. Bearings should be replaced at this time. The wear rate may increase slightly when the backing material is exposed.

When loads are not reversing, bearing wear may not always be reflected in increased clearance. Wear debris collects on the unloaded side of the bearing so that the fit appears to be tight. Unless it is known that clearance will increase with wear, means of wear determination other than checking clearance, or tightness, should be planned.

5-5.2.3 Flexing Elements

The centrifugal force on the blade acts as a thrust load on the pitch hinge. Flexing tension elements are used in a number of helicopters to react this force while also accommodating movement in pitch. The two most common types of flexing elements are metal strap tension-torsion assemblies, shown in Fig. 5-23, and wire-wound tie bars, shown in Fig. 5-24.

5-5.2.3.1 Tension-torsion Strap Assemblies

Many configurations of tension-torsion strap assemblies have been used in addition to those shown in Fig. 5-23. These include different slot arrangements, unslotted straps, straps of different thickness, and assemblies with and without spacers separating the straps.

The tension-torsion assemblies shown have straps of 0.032 in. nominal thickness stainless steel, with slots as shown to reduce the stresses due to torsion. Thin shims separate the straps at the ends to reduce fretting.

The assemblies provide the capability for ±45-deg blade motion under design loads, and in normal operation are cycled approximately ±6 deg during each rotor revolution.

Torsional stiffness of the strap assemblies does not affect the pitch control forces significantly. For example, the larger assembly of Fig. 5-23 has a torsional spring rate of 120 in.-lb per deg under the 85,000-lb centrifugal load of the blade. For the high-speed flight condition the increment of pitching moment contributed by the tension-torsion assembly is less than 3% of the total predicted blade pitching moment.

The strap assemblies of Fig. 5-23 have shown excellent fail-safe characteristics. Fatigue failure is characterized by breaking of a single element of a strap, generally on an outside corner, followed by gradual progression to other outer elements after much continued cycling. Ground-air-ground cycle testing also has resulted in slow failure progression from element to element in the lugs.

The tension-torsion assembly of the OH-6A helicopter is shown in Fig. 5-11. This assembly provides flexibility for flapping as well as for pitch, or feathering. The 15 stainless steel straps carry the centrifugal loads from one blade lag hinge across to the opposite hinge and provide a fail-safe retention system.

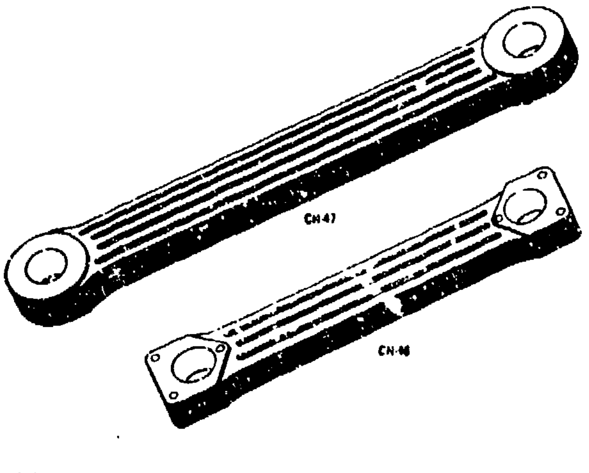


Figure 5-23. CH-46 and CH-47 Tension-Torsion Strap Assemblies

5-5.2.3.2 Wire Tie-bar Assemblies

Very-high-strength, small-diameter wire is wound around end fittings to form a lightweight retention system that is flexible torsionally under blade centrifugal loads. The inherent torsional flexibility can be varied over a range of approximately 10 to 1 if desired, by changing the configuration (Fig. 5-24). Normal torsional stiffnesses, like those of the flexing strap assemblies, are such as to have insignificant effects on control loads.

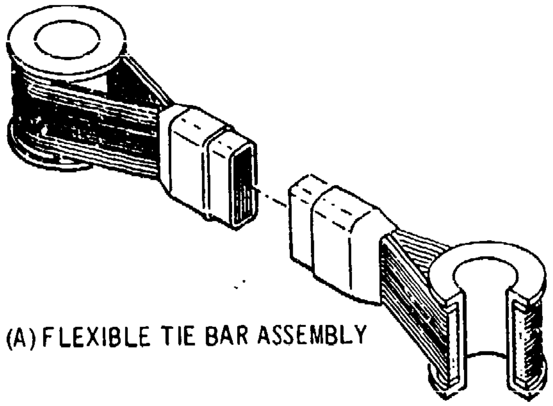
5-5.2.4 Elastomeric Bearings

Elastomeric blade-retention bearings are based on the principle that a thin layer of elastomer will withstand high normal (compressive) forces and still permit high shear deformation (strain). By using alternate layers of elastomer and metal, the blade hinge forces can be carried as compression of the elastomer and the hinge oscillation carried as shear.

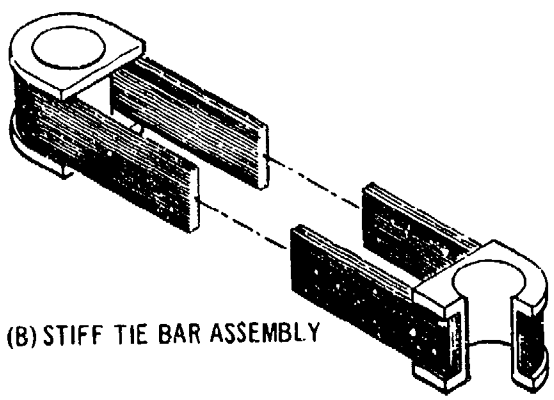
Among the advantages offered by elastomeric bearings are:

1. Elimination of lubrication requirements
2. Improved maintainability and reliability
3. Sand, rain, and dust resistance
4. Compressive loading, giving the ability to carry loads after severe degradation (fail-safe)
5. Surface deterioration as the normal form of wear, giving visible failure warning.

Some of the more common configurations of elastomeric bearings are shown in Fig. 5-25. The cylindrical bearings for radial load and the thrust bearings have been used to replace conventional rolling element bearings as blade retention components. The spherical elastomer permits complete blade articulation — pitch, flap, and lag — in a single bearing, while reacting the blade centrifugal force.

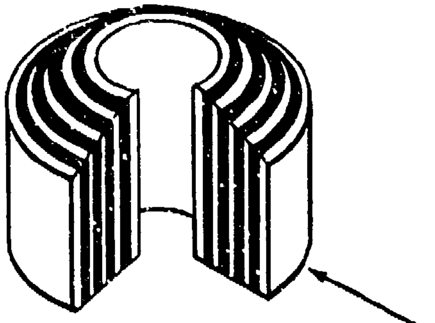


(A) FLEXIBLE TIE BAR ASSEMBLY



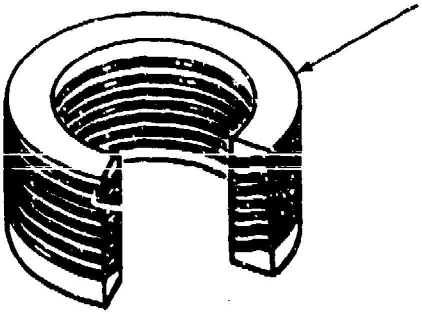
(B) STIFF TIE BAR ASSEMBLY

Figure 5-24. Torsionally 'Stiff' and 'Flexible Wire-wound Tie-bar Assemblies

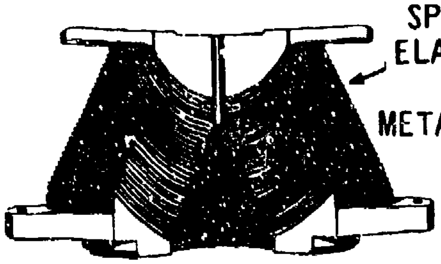


(A) RADIAL BEARING

ALTERNATE ELASTOMERIC AND METAL LAYERS



(B) THRUST BEARING



ALTERNATE SPHERICAL ELASTOMERIC AND METAL LAYERS

(C) SPHERICAL BEARING

Figure 5-25. Elastomeric Bearings

5-5.2.5 Lag Dampers, Lead-lag Stops

The lag damper of an articulated rotor must meet blade stability requirements in ground resonance (par. 5-4.3) and in flight (par. 5-3.6). Two common means of energy absorption in lag dampers are hydraulic shock absorbers (used on the majority of large helicopters (Fig. 5-26)) and friction dampers (spring-load oscillating disks, used in several small helicopters). Although simpler, lighter, and less expensive, friction dampers generally are less reliable. Therefore, the use of friction dampers in new rotors is discouraged.

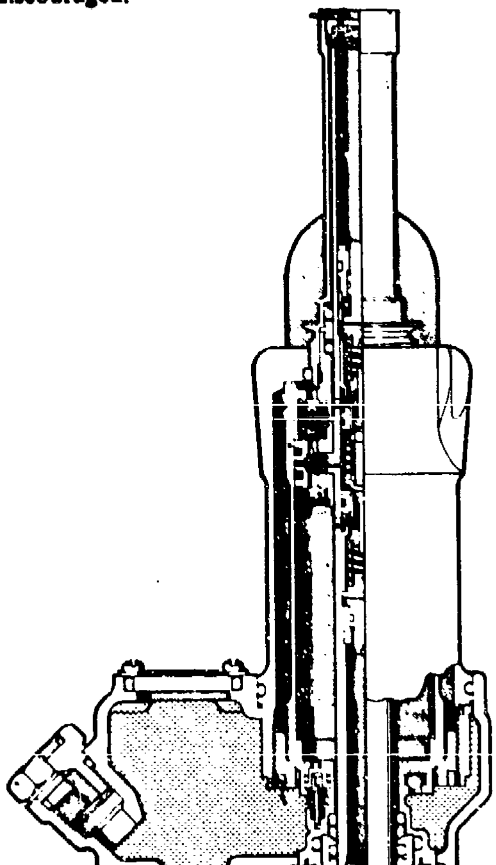


Figure 5-26. Hydraulic Lag Damper

Teflon fabric bearings frequently are used for mounting lag dampers. For satisfactory life with reversing loads, *PV* values (par. 5-5.2.2) should be approximately 1/3 those found satisfactory under nonreversing loads.

The CH-53 has auxiliary pistons in its hydraulic lag dampers. The pistons are pressurized to force all blades to the lead stops so that vibration during rotor startup is minimized.

Hydraulic lag dampers frequently are used as lead and lag stops. Integral hydraulic cushions can be used to reduce the impact of the blade against the stop.

Principal design conditions of the lead and lag stops (whether integral with the damper or not) are:

1. Predictable flight conditions or maneuvers will not cause contact of the lead or lag stops.
2. Lead stops will not yield due to rotor brake application.
3. Lag stops will not yield due to engine starting torque.
4. Lead and lag stops should fail before any dynamic component critical to safe flight yields.

5-5.2.6 Droop and Flap Stops and Restrainers

Articulated rotors require stops to limit the extremes of flapping motion of individual blades. Teetering and gimbal-mounted hubs have a similar requirement, but the motion is that of the complete rotor assembly.

The droop stop must be positioned to allow normal cyclic blade motion in all predictable flight conditions or maneuvers without making contact with the stop. This stop position also must be high enough to prevent blade/fuselage contact in high winds during rotor shutdown and when parked, i.e., when blade centrifugal force does not provide a radial restoring force.

Centrifugally operated droop restrainers often are used to increase blade/fuselage clearances. These restrainers engage at a rotor speed approximately 40-60% of normal during rotor shutdown and restrict blade droop to provide clearances not possible with permanent droop stops.

Flap stops prevent accidental overturning of the blades in very high winds, and are positioned to allow clearance for flight cyclic flapping motions in all flight conditions.

Centrifugal flap restrainers also can be engaged at low rotor speeds during rotor shutdown and, in conjunction with droop restrainers greatly reduce flap hinge motion. Flap restrainers are essential for blade folding to prevent the blades from "elbowing". Elbowing occurs when a blade is folded so that its CG is inboard of the flap hinge axis; if the hinge is free to flap, the blade tip will droop under the blade weight. Flap restrainers can be ground support equipment if blade folding is accomplished only occasionally.

Two common types of centrifugal droop and flap restrainer mechanisms are overcenter linkages and interposer blocks. Both of these use weights to release the mechanism as the rotor speed increases, and springs to re-engage during shutdown.

Another droop restrainer mechanism is a floating (gimbaled) ring below the hub, with projections on the flapping portion of each blade. Cyclic motion of individual blades displaces the ring to permit flapping without restraint, while the ring supports all the blades against collective droop.

5-5.3 CONTROL SYSTEM CONSIDERATIONS

In the selection and design of the blade retention system, consideration must be given to the characteristics of those elements which affect the loads on the rotor control system. Displacement of the blade about the pitch axis is opposed by a friction torque of essentially constant value if angular contact ball bearings are used to react the centrifugal force of the blade. On the other hand, both elastomeric bearings and flexing elements (tension-torsion straps and wire tie-bar assemblies) have the characteristics of torsional springs. When the elements of this type are used to react the blade centrifugal force, the torque opposing angular motion of the blade is proportional to the displacement.

In addition to torques resisting motion about the pitch axis that originate in the blade retention system, there also are torques that depend upon the inertia characteristics of the blades. The first of these, known variously as the propeller moment and the centrifugal feathering moment, is a torque that opposes displacement of the chordwise principal axis of the cross section of the blade out of the plane of rotation. This torque is directly proportional to the difference between the moments of inertia with respect to the principal axes of the cross section and also varies directly with Ω^2. The presence of the torque also has been referred to as "the tennis racket effect". For tail rotors and for main rotors of small helicopters for which the rotational speed is high, it may be desirable to nullify this torque by equalizing the moments of inertia. This may be accomplished by the addition of appropriate balance weights at the root of the blade.

The second torque that depends on the characteristics of the blade is the conventional inertia reaction, proportional to the polar moment of inertia with respect to the pitch axis. This inertial torque varies with, but is opposite in direction to, the pitching (feathering) acceleration. If balance weights are used to reduce or nullify the centrifugal centering moment, the increased polar moment of inertia will result in higher loads in the control system to obtain a given change of blade pitch.

For those rotors in which cyclic pitch is obtained by oscillation of the blade about the pitch axis, e.g., fully articulated and hingeless systems, the dynamic characteristics of the system may be important. The selection of a blade retention system should include the investigation of the response of the blade to the oscillating control force. Systems such as tension-torsion straps or wire tie-bar assemblies that have known torsional spring characteristics may be required to obtain an acceptable relationship between the natural frequency of the feathering motion and the rotational speed.

Control system design considerations are discussed in detail in Chapter 6. Further discussions of both nullification of the centrifugal centering moment and optimization of the natural frequency of the feathering motion are provided by Ref. 45.

5-5.4 BLADE FOLDING

For shipboard operation, or for compact stowage, it is desirable to bring the rotor blades within the fuselage envelope. When the blades are stowed in this manner, there is much less possibility of blade damage when moving the helicopter, or when other helicopters or vehicles are moved in the vicinity. Blade folding is preferable to removing blades; there is less chance of handling damage, and retracking can be avoided. Either manual or powered blade folding systems can be used, depending upon operational requirements.

Consideration should be given to blade folding in the initial design of a rotor system, even if the basic helicopter criteria do not include this as a requirement. Appropriate decisions as to the method of blade attachment, radial location, and clevis clearances of the blade attachment joint will simplify incorporation of folding at a later date.

5-5.4.1 Design Requirements

5-5.4.1.1 Manual Blade Folding

The following provisions are required:

1. Rotor brakes, rotor locks, or means of securing the blades to the fuselage to retain hub azimuth position

2. Pitch locks, locking pins, or fixtures to restrict blade motion about the pitch axis, preventing load feedback into the pitch control system from a folded blade

3. Flap restrainers (articulated, gimbal-mounted,

and teetering hubs). Centrifugally-operated droop and flap restrainers are desired; if not self-contained, special ground support equipment is needed to restrain blade droop.

4. Blade fold hinge, with quick, simple means of locking and unlocking blade motion about this hinge
5. Clamps to restrict lag hinge motion (articulated rotors)
6. Quick-disconnect fittings for attaching handling lines to blade tips (optional)
7. Access to the rotor head (steps, handholds, toeholds, or rungs) and a work platform for performing rotor-head folding operations.

Major steps in a typical manual blade-folding operation of a single-rotor helicopter are:

1. The rotor should be rotated to the proper azimuth position for folding.
2. Rotor brake or rotor lock should be applied, or one blade should be secured to the fuselage to retain rotor hub azimuth position.
3. With an articulated hub it may be necessary or desirable to move some or all blades to a predetermined position about the lag hinge and to lock out any further motion about this hinge.
4. Cockpit cyclic and collective controls should be positioned to the proper setting for folding.
5. Pitch locks should be installed on all blades to be folded.
6. Flap restrainers:
   a. For articulated hubs without automatic flap restrainers, a flap restrainer should be installed on all blades folded 90 deg or more.
   b. For teetering or gimbal-mounted hubs, flap hinge motion should be locked out.
7. Racks should be installed to secure the folded blades to fuselage or other structure, if required.
8. A blade-supporting pole and steadying lines, as required by blade size and accessibility, should be attached. The blade-fold joint should be unlocked, and each blade folded and secured. Blades may be secured in racks, by lines to helicopter structure, or about the fold hinge.
9. If the tail assembly is to be folded, it may be necessary to fold it before the main blades or at an intermediate point in the fold cycle.

5-5.4.1.2 Power Blade Folding

For flight safety it is mandatory that a power blade folding system be properly interlocked to prevent any malfunction from occurring in flight. Interlocks also are necessary to prevent damage to helicopter components due to improper sequencing of the power blade-folding system. It also must be apparent to the flight crew that the blade unfolding sequence is complete and the aircraft is safe for flight.

The following component requirements are necessary for a power blade-folding system:

1. Power rotor orientation mechanism
2. Rotor lock to maintain azimuth position
3. Automatic droop and flap restrainers (articulated, teetering, or gimbal-mounted hub)
4. Control position indicating devices for pilot
5. Power pitch locking device
6. Blade lag hinge positioning device (articulated hubs)
7. Blade fold hinge unlocking device
8. Blade fold actuators.

The power blade-folding mechanism for a typical blade of the CH-46 is shown in Fig. 5-27. An electromechanical actuator housed within each rotor blade folding hinge pin operates a linkage that sequentially inserts pitch lock pins, positions the blade about the lag hinge, releases the blade fold hinge lock, and rotates the blade to the proper fold position.

5-5.4.2 Operational Requirements

Blade folding frequently must be accomplished in an adverse environment with poor lighting, winds, rain, and possible helicopter motion.

Manual blade folding under these conditions can result readily in crew injury and human error. In view of this, design of folding components should consider:

1. Minimizing loose components
2. Minimizing large or special tools
3. Attaching flags to all fixtures (pitch lock pins, etc.) so that it is apparent the helicopter is unsafe for flight
4. Providing adequate access and working areas on the helicopter for performing the folding operations.

Clearance between folded blades and between blades and helicopter structure should be adequate to allow for blade, hub, and drive system deflections under wind loads or due to motions of the helicopter. If the blade-to-fuselage or blade-to-blade clearances are inadequate provisions for blade racks or blade securing lines should be made.

5-5.4.3 System Safety Considerations

In addition to the specific safety consideration discussed in relation to design and operational requirements, the entire blade-folding operation should be reviewed from the system safety viewpoint. The identification of potentially hazardous conditions should be made from the viewpoint of material failure/malfunction, environmental conditions, personnel error, supervisory influence, or any combination of these factors. Maximum effort should be made during the design phase to reduce the hazard of these failure modes. Guides for this safety analysis include MIL-STD-882 and Ref. 46.

5-6 ROTOR BLADES

5-6.1 GENERAL

As described in par. 5-2, design of a helicopter rotor involves the determination of optimum values for each of a number of parameters, including those that define the blade geometry. The blade designer must incorporate strength and stiffness characteristics that will meet the applicable structural design criteria and also will provide acceptable aeroelastic characteristics, and will do so efficiently and economically.

The blade geometry is defined by the parameters of twist, planform, and airfoil section. Selection of values for these parameters generally is accomplished during preliminary design. The type of parametric analysis required to optimize the rotor design

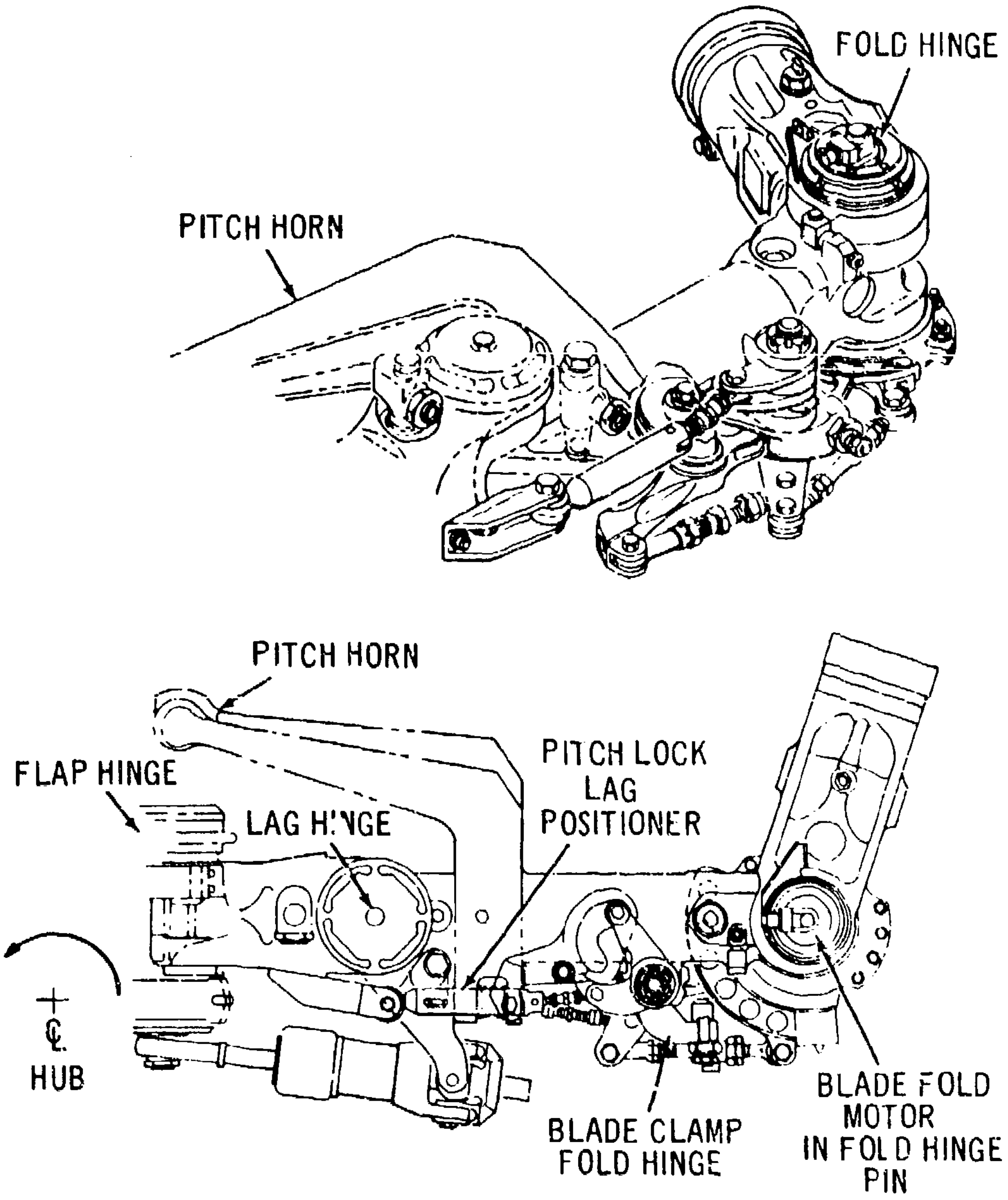

FOLD HINGE
PITCH HORN
PITCH HORN
FLAP HINGE
LAG HINGE
PITCH LOCK
LAG
POSITIONER
℄
HUB
BLADE CLAMP
FOLD HINGE
BLADE FOLD
MOTOR
IN FOLD HINGE
PIN

Figure 5-27. CH-46 Power Blade Folding Mechanism

is described in par. 3-4.1, AMCP 706-201. The considerations pertinent to the three principal flight conditions — hover, high-speed lift, and maneuvering — are reviewed in par. 5-2. In the paragraphs that follow, the significance of the two types of design parameters — aerodynamic and structural — is reviewed, with emphasis on the detail design and manufacture of rotor blades that will satisfy these requirements.

5-6.1.1 Twist

Generally, rotor blades have a linear twist on the order of 4 to 8 deg, and the blade tip angle of attack is less than that at the root (washout). The primary considerations leading to selection of the design value of twist occur in the trade-offs between hover efficiency and the delay of high-speed, retreating blade stall. Thus, for a particular aircraft mission profile that combines both hover and cruise, an optimum twist must be determined that will allow both high hover gross weight and good cruise efficiency.

The effect of twist in the hover mode is to create a more uniform inflow distribution from the blade tip to the root. The so-called "ideal twist" (which results in unrealistic values of twist near the blade root) theoretically would result in a uniform inflow distribution across the rotor. Large amounts of twist, up to 12 deg, approximate this distribution over at least the outboard half of the blade. Twist has the effect of reducing both the induced and profile drag losses of the rotor so that the hover efficiency, generally referred to as Figure of Merit, is increased (see par. 3-2, AMCP 706-201).

The theoretical maximum value for Figure of Merit value is unity. This value can occur only if the rotor has no tip losses and also possesses no profile drag. These conditions cannot occur, so an actual rotor Figure of Merit value always will be less than unity.

The effect of twist in the forward flight mode of rotor blades is to lower the pitch at the tip while maintaining a larger angle near the root. This reduction of tip angle of attack gives a corresponding decrease in the rotor profile drag power, which, in turn, allows a higher forward speed to be obtained.

Aerodynamic parametric studies used to optimize design twist for a particular aircraft also must include torsional deflection in obtaining the section angle-of-attack and the corresponding aerodynamic loading. Effects of drag loads and centrifugal twisting also should be included in the elastic twist angle determination. The forward flight angle-of-attack determination also should include the effect of blade pitch rate (tennis racket effect) on the instantaneous twist angle as it varies around the azimuth. Commonly the inertia contribution — i.e., centrifugal twisting due to blade pitch and pitch rate — will be greater than the aerodynamic twisting moments, and the net torsional deflection is in the nosedown direction.

A further requirement is that at high forward-flight speeds, the advancing blade tip has zero, or near zero, lift load in order that the corresponding high Mach number drag be minimized. For a given amount of twist and a given forward speed, this minimum drag can be met only at one specific gross weight. For operations at gross weights above this value the increase in required collective pitch results in increased blade-tip lift and drag loadings. Also, at gross weights below this minimum, a highly twisted blade tip operates at high forward speed with negative lift on part of the advancing side of the disk. This causes a nosedown control-load pulse, and the aircraft flight envelope may be limited because the resultant vibration exceeds prescribed limits.

The amount of twist that is optimum for forward flight power requirements also is restricted by the relatively linear increase in oscillatory flapwise bending moment with increased twist. In general, the power consumption and blade torsional moments due to compressibility effects can be minimized if, at the design condition, the blade is twisted to produce zero lift on the advancing tip.

5-6.1.2 Planform Taper

As with twist, the effect of planform taper is to give a more uniform inflow distribution across the disk during hover and thus to increase the Figure of Merit. The local induced velocity is proportional to the square root of the blade section lift, which, in turn, is directly proportional to the local blade chord. Thus, by increasing the root chord over that at the tip, the induced velocity over the inboard portion of the disk can be increased, simultaneously increasing the thrust over the inboard portion of the disk. Experimental results have shown, however, that the oscillatory bending moments are increased as the planform taper is increased (i.e., tip chord is much less than root chord).

The higher cost of producing planform-tapered blades has ruled out their general use. In addition, a blade with planform taper requires a thickness taper in order to retain a uniform airfoil section with known characteristics. Also, significant planform taper results in a smaller blade tip cross-sectional area available for tip balance weight placement. Also, excessive amounts of root chord — as dictated from Figure of Merit optimization studies — can cause a premature power limit on forward speed due to an increase in profile power.

5-6.1.3 Airfoil Cross Section

In addition to the usual need for high lift-to-drag ratios, stall angles, and critical Mach numbers, rotor blade airfoils require low pitching moments. Airfoil pitching moment coefficients that vary appreciably with angle of attack give periodic pitch link inputs that are undesirable and that, in turn, can lead to periodic forces and vibrations. Thus, although in forward flight the angle of attack varies with azimuth, it is desirable that the blade pitching moment coefficient not vary. Usually, it is preferred that the pitching moment coefficient be zero so the corresponding loads do not vary with the variations of local airspeed.

The usual starting point in airfoil selection is the minimization of rotor power requirements for the design cruise and hover conditions. Two-dimensional airfoil drag data at the design lift coefficient are used in this determination. The static variation of drag coefficient with Mach number, along with the change in the drag divergence boundary, also is used.

The drag reduction potential of thin airfoils is well known, and has the greatest effect near the blade tip due to the higher Mach number environment there. For ease and economy of manufacturing, a thin airfoil at the blade tip usually is achieved with a uniform root-to-tip thickness taper.

On the other hand, the retreating blade stall and drag divergence characteristics of thick sections are superior to those of thin sections in several series of airfoils. In the low Mach number region of the disk, the thick sections allow a higher lift coefficient to be obtained before the onset of drag divergence. However, the advantage of these thick sections is reversed in the high Mach number environment. Thus, the airfoil section characteristics for the advancing and retreating blades are in conflict.

The addition of airfoil camber and increased leading edge raduis tends to improve the low-speed-stall characteristics of the symmetrical airfoil sections commonly used for rotor blades. At high Mach numbers, the effect of camber is to decrease the maximum obtainable lift coefficient. However, this effect is not too significant because low lift coefficients are desired in the advancing blade, high Mach number region. Camber generally is applied to the forward portion of the rotor blade airfoil cross section in order to retain low pitching moment coefficients. This leads to the "droop snoot" terminology used by at least one contractor. In some instances, the trailing edge is reflexed slightly to counteract an otherwise unavoidable amount of pitching moment and the corresponding cyclic control loads. This method of eliminating undesirable pitching moments may have undesirable effects on blade profile power. At the same time, cambered airfoils extend the low Mach number, retreating blade drag divergence boundary to regions corresponding to greater lift coefficients. However, this beneficial effect disappears as Mach number is increased.

A further benefit of the delay of compressibility effects due to the use of thin, cambered airfoils is that noise levels due to these effects are, in general, decreased for a given flight condition. The noise generated by a blade intersecting a tip vortex is affected only to the extent that the strength of the vortex is affected. This particular form of noise is caused by a trailed tip vortex being intersected by the following blade, with a resultant rapid change of angle of attack. The corresponding pressure change causes the slapping noise that is characteristic of helicopters. At a given flight speed, a lower lift-to-drag (L/D) ratio will reduce the vortex strength and hence lower the noise.

Several airfoil sections that are used or could be used in helicopter blades are shown in Fig. 5-28. The main geometric properties of these airfoils — such as thickness ratio, leading edge radius, and camber — are identified in this figure.

In addition to the characteristics shown, some blades possess a thin, trailing edge extension strip that extends beyond the "true" airfoil trailing edge. In the usual blade, this strip is used as a base to which the upper and lower skins are bonded. The strip can be tailored in length and thickness, or number of laminates, to obtain the desired edgewise stiffness and fatigue properties. Airfoil characteristics shown in Fig. 5-28 also can be used jointly so that the desirable properties of several characteristics can be incorporated into one airfoil. For example, a blade design could be based on a thin airfoil with a droop nose. This would combine the benefits of reduced drag on the high Mach number advancing blade tip with increased maximum lift coefficient on the retreating blade. This hypothetical blade could be modified further with a large leading edge radius, and could have its aft section produced with straight "slab" sides. These changes would improve, respectively, the abruptness of the blade stall characteristics and the ease of manufacturing the blade aft section honeycomb or web structure. Further, the machining of the main bonding molds for a slab-sided blade will be easier, hence, less expensive. The decrease in blade flapwise, edgewise, and torsional stiffnesses caused by this particular geometric shape could be restored with the proper selection and layups of advanced materials such as boron or graphite composites, but the slab-sided airfoil may not provide as high a value of L/D. By use of this type of trade-off procedure, it should be possible to obtain a blade airfoil that represents an optimum configuration for the specific helicopter mission.

Blade tip geometry has been found to have important effects upon overall rotor performance. Early studies generally used constant chord tip covers with flat, semiround, or other types of curvature. These studies indicated that limited performance benefit was obtained with tip covers that had complicated curvatures and were thus difficult to manufacture. The major performance gain of these tip covers, as indicated by the rotor lift to drag L/D usually could be traced to an increase in rotor radius. More recent studies have explored swept leading and trailing edges in an effort to reduce the tip vortex velocity and strength. These results have shown that a planform tapered blade tip does reduce the tip vortex and hence both the corresponding noise and the oscillatory loads originating near the tip of the blade. In addition, the aft sweep of the leading edge delays the effects of compressibility, as with a fixed wing. This allows an increase in forward speed for fixed values of rotor speed and available power. Other attempts at blade tip geometry modification to decrease the tip vortex, such as installing slots or holes, have not proved successful.

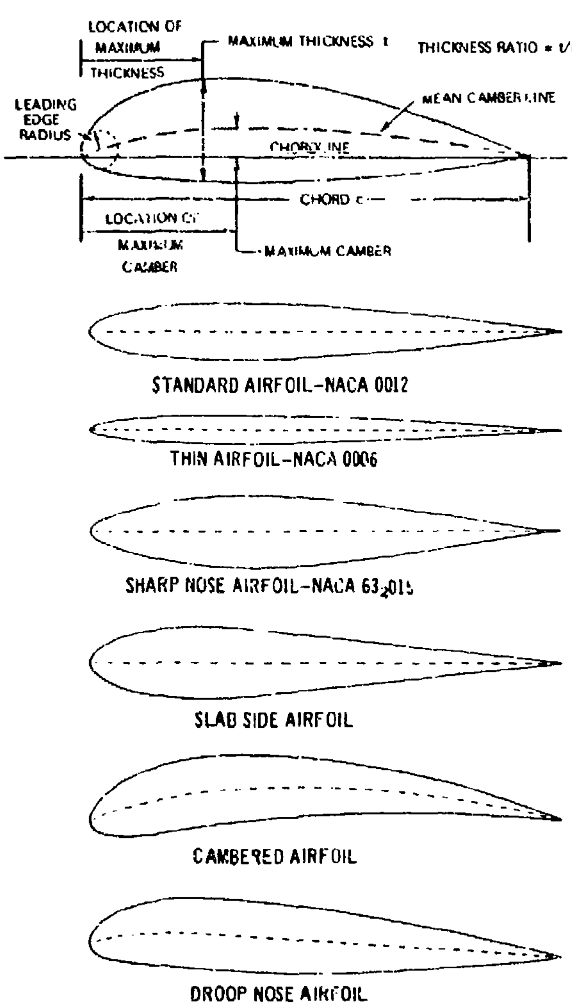


Figure 5-28. Typical Helicopter Rotor Blade Airfoils

The applicability of standard, two-dimensional airfoil data in rotor analysis is questionable whenever the shed tip vortex approaches the following blade. Because the blade tip geometry has such a strong influence on the tip vortex, the allowable spatial relationship between the vortex and blade for future high-speed helicopters should be analyzed for various tip configurations. Recent flight tests have indicated that the oscillatory airloads tend to be concentrated at the blade tips. Most harmonics are characterized by higher loading at the tip due to the impulsive nature of tip vortex interference. In nonsteady maneuvers, the local conditions on the following blade tip are changing rapidly due to the vortex effect on the angle of attack. Therefore, this radial segment of blade does not behave as two-dimensional wind tunnel data would suggest, and time-dependent airfoil section characteristics should be used in the analysis of aerodynamic loads and rotor performance.

Two-dimensional static tests show that the low Mach number stall of thick airfoils of certain series is a leading-edge phenomenon. This stall is characterized by a sudden separation of flow over the entire upper airfoil surface, and results in an abrupt change in the lift curve slope. Also associated with this effect is an instantaneous nosedown pitching moment. No warning occurs that would indicate stall is imminent.

This instantaneous effect does not occur in two-dimensional tests of thin airfoils. Rather, a gradual change in lift curve slope takes place, along with a more gradual increase in nosedown pitching moment due to stall. Many investigations have been conducted as to the effects of nonsteady aerodynamics on various airfoil sections. These tests indicate that, in the dynamic environment of increasing angle of attack with time, stall as indicated by a loss of lift does not occur in the manner predicted from static tests. When a sharp loss of lift does occur due to leading-edge separation, the resulting impact is highly coupled to the dynamic response of the blade in torsion. For instance, a fixed impulse due to a sudden loss of lift on the retreating blade may have a greater effect on a rotor system having both a low control stiffness and a torsionally soft blade. The torsional stiffness of an airfoil cross section is approximately proportional to the square of its area for a given type of cell construction. Thus, a thick airfoil can be manufactured with higher torsional stiffness. Therefore, for the airfoil series with which the thick section receives a sharp impulse due to sudden stall, the torsional response of the blade may be less for a given amount of control system stiffness than when a thin section is used.

Dynamic pitch tests of various airfoils have shown clearly that the increase in the maximum lift coefficient is large for low Mach numbers and decreases as Mach number is increased. Thus, as the retreating blade increases pitch, it can reach a greater angle of attack before stall occurs than that predicted by two-dimensional airfoil tests. The amount of this increase is dependent upon the airfoil geometry, particularly the amount and location of camber and the leading edge radius. The lift coefficient that corresponds to the drag divergence angle of attack obtained from two-dimensional tests, however, remains an adequate indicator for oscillating drag divergence. The more gradual increase in pitching moment due to the trailing edge separation of thin airfoils similarly could result in large losses of lift if these torsionally softer blades untwist sufficiently to precipitate a loss of lift. Sharp increases in torsional loads then occur and can result in the same net effect as leading-edge separation of thick airfoils. It has been found from several experimental sources that cambered airfoils with a slightly increased leading-edge radius possess superior pitching moment delaying characteristics. A large leading-edge radius also helps keep the blade section CG forward, which delays the onset of both pitching moment stall and classical bending-torsion and stall flutter. For airfoils with sharp leading edges and with their maximum thickness further aft, the overall blade CG can be moved forward with a large tip-over balance weight. This is not as desirable as obtaining a more uniform forward CG position from root to tip, since individual blade radial segments may still be acted upon by undesirable moments.

A form of negative damping can occur when the blade twist rate and the loading due to the aerodynamic pitching moment are in the same direction. This may lead to excessive torsional response and to subsequent loss of lift on the retreating blade, characterized by excessive flapwise bending amplitudes. As with torsional stiffness, a thicker blade obviously possesses more flapwise stiffness, for a given type of construction and hence will respond less to a fixed amount of flapwise lift input.

Determination of blade geometry should consider the effects of nonsteady aerodynamics along with blade tip vortex strength and trajectory in conjunction with the objectives of obtaining a high hover Figure of Merit and a low value of cruise power required. Blade twist, chord, and thickness, and the corresponding physical properties of the blade, should be chosen to minimize the responses due to vortex action and aerodynamic hysteresis effects, in addition to resonant conditions. These types of analyses should include the effects on the blade of the entire control system, as well as possible shaft or pylon bending. Dynamic blade stall effects at the first torsional natural frequency and at the once-per-rev rotational frequency of the blade should be included in the blade response analysis.

5-6.2 BLADE CONSTRUCTION

Rotor blade structures may be broken down into three major elements: the spar, the aft section (sometimes referred to as the fairing), and the root end retention. Secondary elements are tip closures and hardware, trim tabs, and tuning weights. While wooden rotor blades still are in use and probably will be used to a limited extent for many years, they are not a factor in significant current or future helicopter development and, therefore, are not addressed in this handbook.

5-6.2.1 Spar

The major load-carrying member of any rotor blade is the main spar, whether it be designed for structure only or also as a part of the aerodynamic shape of the blade. It may be of monolithic construction or may be assembled from two or more components. The predominant types of spars are described in subsequent paragraphs.

5-6.2.1.1 Hollow Extrusion

This extrusion may be a "D" spar, usually consisting of a single cell, or it may be of an essentially trapezoidal shape. The "D" shape conforms to the forward portion of the airfoil shape, with the vertical part of the "D" serving as a shear member. The hollow trapezoidal spar, sometimes referred to as a box beam, may be made to conform to the upper and lower sides of the airfoil surface but requires the addition of a shaped component on the forward side to provide the nose radius of the airfoil. This shaped component usually serves also to provide chordwise balancing of the blade and therefore is made of brass or some other relatively dense material.

5-6.2.1.2 Solid Extrusion

This extrusion is solid in the sense that its cross-sectional outline may be traced without lifting the tracing stylus. It may be referred to as a "C" section, opening toward the trailing edge. As with the "D" section, it conforms to the forward portion of the airfoil shape. Commonly the wall is thickened a considerable amount at the nose to provide chordwise balance and resistance to impact damage. The "C" section may or may not be closed at its aft end with a separately extruded or formed shear web.

The principal advantage of extruded aluminum spars is relatively low cost in production. The "D" and box beam configurations lend themselves to internal pressurization as an in-service inspection system for cracks. The advantage of the "C" section is that its internal surface may be inspected during manufacture. A major disadvantage is that the use of extruded spars is confined to constant-section blades. An added disadvantage is the poor resistance of aluminum to erosion. In low-performance helicopters, where changes in airfoil nose radius are not critical, this problem can be ignored; however, it usually is necessary to cover the aluminum with an erosion-resistant shield at the leading edge of the blade. From the standpoint of efficient design, the fatigue strength-to-weight ratio for aluminum is not attractive.

5-6.2.1.3 Formed Sheet Metal

This type of spar is fabricated from multiple components, the minimum being a "C" section and a shear web. The shear web may be the web of a channel section, with the flanges providing surface area with which to bond or braze the channel into the "C" section to form a "D" shape. Additional webs may be added to made a multicell structure. In most instances, a continuous or segmented balance weight is carried in or near the nose radius of the "C" and may contribute to the overall structure, particularly for chordwise stiffness. There is a wise choice of materials for formed sheet-metal spars, ranging from low alloy steels to any of several types of stainless steels or nonferrous alloys such as beryllium copper.

Among the advantages of this type of construction is the ability to taper the spar in almost any manner desired. Another is the ability to tailor the gages of the different components to achieve a given set of stiffness and strength requirements with greater precision. Perhaps the greatest advantage is the redundancy of the structure. The bondlines between the components are effective crackstoppers so that, even if the "C" spar should fail, the remaining structure can be designed to carry the loads and prevent a catastrophic failure of the blade. Finally, depending upon the alloy and the configuration, the spar can provide adequate erosion protection without an extra shield. Reliability is enhanced in that the quality of the raw material can be closely controlled and inspected prior to spar fabrication. The raw material does not undergo any fundamental change during the fabrication process.

5-6.2.1.4 Round Steel Tube

One of the earliest types of spars for rotor blades was a round steel tube, and certain advantages still exist. Obviously, a round tube cannot be used to constitute a part of the airfoil shape, but must be buried within an enclosing structure or envelope. Inherent in the various processes for producing such spars is the ability to taper both diameter and wall thickness — continuously or in smooth steps — providing considerable latitude in stiffness, mass, and aerodynamic taper of the rotor blade. The heaviest portion of the tube protrudes from the root end of the blade envelope, and may have integral attachment lugs or may simply be a cylinder that accepts a socket type of retention fitting.

Generally, excellent material properties are obtained in tubular spars due to the nature of the cold-working process employed in their fabrication. When this process is accomplished with sufficient precision to avoid stress raisers, high fatigue strength can be obtained. Further, with proper blade design, the material surrounding the spar tube can have sufficient independent strength to make the structure highly redundant. A disadvantage of the completely enclosed spar is the difficulty of access for inspection.

5-6.2.1.5 Formed Metal Tube

An alternative to the round tube is the formed metal tube. Generally, this starts with a round tube which subsequently is formed to either a "D" or an oval shape within the blade envelope. In the former case, the "D" is the forward portion of the airfoil contour. In the latter case, the oval tube is encased within the envelope of the airfoil, much as with the round tube. The oval shape permits a thin airfoil compared to the original tube diameter. In either case, taper of the airfoil is quite difficult to achieve, although the wall thickness can taper so as to give the desired mass and stiffness distribution. The root retention alternatives are identical to those for the round tube. Most of the advantages and disadvantages are the same as for round tubes.

5-6.2.1.6 Molded Reinforced Plastic

Molded reinforced plastic lends itself to almost any geometric spar configuration. High-strength fibers — which may be of various types of glass, graphite, or boron — are imbedded in a matrix, usually of epoxy.

Orientation of the fibers along the length of the spar gives a composite construction that is very strong in axial tension and is light in weight. One successful configuration is much like the solid aluminum extrusion. Others may be "I" beams or variants thereof. The large number of configurations possible include a multicell section with complete shear webs molded integrally inside an airfoil-shaped shell.

One of the greatest attractions of molded plastic is the ability to achieve any desired degree of taper and virtually any desired shape. Another is the ability to wrap each fiber, or filament, around the principal attachment member at the root end so that there are no discontinuities in the load-carrying material. Still another is the relatively wide selection of stiffness/strength/weight ratios that are available through the choice of fiber-reinforcing material and the orientation of the fibers.

A disadvantage of molded reinforced plastic is the difficulty of repeating with precision the properties (density, strength, stiffness) from one unit to another. This problem is being overcome through improvements in the molding process.

5-6.2.2 Aft Section

The aft section, or fairing, of a rotor blade is the aft 70-80% of the airfoil. It consists of upper and lower skins, some type of contour-stabilizing internal member (usually a structural trailing edge strip), and a means of attachment to the spar. This section may make a significant contribution to the beam stiffness and strength of the blade, or, in some cases, it may serve only as a fairing and to transmit the airloads to the spar. There are many different types and variations. The most common are described in succeeding paragraphs.

5-6.2.2.1 Continuous Skins

Continuous skins of sheet metal or fiber-reinforced plastic may extend from the root of the blade to the tip. Regardless of the internal members, continuous skins normally carry a significant amount of the centrifugal loading and a large share of the chordwise bending and torsional stiffness. These contributions can be controlled closely in the case of plastic skins by the selection of the fiber orientation. In this way, a blade can be designed to be torsionally soft and yet very stiff in the flapwise or chordwise direction, or vice versa.

The internal members that tie the upper and lower skins together and maintain the blade contour may be metallic or nonmetallic "I" beams or channels, honeycomb core, foam core, or a series of individual ribs. Blades with a chord of less than 8.0 in. or blades with unusually heavy skins may not require any internal members in the aft section. Very large blades may be constructed with individual sandwich skins both top and bottom, in which case no further reinforcement or stabilization may be necessary.

Spanwise "I" beams or channels in the aft section generally contribute significantly to blade chordwise and torsional stiffness and thus are found more often in the blades of semirigid rotor systems. Channels are adaptable as spanwise members in tapered blades since they can be stretch-formed to the required shape. If they are brake- or roll-formed in a constant shape, they can be placed in a skewed position within the aft section so that they follow a spanwise line of constant blade thickness. However, the use of such internal members often has the disadvantage of complicated internal tooling required for proper positioning, and to supply adequate pressure during adhesive bonding of the assembly.

Honeycomb core as a filler between the top and bottom skins of the aft section is extremely effective in maintaining a stable airfoil contour. Although aluminum alloy honeycomb core is the most common, there is a growing tendency toward the use of nonmetallic honeycomb. The latter has the advantages of being less susceptible to corrosion, relatively resistant to impact, and — where nonmetallic skins also are employed — less susceptible to lightning strikes. Whenever honeycomb is used, careful attention must be given to sealing a blade completely against the entry of moisture, because any water that enters the blade has a tendency to migrate and become entrapped, leading to corrosion and blade unbalance.

Foam core also has been used successfully in blade aft sections. The lightweight foams required in this application are somewhat more susceptible to delamination between skin and core than are the honeycombs, and to failures occurring within the foam itself. Generally, foam cores are pre-cured before blade assembly. Foaming in place is to be discouraged since it is difficult to obtain uniform quality and density.

Individual ribs commonly were used with wooden rotor blades, but seldom are employed with metal or reinforced plastic blades with continuous aft sections. In the latter case, the tooling for installation of the ribs becomes quite complex, and contour stability is difficult to maintain within the weight and balance limitations.

5-6.2.2.2 Segmented Skins

Blades used in fully articulated rotor systems often are constructed with segmented aft sections, sometimes referred to as boxes, pockets, or fairings. Obviously, this type of construction provides for no centrifugal load-carrying ability, and makes little contribution to chordwise stiffness unless each segment is connected by a continuous, structural, trailing edge strip. The skins of the aft section segments may be of metal or reinforced plastic and are stabilized much the same as are the skins in blades with continuous aft sections.

Among the advantages of segmented skins is the ability to replace individual segments in the event of local damage. Because the skins are, in a sense, nonstructural, considerable damage can be sustained without destroying the basic structural integrity of the rotor blade. Also, with this configuration it is easier to achieve blade bending stiffnesses of the values required for the natural frequencies desired in an articulated system. One of the greatest disadvantages in segmented aft sections is the increased difficulty in preventing water from entering the rotor blade. The number of segments may vary from 8 to 20 or more, and each joint between segments must be sealed.

5-6.2.2.3 Wraparound Skins

A special form of continuous aft section is that in which the skin of the rotor blade wraps completely around the nose radius, providing both the upper and lower airfoil surfaces in one piece. This method of construction may be used with any of the previously described internal stabilizing or strengthening members, although generally it is used with a solid extruded spar or a formed-section tubular spar. Such a skin usually is made of aluminum, although the use of other light alloys or fiber-reinforced plastic is not precluded. The method of manufacture normally is to form only the nose radius in the center of the skin material, and to depend upon the spar and/or other internal members to control the remainder of the airfoil contour. A disadvantage of wraparound skins is the difficulty of maintaining close contour tolerances, particularly in nonsymmetrical airfoils. Also, in order to maintain the required weight and balance, the skin normally is too thin to afford protection against erosion of the nose and, therefore, an additional erosion shield is required.

5-6.2.3 Root End Retentions

Root end retentions vary considerably from one blade design to another, depending upon the type of rotor system and the type of blade construction. The main retention bolts or pin(s) provide the interface between the rotor blade and the hub. Because of the high bending and centrifugal loads at this interface, it is necessary to increase the blade thickness to achieve sufficiently high section modulus and bearing area. Commonly, this is accomplished by bonding metal laminates external to the upper and lower surfaces of the blade, and then adding a relatively heavy retention or grip plate external to the stack of laminates. The retention plate contains the main hole(s), which may pass through the blade envelope or through top and bottom lugs that are extensions of the retention plates. Where the bolts pass through the blade envelope, it is reinforced with internal, metal filler blocks that effectively create a solid airfoil section in that region. When a tubular steel spar is used, it may be extended inboard of the blade envelope and be fitted with a socket, or cuff, which is either clamped or threaded onto the heavy root end of the spar. The socket may contain a single retention hole or two holes, depending upon the location and configuration of the lead-lag hinge of the hub. Here, again, the holes are through lugs that are an integral part of the socket and mate with similar lugs on the hub.

5-6.2.4 Tip Closures and Hardware

Almost all rotor blades have some type of fixed and adjustable weights within the envelope at the tip end. Fixed weights are employed to provide adequate rotor inertia, to control flapwise bending frequencies, and to place the static chordwise CG and the nominal dynamic axis in the proper location. Generally, adjustable weights are installed in pairs, displaced equally forward and aft of the design dynamic axis of the blade. These are used to equalize the spanwise mass moment of one blade against another or against a master, correcting for manufacturing tolerances in weight, and also to provide a forward or aft adjustment of the dynamic axis to achieve equal pitching moments from one blade to another. With adequate precision in the tooling and methods of manufacture, the need for either or both of these adjustments may be eliminated.

Tip closures may be simple flat plates or relatively complicated hollow, airfoil-shaped, monocoque shells, usually screwed or riveted to the blade envelope. Various shapes are in use: some simply rounded at the end, some made in the shape of a wedge, and some with very unconventional planforms. Most tip plates or caps have a small protuberance at the extreme tip to facilitate flag tracking of the rotor.

5-6.2.5 Trim Tabs

Rotor blades generally are fitted with ground-adjustable trim tabs. The tab may be an extension of the skin or of the trailing edge filler strip beyond the nominal trailing edge of the airfoil, and may extend for all or part of the span of the blade. A more common type is a relatively short tab located in the vicinity of 75% span. For blade tracking, trim tabs are adjusted by bending them upward or downward as necessary to equalize the pitching moment characteristics of the individual blades.

5-6.2.6 Tuning Weights

Many forms of tuning weights are used internally at various locations along the span of the blade. They are referred to as "antinode" weights because they are placed at the point of maximum deflection amplitude of the blade as it vibrates in various harmonic modes. The purpose is to change the natural frequency of the blade to avoid resonance with any possible forcing frequencies, particularly rotational speed. The weights may be bonded, riveted, or bolted to internal structural members of the blade, or may be suspended at the end of a cable, strap, or rod that is retained at the root end. The latter method of retention precludes high local stresses in the basic blade structure due either to holes, or to centrifugal force because of the concentrated mass. It also precludes high stresses in the weight itself from induced bending, and permits the weight to be made of high-density, nonstructural metal.

5-6.2.7 Design Requirements

Regardless of the method of construction of a rotor blade, the detail design and the selection and distribution of material must satisfy a number of independent and interrelated requirements. The blade geometry having been established, as discussed in par. 5-6.1, additional major considerations are strength, vibration, weight, mass moment of inertia, serviceability, and cost.

As a rule, a rotor blade that is designed to have a reasonable life under the applicable fatigue loading conditions will be structurally adequate for any static conditions. Therefore, major emphasis must be placed on design features that reduce the alternating stresses and make the structure as insensitive as possible to those stresses. Alternating stresses are induced by response of the blades to the periodic airloads, which, in turn, are affected by the blade motion. The blade response is dependent almost entirely upon the mass and stiffness distributions. It is extremely important that these distributions be such as to avoid any bending or torsional natural frequencies that are near resonance with any forcing functions (see par. 5-4.2).

The blade vibration frequencies may be broken down into flapwise, chordwise, and torsional frequencies; these may couple together unfavorably to cause high amplifications of motion and stress. To avoid such unfavorable coupling, it is desirable to be able to change the three stiffnesses independently. Similarly, it is desirable that alteration of the spanwise mass distribution be possible without affecting the chordwise CG. For example, a spanwise antinode weight, if improperly located in the blade, may correct a flapping natural frequency, but may change a torsional frequency so as to cause it to couple strongly with a chordwise bending frequency. Such coupling often results in high-frequency stress amplifications that can seriously limit the life of the rotor blade; but if the vibrations are not transmitted to the aircraft, the condition may not be apparent to the occupants.

This control of natural frequencies is equally important in avoiding excessive vibration of the aircraft and high loads in the control system. Whether or not the rotor blade vibrations will be transmitted to the fixed system is dependent upon the mode of vibration relative to the number of blades in the rotor. Thus, it is necessary to consider the entire system when designing a rotor blade for optimum natural frequency.

In spite of all efforts to avoid amplifications of bending moments by control of natural frequencies, alternating stresses always will exist. It is of prime importance, therefore, that the detail design maximize the tolerance of a rotor blade to these stresses. Materials selected, whether metallic or nonmetallic, must be capable of providing high fatigue strength. To this end, any form of stress raiser — e.g., notch, hole, or sudden change of section — must be avoided in areas of even relatively low alternating stress. Techniques have been developed that now make welding a viable method for fabrication of rotor blades. However, care must be exercised in the placement of the weld, and adequate quality control over the process must be assured. Holes in areas of high stress also can be avoided through the use of adhesive bonding. When the joints are designed with care, stress concentrations virtually can be eliminated. Bonded joints also act as a barrier to the propagation of a crack from one structural member to another. In all of the blade configurations discussed in the earlier parts of this paragraph, adhesive bonding generally is the principal method of joining.

5-6.2.8 Tooling and Quality Control Requirements

Two principal categories of tooling for the construction of rotor blades are the tools for fabricating the main components and those for assembling the blade. In the case of molded fiber-reinforced-plastic blades, these may be combined, and the spar, skins, etc., also may be ma[illegible] [illegible] [illegible]embly tool.

Dies for the manufacture of extruded aluminum spars are relatively inexpensive; however, the machining and other operations involved are likely to offset this cost advantage if the spar is tapered in any way. Frequently, it is difficult to maintain the required tolerances in aluminum extrusions.

Formed sheet-metal spars and shear webs, or longitudinal stiffeners, usually are made in a multistage roll forming mill if the blade is of constant section. Tooling is more expensive than extrusion dies, but is more durable and produces parts to very close tolerances. For tapered blades, it is necessary to stretch-form the parts. Tools and capital equipment for this operation can be quite costly, but very close tolerances can be held.

Tubular metal spars may be made by any of several methods, all of which are some form of swaging. Tooling costs generally are quite high. Quality hazards associated with these processes include mandrel pickup and the enlargement of otherwise negligible or easily removable metal defects.

Adhesive bonding requires large, specialized tools capable of applying accurately controlled heat and pressure while maintaining close dimensional tolerances. These tools may be "unitized"; i.e., they may contain built-in sources of heat and pressure. Heat may come from the electrical resistance "calrod" type of inserts or heating blankets, or may be provided by steam or hot oil passages. Pressure sometimes is applied through pneumatic cells contained in the fixture. Unitized tools have the advantages of being semiportable and of being capable of providing different values of temperature and pressure in different zones, as required for the mass of material and the type of joint in each particular zone. A disadvantage is that each rotor blade type or subassembly requires a completely new tool with heat, pressure, and cooling provisions and relatively complex controls.

The other principal assembly method is the autoclave. Both heat and pressure are provided by this piece of capital equipment, and the tools that hold the blade components and maintain dimensions during bonding are relatively less expensive than comparable unitized tools. However, unless special provisions are made, all areas of the blade receive the same heat and pressure. This can be a distinct disadvantage since more heat input is desirable in a region such as at the root end, where there is considerably more mass, than in a light section of the blade.

A hybrid method of assembly employs a tool that contains its own pressure source, such as pneumatic cells, but that is placed in an oven for heating.

Regardless of the type of tooling, precise controls are required to assure that proper heat and pressure have been applied. Printed chart records are desirable, and provisions should be made for the processing of samples representative of each individual blade assembly that can be tested to destruction. Even after having maintained such control, it is desirable that some form of nondestructive testing (NDT) be applied to the final assembly. The most prevalent NDT method is the ultrasonic scan, which reveals unbonded or poorly bonded joints.

5-6.3 BLADE BALANCE AND TRACK

Individual production rotor blades must have both dynamically and aerodynamically similar characteristics. Dynamic similarity is achieved through maintenance of a specific mass balance by the addition or removal of weights on the blade. Aerodynamic similarity is achieved by maintaining close airfoil and geometric control, or by adjustments, such as with a trim tab. Determination and confirmation of dynamic and aerodynamic similarity are accomplished by physically balancing and tracking each blade against a master blade or set of master blades.

5-6.3.1 Effect of Design

Considerations in obtaining dynamically and aerodynamically similar blades must begin with the design. The selection of materials and the construction of the blade should be made with interchangeability as an ultimate objective. In making material selections, trade-off such as sheet stock versus extruded or forged material must be made. Generally, a weight advantage can be realized by the use of sheet stock. However, the forming of the sheet stock may not produce a close-tolerance airfoil shape. This type of trade-off procedure should be followed for all major components of the blade to insure acceptable balance and track and ultimately, the interchangeability of each blade with other blades of that specific configuration.

Because the control of the weight of individual parts within a close tolerance could result in extremely high costs, some adjustment of the weight of the blade must be provided. This adjustment should permit the addition or removal of weight at the blade tip and, possibly, at the blade root as well. In many blade designs, the adjustable tip weights are installed on at least two separate chordwise attachment points (par. 5-6.2.4). The location of the adjustable weights at the tip takes advantage of the large balance arm about the reference datum, which usually is the center of rotation. Spanwise balance is achieved by adjusting the total weight at both attachments, whereas the chordwise CG is corrected by transferring weights between the chordwise positions. The limits of adjustment are reached when either attachment is completely empty or is completely filled with weights.

To establish individual dynamic balance, both the spanwise and product moments must be controlled to maintain a common dynamic axis for all blades of a particular model. The dynamic axis $\overline{X}$ is expressed as follows:

$$\overline{X} = \frac{\int_{m_e}^{m_R} xy\,dm}{\int_{m_e}^{m_R} y\,dm}, \text{in.} \qquad (5\text{-}8)$$

where

dm = increment of blade mass, slug

e = location of flapping hinge from the center of rotation, in.

m_e = mass of spanwise increment at inboard end of blade (e), slug

m_R = mass of spanwise increment at outboard end of blade (R), slug

R = blade radius, in.

x = chordwise distance from blade leading edge to centroid of mass increment, in.

y = spanwise distance from flapping hinge to centroid of mass increment, in.

Eq. 5-8 implies that the weight of each element or component must be rigidly controlled. However, in practice this is not necessary because a system can be established to match relatively heavy parts with those that are on the light side of the tolerance scale.

Because it would be quite cumbersome to match or select each and every part of the blade assembly, only those components that make up the bulk of the weight need be considered. This method of selective assembly divides the rotor blade into four main components or groups: the spar or spar assembly; the leading edge material, including ballast; the aft section skins and stabilizing material; and the trailing edge reinforcement. These four major components are selected because they comprise the basic structure of a blade and extend the full length of the blade span. Variations in the weights of the remaining parts have little significance in the total weight and balance of the complete blade.

5-6.3.2 Component Limit Weights

By selecting the major components on the basis of their respective weights and moments relative to the available weight adjustments, virtually all blades can be balanced to a master balance blade. The weight variation in each part *shall* be determined by the available capacity of the attachments for adjustable weight. Weight limits for each part may be calculated by assuming that all other parts are of nominal weight and that one of the weight attachments is completely empty or full. When two tip weight attachments are used, the acceptable weight tolerance on forward components (e.g., spar and abrasion strip) is limited by the forward tip weight capacity, whereas the weight tolerance on aft components (e.g., trailing edge and skins) is limited by the aft tip weight capacity.

When this method of weight adjustment is used, the limit weight for each component may be obtained by the solution of simple pairs of simultaneous equations. The equations are set up in terms of spanwise and chordwise (or product) moments where the sums of the moments of empty or full attachments and the two unknown weights are equated to the sums of the nominal moments on the same components, as shown in Table 5-2.

The steps that follow (using data from Table 5-2) show the solution for the minimum and maximum weight for one part (an abrasion strip):

1. Minimum allowable weight (forward attachment assumed full):

a. Spanwise moment is:

$0.19(155.50) + 82.61 W_{s_{min}} + 155.50 W_a = 459.30 + 15.55 + 15.55$

b. Product moment is:

$0.19(155.50)\ (0.75) + (82.61)(0.558) W_{s_{min}} + (155.50)(2.35) W_a = 256.30 + 11.66 + 36.54$

c. Solving these two equations will give $W_{s_{min}}$, the minimum allowable weight for the abrasion strip, lb, if all other components remain at nominal weight

2. Maximum allowable weight (forward attachment assumed empty):

a. Spanwise moment is:

$0.0(155.50) + 82.61 W_{s_{max}} + 155.50\ W_f = 459.30 + 15.55 + 15.55$

b. Product moment is:

$0.0(155.50)\ (0.75) + (82.61)\ (0.558) W_{s_{max}} + (155.5)(2.35 W_f) = 256.30 + 11.66 + 36.54$

c. Solving these two equations will give $W_{s_{max}}$ the maximum allowable weight for the abrasion strip, if all other components remain at nominal weight. In both the solutions the values W_a and W_f, the aft and forward adjustable weights, respectively, must be ≤0.19 lb, the maximum capacity of the adjustable weight attachment. A similar set of simultaneous equations is solved for each of the other three critical components.

Nomograms can be prepared for convenience in the selection of the four or more critical weight components. These nomograms combine into a single graphical format all the minimum and maximum component weights determined by the procedure stated previously. Similarly, these results can be com-

TABLE 5-2. EXAMPLE OF NOMINAL WEIGHT AND CG LOCATIONS

PART		WEIGHT, lb A	CG LOCATIONS		MOMENT	
			SPANWISE, in. B	CHORDWISE, in. C	SPANWISE, lb–in. A x B = D	PRODUCT, lb–in.^{2} C x D = E
ABRASION STRIP		5.56	82.61	0.558	459.30	256.30
[1] ADJUSTABLE TIP WEIGHTS	FORWARD[1]	0.10	155.50	0.750	15.55	11.66
	AFT[1]	0.10	155.50	2.350	15.55	36.54

[1] MAXIMUM CAPACITY 0.19 lb

TABLE 5-3. ROTOR BLADE BALANCE (SAMPLE)

PART		WEIGHT, lb A	SPANWISE MOMENT COEFFICIENT, in. B	SPANWISE MOMENT, lb-in. A x B C	PRODUCT MOMENT COEFFICIENT, in.^{2} D	PRODUCT MOMENT, lb-in.^{2} A x D E
SUB-TOTAL COMPLETED BLADE UNBALANCED		▨	▨	2156.9	▨	3599.6
ADJUSTABLE TIP WEIGHTS ADDED	FORWARD	0.084	155.50	13.1	116.60	9.8
	AFT	0.169	155.50	26.3	365.40	61.8
SUB-TOTAL PRELIMINARY BALANCE BLADE		▨	▨	2196.3	▨	3671.2
PRELIMINARY DYNAMIC AXIS (CHECK)		SUB-TOTAL COLUMN E SUB-TOTAL COLUMN C	=	$\frac{3671.2}{2196.3}$	= ②1.672 in.	
ADJUSTABLE TIP WEIGHTS	FORWARD	- ①0.005	155.50	-0.8	116.60	-0.6
	AFT	- 0.005	155.50	-0.8	365.40	-1.8
TOTAL FOR TEETER BALANCED BLADE		▨	▨	2194.7	▨	3668.8
FINAL DYNAMIC AXIS (CHECK)		TOTAL OF COLUMN E TOTAL OF COLUMN C	=	$\frac{3668.8}{2194.7}$	= ②1.672 in.	

NOTES: ① THIS WEIGHT ADJUSTMENT IS MADE WHEN THE BLADE IS TEETER BALANCED. THE NEGATIVE SIGN INDICATES WEIGHT WAS REMOVED.

② DYNAMIC AXIS AS MEASURED FROM THE LEADING EDGE.

bined into a system employing a digital computer to provide rapid component selection from a number of random-weight parts.

For final balance the spanwise and product moments of all of the blade components, including the paint and adhesive, are obtained for the unbalanced blade. Again, two simultaneous equations can be written. The unbalanced spanwise moment and the forward and aft tip weight moments should be equated to the required spanwise moment of the master blade. The unbalanced product moment along with the forward and aft tip weight moments should be equated to the desired product moment. Solving these equations simultaneously will yield the additional weight required at each location for dynamic balance. A summary of the balance procedure is shown in Table 5-3.

By establishing a weight tolerance for each of the major selective components and using a consistent method of part selection, the blade assembly will, in nearly all cases, balance within the capacity of adjustable weight attachments. Upon final assembly of the blade, it *shall* be balance-checked against a master blade. The tolerance on the actual balance depends on both the type of rotor, e.g., fully articulated or hingeless, and the size of the blade. Tolerances of the order of 10 in.-oz are not uncommon. This physical balance of the blade must be performed on a balance stand capable of registering the blade spanwise moment to within the specified tolerance. Weight should be added or removed as required to balance the new blade. The balance master *shall* be established as that blade to which all other blades of a particular part number or series *shall* be balanced. This demonstration *shall* be accomplished by balancing each blade either directly against the master balance blade or against a calibrated mass balance for which the master blade was the calibration standard.

The spanwise teeter balance discussed previously demonstrates only that the blade will be in flywheel balance; however, the dynamic chordwise balance still may be out of tolerance. Dynamic chordwise balance, therefore, must be checked by tracking the blades at various rpm and collective pitch settings. If the selection of parts was controlled during the fabrication of the blades, a minor adjustment, such as moving adjustable tip weights forward or aft, will correct any dynamic chordwise deviation.

5-6.3.3 Track

To confirm interchangeability, each blade should be tracked prior to its release for installation. However, tracking of hingeless blades is difficult because the deflections at the tip are small. Interchangeability of these blades can be confirmed by using a closely controlled weight and balance system during manufacture. The prerelease tracking of all other blades will be made against master tracking blades. The master blade(s) are blade(s) that have been fabricated as closely as possible to design specifications and to as precise tolerances. These master tracking blades are produced so that, when they are installed, the controls are adjusted to the nominal position. It then can be ascertained how much deviation or tolerance may be allowed on production blades. Consideration of allowable tolerances *shall* include the crew comfort levels defined in MIL-H-8501.

Interchangeability with master blades must be determined either on a tiedown aircraft or on a suitable tower prior to release for random installation. At least one master blade must be tracked with each group of production blades. The blades should be tracked at several rotor speed settings typical of those that will be encountered during operation and at several values of collective pitch, with rotor speed held constant. Track readings *shall* be taken for each blade at each speed and pitch setting. Typical data are shown in Figs. 5-29 and 5-30.

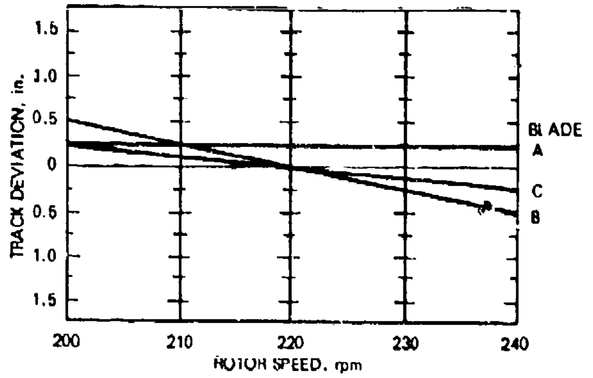


Figure 5-29. Track With Varying rpm (Zero Collective Pitch)

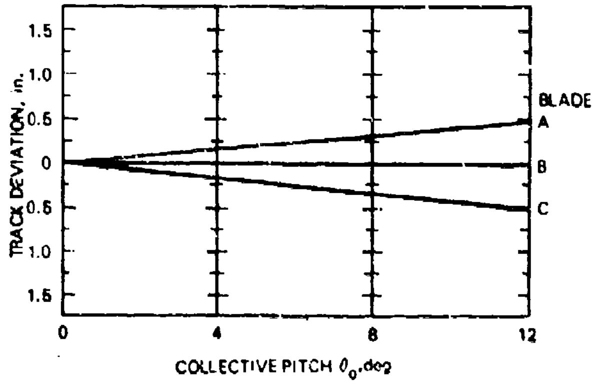


Figure 5-30. Track With Varying Collective Pitch (Constant Rotor rpm)

A typical plot (Fig. 5-29) of the tracking data recorded by any one of several tracking methods during an rpm sweep of an articulated rotor indicates that blade (A) is aerodynamically similar to the master; however, an incidence or pitch adjustment is required to correct the blade track to zero. Blades (B) and (C) can be corrected by downward trim tab adjustment. The appropriate adjustments should be made to bring all blade tracks within the tolerance level compatible with crew comfort levels previously established. This tolerance will depend on rotor size but commonly will be equivalent to differential blade coning angles of the order of 5 min.

A typical plot (Fig. 5-30) of track data during a collective pitch sweep with the same articulated rotor depicts blades (A) and (C) out of track due to dynamic unbalance. This condition is corrrected by moving a portion of the adjustable weight of blade (A) forward while that of blade (C) is moved aft. Blade (B) is seen to be dynamically similar to the master blade without adjustment. Weight adjustments and/or trim tab or trailing edge adjustments will provide blades that are dynamically and aerodynamically alike, permitting interchangeability with all other blades of that configuration. In both Figs. 5-29 and 5-30, the reference master blade is shown as a horizontal straight line through zero without slope.

Several methods of tracking blades may be employed; the accuracy, safety, and reliability of the electronic trackers provide excellent results.

In addition to the flat and collective tracking data, the blade pitching moments should be determined with a suitable calibrated load cell. Pitch link forces should match the master blade pitching moment within an acceptable tolerance for which it can be demonstrated that vibratory levels do not exceed the limits of MIL-H-8501 and that life-limiting oscillatory stresses are not induced.

5-6.4 ROTOR BLADE MATERIALS

As discussed in par. 5-6.2, a relatively broad variety of materials may be used in rotor blade construction. This paragraph considers the major factors that lead to the selection of specific materials, based upon the inherent properties of the materials and irrespective of the details of construction.

Helicopter rotor blades are unique in that many conditions that must be met depend upon various combinations of material properties. A rotor blade must be designed as an integrated part of the complete rotor system. One specific requirement is that the mass moment of inertia of the rotor system must be of at least a minimum value to provide satisfactory autorotational characteristics. This requirement has much influence in the establishment of the minimum mass distribution for the rotor blade. No weight savings can be realized beyond the limit imposed by this requirement; thus, a point exists beyond which an increase in the strength-to-weight ratio of the material cannot reduce blade weight. Although a high strength-to-weight ratio is desirable, more important factors are the ratios of both the fatigue strength and density to the modulus of elasticity of the material.

The total loads on a rotor blade cannot be predicted by a straight forward examination of rotor thrust and centrifugal force. The loads depend upon the response of the blades to the periodic airloads which themselves are affected by the blade motion. The blade response also depends heavily upon the mass distribution. A change in stiffness affects the bending moments, deflection, and radii of curvature of the blade to the extent that the response is changed. It is impossible to predict — without a re-evaluation of the blade response — whether a change in stiffness will increase, decrease, or have no effect on the radius of curvature of the blade. In other words, the radius of curvature of a rotor blade does not have the simple proportional relationship to stiffness that exists in a static structure because the bending moment is a dependent variable. Nevertheless, the following equation from simple beam theory for the radius of curvature r is applicable:

$$\frac{1}{r} = \frac{M}{\Sigma EI}, \text{in.}^{-1} \tag{5-9}$$

and bending stress σ_B in a particular material with modulus E is

$$\sigma_B = \frac{Mc_aE}{\Sigma EI}, \text{ psi} \tag{5-10}$$

Substituting Eq. 5-9 in Eq. 5-10,

$$\sigma_B = \frac{c_aE}{r}, \text{ psi} \tag{5-11}$$

where

c_a = distance from beam neutral axis to outer fiber, in.
E = modulus of elasticity, psi
I = moment of inertia, in.^{4}
M = bending moment, in.-lb

If a stiffness change is made in such a way that the distribution of mass and stiffness is unchanged, the blade response, and thus the radius of curvature, also will be substantially unchanged. Then, as in Eq. 5-11, the blade bending stress will increase in direct proportion to the material modulus of elasticity E. It follows that the most desirable rotor blade material is the one that has the highest ratio of strength to modulus of elasticity. Any material with a high modulus of elasticity that does not have a proportionately high strength is undesirable.

Table 5-4 is a comparison of the ratios of material fatigue allowable (*FA*) to modulus of elasticity *E* for a sample of available rotor blade materials. The comparison uses fatigue strength because this factor is of primary importance in rotor blades. The values given are based on experience with actual structures and are less than the values obtained from laboratory specimen data; however, they are presented here for illustrative purposes only. Specific values of fatigue strength for metals, plastics, and sandwich structures are contained in MIL-HDBK-5, -17, and -23, respectively. Care must be exercised in using any given values for fatigue strength since the configuration of the specific component as well as the necessary manufacturing processes may adversely affect the material properties.

Considering only ratio FA/E as the criterion, Column 3 of Table 5-4 indicates steel is superior to aluminum, and Fiberglas or graphite is superior to either metal. Boron is not particularly attractive. Wood (spruce) is highly fatigue-resistant, but also has disadvantages that preclude serious consideration for present-generation helicopters.

Up to this point, the discussion of materials has dealt with blade bending only. When more than one material is used in a rotor blade, other considerations are necessary since a rotor blade operates in a rotating field. In this condition, strain compatibility determined by the ratio of modulus of elasticity *E* to mass density ρ becomes an important factor.

In a rotating field, the centrifugal force (CF) generated by each blade element is proportional to the mass density of the specific material and the position of the element along the blade radius, or span. When two continuous spanwise members, each of a different material, are side-by-side in a common centrifugal field, each will tend to strain an amount that is proportional to its respective mass density ρ and inversely proportional to its modulus of elasticity *E*. In most cases, the two members are bonded together with an adhesive that can transfer load from one to the other by shear, causing them to strain equally. This being the case, the material with the higher value of the ratio E/ρ will pick up load from the other material and be stressed higher than if it were rotating by itself. Thus, it is desirable that two or more materials, used in combination, have fairly similar values of E/ρ. Column 5 of Table 5-4 indicates that aluminum and steel are highly compatible in this respect, and Fiberglas in combination with steel or aluminum is acceptable. Boron and graphite are compatible with each other, but either should be used with caution in combination with steel, aluminum, or Fiberglas.

TABLE 5-4. COMPARISON OF MATERIAL PROPERTIES

MATERIAL	1 E, 10^6psi	2 FATIGUE ALLOWABLE FA, psi	3 $\frac{FA}{E}$	4 DENSITY ρ, lb/in.^{3}	5 $\frac{E}{\rho}$ 10^6 in.
ALLOY STEEL	29	±30,000	0.0010	0.28	103
ALUMINUM	10	± 6,000	0.0006	0.10	100
"E" GLASS/EPOXY					
UNDIRECTIONAL	6	± 8,700	0.0014	0.065	92
BIDIRECTIONAL	3.5	± 4,200	0.0012	0.065	54
"S" GLASS/EPOXY					
UNDIRECTIONAL	8	± 9,700	0.0012	0.074	108
BIDIRECTIONAL	5	± 4,900	0.0010	0.074	67
BORON/EPOXY					
UNDIRECTIONAL	36	±26,000	0.0007	0.074	486
BIDIRECTIONAL	21*	±13,000	0.0006	0.074	284
GRAPHITE/EPOXY					
UNDIRECTIONAL	30	±40,000*	0.0013	0.053	565
BIDIRECTIONAL	18*	±20,000*	0.0011	0.053	340
SPRUCE	1.4	± 2,000	0.0014	0.016	88

* DATA EXTRAPOLATED AND/OR ESTIMATED FROM NUMEROUS SOURCES.

In view of growing pressures to use advanced composites in aircraft structures, it is appropriate to examine the benefits, if any, to be derived from their application to rotor blades. The outstanding attractions of such materials are very high stiffness, high strength, and low weight. It has been shown that the ratios of these properties — rather than the absolute values — are of prime importance. From the standpoint of fatigue resistance (Column 2, Table 5-4), these materials appear to be very compatible with more conventional materials for use in rotor blades. The question, then, becomes whether there are overriding advantages to be gained from other characteristics, such as ballastic tolerance, or the high value of the ratio E/ρ.

The rotor blade dynamic response is highly dependent upon the rotating natural frequencies of the blade, and it is necessary that the blade be designed to avoid frequencies that are in resonance with any forcing functions. The expression for natural frequency of a rotating beam ω_{R_n} is

$$\omega_{R_n} = \sqrt{a_n^2 \left(\frac{EI}{ml^4}\right) + K_n\Omega^2}\ , \text{ rad/sec} \qquad (5\text{-}12)$$

where

a_n = coefficient which is dependent upon mass and stiffness distribution and has a different value for each mode of vibration, dimensionless

EI = stiffness, lb-in.^{2}

K_n = coefficient dependent upon mass distribution and the mode of vibration, dimensionless

m = mass per unit length of beam, slug/in.

l = length, in.

Ω = rotational speed, rad/sec

For hinged beams, the values a_n and K_n for the first three modes are:

$a_1 = 15.5, a_2 = 50.0, a_3 = 105.0$

$K_1 = 6.38, K_2 = 17.65, K_3 = 3.50$

To examine the effect of stiffness EI on natural frequency, an example is presented: a constant-cross-section blade of 25-ft radius with a weight of 4.0 lb/ft and a flapwise EI of 20×10^6 lb-in.^{2} A tip speed of 680 fps is assumed, giving a rotational speed Ω of 27.2 rad/sec; hence

$$\omega_{R_1} = \sqrt{(15.5)^2\left[\frac{20 \times 10^6}{\dfrac{4 \times (25 \times 12)^4}{12 \times (32.2 \times 12)}}\right] + 6.38(27.2)^2}$$

$$= \sqrt{695 + 4720} = \sqrt{5415}$$

$$= 73.6 \text{ rad/sec}$$

which is

$$\frac{\omega_{R_1}}{\Omega} = \frac{73.6}{27.1} = 2.72$$

or

$$\omega_{R_1} = 2.72\Omega$$

Now, assuming that the stiffness EI is increased to 40×10^6 lb-in.^{2},

$$\omega_{R_1} = \sqrt{1390 + 4720} = \sqrt{6110}$$

$$= 78.2 \text{ rad/sec}$$

Thus, for the first flapwise bending mode on a typical hinged rotor blade, a 100% stiffness change results in a change in rotating frequency $\Delta\ \omega_{R_1} = 4.6$ rad/sec, or 6%. This mode of vibration is critical in an articulated rotor, and the only effective way to control it is by varying the mass distribution, because little can be done by changing the stiffness.

Further examination would show that the first chordwise mode of vibration, as well as the higher modes in both planes, is affected significantly by blade stiffness. For hingeless or semirigid rotors, all modes are affected significantly by blade stiffness. In these types of rotors, the higher modes of vibration are manifest primarily in blade stress levels, as opposed to vehicle vibrations. For multibladed rotors, the higher vibration modes also can contribute significantly to vibrations.

Material selection also can be very important in rotor blade fabrication. It now is possible to produce nonuniform blade cross sections in any of the available materials, although it generally is easier with composites, which can be molded. As in instances where noncompatible values of E/ρ can cause high stresses in flight, it also is important to avoid material combinations with greatly differing coefficients of thermal expansion. Such assemblies can develop high residual stresses as a result of adhesive bonding operations.

Ideally, a rotor blade should be made of materials that are highly resistant to both corrosion and erosion. Corrosion resistance of the nonmetallic composites is highly attractive and can be influential in material selection. In a monolithic composite blade, it is necessary to protect the forward portion against erosion. The most effective materials for this purpose are stainless steel, nickel, or cobalt abrasion shields. Of the elastomeric materials, the urethanes are superior and are very durable when subjected to sand, but generally have been found to have short lives when rain is a significant part of the environment.

It has been determined that rotor blades are vulnerable especially to lightning. To avoid damage to blades subject to lightning strikes, provision must be made for low-resistance paths for the high currents that are characteristic of lightning. The basic lightning protection requirements for all aerospace systems are given by MIL-B-5087. For blades constructed of composite materials — inherently poor conductors — or even those of all-metal bonded construction, adequate protection against lightning damage *shall* be demonstrated by test (see par. 8-9.4, AMCP 706-203.) Valuable preventive design guidelines are given in Chapter 7, AFSC DH 1-4.

5-7 ROTOR SYSTEM FATIGUE LIVES

5-7.1 GENERAL

Critical helicopter components are subject to a load spectrum characterized by a relatively high-frequency oscillating load content. Characteristically, the rotor system — particularly the main rotor — produces and endures the highest cyclic loading. A fundamental design requirement is long life of rotor system components. Resonant conditions that produce high or damaging cyclic stress levels within a component must be avoided. However, compliance with these requirements can be verified only through the correlation of flight test and component fatigue test data. The discussion that follows supplements the description of fatigue life determination given in par. 4-11, AMCP 706-201.

Corrosion has a rapidly degrading effect on the fatigue strength, and related life, of a particular component and this effect is difficult to predict in an accurate quantitative manner. Corrosion-resistant materials and/or proven corrosion protection methods thus should be used to obviate the necessity of considering corrosion in the determination of rotor system component fatigue life.

Fretting is the erosive failure of the metal surface as the result of small displacements of heavily loaded mating parts. All preventive methods practicable should be employed in the design and development phase to preclude the occurrence of fretting between components, particularly critical, highly stressed blade/hub retention areas. Fretting occurs commonly in areas such as tension-torsion strap packs, bearings, (particularly low-angle oscillating applications), retention hole bushings, and blade/hub attachment fittings. The degree of success in preventing the occurrence of fretting is determined through careful inspection of various components that have been subjected to fatigue tests that simulate actual installations and loads.

Rotor system components with long lives can be attained by implementing a combination of techniques in the initial design. Preliminary calculations of rotor system natural frequencies and loads can be made with a reasonable degree of accuracy for a prescribed number of representative vehicle flight conditions. The mission profile specified for the vehicle, coupled with load calculations for specific flight conditions, can define a preliminary load spectrum, consisting of load magnitude and frequency, as well as frequency of occurrence. These data, when combined with section property and therotical stress concentration factors for a component design, can be converted to steady and oscillatory *S-N* (stress versus number of cycles) data. Therefore, preliminary component life can be determined based upon cumulative damage and notched and unnotched material or similar fatigue test data. Coupon fatigue test data must be used with care since these data usually will not reflect accurately the effects of manufacturing processes that are peculiar to a specific component design. *S-N* test data for components of similar design and manufacturing process are more useful in the preliminary determination of component life (see par. 4-11, AMCP 706-201).

Although the previously described method can be employed in the preliminary design phase to predict component life, a more rigorous analysis of component fatigue and flight test data must be performed to determine the final life of the component. In the laboratory fatigue test it is necessary to simulate the actual combined loading conditions, particularly in areas of local attachment or where actual load paths may be in question. For example, at a rotor blade root-to-hub attachment, a meaningful representation of the flight condition includes combined cyclic flap and chordwise moments and shears plus torsion superimposed on the centrifugal force. Because, it is not feasible in many cases to include all associated or influencing components in the fatigue test, it often is important to simulate local flexibilities offered by flexures, bearings, etc., or to simulate local force inputs to the test article such as those offered by pivot point friction or lead-lag dampers.

The failure data acquired for an assembly quite often will involve the failure of only one component of that assembly. This component then becomes the limiting factor in the life of the assembly. If such a component is a replaceable item, it can be replaced periodically during testing, as failures occur, in order to acquire failure data for the longer-life components. Although the individual components of the assembly can be tested separately under simulated loading conditions, testing of the complete assembly usually is preferable in order to include the effects of load transfer between components. For bonded, welded, or otherwise permanently fastened assemblies, individual component test data must be acquired; or *S-N* data of such components in like material, process, and configuration may be employed, if available. These data should be modified by use of the Goodman diagram or other acceptable means to reflect the presence of steady loads as appropriate. Methods of obtaining acceptable component *S-N* data are discussed in par. 5-7.2.

5-7.2 ENDURANCE LIMIT TESTING

5-7.2.1 General

Endurance limit testing generally is required to obtain data adequate to guarantee the service life for rotor blades (see par. 7-4.2.2.2, AMCP 706-203). This is testing in which the material and/or part is subjected to repeated cycles of load, with or without a steady, or constant, load maintained. The endurance limit for most homogeneous, near-isotropic materials has been established, and is defined in MIL-HDBK-5. Fatigue data for plastics and sandwich construction are contained in MIL-HDBK-17 and -23, respectively. In most instances, these data are presented for both smooth and notched specimens. However, with the introduction of many and varied reinforced plastics and advanced composites, applicable fatigue test results are not yet available in the literature. Basic tests thus will be required for such advanced materials.

It is always necessary to exercise care in using data that are related to the shape of material under consideration. For example, the use of sheet stock in rotor blade design dictates the use of tension-tension fatigue data, when available, to predict skin fatigue life or the life of any part made from sheet stock. However, it may be more appropriate to use R. R. Moore rotating beam fatigue data when solid bar or plate stock is integrated into the blade design. Additionally, an applicable notch factor, either inherent in the design or resulting from the manufacturing process is of primary importance.

To confirm a material selection, it may be necessary to conduct coupon tests to substantiate a particular material condition not covered in the current literature. In conducting the coupon fatigue tests, it is extremely important that the test material has been subjected to the processes associated with fabrication of the critical component. In addition, tests *shall* be conducted on the pertinent material shape, using stock, bar, or plate as the design dictates. Other aspects of the proposed component configuration such as edge condition, fillet radii, or sharp bend radii in joggles or over similar fitting areas, should be included in the specimens tested.

Fatigue or endurance tests *shall* be conducted on a sufficient number of coupons for each condition. The number of coupons never *shall* be less than five. However, if the standard deviation of the data points for any test condition exceeds 15% of the mean stress, additional coupons *shall* be tested.

Coupon tests may be conducted on any suitable test fixture or stand capable of applying an alternating load. The alternating load may be superimposed on a steady, or mean, load to produce a load condition as shown in Fig. 5-31. In the normal rotor blade load spectrum, each load condition is a combination of steady and alternating loads. Therefore, the use of this loading condition for coupon testing is recommended.

Coupon design depends upon the type of material being tested. When the material is sheet metal stock some form of the "dog-bone" coupon should be used. The transition from the gage section to the grip area *shall* be such as to eliminate a stress concentration due to section change. An acceptable radius for such a transition is given in Ref. 47. Additionally, care must be taken with the edge condition of sheet stock coupons. The machining of the edge should be controlled so as to prevent thermal degradation of the material near the edge. Edges should be prepared to a polished condition by using fine grit sandpaper or by burnishing. In this manner a base line endurance limit can be established that will permit notch and safety factor reductions from a reliable reference.

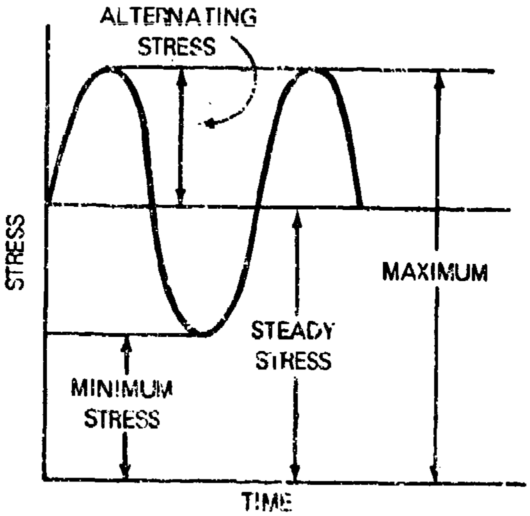


Figure 5-31. Alternating Stress Superimposed on Steady Stress

When stock thickness permits — e.g., in the case of plate, bar, castings, and forgings — it is preferable that the specimens be machined with a round cross section. The circumferential surface should be polished in order to remove any stress concentrations or notch effects.

In order to predict allowable fatigue strength for a part or component, notched specimens should be tested to determine the resulting fatigue strength reduction. Experience has shown that a typical value of this notch factor K_T is about 3.0 for almost all metals. The notch factor K_T, which may be calculated or be based on test data, is the ratio of the peak stresses in notched and unnotched specimens.

Data from tests conducted using the appropriate configuration of coupons will establish an *S-N* curve. In developing each *S-N* curve, a minimum of five specimens should be tested at varying alternating loads with the same steady load.

The stress level at which no failure occurs after 10^7 cycles for ferrous metals establishes the endurance limit of the material. A family of *S-N* curves, each for a different steady load will provide sufficient data to plot a Goodman diagram (see par. 4-11, AMCP 706-201). The Goodman diagram, in turn, will permit the conversion of a particular load condition into an equivalent load condition of different steady and alternating load levels. MIL-HDBK-5 presents the data in a constant-life diagram rather than the Goodman diagram. In any case, a diagram constructed from coupon data should be revised as component and assembly test data are generated.

5-7.2.2 Nonmetals

The use of plastics reinforced with glass, graphite, and/or other advanced composite materials in the construction of rotor blades will require the development of both *S-N* curves and rational Goodman diagrams. Additionally, the processing of these materials is subject to variations among manufacturers. Therefore, care must be exercised that test specimens are representative of the material and processes to be used in the blade construction. Particular attention should be given to fiber orientation with respect to the principal axis of loading.

The establishment of a family of *S-N* curves similar to Fig. 5-32 is an acceptable method of determining allowable fatigue strength for a particular reinforced plastic or advanced composite material (ACM). These fatigue strengths, or endurance limits, *shall* be

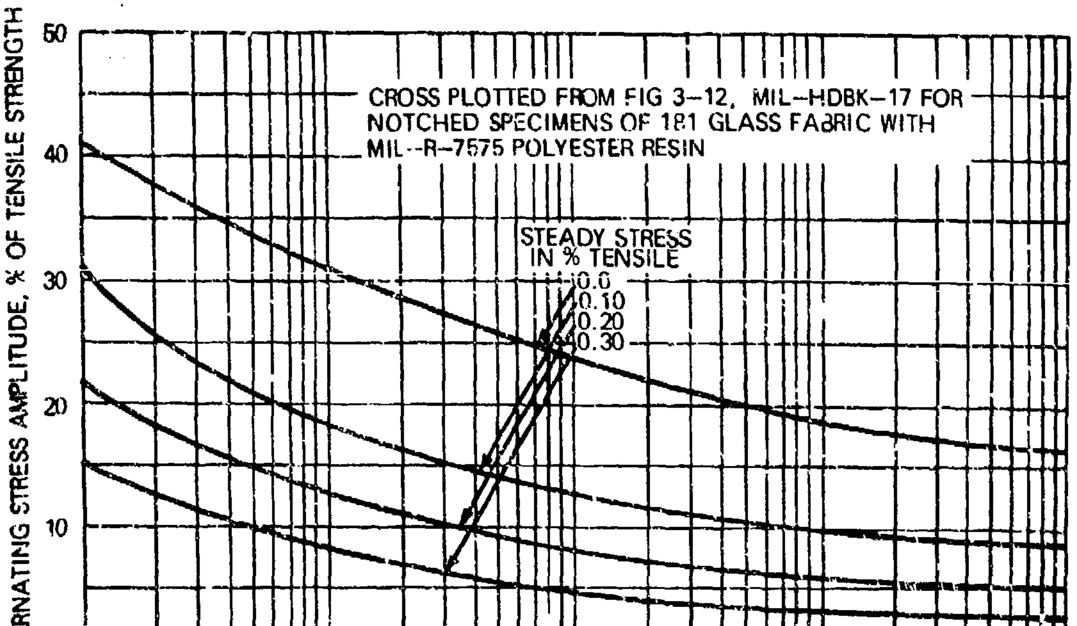


Figure 5-32. Alternating Stress vs Cycles at Various Steady Stress Levels (Cross Plotted from Fig. 3-12, MIL-HDBK-17 for Notched Specimens of 181 Glass Fabric With MIL-R-7575 Polyester Resin)

established for a particular steady load at a minimum 5×10^7 cycles without failure.

The objective of the coupon test is to establish or verify the endurance limit of a material for several combinations of steady and alternating loads. Because it would be unconservative to extrapolate an equivalent alternating stress from a combined steady and alternating stress for reinforced plastics, a family of curves similar to those shown in Fig. 5-32 will be required to evaluate fatigue damage.

5-7.2.3 Structural Members

Following the establishment of material endurance limits from coupon test data, those parts that carry primary and secondary oscillatory loads should be tested. Components such as the blade spar, which may be one continuous member or a built-up section, should be tested thoroughly prior to their incorporation into the complete assembly. Testing of such parts and subassemblies will provide test data valuable for further adjustment and refinement of the stress diagrams obtained from coupon tests. Additionally, testing of critical structural members such as the spar reduces the cost of testing full-size blades or blade sections. The discussion that follows supplements the test requirements delineated in Chapter 7, AMCP 706-203.

In many blade configurations, a full-length member such as the spar lends itself to electromechanical vibratory testing or other simple loading methods involving minimum fixtures. The extreme fiber stress due to flap bending often is experienced directly by the spar, while — because of a location close to the neutral axis in the chordwise bending plane — the effect of loading in this plane may not be significant. To obtain usable data, the part or subassembly must be instrumented and calibrated to known load conditions prior to conducting the fatigue test.

A minimum of three specimens of each significant structural member should be tested to ascertain the fatigue strength in the manufactured condition. It is extremely important that the processing of these specimens be identical to that of the final production unit.

The data generated from part or subassembly tests will compare with coupon data discussed previously. The shape of the *S-N* curve for most metals is shown in MIL-HDBK-5. A rational method of curve fitting such as is described in Chapter 9, MIL-HDBK-5, *shall* be used when no reference curves are available. The data obtained by using an electromechanical test machine would be simple alternating stress (zero steady stress). These data can be used to refine the Goodman diagrams in a manner similar to that shown in Ref. 48. Additional data from tests of other parts of the same material may be used to define further the allowable fatigue envelope.

In many instances the retention holes of the root blade fitting receive special processing. Test data to substantiate the endurance limit of the root retention fitting can be obtained by testing the individual fitting rather than the entire blade or root section. The effect of bearingizing, (a special rolling treating of the bearing surface of a hole) shot peening, or other such treatment to improve the fatigue life should be evaluated at this time. Component tests of specimens selected as beyond normal tolerance also can be used to provide data to assist in the establishment of limits of allowable defects and of overhaul and repair criteria. To the maximum extent possible, test loads *shall* simulate the condition(s) experienced in flight test and be considered in the analysis. However, where well-defined stress diagrams exist, a combination of steady and alternating loads that may be converted to an equivalent alternating stress condition should be selected.

Testing of the extreme aft section member, whether or not it includes a trailing edge strip or other reinforcement at the aft terminus of the skin, likewise will provide valuable data for service life prediction. Because this member experiences the maximum fiber stress in the chordwise bending plane, tension-tension fatigue loading will provide acceptable data. Due to the relatively sharp contour presented by the trailing edge, failure may be precipitated at a relatively low stress level by a small nick or scratch. Although the principal stress is due to bending, the critical stress may be simulated as a tensile stress due to the small gradient. If practicable, it is advantageous for the test loads to duplicate the predicted stress combination in the trailing edge.

Additional parts or subassemblies peculiar to a specific design may warrant special endurance limit testing. Among such parts are the tip and/or inertia weight attachment fittings. These parts may be subjected to high-amplitude, low-cycle fatigue resulting from the start/stop centrifugal force and the attendant secondary moment and/or shear loads. The endurance limit for the fittings and attachments may be confirmed by duplicating the load conditions experienced in service. Other components and/or subassemblies should be tested whenever the construction of the blade does not permit accurate or reliable analysis.

5-7.2.4 Determination of Fatigue Life

Endurance limit testing *shall* provide the fatigue data necessary to permit the determination of a service life. Service life determination *shall* consider, as a minimum, the flight maneuvers and loading conditions of a realistic mission profile and the resulting frequency of occurrence of damaging stress cycles (see par. 4-11, AMCP 706-201).

5-8 PROPELLERS

5-8.1 GENERAL

The essential elements of propeller design are described in the paragraphs that follow. Included are a discussion of propeller dynamic behavior and how it is handled in design; information on the detail design of hubs, actuators, controls, and blades; and a description of how test data are used to verify that the propeller has a satisfactory fatigue life.

In many respects, the propeller design process is much the same as the design of a helicopter rotor. However, because of differences in the technology and therefore in various details of the process, this discussion for the most part is independent of the description in prior paragraphs of the rotor design process. Also, the design requirements specific to propellers generally are beyond the scope of this handbook. Therefore, the paragraphs that follow are only descriptive of the process and are provided for assistance in the integration of propellers into the design of compound helicopters.

Almost all propeller technology has developed from design work and experience with conventional aircraft applications. However, the information presented here is applicable to propellers for helicopters as well. Where appropriate, there are special comments relative to helicopter applications. Propellers of metal or composite material, with hydraulic means for controlling blade angle, are emphasized. Information on other kinds of propellers, such as fixed-pitch wooden versions or those with electrical blade angle actuation, may be found in ANC-9.

The preliminary design procedure for choosing a propeller is described in considerable detail in par. 3-3, AMCP 706-201. The generalized performance and weight methods given therein allow an examination of all pertinent variables so that the best configuration can be selected. The best configuration is usually a compromise that depends upon the relative importance of cruise performance, takeoff thrust, and other characteristics. This systematic method of propeller selection has proven successful for fixed-wing aircraft and can be expected to provide the basis for the proper choice of propellers for helicopters.

Besides the fundamental performance parameters, noise frequently plays a major role in the selection of a propeller configuration. If noise is an important design criterion, some further compromise may have to be made in both performance and weight as quiet propellers generally require low disk loading and tip speed.

During preliminary design, the propeller diameter, number of blades, activity factor, integrated design lift coefficient, and rotational speed will be selected. The planform and twist distribution also will be selected, and the airfoil type and camber distribution defined. The significant performance parameters and the aerodynamic loads under important operating conditions then are computed for use in the mechanical design of the propeller.

5-8.2 PROPELLER SYSTEM DYNAMICS

5-8.2.1 Vibratory Loads

The structural design of a propeller is determined primarily by its aerodynamic configuration requirements, and by the structural capacity required to handle the aerodynamic loads. Although the centrifugal and steady aerodynamic loads must be taken into account, usually it is the vibratory loads that dominate the structural design. Basic vibratory loads originate from several sources, primarily the following:

1. Aerodynamic
2. Engine (These excitations, which generally are experienced with piston engines, are essentially of no significance with turbine engines and are not discussed here. The subject is treated briefly in Ref. 49.)
3. Gyroscopic and inertial
4. Stall flutter.

Vibratory aerodynamic blade loads are a result of the propeller operating in a nonuniform flow field, which causes the aerodynamic lift on each blade section to vary as the blade rotates. For conventional aircraft, the nonuniform flow field is primarily an angular inflow into the propeller disk resulting from the attitude of the aircraft, which varies with flight speed and gross weight. For helicopters, some of the factors that can cause propeller flow aberrations in direction, velocity, and density are listed in Fig. 5-33.

For normal flight operating conditions, the nonuniform propeller flow field is steady, and the variation in aerodynamic blade forces as the blade rotates is periodic. The blade forces at each azimuthal and radial position may be calculated by standard aerodynamic techniques such as are used for propeller performance computation. The harmonic components of the loading may be evaluated by Fourier analysis of the periodic loading. These harmonics are the P-order aerodynamic excitations — 1P, 2P, 3P, etc. — where P is the propeller rotational frequency. Although all of these excitations cause blade stress, the strongest and most important is that due to 1P (see Fig. 5-34) provided the dynamic design of the propeller system is handled properly.

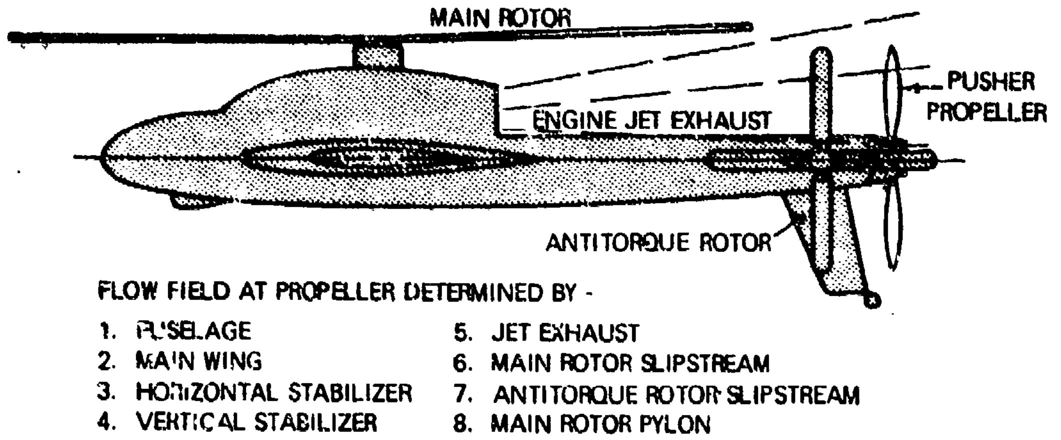

MAIN ROTOR
PUSHER PROPELLER
ENGINE JET EXHAUST
ANTITORQUE ROTOR
FLOW FIELD AT PROPELLER DETERMINED BY -
1. FUSELAGE
2. MAIN WING
3. HORIZONTAL STABILIZER
4. VERTICAL STABILIZER
5. JET EXHAUST
6. MAIN ROTOR SLIPSTREAM
7. ANTITORQUE ROTOR SLIPSTREAM
8. MAIN ROTOR PYLON

Figure 5-33. Propeller Flow Field for Compound Helicopters

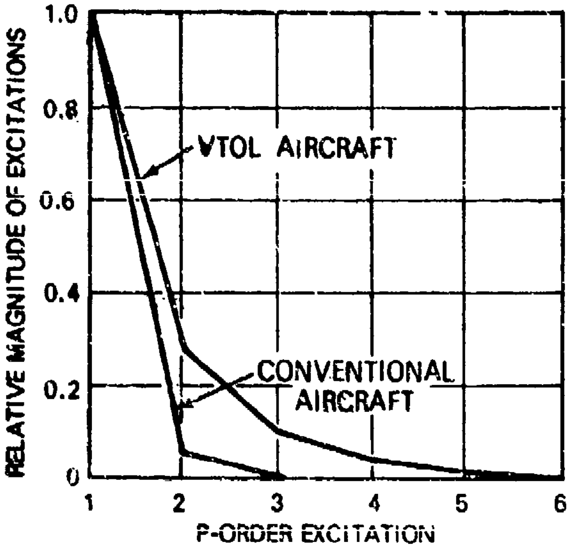


Figure 5-34. Comparison of P-order Excitations

The development of 1P-blade loads from angular inflow is depicted in Fig. 5-35, where blade #1 is in the retreating position relative to the inflow with reduced angle of attack and relative velocity and blade #3 is in the advancing position, with increased angle and velocity. This figure shows that although the resulting loads on the blades and propeller shaft vary at a frequency of 1P, the moment and side force loads on the airframe are always in the same direction. These airframe loads are steady if there are three or more blades. The variation in lift experienced by a blade section rotating around a propeller centerline inclined to the airflow is proportional to the product of the inflow angle A and the square of the aircraft indicated airspeed V_i, i.e., 1P-blade excitation, is proportional to AV_i^2. The dimensionless excitation factor EF is defined as

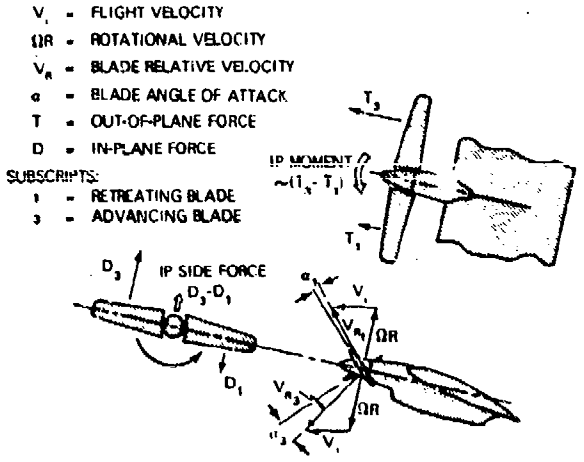

V₁ - FLIGHT VELOCITY
ΩR - ROTATIONAL VELOCITY
V_R - BLADE RELATIVE VELOCITY
α - BLADE ANGLE OF ATTACK
T - OUT-OF-PLANE FORCE
D - IN-PLANE FORCE
SUBSCRIPTS:
1 - RETREATING BLADE
3 - ADVANCING BLADE
T₃
T₁
1P MOMENT ~(T₃ - T₁)
D₃
IP SIDE FORCE
D₃-D₁
D₁
α₁
V₁
V_R1
ΩR
V_R3
ΩR
α₃
V₁

Figure 5-35. Propeller 1P Loads from Nonaxial Inflow

$$EF = A\left(\frac{V_i}{348}\right)^2 \text{, d'less} \qquad (5\text{-}13)$$

where

A = propeller inflow angle, deg
V_i = indicated airspeed, kt

An alternative expression used to indicate the severity of 1P aerodynamic excitation is Aq. The relationship between the two expressions is

$$Aq = 409\ EF \qquad (5\text{-}14)$$

where

q = dynamic pressure, lb/ft²

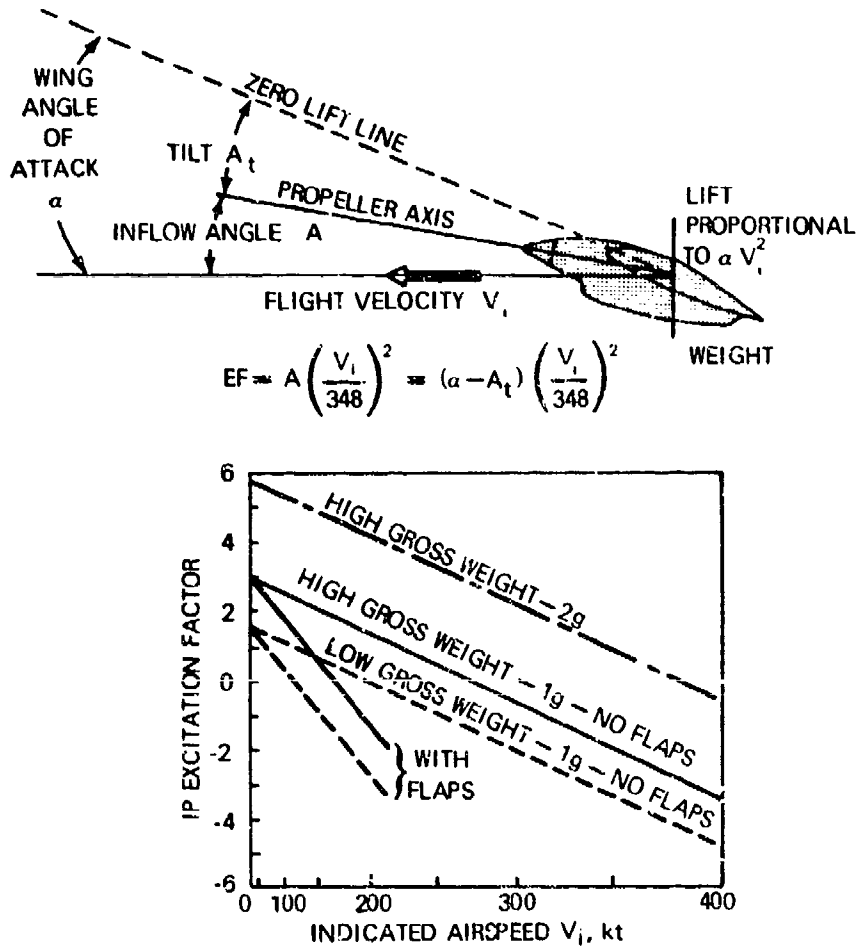


Figure 5-36. 1P Excitation Diagram for Typical STOL Aircraft

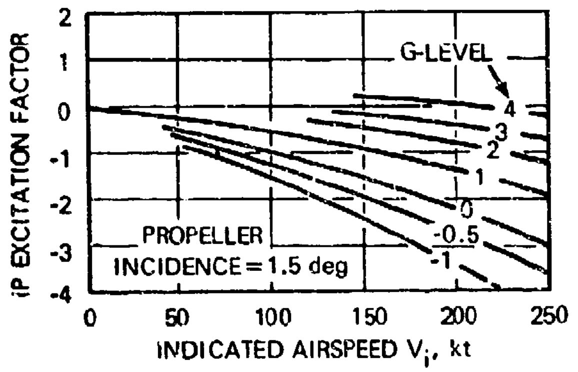


Figure 5-37. 1P Excitation Diagram for Helicopter With Pusher Propeller

In a fixed-wing aircraft, the effective angular inflow into the propeller is a function not only of the aircraft attitude, but also of the wash effects of the wings, fuselage, nacelles, stores, jets, etc. In general, these inflows vary approximately as the square of the aircraft velocity, so that the variation of the blade excitation *EF* can be depicted as shown in Fig. 5-36 for a fixed-wing STOL aircraft. The ordinate intercept is determined by the gross weight and the wing area, and the slope of the *EF* line is a function of the tilt of the propeller axis relative to the wing zero-lift line. (As indicated, wing flaps shift the zero-lift line and therefore the slope of the *EF* line.) It is customary in aircraft design to consider the 1P excitation factor when the nacelle alignment is being chosen. With an advantageous nacelle tilt, the excitation factor at high speed may be no higher than at low speed. For a pusher propeller on a helicopter, the *EF* diagram may take a form such as is shown in Fig. 5-37. In this case, it would be possible to reduce the 1P excitation by tilting the propeller axis so as to obtain virtually no angular flow into the propeller over the entire operating range for a given load factor level. The ordinate intercept is zero because the propeller is not affected by the wing, and the main rotor is the lifting device at low speeds.

Although Figs. 5-36 and 5-37 — which consider only the 1P excitation caused by angular inflow in the pitch direction — indicate speeds at which the 1P excitations are zero, this, in fact, seldom occurs because of the presence of yaw washes in addition to the pitch washes considered previously. In addition, if the mounting of the propeller is flexible, variation in the nacelle alignment must be included in the propeller load analysis as an aeroelastic effect.

Once the aerodynamic environment at the propeller plane has been defined, the aerodynamic blade loads can be calculated for various azimuthal and radial positions as indicated. However, because the blade deflects somewhat in the presence of these loads and thereby changes its angle of attack, the actual loads are slightly different from those for a rigid blade. The computation of the actual loads must take into account the derivative of blade load with blade angle changes.

5-8.2.2 Critical Speeds and Response

The response of the propeller blades to the vibratory aerodynamic excitation loads described in the preceding paragraph is determined by the structural and dynamic characteristics of the propeller system. The response, in turn, determines the stresses in the blades and the loads and stresses in the barrel, propeller shaft, and the aircraft itself. The dynamic characteristics of the propeller are described best by defining its critical speeds for the various aerodynamic excitation orders i.e., the rotational speeds at which the frequency of the aerodynamic excitation coincides with a natural propeller blade frequency. The relationship of P-order excitation and propeller blade frequency commonly is shown in a critical speed diagram such as that of Fig. 5-38. At propeller critical speeds, there may be high dynamic magnification of the aerodynamic loads. Therefore, the propeller system should be designed so that the lower, stronger, critical speeds do not fall within the operating speed range of the propeller. The operating range in the typical diagram of Fig. 5-38 may be seen to be free of critical speeds up to 8P.

The dynamic characteristics of the propeller system depend upon the number of blades and the mode of vibration associated with the aerodynamic order of excitation. The three basic modes for a four-bladed propeller — whirl, symmetrical, and reactionless — are illustrated in Fig. 5-39, which also shows how the engine can participate in the system response. For propellers with three or more blades, response to the 1P aerodynamic excitation does not involve the nacelle or the aircraft, because the resulting loads on the aircraft are steady. Hence, a conventional forced blade response (assuming a fixed hub) can be used (Refs. 50 and 51). Such a program must include the effects of the blade torsional dynamics and blade retention stiffness, and determines not only the 1P-blade loads and stresses, but also the resultant steady loads on the nacelle-aircraft structure. In order to avoid 1P-magnification, the blades should have a high first mode frequency and be torsionally stiff.

For four-bladed propellers, the 2P-, 6P-, etc., aerodynamic loads excite blade modes that are reactionless with respect to the aircraft, as shown in Fig. 5-39. Thus, the dynamic characteristics and response of the propeller blades to these excitations do not involve the nacelle-aircraft system and can be analyzed using the same analysis as used for the 1P-excitations. However, because of the low damping associated with these reactionless modes, it is important to place their critical speeds, particularly the 2P, outside of

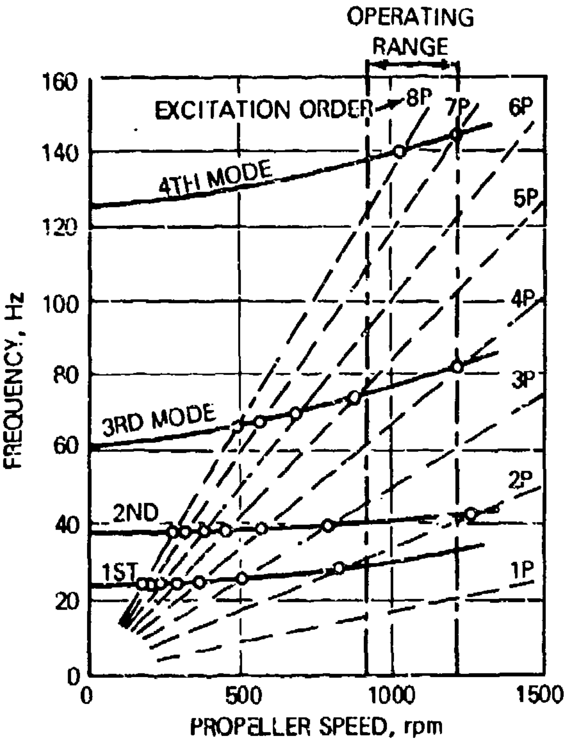


Figure 5-38. Propeller Critical Speed Diagram

the operating speed range with at least a 10% margin.

Because the values of critical speed change with blade angle due to the blade twist and centrifugal effects, blade angle should be considered in the evaluation of reactionless mode critical speed relative to the operating range. Also, the effective retention stiffness differs for the three kinds of propeller modes shown in Fig. 5-39 because of structural coupling within the hub. This effect must be included in dynamic and response calculations for the blade. In general, the retention stiffness is lowest for the reactionless modes and highest for the symmetrical modes.

Propeller aerodynamic excitations with a frequency order of one greater or one less than integer multiples of the number of propeller blades combine at the propeller hub to produce backward or forward whirl modes of the propeller, respectively. Because of this whirling action and the rotation of the propeller, these aerodynamic excitations appear on the gearbox-aircraft system as rotating shear and moment loads at frequencies corresponding to multiples of the number of blades. For example, in a three-bladed propeller, excitations at frequencies of 2P and 4P are felt by the gearbox as a 3P-whirl.

This interaction of the propeller dynamic system with the aircraft system must be taken into account in calculating the propeller blade whirl mode critical speeds. This can be done by a complete coupled analysis of a rotating, flexible propeller attached to a stationary aircraft dynamic system. It also can be calculated by first determining the variation with frequency of aircraft system whirl impedance, e.g., angular and radial deflection of the propeller shaft for unit shear and moment whirl loads, and then including the aircraft impedances in the propeller critical speed analysis.

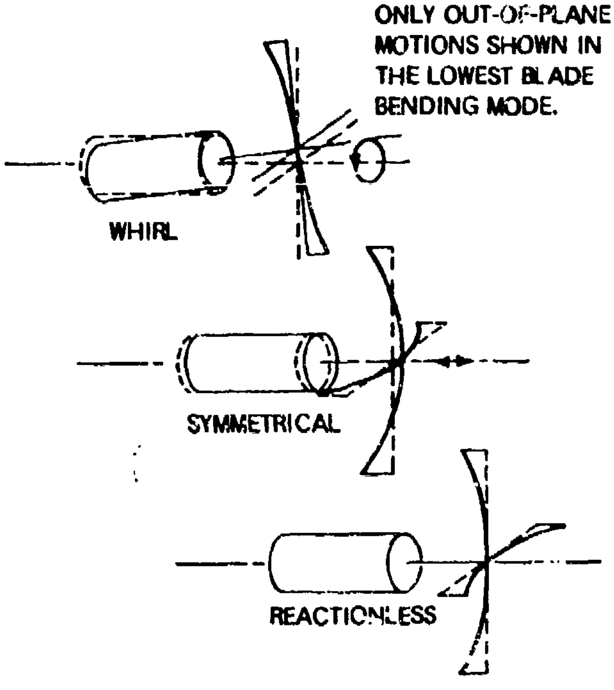

ONLY OUT-OF-PLANE MOTIONS SHOWN IN THE LOWEST BLADE BENDING MODE.
WHIRL
SYMMETRICAL
REACTIONLESS

Figure 5-39. Propeller Vibration Modes

Aerodynamic excitations at frequencies that are multiples of the number of blades excite the propeller in a symmetrical mode, producing vibratory fore-and-aft and torque loads at the same frequency on the gearbox-aircraft system. Just as with the whirl modes, dynamic characteristics of the aircraft and transmission system must be included when symmetrical mode propeller critical speeds are computed. Again, this can be done with a coupled analysis, or by the impedance technique discussed previously.

Because the torsional impedance of a transmission system usually is low, symmetrical blade modes that are primarily inplane (putting vibratory torques on the shaft) will have considerably higher critical speeds than would be calculated for a fixed hub.

Because of the centrifugal stiffening effect and the twist of the blade, propeller critical speeds will vary with blade angle. This effect must be considered in placing the critical speeds properly. In general, it is customary to place the lower order whirl and symmetrical critical speeds at least 5% out of the normal operating range. Less margin is needed for these critical speeds than for the reactionless modes because of the much greater damping supplied by structural interaction with the aircraft system.

Once the dynamic characteristics of the propeller system have been determined, the magnitude of the response to the various propeller aerodynamic excitations can be determined. For modes that are being excited well below their critical speeds, a real-variable response analysis may be used (ANC-9). This always is possible for 1P-aerodynamic excitation and sometimes for 2P.

System response of the higher order aerodynamic excitations may be determined by an energy method, using the calculated normal modes of the propeller and assuming the structural and aerodynamic damping from experience. Another method is to use response analysis, such as given in Refs. 50 and 51, with complex variables so as to include structural and aerodynamic damping. The former (energy) method uses the normal modes and natural frequencies obtained from dynamic analysis of the propeller system, and determines the response of the blade to a particular aerodynamic excitation order by equating the energy dissipated through damping with the energy introduced by the excitation. Experience shows that the effective overall damping, aerodynamic plus structural, varies with the type of vibration mode, being about 0.02 to 0.04 of critical for reactionless modes, and about 0.04 to 0.06 for whirl and symmetrical modes.

From the response of the blades to the various aerodynamic excitations, one can determine the blade stresses, retention and shaft loads, and, finally, the loads applied to the gearbox and aircraft. Certain excitation orders put vibratory torque, but not vibratory bending moment, on the gearbox; others do the opposite. Also, 2P-excitation on a four-way (four-blade) propeller puts no load at all on the gearbox, as this is a reactionless mode, i.e., all the loads are reacted within the hub.

A propeller must have the structural capacity to withstand the combined loading from its response to all of the aerodynamic excitation orders superimposed.

5-8.2.3 Gusts and Maneuvers

Gusts and maneuvers can have significant effects upon propeller vibratory loads. The more obvious effects are caused by changes in the aerodynamic flow fields and the consequent excitations to which the blades are subjected. Secondary effects are the result of gyroscopic motion and inertia forces. Because the basic frequency of the vibratory loads is the propeller rotational speed, many stress cycles can be accumulated on the propeller during a gust or maneuver. This is in contrast to nonrotating airframe components, which are subjected to only one major load cycle during a gust or maneuver.

A gust can have velocity components in three directions: longitudinal, lateral, and vertical. The longitudinal, or fore-and-aft, component essentially is parallel to the flight path of the aircraft, and, therefore, subjects the propeller and airframe to a change in dynamic pressure. The steady torque and thrust on the blades change with a suddenness that depends upon the rise time of the gust. These changes in load can be relatively high for the large propellers used in V/STOL aircraft because the blades of these propellers are operated at relatively low angles of attack. The change in dynamic pressure also has a direct effect on the 1P excitation factor, and the 1P-stresses are affected accordingly. A longitudinal gust changes the lift on the aircraft, thus imparting vertical accelerations and changing the wing circulation, which, in turn, has an effect on the flow field.

The lateral component of a gust can be treated as a change in yaw inflow to the propeller. The yaw inflow is added vectorially at right angles to the normally considered pitch inflow. Hence, the total 1P-inflow angle is affected. Likewise, the dynamic pressure is changed by the cross-flow component, but for the same gust velocity the wing lift is affected less by a lateral gust than by a longitudinal or vertical gust.

Vertical gusts have direct effects upon the pitch component of the inflow angle, the dynamic pressure, and the wing lift. Each of these factors influences the flow field and, consequently, the excitations and loads. As in the case of a lateral gust, the vertical component is added vectorially to the forward airspeed. This changes the magnitude and direction of the velocity inflow.

The current method for determining propeller vibratory loads during gusts uses a quasi-steady-state analysis to evaluate the flow field and aerodynamic excitations. Although the propeller speed and blade angle may change, depending upon the rise time of the gust, it is expedient and conservative to assume a step change in the inflow to the propeller. In other words, the propeller is assumed to be placed suddenly in a different aerodynamic environment without any change in blade angle or propeller rotational speed, and the propeller loads and responses are determined in the manner discussed in the preceding paragraphs.

Propeller vibratory loads incurred during maneuvers are determined by using essentially the same procedures as for gusts, with the exception that the maneuvers analyzed generally are limited to those involving vertical load factors.

Aircraft design specifications (MIL-A-8860 series) do not include the time duration of each maneuver nor a breakdown of the maneuvers as functions of airspeed. The maneuver spectrum (see par. 4-11, AMCP 706-201) must be available to the propeller designer so that he can assure that the fatigue lives of the propeller components will be satisfactory.

In the design analysis, blade vibratory loads for maneuvers involving vertical load factors are calculated using the procedures given in the preceding paragraphs and considering that the effective gross weight of the vehicle is its actual gross weight multiplied by the vertical load factor.

Maneuvers influence blade vibratory loads not only by changing the aerodynamic flow fields but also by the resulting effects of gyroscopic motion and inertia forces. In a pullout or pushover maneuver, the angular velocity of precession Ω_1 is equal to

$$\Omega_1 = \frac{(n_z - 1)g}{V}, \text{ rad/sec} \qquad (5\text{-}15)$$

where

n_z = load factor, dimensionless

V = flight speed, fps

The resulting blade loads can be calculated using a procedure like that used for calculating the response due to 1P-aerodynamic excitation. For this analysis, the load is a function of the mass distribution of the blade and is applied perpendicular to the plane of the propeller (out-of-plane). Like 1P aerodynamic excitation, gyroscopic motion induces a 1P-moment on the propeller shaft, which, for blades having three or more blades, exerts a steady bending moment M on the aircraft as expressed in

$$M = I_P \Omega_1 \omega \text{ ,ft-lb} \qquad (5\text{-}16)$$

where

I_P = propeller mass, mass moment of inertia, slug-ft²

ω = propeller speed, rad/sec

Inertia loads result from the vertical load factor applied to the propeller. As in the gyroscopic analysis, the load is a function of the mass distribution of the blade, but in this case it is applied inplane. The 1P shear force F on the shaft is simply

$$F = n_Z W_P \text{, lb} \qquad (5\text{-}17)$$

where

W_P = weight of the propeller blades and hub, lb

There may be other special occasions where loads due to maneuvers should be considered. For instance, a tail propeller of a helicopter may be subjected to large precession rates in yaw while hovering.

5-8.2.4 Stall Flutter

Propeller blades must be designed not only to handle the applied aerodynamic excitation loads and to have the appropriate dynamic characteristics, as discussed in the preceding paragraph, but they also must be designed to be free of flutter. Classical bending-torsion flutter is not of concern because of the large separation between the fundamental bending and torsional frequencies of propeller blades (Ref. 52). However, stall flutter is a major concern because of its potentially destructive torsional vibration. There are two apparent causes of high torsional blade vibration: aerodynamic hysteresis and Karman vortices (Ref. 53). Aerodynamic hysteresis can cause divergent, self-excited torsional vibration and is, therefore, true flutter. The Karman vortex excitation, however, is not true flutter but a forced excitation. It nevertheless is similar to hysteresis stall flutter and can cause large amplitudes of structural response and possible failure.

Torsional dynamic divergence due to stall flutter occurs because of the phase lag in the aerodynamic circulation variation with airfoil torsional motion. Because the vortex formation must travel to infinity before full circulation develops, the airfoil angular motion tends to lead the aerodynamic change in moment about the elastic axis. When the airfoil motion and phase lag combine appropriately, aerodynamic energy is fed into the structural system and self-excited divergent torsional blade oscillation occurs at the fundamental torsional frequency of the blade.

Although methods have been developed for analytically predicting stall flutter (Ref. 54), experience shows that a general understanding of stall flutter and empirical relationships usually is sufficient to evaluate whether a given blade design will be subject to this phenomenon. Tests and analyses have shown that stall flutter is dependent primarily upon three factors: the reduced frequency, the blade angle, and the airfoil Mach number (Ref. 55). The effects of Mach number can be combined with the reduced frequency to give the stall flutter parameter SFP.

$$SFP = \frac{\omega_t b_s}{aM\sqrt{1 - M^2}} \text{ , d'less} \qquad (5\text{-}18)$$

where

ω_t = natural torsional frequency, rad/sec

M = local Mach number, dimensionless

b_s = blade semichord, ft

a = speed of sound, fps

When full-scale and model blade stall flutter test results for many propellers under static conditions are combined in a plot of SFP versus blade angle, points indicating the onset of flutter form a general trend, as shown in Fig. 5-40. The envelope of these flutter points may be used as a design basis. Although Ref. 55 shows that a blade whose SFP is greater than 1.0 will not flutter regardless of blade angle or power, Fig. 5-40 shows that, for low blade angles, a blade may have an SFP of less than 1.0 without being susceptible to stall flutter.

Because the design line in Fig. 5-40 is drawn without regard to such secondary effects as camber, thickness, planform, sweep, and center of twist, it is, in general, conservative; i.e., although the SFP of a blade lies under the curve, the blade will not necessarily flutter. In general, increasing the camber and thickness and shifting the center of twist forward will increase the blade angle at which flutter occurs. The effects of planform and sweep are more difficult to assess, because the stability of the blade involves the integrated effects over the entire blade. Thus, although some blade sections are stalled, the blade itself will be stable unless the integrated energy fed into the blade is greater than the structural damping present.

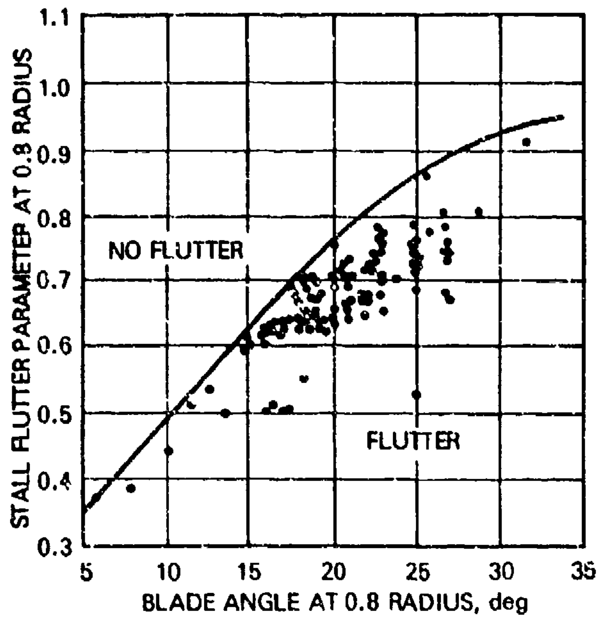


Figure 5-40. Stall Flutter Design Chart

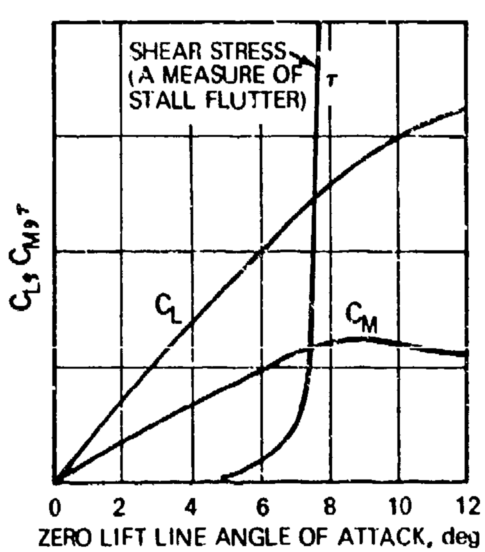


Figure 5-41. Airfoil Characteristics and Stall Flutter

A blade also may be evaluated for stall flutter by analysis of the characteristics of the moment coefficient C_M of the airfoil. As shown in Fig. 5-41, the onset of stall flutter usually occurs when the airfoil is operating near the peak of the C_M curve with respect to the torsional elastic axis, and not the lift coefficient C_L curve. Thus, stall flutter occurs only at high thrusting or blade load conditions — both forward and reverse. Because, for a given blade angle, airfoil camber increases the thrust or C_L without changing C_M, increasing the airfoil camber is one way of increasing the forward thrust or power at which a blade will be subject to stall flutter. However, in all cases, if the *SFP* is greater than 1.0, the blade will not be susceptible to true stall flutter regardless of blade angle or loading.

The other possible cause of high torsional blade response, and also bending response, is Karman vortex excitation. When the vortex excitation frequency, which is proportional to $(aM\sqrt{1-M^2})/b_s$ coincides with the torsional or bending natural frequency of the blade, significant blade response may result. The frequency expression is the reciprocal of the *SFP* divided by the torsional frequency, i.e., $(SFP/\omega_t)^{-1}$. Although the coincidence of the natural blade torsional or bending frequencies with the Karman vortex excitation frequency can result in significant blade response, particularly at very high blade angles of attack or blade stall, the response is not divergent. This type of response can be experienced at high blade angles even by blades whose *SFP* is greater than 1.0.

Apparent from this discussion is the desirability of designing propeller blades with high torsional frequencies so that their *SFP* is greater than 1.0. If this is done, divergent stall flutter is avoided. Also, the higher the *SFP*, the less likelihood that Karman vortex excitations will be a problem at very high blade loads. In general, solid propeller blade construction, e.g., using aluminum, gives relatively low torsional natural frequencies and *SFP* values below 1.0, so care must be exercised that these blades are not operated in the stall flutter zone of Fig. 5-46. Although composite monocoque blade construction has better torsional characteristics than solid construction, it frequently gives *SFP* values below 1.0. In general, propeller blades consisting of a structural spar and a thin composite airfoil shell have *SFP* values of much greater than 1.0 and are not susceptible to stall flutter.

5-8.2.5 Propeller Roughness

A propeller can apply loads to the aircraft with resulting vibration or roughness that is unacceptable to the aircraft structure or the occupants. These vibratory excitations stem primarily from two sources: excessive mass and aerodynamic unbalance of the propeller, and an undesirable combination of nonuniform flow field and propeller dynamic characteristics. The former shakes the aircraft at a frequency order of 1P, whereas the latter shakes the aircraft at frequencies that are multiples of the number of the blades — e.g., 4P, 8P for a four-bladed propeller. Through proper design, manufacture, and assembly of the propeller, the detrimental effects of propeller roughness can be minimized. The importance of propeller balance and higher order excitation to aircraft roughness is dependent upon the propeller mount design and its integration with the overall aircraft dynamic system, which determines the damping and transmissibility of these excitation loads to the airframe.

5-8.3 PROPELLER HUBS, ACTUATORS, AND CONTROLS

The design methods described in this paragraph relate primarily to the propellers of one manufacturer. In these propellers, pitch change actuation is hydraulic, but many of the design aspects apply to almost any configuration. The discussion is limited in scope inasmuch as propeller design, development, and manufacture usually are performed under subcontract, or under separate prime contract, and provided as Government-furnished equipment (GFE).

5-8.3.1 Propeller Barrel and Blade Retentions

The shape of the propeller barrel is determined by the blades. A split, or two-piece, barrel is used for high power propellers with solid aluminum blades; a one-piece barrel with a single integral blade retention race is used for low power propellers with solid aluminum blades; a one-piece barrel with multiple integral races is used for propellers whose blades are made with a steel core and a Fiberglas shell. The last type generally is used for helicopter installation and is the type that is discussed.

The barrel assembly consists of the barrel, blade retention balls, seals, and clamps. The barrel itself is a one-piece, vacuum-melted steel forging incorporating the bearing race for blade retention. The tailshaft of the barrel extends into the gearcase and is driven directly through a splined joint. The tailshaft is supported on two radial roller bearings — front and rear — so that propeller moments are reacted directly into the gearcase housing. Propeller thrust loads are reacted through an angular contact bearing at the front of the tailshaft.

5-8.3.1.1 Barrel Loading

The following loads are considered when designing a propeller barrel:

1. Centrifugal force. Blade centrifugal force is transmitted to the retention, barrel arm, and front and rear rings, where it is reacted elastically within the barrel. For purposes of analysis, the barrel is considered to contain a front and a rear structural ring.

2. Steady bending moment. Steady bending moment is due to aerodynamic loading on the blade, or to the blade being offset from a radial line. The moment is transmitted from the blade to the retention, to the barrel arm, to the front and rear rings, and into the tailshaft. The inplane component is reacted by the splined joint, and the out-of-plane component is reacted elastically within the barrel.

3. Propeller thrust. Thrust is transmitted from the blade to the retention, to the blade arm, to the front and rear rings, and to the tailshaft where it is reacted by the tailshaft thrust bearing.

4. Propeller torque. Torque is transmitted from the blade through the retention to the blade arms and into the tailshaft, where it is reacted through the drive spline.

5. 1P-aerodynamic bending moment. The inplane component of 1P-vibratory loading is transmitted from the blade to the retention, to the barrel arm, and into the front and rear rings. There, because of the unsymmetrical load phasing among the blades, it is reacted elastically within the barrel. The out-of-plane component contributes to a combined bending moment on the tailshaft, reacted by front and rear radial bearings.

6. 1P-aerodynamic side force. This force is transmitted like 1P out-of-plane moment and is reacted by the tailshaft radial bearings.

7. Higher-order vibratory loads. Moments and forces at higher orders (2P, 3P, 4P, etc.) combine in various patterns depending upon the load phasing among the blades. According to their patterns, these loads may be reacted elastically within the barrel or transmitted to the tailshaft bearings and reacted as a bending moment or a fore-and-aft or side force. They also combine in a vibratory torque.

8. Gyroscopic moment. Bending moment from gyroscopic action affects the barrel like 1P-aerodynamic moment.

9. Propeller effective weight. The side force from propeller weight, multiplied by the aircraft vertical load factor, affects the barrel like 1P-areodynamic side force.

10. Bearing pressfit loads. The tailshaft bearings are pressfit onto the tailshaft and impose a compressive load locally.

11. Axial preload. Tensile load is imposed on the tailshaft by axial preloading of the shaft bearing against a shaft shoulder.

12. Blade twisting moment. A combination of centrifugal, frictional, and aerodynamic effects, produces a twisting moment around the blade pitch change axis. This is transferred indirectly to the barrel as a couple between the blade retention and the pitch change actuator.

5-8.3.1.2 Loading Definition

In designing the barrel for a specific configuration, the various loads described in the preceding paragraph must be defined for the applicable aircraft mission profile. Loadings involved in demonstration and qualification tests also must be considered. Applicable specifications *shall* be defined by the propeller procurement specification.

From the analysis of propeller design parameters and dynamics, a summary of significant loads is prepared for selected conditions. These loads are the basis for the structural analysis of the barrel.

5-8.3.1.3 Barrel Structural Tests

After the barrel has been manufactured, the stresses at critical locations are measured under various load conditions to assure structural integrity. The barrel is strain-gaged, and an experimental stress analysis is performed with axial loads applied to simulate centrifugal loads and bending moments, both static and vibratory.

An assembly with cylinderical test bars instead of blades is tested dynamically to determine its natural frequency, so that the retention spring rate can be deduced. This is compared with the calculated values.

The barrel also is fatigue tested to determine its actual margin of safety under the design loads.

5-8.3.2 Propeller Actuators and Controls

Propeller pitch change actuators and control systems adjust and maintain blade angle according to one of several control modes, as required by aircraft and engine operating conditions. One control mode commonly used is constant-speed governing, in which a selected propeller rotational speed is held constant by a governor that raises or lowers blade angle in response to changes in forward speed or applied power. This is the control mode used almost universally in flight operation of conventional aircraft, where it allows the pilot to select the most efficient combination of propeller and engine operating conditions. The other common control mode is beta control, in which the control sets a selected blade angle β, in response to direct pilot control or to the output of a coordinated engine or aircraft flight control system. Beta control often is used for propeller reversal during landing, in low-thrust ground operation, and for propellers used as primary aircraft flight controls, such as in VTOL aircraft. Many propellers have a combination control system that uses either constant-speed governing or beta control as required.

The pitch control system has two basic components: the control and the actuator. The control receives signals from the pilot or another control system or from the propeller itself, and transmits a signal to the actuator to change blade angle as necessary. The actuator converts this signal into a mechanical action to move and maintain the blade angle.

5-8.3.2.1 Control Configurations

Several arrangements are used for control components. In one common configuration, the control assembly is nonrotating and mounted near the propeller on the gearcase or surrounding the barrel tailshaft. The output of the control is a hydraulic flow that is transferred to the pitch change actuator in the barrel assembly through transfer bearings. The rate of flow controls the rate of blade angle motion. In another configuration, the stationary portions of the control produce a mechanical signal directly related to the desired blade angle. This signal is transferred to the barrel assembly through a mechanical bearing or a differential gear train, and the rotating assembly contains the hydraulic valves and pumps required to drive the pitch change actuator.

5-8.3.2.1.1 Constant-speed Governors

The input to a constant-speed governor is a signal calling for a desired propeller speed. This signal may come directly from the pilot or from another control system, such as a synchronizer or a coordinated power management control. The desired speed is compared with the actual speed by a device such as a speed-set spring balanced against a set of rotating flyweights driven by the propeller or engine. If an off-speed condition occurs, the device puts out an error signal, commonly in the form of a pilot valve displacement. The valve displacement, in turn, meters oil to the pitch change actuator, raising or lowering the blade angle to slow down or speed up the propeller to correct the off-speed.

In forward-thrust operation, blade twisting moment always is toward low pitch, and the actuator loads always are in one direction. The governor, then, needs to meter high-pressure oil only to one side of the actuator piston to raise the blade angle; letting oil drain back to the sump permits the twisting moment to lower the blade angle. A governor with only this function is called a single-acting governor.

A double-acting governor can direct high-pressure oil to either side of the actuator piston. This capability is required if the blades are to be controlled in reverse thrust operation or to be unfeathered.

In many constant-speed governors, the pilot valve positioned by the speed-sensing device meters actuator oil flow directly. For some designs, however, large oil flows are required and use of a simple pilot valve would cause large hydraulic forces. Speed sensing-accuracy will be affected adversely unless correspondingly large speed-set spring and flyweight forces are used. To avoid the weight and size of such a design, a servo-type governor may be used. In this design, the speed-sensing device positions a small pilot valve that controls only the flow to a servo piston, which, in turn, positions a servo valve that meters the main oil flow. Because the pilot valve is isolated from strong hydraulic forces, it and the speed-sensor can be made light and sensitive.

5-8.3.2.1.2 Beta Control

With beta control, the input to the propeller control is a signal calling for a desired blade angle. In recent configurations, the signal is transferred mechanically to the rotating components, where it positions a distributor valve spool. The sleeve of this valve is positioned mechanically as a function of blade angle and, if there is a disparity between desired and actual blade angle, the distributor valve directs high-pressure oil to the appropriate side of the pitch change actuator. When the blades change pitch, the valve sleeve moves as well, until the desired blade angle is reached and the valve is closed.

5-8.3.2.2 Hydraulic System

Some propeller control systems use engine or gear-case lubricating oil, boosted in pressure by an extra pump, or oil from the aircraft hydraulic system. To minimize contamination and improve reliability, other propeller designs incorporate an independent hydraulic system. The complete hydraulic system contains the following basic components: a sump, a pump, a filter with a bypass valve, a relief valve, the control valve, and the pitch change actuator. Schematically, the actuator is a linear hydraulic piston with a mechanical device to convert the linear piston motion to rotary motion of the blade. The maximum operating pressure (relief valve setting) may be 1200 psi for a simple system or up to 3000 psi for a system where weight is critical.

A schematic diagram of the complete control and hydraulic system for a double-acting governor is shown in Fig. 5-42.

5-8.3.2.3 Auxiliary Functions

In addition to constant-speed governing and beta control, some propellers provide various auxiliary functions. Two of these — feathering and pitch lock — are safety items.

Propellers are feathered by turning their blades to a 90-deg blade angle — i.e., edgewise to the relative wind — in order to bring a disabled propeller or engine to a stop for minimum drag.

The pitch lock acts to prevent further decrease in pitch travel when normal blade angle control is lost or a preset maximum rotational speed is exceeded. Without this feature, a loss of hydraulic pressure could allow the blades to drop to low pitch in flight, causing a dangerous windmilling overspeed.

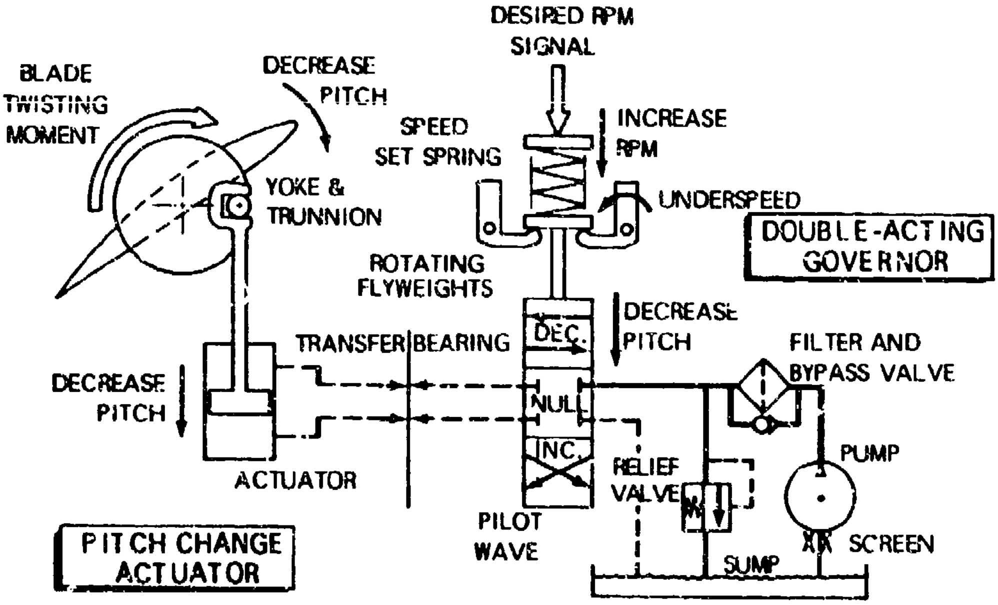

DESIRED RPM SIGNAL
BLADE TWISTING MOMENT
DECREASE PITCH
SPEED SET SPRING
INCREASE RPM
YOKE & TRUNNION
UNDERSPEED
DOUBLE-ACTING GOVERNOR
ROTATING FLYWEIGHTS
DECREASE PITCH
DEC.
TRANSFER BEARING
FILTER AND BYPASS VALVE
DECREASE PITCH
NULL
INC.
RELIEF VALVE
PUMP
ACTUATOR
PILOT WAVE
SCREEN
SUMP
PITCH CHANGE ACTUATOR

Figure 5-42. Propeller Control System Schematic

5-8.3.2.4 Control Performance

A propeller control, in combination with the engine control, must set and maintain the operating conditions directed by the pilot. It must respond quickly and accurately to pilot commands and be unaffected by undesirable disturbances. Control performance is summarized in three parameters: accuracy, stability, and transient response. In order to define these characteristics, it is necessary to study the behavior of the entire propulsion system over its complete operating range, using various analytical techniques.

The overall transient performance of an aircraft propulsion system generally is analyzed by using a nonlinear dynamic simulation of the overall system. This involves detailed nonlinear dynamic equations for the engine and the engine control, as well as for the propeller control. This simulation requires inputs that define the ambient conditions surrounding the engine and propeller — such as pressure, temperature, and flight speed — and the pilot-initiated inputs that specify the desired operating point. This type of program provides estimates of thrust, propeller speed, engine speed, fuel flow, and many other variables as a function of time for various types of pilot command signals or external disturbances.

Fig. 5-43 shows a simplified block diagram of a turboprop propulsion system and indicates some of the typical input parameters to the engine and propeller controls. Each of the blocks contains a matrix of nonlinear differential equations that are used to define the steady-state and transient behavior of that component.

A simpler analysis, providing a good insight into the basic control requirements, can be made by linearizing the important system parameters at various discrete operating conditions and examining the system behavior for small disturbances around these operating points. This permits the application of classical servomechanism theory to the design of the control system.

Fig. 5-44 shows a typical linearized block diagram for constant-speed control of a turbine-driven propeller system. This integral control system will move the blade angle until the speed error goes to zero, thus assuring that sensed speed is equal to desired speed under steady-state conditions. The governor shown in Fig. 5-42 is such a control.

Whenever rapid transient response is important, the control designer may provide lead compensation to improve the speed of response and the stability characteristics of the system. This lead characteristic compensates for the lag time constant between blade angle change and speed change. In effect, it provides an anticipatory signal for control that helps to minimize overshoots and provides good system stability.

5-8.3.2.5 Control Reliability

Because control malfunctions can cause serious trouble, reliability is a vital part of propeller control design. There are two different philosophies involved in design for reliability: safe-life and fail-safe.

The safe-life theory requires that the probability of a catastrophic malfunction be exceedingly remote. The design must have a sufficient margin of safety for all operating conditions, both normal and abnormal. Proof that a system enjoys this level of reliability requires extensive testing and thorough analysis of service experience.

The second method, fail-safe, requires that no reasonably probable single malfunction be allowed to cause unsatisfactory operation. The design must include safety devices and redundant components in order to tolerate single failures. Some propellers have dual hydraulic systems, both controls and actuators, to provide fail-safety. This is the case especially for propellers used for primary aircraft control as well as for propulsion.

An important part of design for reliability is the Failure Mode and Effect Analysis (FMEA). Such an analysis will help to reveal areas needing better safety features. It also will provide an improved understanding of possible malfunctions so that their consequences on aircraft operation can be appraised jointly by the airframe designer and the propeller manufacturer.

5-8.4 PROPELLER BLADES

5-8.4.1 Blade Geometry

As explained in par. 5-8.1, the aerodynamic size of a propeller is chosen initially with the help of parametric performance studies in which the trends of performance and weight are evaluated for various combinations of characteristics. For the parametric studies, a number of important but secondary aerodynamic design details are assumed to be in a "standard" condition, to be evaluated later.

For the complete blade design, the secondary aerodynamic details — i.e., thickness, planform, twist, airfoil type, and camber — are chosen first from past experience. Major consideration is given to the anticipated severity of structural loading and the type of blade material and construction planned. Then — for the specified operating conditions — aerodynamic performance and loading, and the blade structural response to the loads, are computed. With these results, changes are made in the design details and the analyses are repeated until the desired levels of aero-

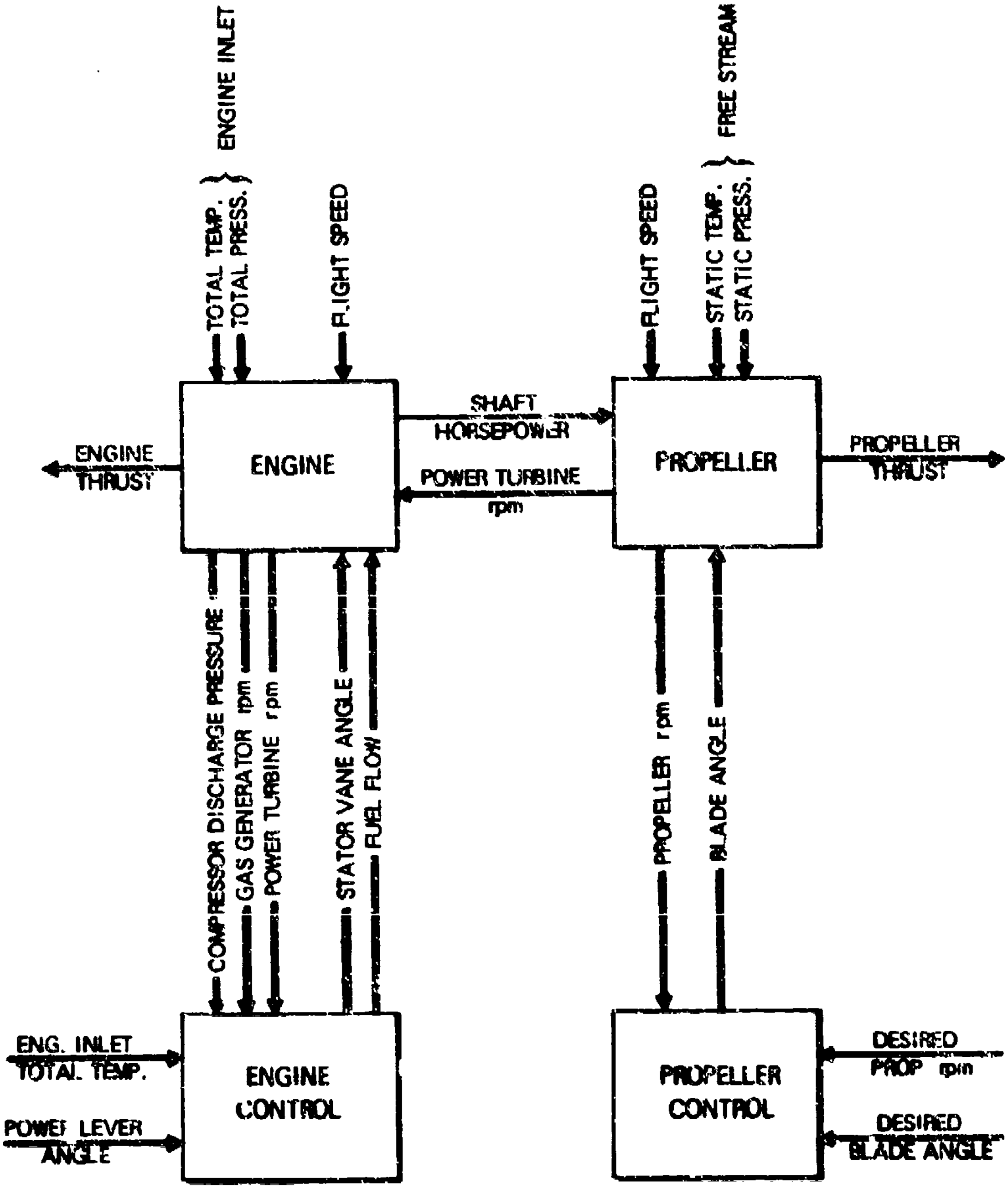

TOTAL TEMP.
TOTAL PRESS.
ENGINE INLET
FLIGHT SPEED
FLIGHT SPEED
STATIC TEMP.
STATIC PRESS.
FREE STREAM
ENGINE THRUST
ENGINE
SHAFT HORSEPOWER
POWER TURBINE rpm
PROPELLER
PROPELLER THRUST
COMPRESSOR DISCHARGE PRESSURE
GAS GENERATOR rpm
POWER TURBINE rpm
STATOR VANE ANGLE
FUEL FLOW
PROPELLER rpm
BLADE ANGLE
ENG. INLET TOTAL TEMP.
POWER LEVER ANGLE
ENGINE CONTROL
PROPELLER CONTROL
DESIRED PROP rpm
DESIRED BLADE ANGLE

Figure 5-43. Simplified Propulsion System Block Diagram

$\frac{\partial Q_P}{\partial N_P}$ = CHANGE IN PROPELLER TORQUE WITH N_P

$\frac{\partial Q_E}{\partial N_P}$ = CHANGE IN ENGINE TORQUE WITH N_P

$\frac{\partial Q_P}{\partial \beta}$ = CHANGE IN PROPELLER TORQUE WITH BLADE ANGLE

S = LAPLACE OPERATOR

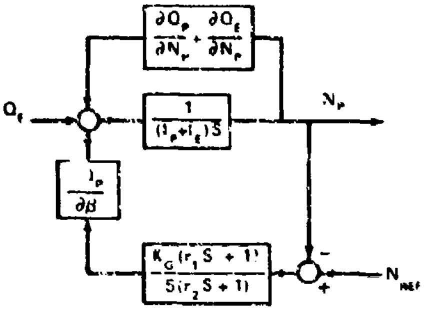

$\frac{\partial Q_P}{\partial N_P} + \frac{\partial Q_E}{\partial N_P}$
Q_E
$\frac{1}{(I_P+I_E)S}$
N_P
$\frac{\partial Q_P}{\partial \beta}$
$\frac{K_G(\tau_1 S + 1)}{S(\tau_2 S + 1)}$
−
+
N_{REF}

Figure 5-44. Linearized Propeller Control Block Diagram

dynamic performance and blade stress are achieved. For efficiency and speed, these analyses are performed with the help of a high-speed digital computer.

Of the secondary details, thickness and planform are the most important to propeller strength and weight. The choice of blade thickness and its distribution along the blade is affected by structural capacity requirements and also by considerations of vibration-critical speed location. The aerodynamic requirement to minimize profile drag losses puts a constraint on blade thickness, and special problems such as tip compressibility also may affect the design. The choice of blade planform shape is affected by the choice of blade material and construction; with spar-and-shell Fiberglas blades, a tapered planform (wider inboard) lends itself to both minimum weight and reduced noise. The selected activity factor usually limits the amount of taper — the higher the activity factor, the less the taper. For solid aluminum blades, a planform with an elliptical or round tip often is used for reduced noise.

5-8.4.2 Blade Construction

Viewed as a structure, a propeller blade is a rotating, cantilever beam subjected to two major classes of loading: inertial forces and aerodynamic forces. The inertial forces consist of steady centrifugal loads and vibratory reactions; the aerodynamic forces are both steady and periodically varying. Steady loads are important to the blade design, but vibratory loading usually is the dominant influence. Even without dynamic magnification, vibratory loading at one cycle per revolution (1P) can be very significant. In addition, if the frequency of aerodynamic load variation approaches one of the natural frequencies of the propeller, considerable magnification of the cyclic loading in the blade can result. Much effort is spent in blade structural design to limit this dynamic magnification by properly positioning the bending and torsional frequencies.

In general, there are six major structural aspects that must be considered in designing propeller blades:

1. Axial load capacity
2. Bending capacity
3. Bending stiffness
4. Bending frequencies
5. Torsional stiffness
6. Torsional frequency.

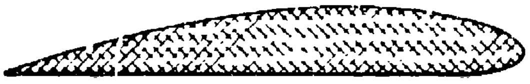
The objective of the structural designer is to obtain, by judicious use of various materials and configurations, the lightest and best structure possible within the geometric constraints defined by aerodynamic requirements.

5-8.4.2.1 Types of Blade Construction

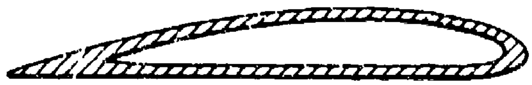
The simplest construction is the solid blade, as shown in Fig. 5-45(A). With the proper choice of material, this construction has provided an acceptable balance of weight and structure for many years on conventional aircraft. Its advantages are simplicity and low cost. Its disadvantages are inefficient use of material in the center of the cross section, particularly in bending, and the fact that the primary structure is exposed to service-inflicted foreign object damage (FOD). Unfortunately, propeller blades often are subjected to the impact of various objects, ranging from sand and dust to stones and birds. Many of these impacts are capable of inflicting surface damage that propagates as fatigue cracks because of the cyclic stresses caused by the unsteady loadings. Recognizing the potential of FOD has been shown repeatedly to be essential to acceptable service performance. With solid construction, the designer must select materials with low notch sensitivity and low crack propagation rates, and must observe conservative stress limits.

The simplest form of hollow-blade construction is shown in Fig. 5-45(B). This is a fully stressed skin of monocoque construction in which the central material of the blade section, which contributes little to beam strength or stiffness, is absent. The simple hollow blade construction has the potential for substantial weight reduction, but two significant problems have interfered with realizing this potential. First, foreign objects that only gouge the surface of solid blades can both gouge and dent the hollow sections. The dent causes an additional concentration of stress in the same area as the gouge. Therefore, the minimum wall thickness is determined not by gross structural considerations but rather by the required resistance to FOD. Because impact velocities are greatest at the tip, thick walls are required there; from the standpoints of centrifugal load and frequency placement, this is an undesirable region for added weight. The second problem of simple hollow blades occurs at the leading and trailing edges, which form the only shear load path between the thrust and camber faces. Because the faces intersect at a sharp angle, the bend radius is small. The resulting geometric stress concentration often is unsatisfactory, and additional material must be added at the edges to increase the radius and reduce the local stress. This added material not only increases the weight, but also reduces the blade natural frequencies, particularly in torsion.

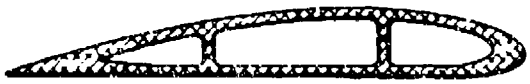
Some of the problems of simple hollow blade sections are alleviated by a modified monocoque construction, such as the ribbed section shown in Fig. 5-45(C). One or more ribs are added to supply additional shear load paths and to reduce denting. The exact proportioning of sheet and rib thickness and spacing along the length of the blade is a combination of analysis and experimentation. Once perfected, the resulting blade generally is much lighter than a solid version. The primary structure, however, still is exposed to FOD, and cracks caused by such damage can propagate across the entire section.



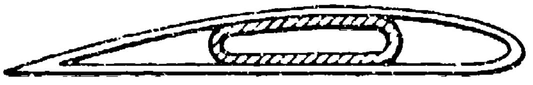
(A) SOLID



(B) MONOCOQUE



(C) MODIFIED MONOCOQUE



(D) SPAR–SHELL

Figure 5-45. Typical Blade Cross Sections

One way to solve this remaining deficiency is to build what is in principle a modified monocoque section in two pieces. Fig. 5-45(D) illustrates the cross section of such a blade construction. In this approach, the central tubular member, or spar, is made as a tapered-wall, varying-diameter tube and then is flattened and twisted to shape. The outer aerodynamic contour or shell then is bonded to the spar. This construction allows considerable design flexibility. The spar and shell wall thicknesses can be varied independently to achieve the required mass and stiffness distribution, and the spar position within the shell can be varied as well. Additional refinements, such as chordwise wall thickness variations in both shell and spar, also can be achieved. Most important, with the separation of the aerodynamic shell and structural spar, different materials may be considered for each item. With the proper choice of material, damage isolation can be achieved; i.e., the effects of service inflicted damage can be limited to the shell, allowing adequate time for detection and repair.

A typical spar-shell blade is shown in Fig. 5-46. Because the spar is formed from a tube, the transition from the airfoil contour to the round retention section is natural and convenient. The spar is continuous from retention to tip, with no joints of any sort, and the shell is bonded to the spar over the entire blade length. With the large joint area and the continuous transfer of load from the shell to the spar, the bonded joint is loaded lightly. A filler material often is used in the cavities of the blade. This type of blade construction has been very satisfactory, with substantially lower weight than solid blades.

The use of hollow blades to achieve these improvements, however, introduces a number of special structural considerations in addition to the six major aspects mentioned previously. These are:

1. Secondary structural action in the transition region between shank and airfoil, where local bending of the spar and shell walls can occur since the gross bending tends to straighten or deform the longitudinal profile of the walls.

2. Panel vibration of local areas of the spar or shell, especially if they are unsupported

3. Shear flow, from the inability of the major shear loads in a hollow blade to pass directly across the blade thickness. Instead, these loads must follow the structural material around the cavities.

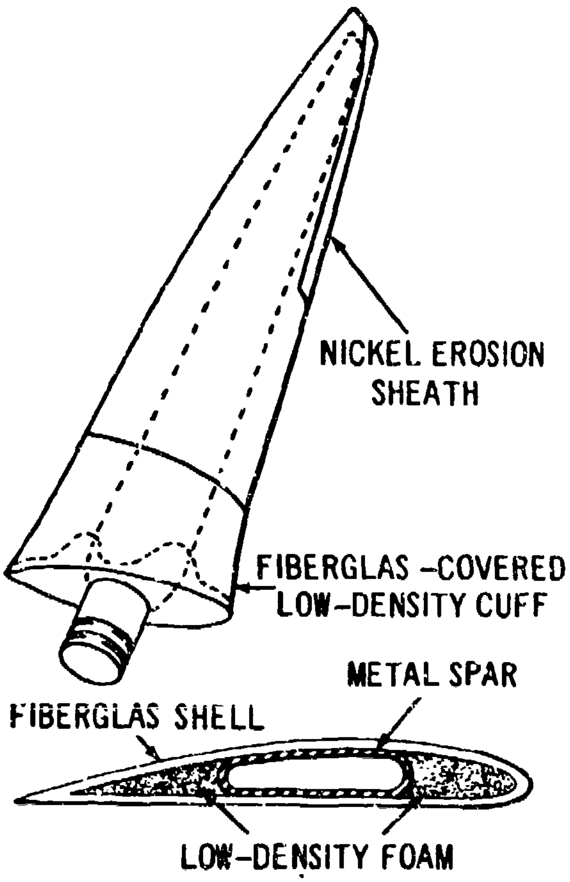


Figure 5-46. Typical Spar-shell Blade

4. Filler stressing, from carrying some of the shear loading across the blade thickness and from reactions to cross section deformation
5. Shear lag effects, in which regions of the surface distant from a rib tend to duck the bending load
6. Chordwise deformation and stress from pressure and inertia loads on the airfoil, as well as from shear lag effects
7. Structural stability — the avoidance of buckling
8. Joint stressing, especially at the inboard end of the shell.

These possible problems must be considered carefully in design, and require judicious choice of configuration, filler, and joint design.

Additional information on types of propeller blade construction, including both wood and metal blades, may be found in ANC-9.

5-8.4.2.2 Manufacturing Processes and Tooling

More so than with many structures, the design of light, strong propeller blades has been affected by the manufacturing processes available. For hollow blades, especially, detail blade design must represent an intimate combination of functional requirement and manufacturing capability.

Monocoque blades are made by manufacturing procedures that fall into two general categories: flattening and twisting tubes into shape, and joining a number of individual pieces.

In the tube-flattening process, a round tube is drawn, extruded, or reduced in some manner from the billet so that its wall thickness tapers as required and its perimeter is compatible with the desired airfoil sections. The shank region is swaged down to the proper size, and then the tube is pressed and twisted into shape in dies. If the blade is to be tapered, the trailing edge of the pressed tube is trimmed and welded.

Some hollow blades are made from only two pieces, one containing the shank region and one surface of the blade, the other the second surface. The two pieces are welded together at the leading and trailing edges, and a brazed fillet added to strengthen the edges. The pieces to be assembled are milled, ground, and press-formed into the proper thickness and shape before joining. Other blades have been welded from smaller segments, with both chordwise and longitudinal welds.

Hollow blades with ribs are made essentially the same as the blades just described. In this case, one or more of the pieces to be welded or brazed is milled to contain the central ribs.

The manufacture of spar-shell blades varies according to the material used. These blades have been produced both in all-metal configurations and in configurations with steel spars and Fiberglas shells. Current development work is demonstrating the promise of new composite materials for the spar or the shell or both.

Metal spars are manufactured by processes similar to those used for flattened-tube monocoque blades.

Internal and external peening with metal shot or glass beads are used for strength improvement. The retention bearing raceways then are machined on the root of the spar. These raceways are integral with the spar material, and must be hardened locally by carefully controlled flame or induction methods. Finally, the finished spar may be plated for corrosion protection.

Metal shells are formed from polish-ground sheets, which are folded around the leading edge and seam-welded at the trailing edge and the tip. As with the spars, the final shape is obtained by hot-forming.

Fiberglas shells normally are laid up on metal mandrels, either by a wet layup process or by using preimpregnated fabrics. Some Fiberglas shells are made in one piece, folded around the leading edge and open at the trailing edge, which is joined when the spar and shell are joined. Others are made in two pieces, with bonded joints at both the leading and trailing edges.

The spar and shell are joined by brazing, for all-metal blades, or by adhesive bonding, for the Fiberglas shells. Brazing is performed in dies in a brazing furnace, with pressure in the spar to hold the joint in intimate contact. Adhesive bonding of Fiberglas shells may be accomplished with heat and pressure in a pair of dies, but the more common practice is to use a die only for the camber faces of the blade and to apply pressure by vacuum bag or autoclave.

The assembly of a spar-shell blade also requires the installation of lightweight filler material. Some filler pieces are precast to shape or cut from balsa wood or honeycomb; these are installed as parts of the assembly when the spar and shell are joined. Cast-in-place filler also is used; this is poured into the blade cavities and cured after the spar-shell joint is made.

Spars and shells of advance composite materials, in the present state of the art, usually are laid up by hand in tape form on a mandrel. When the composite matrix is resin, the material is cured with the mandrel as a die and autoclave pressure, or with an outside die and inflatable bladders inside. When the matrix is metal, the material is compacted and diffusion-bonded in matched metal dies.

With advanced composites, selected materials and fiber orientations can be used in the various layers of the spar and shell. Numerically controlled tape-laying machines are being used in current development programs to facilitate the efficient manufacture of these new blades.

5-8.4.2.3 Quality Control

The production quality of a propeller blade has a considerable effect on allowable stress levels, because fatigue stresses are critical. Therefore, it is important that blades be inspected carefully for material quality, surface condition, and — in the case of spar-shell blades — quality of bonded or brazed joints. In addition to verifying those attributes affecting strength, inspection must insure that the blades are correct dimensionally. Airfoil dimensions are used to assure that each blade will produce its design thrust performance; an individual performance test is both impracticable and unnecessary. Airfoil dimensions of individual blades also are used to check aerodynamic balance among a set of blades.

Material quality is verified by certification of each lot of forgings, including tensile and chemical tests of sample tabs, and by hardness tests on each piece. Processes affecting strength, such as cold rolling and shot-peening, are subjected to thorough process control. This includes frequent inspection of the machines and techniques involved, as well as periodic destructive examination of sample blades.

The surface condition of propeller blades should be subjected to careful visual scrutiny. In addition, spar-to-shell joint quality is assured by process control and by nondestructive testing techniques such as x-ray, ultrasonic scanning, and tap testing. These nondestructive methods also assure the quality of the blade filler material.

5-8.4.3 Blade and Propeller Balance

Propeller roughness due to unbalance is caused primarily by deviations from tolerances in the three main propeller components: blades, hub assembly, and spinner. The most important source of propeller unbalance is the blades, because of their large radius and mass, and their aerodynamic characteristics. Four factors affect the amount of blade unbalance: mass force, mass moment, aerodynamic forces, and aerodynamic moment.

To obtain realistic estimates of probable unbalances for a particular propeller design, the effects of the various independent dimensional tolerances must be evaluated statistically for each of the three major components and then combined statistically for the overall propeller. Analytical techniques and computer programs have been developed for estimating and assessing unbalance for numerous propeller installations with good success. A discussion of the various aspects of propeller balance is given in Ref. 56. Additional comments appear in ANC-9.

In general, blade balance is achieved in manufacturing and assembly by the following steps:

1. Mass force unbalance is controlled by horizontally balancing the blades against a master blade within 0.002 in. times the blade weight with a minimum tolerance of 0.10 in.-lb.

2. Mass moment unbalance is controlled by vertically balancing the blades against a master blade within 0.004 in. times the blade weight with a minimum tolerance of 0.20 in.-lb (MIL-P-26366). The vertical balance should be accomplished for two orthogonal blade angular positions. To achieve mass balance, small weights are added, usually in the blade root region.

Aerodynamic balance of propeller balance usually is controlled by holding propeller airfoil shape and by installing the blades with their reference stations within 0.2 deg of each other. In many installations, however, adequate control cannot be achieved without selective assembly. In these cases, the weighted average blade angle error must be compared to a specification maximum. The weighting factors for the angle errors along the blade are based on the aerodynamic loading for a specific operating condition for either force or moment unbalance. Usually, correcting aerodynamic force unbalance also will correct moment unbalance satisfactorily. The weighted average blade angle error may be obtained by manually averaging the weighted blade angle errors at various stations along the blade, or by using an automatice blade aerodynamic balancing machine.

Hub and spinner unbalances also can influence the overall propeller balance. Even though the hub is statically balanced about its axis to 0.0005 in. times its weight, and the spinner is both statically force balanced about its axis to 0.0005 in. times its weight and dynamically moment balanced, close dimensional control and indexing must be maintained to achieve good propeller balance. Critical factors include out-of-plane and inplane blade retention squareness, axial positioning of the blade, eccentricity and squareness of the hub retention on the propeller shaft, and spinner mounting runout and tilt. The balance requirements of the hub and spinner usually are met by removing material or adding balance weights.

After the parts of a propeller are balanced separately, the assembled propeller is balanced statically in either the horizontal or vertical position to 0.0005 in. times the propeller weight by adding balance weights to the hub. The blades should be at cruise flight angle during this final balance to obtain the smoothest operation in flight.

If additional balancing is required, it may be performed on the aircraft dynamically by using systematic trial weight methods or special instrumentation, such as the pulse synchronizer unbalance indicating (PSUI) unit or a vibration analyzer. Because of differences in blade angular position and loading, and possibly in nacelle system response in flight, it may be necessary to supplement ground dynamic balancing of the propeller with inflight dynamic balancing.

5-8.4.4 Blade Materials

The continuing development of new materials and construction techniques for propeller blades has permitted substantial weight reductions, as summarized in Fig. 5-47. Advanced composite blade construction has the potential for even more improvement.

The paragraphs that follow deal with various kinds of blade materials. Additional information is contained in ANC-9.

5-8.4.4.1 Hollow Blades

Steel suitable for one-piece hollow blades, or for the spar of spar-shell blades, are low-alloy steels equivalent to AISI 4350, vacuum melted, in both the 36-40 R_C hardness and the 40-44 R_C hardness ranges. These steels must be protected from corrosive environments. The leading edge or the entire airfoil may be protected with erosion-resistant coatings or platings with less-durable paint coatings on the internal surface.

Impact damage is a serious problem for a one-piece hollow steel blade. Wall thickness that may be adequate for carrying structural loads may be thin enough to dent locally. The strength reduction for local impact damage is the combined effect of the gouge stress concentration, the local plastically deformed material, and the stress-raising action of the dent. Frequent inspection and local removal of gouges and their plastically deformed surrounding material can be used to protect against the effects of this type of damage. Another method is to protect the blade with a hard, damage-resistant plate such as nickel or chrome. However, the strength-reducing effects of the hard plating on the steel structure must be considered in the initial design.

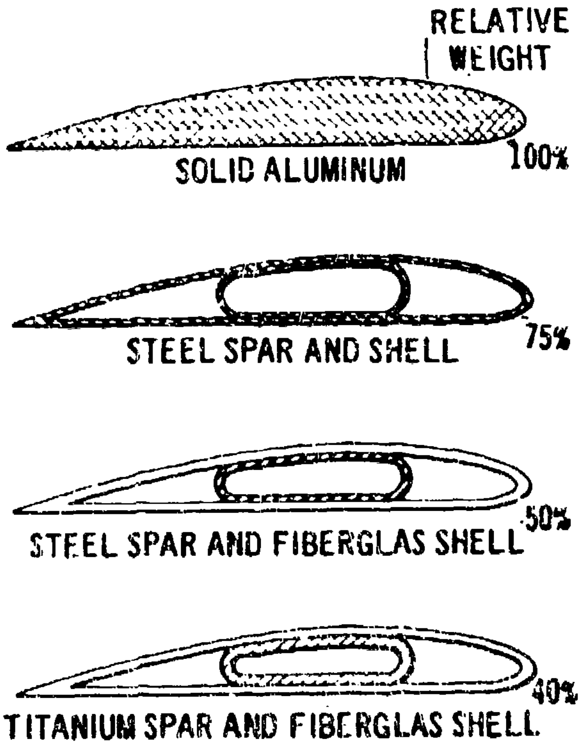

RELATIVE WEIGHT
SOLID ALUMINUM
100%
STEEL SPAR AND SHELL
75%
STEEL SPAR AND FIBERGLAS SHELL
50%
TITANIUM SPAR AND FIBERGLAS SHELL
40%

Figure 5-47. Blade Materials and Weight Reduction

When spar-shell construction is used, with the spar as the major load-carrying member and a surrounding shell of a different material, the spar is protected from erosion and impact damage by the shell. Internal surfaces of the spar must be protected, however. Glass cloth or fiber-reinforced plastic materials can be used for the aerodynamic shell and bonded to the spar with adhesives. The elastic and mechanical properties of the shell can be tailored by selected orientation of the lay-up.

Hollow titanium load-carrying spars are being developed, using 6Al-4V alloy. Because of the excellent corrosion resistance of titanium alloys in general, no corrosion protection is required. An airfoil envelope adhesively bonded to the spar provides both erosion and impact damage protection. Therefore, no protective platings that can degrade the fatigue strength of titanium need be considered in this type of construction.

5-8.4.4.2 Composite Materials

Glass fibers in resin matrices, either filament-wound or in cloth lay-ups, have been used widely for various structures. Glass cloth in an epoxy matrix is used as the aerodynamic shell on hollow spar-shell blades with metal spars. Although these lay-ups do not exhibit high strengths, they are free from corrosion and are relatively insensitive to the effects of impact damage. Adequate protection from light particle erosion can be provided by a metal sheath bonded adhesively to the leading edge and extending over portions of the face and camber sides.

Boron and carbon filaments in an epoxy matrix and boron in an aluminum alloy matrix are still in the developmental stage. Together with Fiberglas, these advanced composites provide the designer with an essentially unlimited range of raw materials for monocoque or spar-shell blade structures.

The highly directional properties of fibrous composites make them both promising and troublesome. They cannot be considered as simple substitutes for isotropic materials. The designer must take advantage, as much as possible, of the directional properties of the composites because the great gains in unidirectional strength and stiffness are reduced if the structure must carry multidirectional loads. Maximizing structural efficiency of composites requires that the reinforcements be oriented and proportioned accurately and economically to sustain the design loads. This requires detailed knowledge of both failure modes and consistent deformation.

With metal matrix composites, this maximization is a little easier because of the comparatively good transverse strength of the matrix material. The problem is finding satisfactory techniques for fabricating individual components from the tapes. Brazing and diffusion bonding are two processes with great potential that possess the capability of being developed into highly reproducible and economical manufacturing processes. Dramatic increases in strength-to-weight and stiffness-to-weight ratios over either steel or titanium are possible with the new advanced composites.

When loading is essentially unidirectional with the reinforcing fibers and the matrix material is not highly stressed, an epoxy matrix appears preferable because of its lighter weight. However, a metal matrix offers higher strength and stiffness, where needed. Also, the allowable strength and design modulus of material with an epoxy matrix normally must be adjusted downward to allow for moisture absorption in service and for the gradual modulus decrease under continuous cyclic stressing.

5-8.4.4.3 Filler Material

Rigid urethane foam materials with a density in the range of 6-12 lb/ft^3 currently are used as the structural filler in lightweight blades. The foam materials of particular interest in this category employ a polyester resin base and are of the pour-in-place type, blown with CO$_2$. Though anisotropic in nature, these materials combine low thermal conductivity and high strength-to-weight ratios with adequate moisture resistance, excellent adhesion, and satisfactory temperature stability. Of particular importance in blade fabrication is the ability of these foams to be installed in-place in spite of complex cavity configuration.

5-8.4.4.4 Structural Adhesives

Modified epoxy-based adhesives are the materials most frequently used both for the matrix in Fiberglas-reinforced blade shells and for the bonding agent in Fiberglas-to-metal and metal-to-metal bonded joints. These adhesives are the most versatile and provide the best balance of properties for adhesive bonding. For propeller blade bonding applications, these adhesives are used in supported film form. In this form, processing is simplified, bond thickness is predictable consistently, low cure shrinkage is exhibited, excellent wettability on many types of materials is achieved, and a high load-carrying capability is provided.

5-8.5 PROPELLER BLADE FATIGUE LIVES

Propeller components must be designed so that vibratory stresses are within the fatigue strength of the parts. Because of the importance of propeller structural integrity to the safety of flight, and because propeller dynamic behavior is among the more difficult aspects of propeller design, determination of blade fatigue strength must be based upon both comprehensive fatigue tests of full-scale blades and vibratory blade stresses measured on the aircraft under conditions representative of service operations.

The number of vibratory stress cycles accumulated in the service life of a propeller is so great that vibratory stresses below the endurance limits are necessary for most operating conditions. For instance, assuming a 1P vibratory stress and a rotational speed of 1500 rpm, 9×10^6 cycles are accumulated in 100 hr of service.

Some momentary or intermittent operating conditions can occur in which stress amplitudes exceed the endurance limit. Each cycle of such stress uses up some of the fatigue life of the propeller, and it is necessary to establish conservatively that the accumulation of these cycles can be tolerated. Current practice is to apply Miner's Rule for cumulative fatigue damage.

Vibratory stress limits must be derived primarily from controlled laboratory tests for full-scale propeller blades, supplemented by specimen tests.

Ground and flight measurement of propeller vibratory stresses during aircraft operation is described in Chapter 8, AMCP 706-203. The instrumentation required, many of the considerations involved in planning and executing a vibratory stress survey, and the interpretation of results are included. Certain of these subjects are summarized in the paragraphs that follow, with emphasis on the interpretation of results. An example of the application of Miner's Rule is included by reference.

5-8.5.1 Endurance Limit and Other Structural Testing

5-8.5.1.1 Specimen Tests

Considerable information about the fatigue strength of propeller blade material can be obtained from specimen tests. A discussion of types of testing, number of samples, and statistical interpretation of fatigue test data can be found in Ref. 57.

5-8.5.1.2 Full-scale Tests

Specimen testing can complement, but never replace, testing of full-scale production components. Because of the larger size, fabrication differences, and differences in stress state, full-scale components tend to have both lower mean fatigue strengths and greater scatter than conventional laboratory specimens. A typical illustration of this difference is shown in Fig. 5-48, where the mean fatigue strength of full-scale components at larger numbers of cycles is about one-half that of the specimens. The data scatter

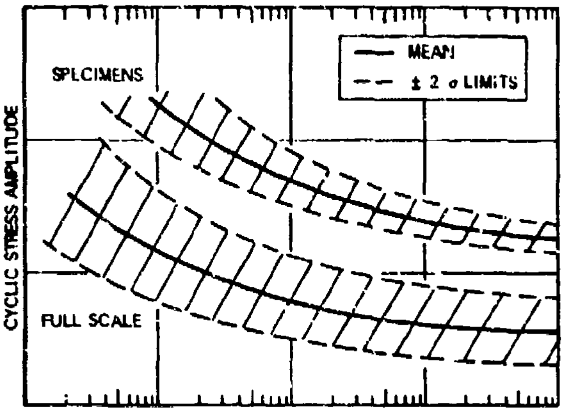


Figure 5-48. Fatigue Strength Difference Between Specimen and Full-scale Tests

(represented by the coefficient of dispersion; i.e., the ratio of the standard deviation in fatigue strength to the mean fatigue strength, at a particular number of cycles) is generally in the range of 5-10% for specimens, but may be as high as 15-20% for full-scale components.

The types of full-scale tests to be conducted depend upon previous tests on similar components, anticipated service loadings and environments, and related service experience. The various regions of the blade — tip, mid-blade, airfoil transition, shank and retention — must be considered as to their shapes and fabrication details, their steady and vibratory loadings, and their environmental exposures. The requirements for fatigue testing of propeller blades are similar to those for rotor blades discussed in par. 5-7 and in Chapter 7, AMCP 706-203.

5-8.5.2 Flight Loads Test Data and Fatigue Life Determination

5-8.5.2.1 Aircraft Tests

The instrumentation required for measuring blade vibratory stresses in flight is described in Chapter 8, AMCP 706-203.

In general, survey tests are programmed to encompass all significant service operating conditions, with adequate allowance for variability and, where possible, for future change and growth. The results of a propeller vibratory stress survey customarily are summarized in plots of vibratory stress against the most pertinent variable — such as propeller speed, airspeed, power or time. The curves usually are selected to show the highest stresses in the tip, the mid-blade, and the shank regions of the blade. An example of a stress summary plot is shown in Fig. 5-49. A full set of such curves will form the basis for the fatigue life analysis.

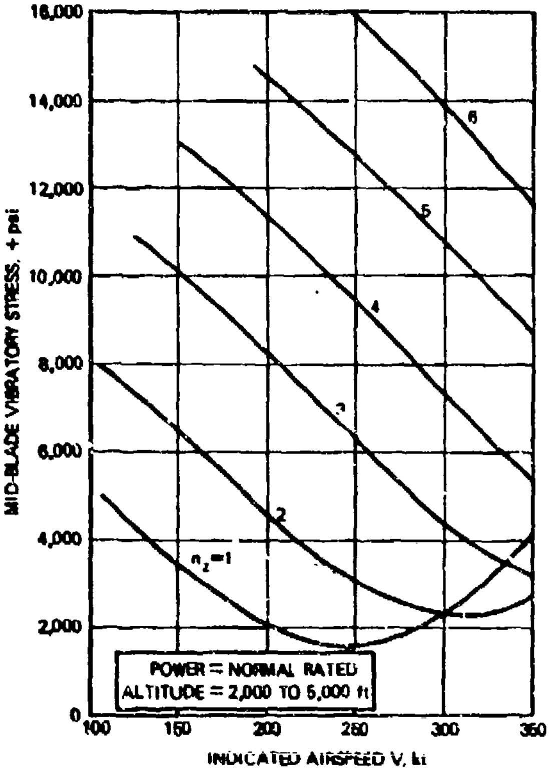


Figure 5-49. Typical Stress Summary Curves

5-8.5.2.2 Interpretation of Results

To determine structural integrity, the measured stresses in various regions of the blade — after due consideration of background data and allowance for change — must be compared with appropriate material strength information. The strength data are established from full-scale and specimen testing. In choosing appropriate strength data, the level of mean stress from design computations is sufficiently accurate.

For operating conditions considered to be essentially continuous, the stress levels must be below the endurance limit. Normal cruise flight must be treated as a continuous operating condition, and, in some installations, normal climb and descent should be treated similarly.

A further consideration, after establishing that continuous operating conditions produce acceptable stresses, is to determine whether any conditions exist where the dynamic response of the propeller might be poorly repeatable, i.e., conditions where the vibratory stress — though it be low as tested — might become too high under expectable variations in circumstances. Such conditions might include operation at certain critical speeds in a random ground wind environment, or some instances of stall flutter. In such cases, it may be necessary to avoid specific operating regions in order to assure structural integrity.

For conditions, such as takeoff and inflight maneuvers, where vibratory stress levels exceed the endurance limit, the cumulative fatigue damage must be shown to result in an acceptable fatigue life for the propeller. Fatigue life determination is discussed in Chapter 4, AMCP 706-201.

5-9 ANTITORQUE ROTORS

5-9.1 GENERAL

The tail rotor of a single-rotor helicopter is designed to provide thrust for counteracting main rotor torque at all flight conditions and to provide variable thrust for control in both the torque and antitorque directions. Yaw control is effected by variation and modulation of the collective pitch setting of the tail rotor. The collective pitch is controlled by directional control pedals and normally is adjusted so that, at the design point hover condition, the pedals are in a neutral position. For single-rotor helicopters with the advancing blade on the right, (the conventional direction of main rotor rotation), the left pedal increases tail rotor (positive to the right) thrust, producing a left yaw. Likewise, the right pedal will produce a right yaw, with the tail rotor going to lower values of thrust and finally into "reverse thrust".

The tail rotor design goal is to produce, with minimum power and weight, the thrust necessary to meet the control and antitorque requirements. Tail rotor requirements must be met without the occurrence of any undesirable vibration, whirl, or shake characteristics.

The tail rotor is designed for the most severe ambient conditions including helicopter critical hover altitude and temperature, and the critical altitude for the engine.

The maximum thrust that the tail rotor must provide without blade stall is that required to counteract main rotor torque, while also providing the specified positive yaw acceleration and overcoming tail rotor gyroscopic precession effects, in the maximum specified cross-wind. Provision must be included to counteract disturbances such as gusts. Consideration also must be given to the thrust loss due to interference with the vertical fin, main rotor, and other parts of the helicopter.

Experience has shown that if the tail rotor thrust requirements at the critical low-speed conditions (e.g., hover plus left yaw into a side wind at minimum density altitude) are met, all requirements for forward flight usually will be satisfied. However, the tail rotor thrust capability still [illegible] be analyzed under forward flight conditions, including critical maneuvers. Without such verification, it may be necessary to restrict the helicopter operational envelope or to redesign the tail rotor when deficiencies are discovered during subsequent flight test.

Prior to final definition of the maximum required tail rotor thrust, consideration should be given to increased thrust requirements resulting from the increased engine power available with engine growth.

Refs. 58 through 61 give additional data on tail rotor design. There currently are no Military Specifications applicable specificallly to the design of antitorque rotors.

5-9.2 TYPCIAL ANTITORQUE ROTORS

Tail rotors in current use employ from two to six blades. However, there is no basis for limiting the number of blades. The blades are retained in a hub to allow collective pitch change ranging from positive to negative angles. Two-bladed tail rotors usually are designed with teetering blades, whereas rotors with three or more blades have individual hinges. In either type, δ_3 commonly is used to control the flapping magnitude. In tail rotors with teetering blades, the precone required for bending moment reduction also can be used to compensate for the aerodynamic and centrifugal twisting moment that drives the blade to low collective pitch angles. The blades may be retained in the hub by bearings, tension-torsion straps, or elastomeric bearings.

The blades used for tail rotors are much stiffer in all modes than those used for main rotors. For instance, in the torsional mode, the combination of blade and control stiffness generally is five to six times greater than the stiffness in main rotors. This, combined with the higher relative mass and high bearing damping, usually eliminates the mass balance requirement for preventing flutter and divergence. (Ref. 59).

5-9.3 TAIL ROTOR DESIGN REQUIREMENTS

The tail rotor *shall* produce the thrust necessary for helicopter yaw control and the antitorque requirements of the main rotor. This thrust must be produced when operating in the flow field and dynamic environment found with tail rotor helicopters; e.g., in the presence of a tail fin, in the wake of the main rotor at sideward velocities, near the ground, and when the helicopter has a positive or negative yaw rate and/or acceleration.

Based upon the geometric conditions between the tail rotor and the rest of the helicopter (Fig. 5-50), the tail rotor thrust T_{tr} required is found from Eqs. 5-19 and 5-20 with X and R_{mr} measured in feet:

$$Q_{mr} = \frac{550\,HP_{mr}\,R_{mr}}{(\Omega R)_{mr}}\,,\ \text{ft-lb} \qquad (5\text{-}19)$$

$$T_{tr} = \frac{Q_{mr} + I_{zz}\ddot{\Psi}}{X}\,,\ \text{lb} \qquad (5\text{-}20)$$

where

I_{zz} = mass moment of inertia of helicopter in yaw, slug-ft^2

X = distance from centerline of main rotor to centerline of tail rotor, ft

$\ddot{\Psi}$ = yaw acceleration, rad/sec^2

Q_{mr} = main rotor torque, ft-lb

R_{mr} = main rotor radius, ft

$(\Omega R)_{mr}$ = main rotor tip speed, fps

T_{tr} = tail rotor thrust required to compensate for main rotor torque, and to provide required yaw acceleration, lb

The tail rotor thrust calculated in Eq. 5-20 is the net thrust, taking into account all interference losses due to the presence of vertical tail fins, flow field, and dynamic effects. The rotor will be designed to meet the thrust requirement found in Eq. 5-20 at the critical hovering temperature and altitude or the critical engine altitude. Normal forward flight conditions are of secondary importance since the vertical tail fin usually is designed to unload the tail rotor at cruise. High-speed autorotations and rolling pullouts are two critical exceptions.

In addition to producing the thrust required, the tail rotor *shall* be designed for ease of control, manual or boosted. The rotor should be designed so that a linear control will be obtained in all flight conditions where the tail rotor does not operate in the vortex stage. The natural frequency of the rotor should not correspond to any critical frequencies occurring in the rest of the structure. In addition to normal structural integrity, tail rotor design *shall* consider aeroelastic problems.

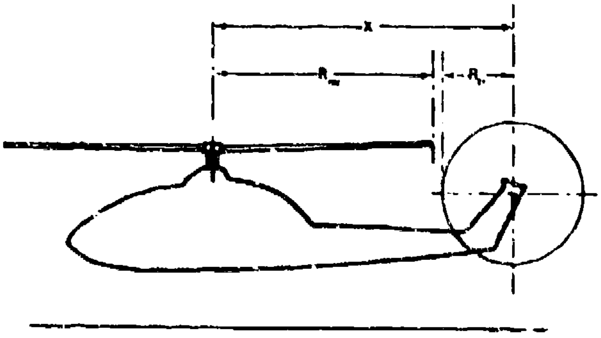


Figure 5-50. Geometric Data

5-9.4 INSTALLATION CONSIDERATIONS

During design of the tail rotor, consideration *shall* be given to the actual installation on the vehicle in determining the required thrust. The rotor may be designed to operate with the tail fin downstream (tractor configuration), or with the fin upstream (pusher configuration). The rotational axis of the tail rotor also may be canted with respect to the fin to obtain a life component from the thrust vector. The location of the rotor will have been selected to provide adequate clearance from the ground and other parts of the helicopter, and with provisions for the safety of ground personnel (see Chapter 13, AMCP 706-201).

5-9.4.1 Tractor Configuration

In the tractor configuration, the fin produces a blockage that causes a thrust loss. Tests (Ref. 58) indicate that the net thrust available to satisfy helicopter requirements may be estimated from

$$T_{net} = T\left(1 - \frac{0.75S}{A}\right), \text{ lb} \qquad (5\text{-}21)$$

T_{net} = thrust for control and antitorque, lb
S/A = ratio of disk area blocked by the fin to total disk area, dimensionless

5-9.4.2 Pusher Configuration

In the pusher installation, the production of thrust creates negative pressures on the fin and tail boom on the side adjacent to the rotor. The integral of these pressures over the affected area produces a force that must be subtracted from the rotor thrust. This force can run as high as 20% of the tail rotor thrust, but can be reduced by increasing the axial distance between the rotor and the fin. At a distance corresponding to 60% of the tail rotor radius, this loss is only 1-2%. The effect of distance for both tractor and pusher configurations is shown in Fig. 5-51, taken from Ref. 58.

It is possible to design the pusher configuration with lower losses than occur in the tractor installation. However, because of flow blockage, rotor performance is influenced to a greater degree by wind effects. Therefore, the pusher configuration should be used with caution.

5-9.4.3 Operational Considerations

When the tail rotor is operating at a yaw rate, a moment is required to precess the gyroscopic forces. This moment is a function of yaw rate $\dot{\Psi}$, tail rotor angular velocity Ω_{tr}, and polar moment of inertia I_p, and must be produced by aerodynamic forces applied 90 deg ahead of the direction of precession in the case of rotors with flapping blades. This is accomplished by a lag of the tip path plane with respect to the control axis, which produces an equivalent to cyclic feathering. As a result, one side of the disk is loaded more highly than the other; if blade stall is encountered, the additional precessional moment must be produced by the unstalled side. This effectively reduces the thrust capability of the tail rotor. The rotor blade must be sized to operate at lift coefficients below the value for stall throughout the operating range. The increased loading caused by rotor precession must be provided for in this sizing.

The effects of operation in a side wind also must be considered on the basis of a uniform variation of thrust with pedal position. When rotor-induced velocity approaches sideward velocity, the rotor will encounter the vortex ring state. This characteristic, shown on Fig. 5-52 (from Ref. 58), gives undesirable flying qualities and generally is avoided by pilots. It is preferable that the induced velocity (disk loading) be sufficiently high that the vortex ring state is not approached until sideward velocity exceeds 35 kt.

5-9.4.4 Direction of Rotation

When the helicopter is in rearward flight near the ground, the characteristics of a tail rotor installation designed with the top blade moving forward produce undesirable flying qualities. The tail rotor can encounter a large ground vortex produced by the main rotor, which causes nearly a 20% decrease in tail rotor

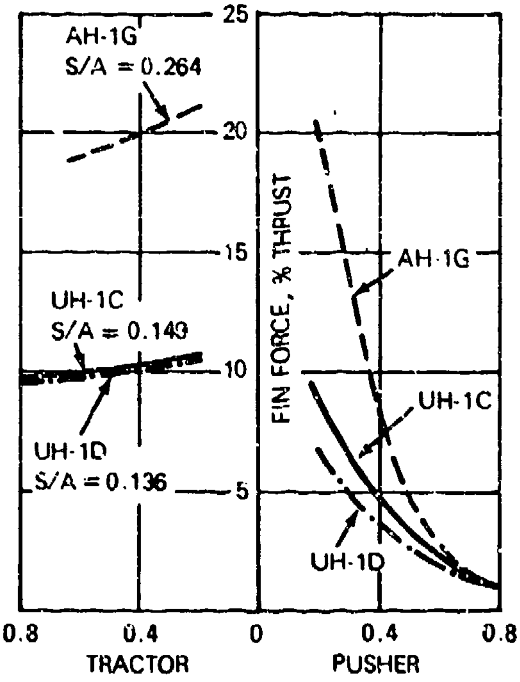


Figure 5-51. Fin Separation Distance/Rotor Radius

thrust and an adverse fin force of nearly 25% of the remaining tail rotor thrust. This decrease of effectiveness results in an increase in required power, and could influence the rotor solidity required to prevent blade stall (Ref. 61). Tests have shown substantial improvement in control capability in this critical condition when the direction of rotation is reversed (Ref. 63). This improvement apparently is due to the higher relative velocity of the airflow over the tail rotor because the tail rotor is turning into, or against, the main rotor downwash instead of turning in the same direction. Therefore, tail rotors normally are designed to rotate with the bottom blade moving forward.

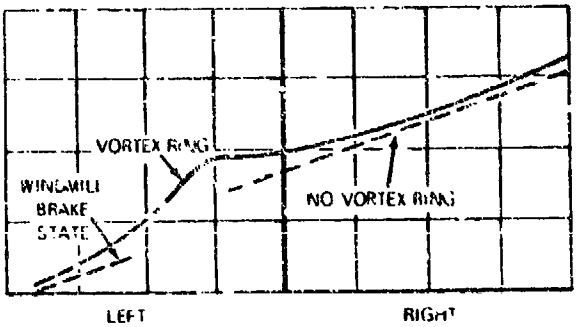


Figure 5-52. Sideward Flight Velocity

5-9.4.5 Engine Exhaust

Possible drastic reductions in tail rotor thrust due to the ingestion by the tail rotor of hot exhaust gases (with resultant reduced air density) should be avoided by careful design and placement of engine exhausts, and by subsequent testing.

5-9.5 TAIL ROTOR DESIGN PARAMETERS

The physical size of a tail rotor will depend upon the loading, tip speed, blade twist, and airfoil section. Because these parameters influence the installed weight, it is necessary to determine their interrelationship in order to optimize the design.

5-9.5.1 Tail Rotor Disk Loading

As in the main rotor, the power required to generate thrust depends upon the disk loading, and thus on the diameter selected. As noted in Fig. 5-53, the power required increases with increased disk loading; i.e., thrust loading, lb/hp, decreases. To minimize the power required, low disk loadings are indicated. However, as noted in par. 5-9.4.3, it is preferable that the disk loading be sufficiently high to provide an induced velocity greater than 35 kt during

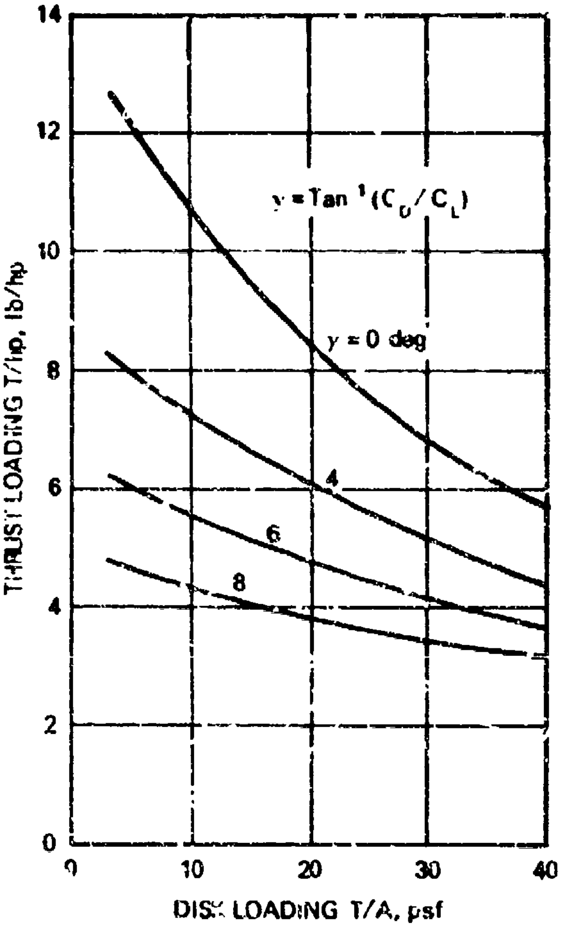


Figure 5-53. Tail Rotor Performance, Four Blades

sideward flight to the left. Tail rotor weight and boom size increase as disk loads decrease. Thus, the final selection of tail rotor disk loading will depend upon the overall trade-off of required power versus tail rotor diameter.

5-9.5.2 Tail Rotor Tip Speed

Tail rotors are designed to operate at tip speeds of 600-800 fps. For a given thrust requirement, tail rotors operating at low tip speeds will need higher solidities to obtain the required operating C_L. The low tip speed also increases the torque of the drive system. These factors both increase the overall weight of the antitorque system. Higher tip speeds can result in blade aerodynamic compressibility losses with corresponding power losses, high control forces, blade leading edge erosion problems, and higher noise levels (Fig. 5-54).

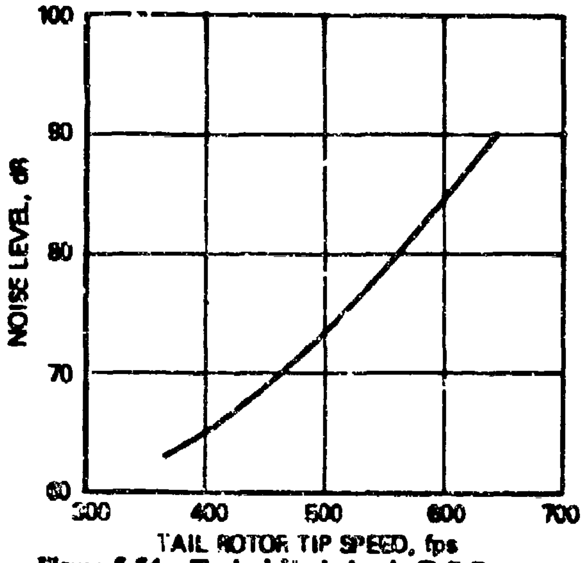


Figure 5-54. Typical Variation in Tail Rotor Noise Level

5-9.5.3 Blade Number and Solidity

The rotor solidity σ can be split up in any way between blade chord and blade number. Usually, it is best to keep the value of individual blade solidity $\sigma_b = c/b$ between 0.03 and 0.06 for reasons of structure and retention. The blade number b can be any value.

The rotor solidity required depends upon tip speed, disk loading, maximum thrust required, and the airfoil section chosen. The choice of the airfoil affects the maximum operating C_L, thus directly influencing the solidity.

The product $(c_e b)$ of the required blade chord c_e times the blade number b can be found from Eq. 5-22, knowing the maximum operating tip speed, helicopter yaw rate, fin blockage effect, and thrust requirement:

$$c_e b = \frac{6K}{C_{L_{max}}\rho(BR)^3\Omega^2} \times \left(T_Q + \frac{I_{zz}\ddot{\Psi}}{X} + \frac{8I_p b\Omega\dot{\Psi}}{3BR}\right), \text{ ft} \quad (5\text{-}22)$$

where

b = number of tail rotor blades
B = blade tip loss factor, dimensionless
c_e = effective blade chord, ft
I_p = polar moment of inertia per blade, slug-ft^2
K = ratio of total tail rotor thrust to net tail rotor thrust, dimensionless
R = tail rotor radius, ft
T_Q = tail rotor thrust to compensate for main rotor torque, lb
X = distance from centerline of main rotor to centerline of tail rotor, ft
ρ = air density, slug/ft^3
$\dot{\Psi}$ = yaw rate, rad/sec
$\ddot{\Psi}$ = yaw acceleration, rad/sec^2
Ω = tail rotor rotational speed, rad/sec

Eq. 5-22, which is derived from Ref. 58, considers the blade lift required for tail rotor precession and to provide a specified acceleration in yaw.

5-9.5.4 Twist

Blade twist decreasing from inboard to outboard is used to improve the spanwise load distribution (and, therefore, propulsive efficiency) of the tail rotor in the static or low-speed flight cases. The twist required to maximize propulsive efficiency becomes greater with increasing disk loading. However, although useful in optimizing rotor performance, blade twist can have adverse effects on the reverse thrust characteristics of a given rotor.

5-9.5.5 Blade Airfoil Section

The choice of the airfoil section influences overall size and performance. Maximum thrust is determined by the maximum operating lift coefficient of the section, and the section choice usually is based upon pitching moment characteristics. Many tail rotors use NACA 0012 and 0015 airfoil sections, which have a pitching moment of essentially zero over the usual operating range.

Because the size and weight of tail rotors are directly dependent upon the maximum operating lift coefficient, the use of cambered airfoils is being considered. The more important characteristics of two cambered sections considered suitable for tail rotors are shown in Table 5-5 along with the characteristics of the symmetric NACA 0012 for comparison.

The NACA 23012 airfoil is typical of a class of airfoil sections whose camber is primarily over the forward section of the airfoil. Compared with the symmetrical airfoil, this results in an improved $C_{L_{max}}$, with only a small increase in pitching moment coefficient C_M. The NACA 64,-412 is typical of the laminar-flow airfoils (NACA 63, 64, 65, and 66 series) evolved to provide low values of minimum section drag coefficient $C_{D_{min}}$ by maintenance of laminar flow over much of their surface, along with good high-speed lift and drag characteristics. Due to the fact that the camber is distributed over the entire chord, the pitching moment coefficient is quite large, as shown in Table 5-5.

The values of $C_{L_{max}}$ are shown for comparison only. Values used for design must include correction for operating values of Mach and Reynolds numbers and for leading edge roughness. Failure to apply necessary corrections to airfoil section data will result in deficient tail rotor capability.

By suitable design and balancing of the rotor blade, negative (nosedown) pitching moment can be reduced. For example, by using a suitably stiff blade with preconing, and by balancing the blade well aft of the aerodynamic center (Fig. 5-55), it may be possible to use the component of centrifugal force normal to the blade chord against the negative aerodynamic pitching moment. However, when the CG is moved aft the centrifugal centering moment (tennis racket effect) may be increased and the desired increase in noseup moment may not occur. Another method of handling the pitching moment is by using boost. In any case, control boost may be necessary in large helicopters to obtain the required time response.

5-9.6 TAIL ROTOR PERFORMANCE

Once the tail rotor configuration has been selected, the performance can be estimated for the zero velocity condition by using the same procedures as for a hovering main rotor (see par. 3-2, AMCP 706-201).

TABLE 5-5
AERODYNAMIC CHARACTERISTICS OF SEVERAL AIRFOIL SECTIONS SUITABLE FOR TAIL ROTOR BLADES

AIRFOIL	$C_{l_{max}}$	C_M
NACA 0012	1.58	0
NACA 23012	1.78	−0.015
NACA 64 412	1.67	−0.079

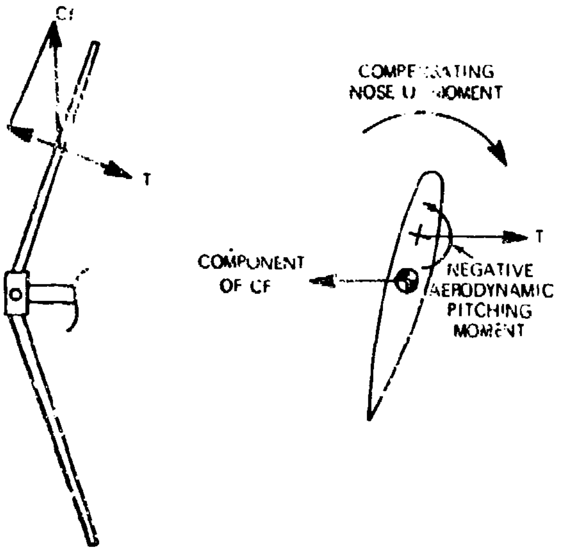

Cf
T
COMPENSATING NOSE UP MOMENT
T
COMPONENT OF CF
NEGATIVE AERODYNAMIC PITCHING MOMENT

Figure 5-55. Compensation for Negative Pitching Moment With Coning Angle and Aft Blade CG

5-9.7 STRUCTURAL CONSIDERATIONS

5-9.7.1 Structural Dynamics

The placement of the natural frequencies of the tail rotor is extremely important in insuring the structural integrity of the system. The effects of main rotor aerodynamic excitation must be considered. For instance, the main rotor may contribute frequencies at the tail rotor corresponding to one, two, four, and six per rev. A three-per-rev frequency in the rotating system also may be found. A summary of the excitation sources is shown in Table 5-6 (from Ref. 59).

Guidelines for placement of the tail rotor natural frequencies have been developed (Ref. 59) and are:

1. All vibration modes occurring below 150 Hz should be considered.

2. The natural frequencies of the tail rotor should not be coincident with, nor in close proximity to, any exciting force frequencies for steady-state operating conditions, including ground idle.

3. Natural frequencies coincident with the excitation sources shown in Table 5-6 should be avoided for at least the first two modes.

Additional details of the placement of these natural frequencies may be found in Ref. 59.

5-9.7.2 Structural Loading

The inclusion of the first four lowest frequency modes (out-of-plane and inplane) is sufficient to represent steady-state rotor behavior. The aerodynamic blade loads may be calculated by classical techniques, with the local blade segment aerodynamic coefficient defined as a function of Mach number and angle of attack for the design operating condition (see par. 3-2, AMCP 706-201). The effects of induced velocity, main rotor velocity, fin interference, and elastic feedback velocity, as appropriate, must be included to obtain the proper overall loads.

TABLE 5-6 SUMMARY OF TAIL ROTOR EXCITATION SOURCES

SOURCE	FREQUENCIES	BLADE MODE
ANISOTROPY, UNBALANCE, OUT-OF-TRACK	$nb\Omega_{tr}$	INPLANE
STEADY STATE FIXED SYSTEM EXCITATION	$\Omega_{mr} \pm \Omega_{tr}$ $nb\Omega \pm \Omega_{tr}$	INPLANE
	Ω_{mr} $bn\Omega_{mr}$	OUT-OF-PLANE
TRANSIENT FIXED SYSTEM EXCITATION	$\omega_{f_i} \pm \Omega_{tr}$	INPLANE
	ω_{f_i}	OUT-OF-PLANE
MAIN ROTOR AERO EXCITATION	$b\Omega_{mr} \pm \Omega_{tr}$ $n\Omega_{mr} \pm \Omega_{tr}$	INPLANE
	$nb\Omega_{mr} \cdot \Omega_{mr}$	OUT-OF-PLANE

5-9.7.3 Blade Structural Analysis

When blade sections, stiffness, and mass distributions have been selected and the externally distributed loads calculated, the next task is to establish the structural integrity of the blade. The internal strain (stress) distribution among the elements of the blade is computed; the resultant stress levels are compared with allowable levels that test and/or experience have shown will preclude failure during the life of the system.

There are two general categories of design loading conditions considered in blade design: ultimate conditions and fatigue conditions. The blade must have an ultimate strength 50% greater than the highest peak load anticipated during the lifetime of the system. The blade also must have fatigue strength sufficient to prevent a failure due to alternating loads. Experience has shown that fatigue usually is the more critical design condition.

In fatigue analysis, a spectrum of fatigue loads and their expected number of occurrences in the lifetime of the helicopter is developed. This is derived from a mission analysis that accounts for all maneuvers, turbulence, climbs, descents, taxi loads, and ground-air-ground cycles (see Chapter 4, AMCP 706-201 and Chapter 5, AMCP 706-203).

The fatigue strength of the component generally is given in the form of a *S-N* curve derived from test data. The magnitude and shape of the curve are a function of material, stress concentrations, environmental conditions, and magnitude of concurrent steady stress.

The influence of steady stress on fatigue strength may be considerable and must be considered in fatigue analysis. In tail rotor blades, the steady stress generally is equal to the alternating stress, and this reduces the allowable alternating stress by approximately 20%, depending on the material.

As with main rotors, tail rotor blade design is an iterative process of developing a blade with adequate aerodynamic, physical, and dynamic characteristics, and fatigue integrity. If, at any stage in the design process, an inadequacy is detected, the design process begins once again until all required characteristics are achieved.

5-9.7.4 Aeroelasticity

Problems due to aeroelasticity have been encountered in the installation of tail rotors. These problems have resulted in undesirable flying qualities at high speeds. An example of such a problem, termed "tail wagging", was encountered on an experimental helicopter. The rotor was mounted on a fin that projected above the tail boom. A natural frequency of the boom in torsion was coupled to the change in rotor thrust due to lateral velocity (inflow) so that negative damping was encountered. The change in tail rotor thrust due to the motion is proportional to lateral flapping b_1 and to the pitch-flap coupling tan δ_3. It was found in this case that negative δ_3 was the best way to damp the system. This type of problem can be avoided only through careful consideration of the dynamic characteristics of all components of the helicopter.

5-9.7.5 Flutter and Divergence

Tail rotors are not scaled-down main rotors. Therefore, flutter and divergence problems generally are not as severe as for the main rotors. Usually, the tail rotor blades are much stiffer than those in main rotors due to the operating environment. Also, the relative inertia, as expressed by Lock number, is of the order of 2.5 times that of the main rotor. Finally, aspect ratios of tail rotor blades are much lower. Although these factors reduce the tendency toward flutter and divergence, problems with tail rotors have occurred, so the proper combination of δ_3 and pitch link stiffness must be used. The methods discussed for the main rotor (par. 5-4) and the data in Ref. 59 may be used to determine the design details necessary to eliminate this problem

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CHAPTER 6
FLIGHT CONTROL SUBSYSTEM

6-0 LIST OF SYMBOLS

L_p = roll damping; derivative of rolling moment with respect to roll rate; ft-lb/rad-sec
L_β = dihedral stability; derivative of rolling moment with respect to yaw angle, ft-lb/rad
L_δ = roll control sensitivity; derivative of rolling moment with respect to control input, ft-lb/in.
M_q = pitch damping; derivative of pitching moment with respect to pitch rate, ft-lb/rad-sec
M_u = speed stability; derivative of pitching moment with respect to forward velocity; ft-lb/fps
M_α = pitch stability; derivative of pitching moment with respect to pitch angle, ft-lb/rad
M_δ = pitch control sensitivity; derivative of pitching moment with respect to control input, ft-lb/in.
N_r = yaw damping; derivative of yawing moment with respect to yaw rate, ft-lb/rad-sec
N_β = yaw stability; derivative of yawing moment with respect to yaw angle, ft-lb/rad
N_δ = yaw control sensitivity; derivative of yawing moment with respect to control input, ft-lb/in.
n = number of pitch links, dimensionless
n_1 = number of nonredundant components having the failure rate P_1
n'_1 = number of redundant components having the failure rate P_1
n_2 = number of nonredundant components having the failure rate P_2
n'_2 = number of redundant components having the failure rate P_2
P_x = failure rate of aggregate of components, hr^{-1}
q = dynamic pressure, psi
Z_w = vertical damping; derivative of vertical force with respect to vertical velocity, lb/fps
Z_δ = vertical control sensitivity; derivative of vertical force with respect to control input, lb/in.
ζ = damping ratio, dimensionless
θ = generalized rotation about any axis, rad
Ω = rotational speed, rad/sec
ω_n = undamped natural frequency, rad/sec
ω'_l = natural frequency of inplane (lead-lag) motion of rotor blade, rad/sec

6-1 GENERAL

Flight control system detail design involves translating a helicopter preliminary design, which has been shown to satisfy the general stability requirements stated in Chapter 6, AMCP 706-201, into a refined configuration that meets the mission flying-quality requirements. The paragraphs that follow discuss the influence of helicopter stability requirements upon the detail design of the flight control system.

6-1.1 DESIGN METHOD

The airframe manufacturer — acting as the system integrator of airframe, controls, and stability augmentation subsystem — should conduct iterative and/or competitive trade-off studies. These studies will evaluate the performance, cost, safety, reliability, and maintenance characteristics of one or several control systems as they relate to the mission requirements and flight path stability specifications referred to within this chapter.

6-1.1.1 Point of Departure

Typically, the preliminary design results in a definition of the flight controls and a first estimate of stability augmentation subsystem characteristics that are believed to be sufficient to permit compliance of the helicopter with the stability and control specifications. These preliminary design data include: control kinematics as limits on rotor blade or aerodynamic control surface travel, general arrangement of controls, and mechanical advantages. The preliminary design serves as a base point from which design alterations are proposed for the purpose of improving system capability, reliability, maintainability, and cost. These design alternatives then are subjected to a detail design trade-off study for assessment. Typical considerations to be reviewed are:

1. The level of helicopter stability required, with or without augmentation
2. The parameters that should be controlled, and the characteristics of these control loops
3. The automated tasks or autopilot (pilot relief) functions that should be provided
4. The need for maneuver augmentation to provide satisfactory helicopter response to pilot control inputs as well as to external disturbances
5. The use of single-, dual-, or multichannel redundant systems; augmentation actuator location; and whether augmentation inputs should be in series with or parallel to the pilot's inputs

6. Control system ballistic tolerance to anticipated threats.

A synopsis of automatic control system requirements is presented in MIL-C-18244.

6-1.1.2 Mission Requirements and Flight Envelope

MIL-H-8501 designates conditions for which the stability of the helicopter must be evaluated. The following are additional parameters to be included for each configuration applicable to the assigned missions:

1. CG limits in the vertical, lateral, and longitudinal directions and their allowable variation in comparison with other parameters (CG limits are a function of gross weight for conventional helicopters)
2. Allowable rotor speed variations
3. External load configuration.

Because stability is so dependent upon the factors cited, their effects should be evaluated throughout the development phase, and during analysis and testing. Compliance with stability specifications should be evaluated, and all the factors weighed as to minimum requirements.

6-1.1.3 Basic Helicopter Data

During analysis and flight testing, data on the stability characteristics of the basic, unaugmented airframe should be established. The helicopter system is likely to undergo many changes in its automatic controls, but only limited airframe changes, during its life cycle.

6-1.2 ANALYTICAL TOOLS

There are two basic forms of mathematical representation for assessing vehicle trim and stability: small-perturbation equations and total-force equations.

The typical small-perturbation equations noted in par. 6-2, AMCP 706-201, expedite the assessment of stability at one flight condition, and can incorporate nonlinear control loops readily. These equations also are adaptable to parametric studies using analog or digital computers.

Total-force equations completely describe the absolute forces acting upon the helicopter. However, they require a rather large complement of either analog or digital computer equipment for their solution. This type of solution is necessary when investigating large variations in flight conditions (i.e., speed, attitude, and large bank angles).

Wind tunnel testing should be conducted in order to refine the mathematical model. Information concerning aerodynamic characteristics of the fuselage, steady-state rotor characteristics, and rotor downwash effects are obtainable from this testing.

6-1.3 SIMULATION AND TESTING

Other than mathematical representation and wind tunnel investigation, key tools in the iterative method of developing satisfactory helicopter flying qualities are pilot-in-the-loop flight simulation, bench testing, hardware-in-the-loop simulation, and prototype flight testing. These tests provide data from which the analytical model can be updated and revised, if necessary.

Ground-based, piloted flight simulation provides an evaluation of man/machine interface problems and an estimate of the flying qualities prior to the flight of the prototype. It also allows additional lead time for preparation of solutions to any potential problems indicated by simulation. A simulation capable of evaluating many aspects of flight control is discussed in Ref. 1.

6-2 STABILITY SPECIFICATIONS

Because much effort is spent evaluating the handling qualities of a helicopter, it is necessary to attempt to define "good handling". Possibly there is no single definition, but helicopter dynamic stability certainly influences the pilot's ability to control the vehicle and his acceptance of its characteristic responses to external disturbances. Of special interest are the magnitude of the response, the rapidity with which a steady attitude or trim is regained, and the existence of any oscillations or lightly damped reactions. Engineers work to quantify the rating of handling qualities and dynamic stability so that they are able to provide future vehicles with desirable stability and to make predictions or comparisons for existing helicopters. The latter includes prediction with sufficient accuracy to supplement flight testing activities and to expand the engineer's understanding of safety-of-flight topics.

6-2.1 CRITERIA AND METHOD OF ANALYSIS

The following list presents the more significant topics relating to helicopter handling qualities:

1. Control power, sensitivity, and interaxis coupling
2. Inherent or augmented static stability and damping
3. Characteristic roots
4. Type of automatic control system and variables controlled
5. Force feel
6. Magnitude of response
7. Influence of control components upon stability.

MIL-H-8501 provides workable guidelines for establishing handling qualities. Complying with the specifications, however, does not necessarily produce the dynamic stability levels that result in the best handling qualities, nor does it insure mission success.

Considerations pertinent to improving helicopter mission effectiveness, along with methods of analysis, are reviewed in the paragraphs that follow.

6-2.1.1 Control Power and Damping

Par. 6-3.1, AMCP 706-201, presents charts illustrating control power and damping requirements for operations under both instrument flight rules (IFR) and visual flight rules (VFR). During preliminary design, specification compliance is obtained by providing values within the boundaries shown on these charts. It is convenient to estimate the augmented damping due to stability augmentation systems and to plot these estimates on the charts.

The minimum angular displacement response for each control input shall be as specified in MIL-H-8501 for VFR or IFR conditions, as applicable. This requirement places a constraint upon the maximum augmentation loop gain, including any pilot maneuvering compensation, because a high gain reduces angular rate, and, therefore, angular displacement at a given time. These criteria are constructed as control sensitivity and damping curves for the pitch, roll, and yaw control axes in par. 6-3.1, AMCP 706-201. Although gain setting by these criteria is not sufficient, and requires other analyses, this method does serve as a first estimate for rate control loop setting. This type of rating is not applicable for attitude control loops, which must be evaluated for their specification compliance by using transient analysis.

Vertical control sensitivity Z_δ and vertical damping Z_w likewise are rated in par. 6-2.5.3, AMCP 706-201. These parameters are fixed by performance considerations unless they are altered by augmentation controls. Use of vertical augmentation is probably due to the present trend to incorporate hover-hold systems which add control loops that regulate vertical velocity and position. In hover, the thrust lever commands vertical velocity with a first-order time lag equal to the reciprocal of the vertical damping. This lag for a cargo helicopter is about 5 sec ($Z_w = 0.2$ sec^{-1}). This is long enough for pilot opinion and workload to be influenced markedly by a 50% or greater reduction in time constant.

For certain helicopter types (e.g., tilt-rotor vehicles), the rotor is not an effective source of rolling moment throughout the flight envelope. Therefore, it is necessary to employ ailerons, or some other roll control device, in cruising flight, where the rotor shaft axis essentially is parallel to the flight path. In practice — since this type of helicopter accelerates from hover to cruise — the rotor slowly is phased out as a roll-control source, with the roll-control function of the ailerons progressively increasing with airspeed and/or vehicle configuration variation. A similar situation exists with respect to the pitch, yaw, and altitude controls. Fig. 6-1 is an illustration of typical control function scheduling for the roll axis.

No concise flying quality specification exists for nonhelicopter rotorcraft. Therefore, the procuring activity generally specifies some composite application of MIL-H-8501 and MIL-F-8785 as a function of airspeed or load distribution among the lift producers.

6-2.1.2 Characteristic Roots

Any mode of response detectable and/or directly controllable by the pilot *shall* exhibit a level of dynamic stability at least equal to that specified in MIL-H-8501. This specification primarily addresses the longitudinal axis for VFR, but it should be considered a minimum requirement for any axis.

There also are requirements for stabilizing such response modes as are not under the direct control of the pilot, or are controlled by him only indirectly such as by changing flight conditions or system settings. Examples include modes influenced by stability augmentation control loop coupling, potential aeromechanical resonance, and external sling load dynamics. These modes should meet the MIL-H-8501 criteria for applicable VFR or IFR conditions, except that where MIL-H-8501 allows divergence or does not specify a criterion, some minimum acceptable damping should be designated (e.g., 0.03 damping ratio, which is the minimum for flutter in MIL-A-8870).

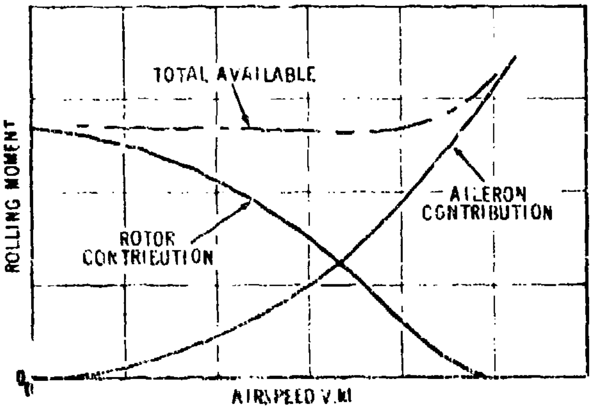


Figure 6-1. Typical Control Function Scheduling for a Tilt-rotor Aircraft

MIL-H-8501 requirements are specified in terms of the time necessary for the motion to double its amplitude or to achieve half amplitude, and therefore are adaptable readily to system analysis techniques. Solutions to the linearized vehicle equations of motion exhibit both frequency and rate of decay characteristics independent of the manner in which the vehicle is disturbed. Changing the type of disturbance changes only the magnitude of the response; the frequency and rate of decay (measures of stability) remain unaffected. The characteristic roots of the equations of motion are a measure of this stability, and may be plotted on the complex plane. These plots have the same format as do root locus plots used by servo system analysis.

6-2.1.2.1 Root Plots

A characteristic root plot for an unaugmented helicopter is shown in Fig. 6-2. The vertical scale is the damped natural frequency or complex part of the root, the left half (negative abscissa) designates the stable real part of the root with the time to half amplitude as scaled, and the right half (positive abscissa) designates an unstable root location with time to double amplitude as indicated.

The VFR criterion boundary given in MIL-H-8501 also is plotted in this figure. Root locations to the left of the boundary line satisfy the specification criteria. The second-order system damping ratio, and either the undamped or damped natural frequencies of the plotted roots, are obtainable. The undamped natural frequency ω_n is the magnitude of the radius vector. Experimental data or computer solutions of the equations of motion may be plotted directly in this format so as to indicate system proximity to a specification limit. Experimental data are estimated either by using a step response and extracting the damping and frequency (by assuming a second-order system), or by analog computer matching.

Typical plots are shown in Fig. 6-2 for a range of airspeeds from hover to cruising flight. In hover the helicopter exhibits an unstable oscillatory mode of 8 sec to double amplitude, and a single convergent aperiodic root. As the airspeed increases, the single root moves to the right and becomes a divergent root while the other oscillatory root becomes stable and decreases its frequency.

6-2.1.2.2 Modes and Required Damping

The damping requirement for either VFR or IFR frequently may be exceeded with only minor complication and increase in cost. This damping may be increased for three reasons:

1. Generally, well-damped roots are less likely to couple with other modes of response, or to respond to periodic disturbances, than are lightly damped roots.
2. Good damping reduces pilot workload.
3. Augmentation systems having good damping are less prone to limit-cycle oscillations with resultant minor subsystem deterioration.

Damping requirements usually are associated with a particular mode of response, and those for helicopters are almost identical to those for fixed-wing aircraft, except for the hover condition.

A helicopter with stability augmentation exhibits in hover a longitudinal, short-period mode, which is predominantly a pitching response. The mode should be adequately damped since it is the pilot's prime control mode both in hover and in forward flight. The pilot rarely will rate the vehicle as satisfactory (Cooper Rating of 3.5) unless the modal damping is greater than 0.3, and will accept a value of 0.3 only when the frequency is the most favorable for a given flight test experiment.

There is no target value for the short-period natural frequency; this might be established best in simulation and prototype flight test. There are upper limits, however, because increasing the frequency tends to aggravate any existing control cross-coupling or coupling into other modes.

There *shall* be no objectionable flight characteristic attributable to poor speed stability (phugoid mode). Any long aperiodic or long-period (greater than 10 sec) oscillatory divergences, as allowed in the present MIL-H-8501, compromise speed stability and thus place an added workload on the pilot. Stick-fixed speed stability demands that all roots exhibit stable damping. Helicopter lateral-directional flight dynamics generally exhibit roll subsidence, spiral, and Dutch roll modes similar to those of fixed-wing aircraft. The spiral and Dutch roll modes may occur at two or more frequencies in helicopters with lateral axis stability augmentation.

Roll subsidence is an aperiodic response, with the time constant set primarily by the amount of roll damping. The higher the damping, the shorter the time constant. The requirement for minimum damping is set by MIL-H-8501, e.g., for a typical, light (2500-lb) vehicle under IFR conditions, the requirement is a 0.3-sec time constant. For cargo helicopters, the IFR specification allows time constants in excess of 1 sec. Test results from a nap-of-the-earth flight typical of a weapon or reconnaissance helicopter show a distinct advantage in reducing the roll control time constant from 0.3 to 0.12 sec. This test and supplemental data on roll control response are detailed in Refs. 2 and 3. The value of roll damping

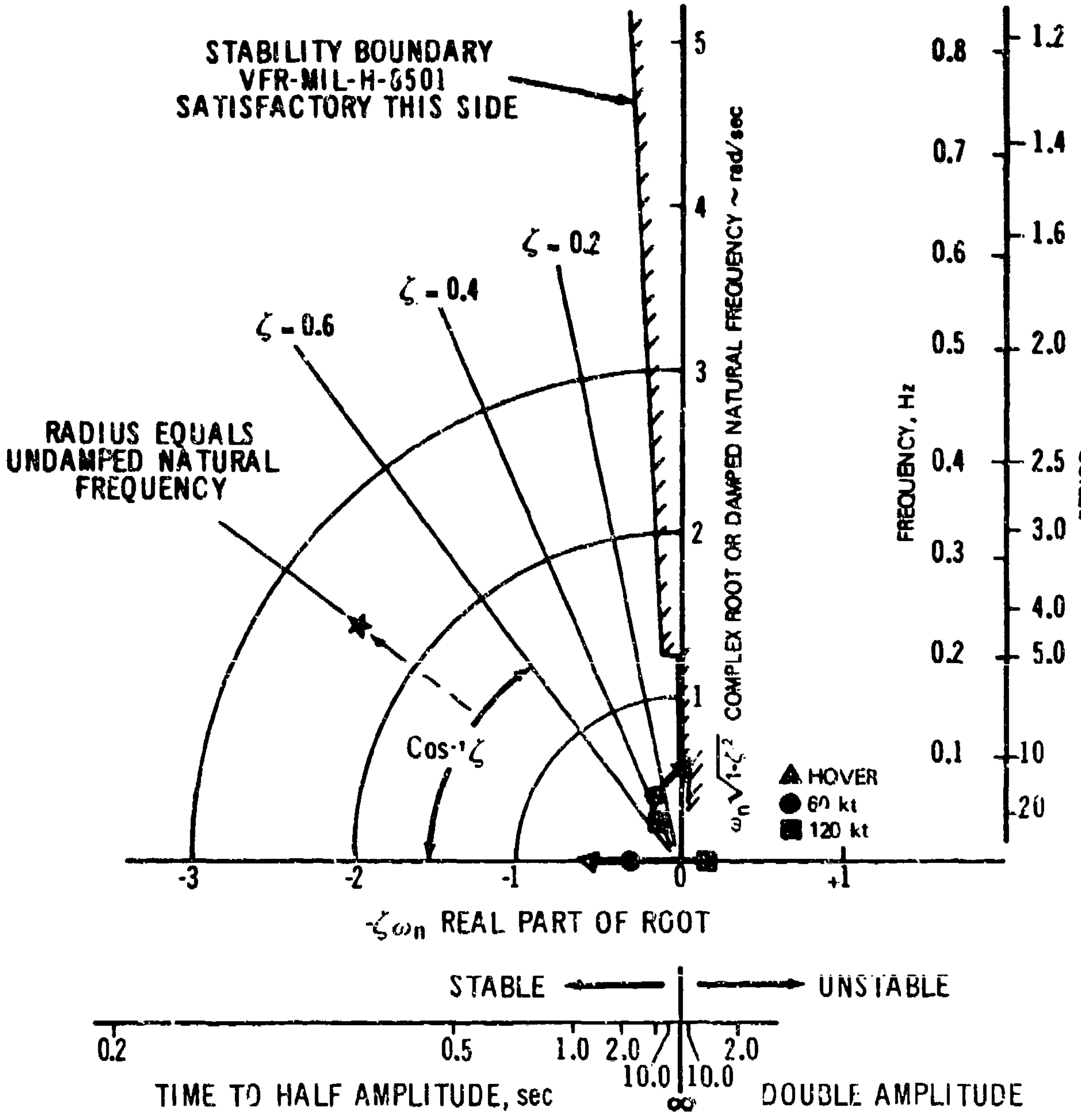


Figure 6-2. Characteristic Root Plot

can be assessed more effectively by the roll subsidence time constant. For missions requiring agility, the time constant should be minimized, consistent with other factors such as pilot acceleration environment and lateral pilot induced oscillation (PIO).

In the text referenced, the low roll time constant was obtained with a rigid-rotor vehicle. However, the same response can be obtained with a conventional, articulated rotor with stability augmentation. Root and transient analyses can indicate the augmentation loops needed in order to reduce this time constant. Because of the obvious requirement to minimize time delays, any lags in roll control actuation also should be reduced as much as is practicable.

At speeds above 40 kt, other lateral modes of the helicopter are comparable to those of fixed-wing aircraft. Unfortunately, MIL-H-8501 presents no VFR stability criteria. Poor, and even unstable, Dutch roll characteristics can occur with MIL-H-8501 compliance. Poor Dutch roll stability under conditions requiring manual heading control puts an excessive burden upon the pilot. The requirements for spiral and Dutch roll stated in pars. 3.7.4.4 and 3.7.4.5 of Ref. 4 should be used instead. These are similar to fixed-wing specifications, and can be met with rudimentary stability augmentation. Any coupling of the lateral roll subsidence and spiral modes (commonly referred to as a "spiral-roll" or "lateral phugoid" mode) should not be detectable by the pilot.

6-2.1.2.3 Inherent Airframe Stability

The unaugmented (inherent) stability of the helicopter *shall* be considered fully with respect to vehicle configuration and flight envelope limits, even though augmentation is planned. This is necessary because situations can arise in which augmentation system failures require the pilot to fly only with the inherent vehicle stability. A cost-effectiveness and performance-effectiveness design trade-off exists between the degree of flight control system sophistication and the degree of inherent stability provided by the basic helicopter configuration. In effecting such a trade-off study, the level of stability required under failure conditions *shall* be defined since it may be subject to variation with the flight control system concept used.

For example, consider the following three relationships between augmentation sophistication and inherent stability:

1. A reliable, multiredundant augmentation system can tolerate a significant degree of inherent instability. A vehicle so equipped, however, may not be capable of flight with all augmentation switched off.
2. A single-channel augmentation system will require inherent vehicle stability suitable for compliance with the VFR requirements of MIL-H-8501 under augmentation system failure conditions.
3. A dual-channel system will require an inherent stability level somewhere between the first two cases. A hardover failure of one channel (while operating in the normal dual mode) can produce a vehicle response that causes the remaining operating channel to experience a saturation in opposition to the failure. Such a failure, with inherent airframe instability, results in a rapid divergent vehicle response, whereas the vehicle response with increased positive inherent stability becomes slower and more easily controllable by the pilot.

A method for improving the inherent longitudinal stability of single- or tilt-rotor vehicles is the addition of a horizontal tail; "differential delta three" (rotor blade flap-pitch coupling on the forward rotor) can be used with tandem-rotor vehicles. Lateral-directional stability improvement is afforded by the tail rotor and a vertical tail surface on single- or tilt-rotor machines, whereas fuselage aerodynamics (including the aft pylon) is the principal source of improvement for tandem-rotor vehicles.

6-2.1.2.4 Variation of Parameters

The effect of added stability augmentation upon the characteristic roots should be reviewed thoroughly on the root plots, especially with regard to changes in augmentation gains, time constants, airspeed, and other flight conditions. This information will help save time and cost by indicating regions of critical stability and regions where stability is too sensitive to certain parametric variations.

The root plot formats important to design are:

1. Basic helicopter versus airspeed, at critical gross weights and at both extremes of the CG range
2. Basic helicopter versus airspeed, as in Item 1, but at the upper limit designated for the mission
3. Helicopter with stability augmentation for the conditions in Items 1 and 2.

6-2.1.3 Type of Control

The types of stability augmentation applicable to a given design vary in the parameters they control and, therefore, in how they aid the pilot.

Common augmentation control types are:

1. Rate
2. Attitude and trim
3. Altitude and/or airspeed hold
4. Heading
5. Hover position
6. Special augmentation.

Rate controls provide improved damping of all augmented degrees of freedom (potentially six, including three linear and three angular). This type of augmentation aids the pilot in coping with the short-period responses, but does not prevent long-term roll drift and possibly a sluggish or unstable speed hold, even with a stable stick gradient.

The addition of attitude loops and trim functions (such as lateral accelerometers and speed-hold loops) aids the pilot by providing long-term trim-speed hold and strong sideslip roll attitude and pitch control. The pilot then must provide only the power setting, altitude, and heading control. Finally, the use of altitude-hold and heading-hold removes the need for pilot input to the controls. Hover-position hold loops, and controls designed to impart stability to external sling loads in forward flight, are examples of special augmentation controls.

6-2.1.4 Transient Response

The characteristic root analysis only partially describes the handling qualities of a helicopter. The magnitude of vehicle response to either pilot inputs or external disturbances also must be determined.

For pilot control inputs, the control sensitivity requirements are given by MIL-H-8501 for all but the vertical axis. Some complication arises with the introduction of stability augmentation loops, which may tend to reduce the control sensitivity. This effect can be overcome by the incorporation of control input feed-forward arrangements. This approach not only allows use of high-gain augmentation for minimum response to external disturbance, but also provides the desired sensitivity to pilot inputs.

In addition to specifications for longitudinal control sensitivity, MIL-H-8501 presents a criterion for maneuver stability. It states that, following a specified step input, the acceleration and pitch rate responses *shall* be concave downward and convergent in not more than 2 sec after the start of the input. A shorter time is desired for attaining a stabilized value of acceleration, especially where maneuvering capability is essential to the mission. Analysis can indicate methods of maximizing this performance by considering various combinations of feed-forward and loop closures. Such analysis should include any significant structural dynamics. For example, elasticity of the rotor mounting causes a major increase in the time required for the response to become concave downward (Ref. 2).

It also is desirable to analyze the transient response to pulse inputs. This response will indicate the helicopter behavior in turbulent air. An assessment of the attitude time histories, and the time interval required to reacquire the initial trim flight condition with alternate stability augmentation schemes, is recommended. For augmentation systems employing some form of pilot control feed-forward, the pulse should be inserted downstream of the augmentation system so as to simulate a turbulence encounter since pilot pulse imput into such a system is not equivalent to an atmospheric pulse.

In addition to the transient response analyses noted, transients associated with the following maneuvers should be assessed with respect to SAS authority, control margins, and vehicle capability:

1. Jump takeoff
2. Rapid acceleration from hover to maximum level speed
3. Quick stop
4. Autorotation entry and recovery
5. Hovering turns
6. Pedal-fixed turn entries and recoveries
7. Fixed collective, constant speed turn entry and recovery.

The characteristics of residual, limit-cycle oscillations *shall* be maintained within acceptable levels. Table 6-1 presents a set of allowable limits for use by the designer.

Fig. 6-3 is a graphical presentation of allowable limits for the pitch axis. The acceleration requirement establishes the high-frequency limit, angular rate establishes the middle-frequency range, and angular displacement establishes the low-frequency limit.

6-2.1.5 Other Factors

The stability and control requirements specified or derived for the helicopter represent the design objectives for the flight control system. Therefore, each potential flight control system design must be investigated to insure that it does not violate these operational requirements. Often, a stability augmentation system will exhibit satisfactory performance for small-perturbation maneuvers or disturbances, but will impair vehicle stability severely during large disturbances and atmospheric turbulence because of its rate-limiting or saturation characteristics. Then, it becomes necessary to establish the operational limits of the flight control system through analysis and experiment. If the system limits vehicle stability, it must be reconfigured appropriately in order to assure specification compliance. Flight conditions for which such studies must be made include high-rate vehicle motions, regions where stability may be sensitive or highly nonlinear, and turbulent air.

The influence of flight control system sensor outputs upon vehicle stability also must be considered and monitored in order to assure proper design and operation. For example, a vertical gyro employed as a pitch angle sensor in a body axis coordinate system will introduce significant inter-axis coupling at large bank angles, because the gyro operates in an earth-axis coordinate system.

The effect of atmospheric or self-induced turbulence upon helicopter stability and control requires significant attention. A vehicle that has satisfactory handling qualities in calm air may exhibit large attitude and rate responses, including poor speed hold, in gusty air due to poor modal damping or to control loop nonlinearities. Response to atmospheric turbulence *shall* be evaluated over the entire flight envelope. MIL-F-8785 provides criteria suitable for gust analyses in cruising flight. Attention also *shall* be given to special operating requirements, such as an external cargo hookup, where gusts or wind shifts often increase the demand upon the position stability augmentation system.

Many nonhelicopter rotorcraft are subject to stability degradation from self-induced disturbances close to the ground during hover or low-speed operations. This phenomenon, often termed "skittishness", is believed to be caused by the rotor wake reflecting from the ground and reimpinging upon the fuselage or wings, or being reingested as rotor inflow. With certain combinations of vehicle configuration and flight operating variables, this reimpingement and reingestion becomes oscillatory, and results in large random responses. If the flight control system is forced to suppress skittishness, the designer must insure that satisfactory stability augmentation and control margin levels are produced.

TABLE 6-1. MAXIMUM AMPLITUDES OF LIMIT-CYCLE OSCILLATIONS

AXIS	OSCILLATION ANGLE	OSCILLATIONS ANGULAR RATE	LINEAR ACCELERATIONS
PITCH	±3 mils MIL-F-8785	±0.5 deg/sec USAAML-TR-65-45	
ROLL	±3 mils MIL-F-8785	±0.15 deg/sec MIL-F-9490	
YAW	±3 mils MIL-F-8785	±0.10 deg/sec MIL-F-9490	
LATERAL (DUE TO OSCILLATION)			±0.01 g MIL-F-9490
VERTICAL (DUE TO OSCILLATION)			±0.02 g[1] MIL-F-9490 ±0.05 g MIL-F-8785

[1] DISREGARD FOR PURE HELICOPTER ENVIRONMENT

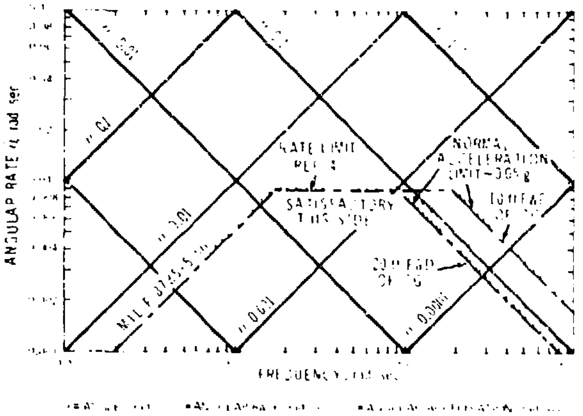


Figure 6-3. Allowable Pitch Control System Residual Oscillations

6-2.2 AUTOROTATION ENTRY

Most low-disk-loading helicopters have the capability for stabilized autorotation in the event of power failure but the vehicle characteristics must allow safe entry into this condition. During entry, the characteristics of the vehicle *shall* provide a reasonable pilot reaction time from the point of power failure to initial corrective action (1 sec minimum, 2 sec desired), and should permit a pilot of average ability to maintain control with adequate margin. Once stabilized in autorotation, the vehicle should be capable of mild maneuvers.

MIL-H-8501 states that the rotor speed *shall not* fall below a safe value, including that needed to maintain hydraulic and electrical power, during entry into autorotation. Other factors to be reviewed in connection with this maneuver are:

1. Margin of control power available to overcome disturbances, especially near zero load factor and for vehicles with fixed wings
2. Restriction due to blade stress limits
3. Flapping and blade clearance with respect to the airframe
4. Buffeting due to wing wake
5. Interaction of stability augmentation system
6. Magnitude of control trim change resulting from collective pitch reduction necessary to enter autorotation.

Simulation provides an excellent method of evaluating the average pilot's ability to effect the autorotation maneuver, and of defining potential improvements.

Refs. 5 and 6 describe the difficulties of obtaining a satisfactory time delay in the event of total power failure at an airspeed near 200 kt. With dual engine installations — where the probability of sudden, simultaneous (less than 2 sec) failures is extremely remote — it may be reasonable to consider only single-engine failures.

6-2.3 SYSTEM FAILURES

When a stability augmentation system (SAS) is used — whether it is electronic, fluidic, or mechanical — the potential hazard of a hard-over component failure exists. MIL-H-8501 requires that the pilot be able to delay a corrective control input for 3 sec without the response exceeding an angular rate of 10 deg/sec or a ±0.5 g change in normal acceleration.

The influence of flight conditions and CG position upon the severity of the response to an SAS failure should be reviewed. Frequently, an aft CG position coupled with flight operation near the blade or rotor limits is the most critical situation. Stress levels upon recovery from a failure may be an additional factor in the ability to satisfy the failure requirements.

Responses to SAS failures can be reduced in magnitude by the following methods:

1. Reduction in augmentation system authority
2. Increase in inherent airframe stability
3. Multichannel redundant systems.

Selection from among these methods during the design process is done after due consideration of the other flight control system requirements such as performance (especially the authority needed to meet gust and maneuvering requirements), reliability, maintainability, and cost.

Failure-effect studies, which note the consequence of each component failure, should be conducted in an organized manner. These must identify:

1. Any failure that cannot be tolerated, such as an oscillation due to loss of feedback
2. Any compromise in control margin
3. Failure causing multiaxis response too difficult for the pilot to control
4. Ability of the pilot to switch out failures
5. Consequence of subsequent failures.

As an example of Item 5, after the first failure in a dual system, the remaining system must meet the failure criteria or the flight envelope must be restricted so as to meet the failure requirements.

Pilot-in-the-loop simulation is a valuable tool for evaluating a wide variety of failure modes and their impact upon detail system design.

6-3 STABILITY AUGMENTATION SYSTEMS

6-3.1 GENERAL

Par. 6-2 contains numerous references to stability augmentation systems (SAS). Owing to the inherently poor stability of a helicopter rotor, satisfactory flying qualities have been achieved in many cases by altering the inherent characteristics artificially. Such techniques are called mechanical stability augmentation.

The pilot workload associated with early helicopters was very heavy. The handling qualities requirements of MIL-H-8501 have been developed not only to reduce this workload, but also to increase the mission capability of the helicopter. The result, however, is that it is virtually impossible to satisfy these requirements without modifying the inherent characteristics of the helicopter with a rather sophisticated SAS.

6-3.1.1 Bell Stabilizer Bar

Perhaps the earliest mechanical SAS is the Bell stabilizer bar. A bar with weights on the ends is mounted pivotally upon the rotor shaft at right angles to the two-bladed, teetering rotor (Fig. 5-7). Mixing levers are connected to the bar, and through pushrods, to both the swashplate and the blade pitch arms. Cyclic pitch input to the blades is the sum of pilot control input and stabilizer bar teetering motion. Viscous dampers, connected from the stabilizer bar to the rotor shaft, control the rate at which the plane of rotation of the bar and rotor follows or lags the tilt of the rotor shaft. This lag in tilting stabilizes or damps the helicopter pitch and roll motion. Additional data may be obtained from Ref. 7, and pars. 6-2.4.3.2 and 6-4.2.1 of AMCP 706-201.

6-3.1.2 Hiller Servo Rotor

Another early mechanical SAS is the Hiller servo rotor. The two-bladed, universally mounted, underslung rotor has a gyro bar fastened to the hub at right angles to the blades. On each end of the gyro bar is a short paddle blade with airfoil cross section, whose pitch is controlled cyclically by the swashplate. Cyclic pitch imparted to the servo rotor tilts its plane of rotation, resulting in a cyclic pitch input to the main rotor blades. Stabilization results from the lag in the response of the servo rotor to tilting of the rotor shaft, and the consequent pitch and roll damping due to the lagged response of the main rotor. Additional data may be obtained in Ref. 8 and pars. 6-2.4.3.2 and 6-4.2.1 of AMCP 706-201.

6-3.1.3 Mechanical Gyro

Refs. 9, 10, and 11 discuss two applications of intermediate-size (10-15 lb) gyros in mechanical SAS's. The first, produced by Cessna, is a rate gyro that is connected mechanically in series with the input from the pilot control stick to the control boost actuator. This system acts primarily to damp roll motions. The second, the "Dynagyro" by Dynasciences, is a two-axis, hydraulically driven unit with rotating damping arranged so as to align the gyro wheel slowly with its mounting reference in the fuselage. Outputs of the gyro, i.e., its pitch and roll displacements relative to its mounting, are fed into hydraulic boost actuators that are connected in series with the pilot's cyclic pitch boost actuators in the respective directions. This design is similar in principle to the Bell stabilizing bar, except that blade pitching moments are prevented from feeding back into the gyro. The Dynasciences SAS also includes a hydraulically driven rate gyro mechanically coupled into the hydraulic boost actuator that controls tail rotor collective pitch. This provides yaw damping. No electrical power is required in either system.

Ref. 12 describes an all-mechanical yaw rate gyro for single-rotor helicopters. The gyro, located at and driven by the tail rotor, tilts about a longitudinal axis in response to a yawing rate of the helicopter, and mechanically changes tail rotor collective. Rudder pedal displacement moves the reference point of the gyro centering spring, biasing the system for turns.

6-3.1.4 Lockheed Control Gyro

In later versions of the Lockheed control gyro, a gyro bar, consisting of as many arms as the rotor has blades, is mounted universally upon the rotor shaft above the rotor hub. Pitch links connect each arm to a pitch arm on the following blade. Push rods also connect each arm to a point directly below on the swashplate. Spring capsules in the linkage between the swashplate and the control stick enable the pilot to exert a moment upon the swashplate. This moment is proportional to stick displacement, and is transferred to the control gyro, which precesses in the appropriate direction 90 deg of rotor rotation later. The tilt of the gyro results in an input of cyclic pitch to the main rotor blades. Rotor tilt and fuselage tilt follow because of the relatively high flapping natural frequency of the hingeless blades. Additional data about this system may be obtained in Ref. 13 and in par. 6-2.4.3.2, AMCP 706-201.

6-3.1.5 Electrohydraulic SAS

In order to achieve acceptable handling qualities, many helicopters use electrically driven and sensed rate gyros to measure rates of pitch, roll, and yaw (Ref. 14). These rate signals are amplified, shaped, cross-coupled where appropriate, and fed into electrohydraulic servo actuators in series with the conventional control boost actuators.

6-3.1.6 Fluidic and Hydrofluidic SAS

The fluidic SAS, which is operated by air or liquid, is analogous to the electrohydraulic SAS and may be substituted for it (Refs. 15 and 16). The fluidic SAS, with specially developed angular rate sensors having no moving parts and with integrated circuits having no external plumbing, offers advances in reliability and significant savings in cost and weight. However, it represents an advanced state-of-the-art, and it still may suffer from problems such as leakage, temperature sensitivity, and null shift of the sensors.

6-3.1.7 Flapping Moment Feedback

Rigid-rotor helicopters exhibit strong noseup pitching moments with an increase in speed or in upward gust encounters. One method of counteracting this tendency is to sense the pylon bending moment and to apply cyclic pitch in such a direction as to reduce the moment. If the pylon is flexible, its deflection due to rotor moment can be connected mechanically into the cyclic pitch loop at the proper phase to reduce the deflection (Ref. 17). This concept has not yet been developed fully.

6-3.2 CRITERIA FOR SELECTION

6-3.2.1 Augmentation Requirements

It is virtually impossible for a helicopter to comply with the handling quality requirements of MIL-H-8501 without some type of SAS. Selection of the type of system to be installed requires evaluation of the deficiencies of the unaugmented, or inherent, characteristics. The evaluation criteria include both the specification requirements and the requirements imposed by the missions assigned to the helicopter.

Refs. 18 and 19 discuss the tailoring of helicopter handling qualities to mission requirements exceeding those set forth in MIL-H-8501. Par. 6-3.1, AMCP 706-201, presents recommendations for control power and damping.

High-performance attack and troop support helicopters require high control power in order to achieve the necessary maneuverability. Good damping in roll, pitch, and yaw also is required in order to prevent the helicopter from being oversensitive and difficult to hold in a given attitude. Furthermore, helicopters become more divergent at very high speeds, with the result that speed compensation of the stabilization system may be required. By means of simulation studies with alternate helicopter/SAS combinations, it is possible to determine a range of gains for the SAS that will cover the extremes of operational requirements. During flight test of SAS prototypes, adjustment capability can be provided by means of calibrated potentiometers or resistors (decade boxes). Final values for system gains should be based upon adjustments made under actual flight conditions duplicating those of the required mission. The test program also will establish whether or not the gains can be constant, or if they must vary with flight speed, gross weight, or any other parameter.

For a small observation helicopter, the requirements of MIL-H-8501 generally are adequate, and the simplest mechanical SAS may be sufficient to meet them.

6-3.2.2 Helicopter Size

The general category of SAS, mechanical or power-assisted, to be used is determined by helicopter size. Only the smallest helicopters can use all-mechanical systems, because the rotor feedback forces that the SAS must overcome are correspondingly small. In helicopters with power-operated controls (par. 6-4), the SAS need not operate directly upon the rotor but can operate at a much lower force level in the control system below the power actuators (between the pilot's stick and the actuators). In practice, if both hydraulic and electrical power are available, the SAS gyros are made as small as possible and their output signals are amplified (electrically and/or hydraulically) to the power level required to provide inputs to the control actuators. Dual or triple electrical SAS's can be provided below the final rotor control actuator with less weight than a single mechanical system.

6-3.2.3 Type of Rotor System

It is possible to use the Bell stabilizer bar or the Hiller servo rotor with rotor systems having more than two blades (Ref. 20). However, some of the obscure refinements or kinematic relationships necessary for the success of the system may be overlooked. For example, the orientation of the gimbal pivots on the Bell rotor is critical in order to prevent driving torque from acting about the feathering axis. In the Hiller system, the amplitude and phase of the feedback of blade flapping into the cyclic pitch control of the servo rotor paddles are very important to the effectiveness of the system.

The Lockheed control gyro, which is precessed by forces applied through springs, is applicable only to hingeless rotors or those with an equally high flapping natural frequency. In order for a flapping or teetering rotor to exert a moment upon the fuselage, it would have to tilt relative to the rotor shaft. To sustain this tilt, the control gyro would have to be tilted by an equal or greater amount, depending upon the linkage ratio. This control gyro tilt, 90 deg out of phase with the swashplate tilt, would alter the phase of the maximum spring-applied force upon the control gyro, causing it to nutate toward the swashplate tilt, and eventually to line up parallel to the swashplate. In the case of hingeless rotors, with their high control power, the required amplitude and/or duration of control gyro tilt are too small to permit any noticeable gyro precession.

In general, whenever a rotor-mounted SAS is modified from its original form, an extensive program of developmental and qualification testing is necessary. The internally mounted electronic, hydraulic, or fluidic SAS's, which are more flexible and less dependent upon rotor dynamics, are more adaptable to any rotor system. The electronic SAS gains are adjustable individually in pitch, roll, and yaw directions; can be made variable with airspeed; and can be cross-coupled if desired to compensate for adverse airframe cross-coupling.

6-3.2.4 Helicopter Configuration

Single-rotor helicopters can be equipped with any type of SAS that is compatible with helicopter size and the type of rotor used. This is because pitch and roll attitudes both are controlled by cyclic pitch inputs to the main rotor. The yaw SAS, if used, operates by controlling the collective pitch of the tail rotor. Thus, each SAS input to the helicopter control system is independent.

On the other hand, tandem-rotor helicopters obtain longitudinal control by use of differential collective pitch of the two rotors. Obviously, any longitudinal SAS will be required to change the thrust of one or both rotors. Rotor-located, mechanical SAS's — such as those that are used by Bell, Hiller, and Lockheed and that affect only cyclic pitch — are not adaptable readily to tandem-rotor helicopters. The necessity for mixing all controls from the cockpit of a tandem helicopter before they are impressed upon the rotor makes it more straight-forward to introduce SAS control inputs in series with the cockpit controls before mixing. However, in large helicopters that contain many linkages in the control system, even normal amounts of play in these linkages may detract from SAS performance. Therefore, the SAS output signals for the respective axes should be mixed electrically in the same manner and proportion as are the mechanical controls. Then the SAS control inputs may be introduced at the input to the upper rotor control actuators.

6-3.2.5 Suppression of Structural and Rotor Mode Responses, Vibrations, or Gusts

Helicopters whose blades have an inplane natural frequency below the rotor speed consequently have high response to horizontal pylon forces at frequencies of rotor speed plus lag frequency ($\Omega = \omega_\zeta$). If any of the airframe modes of vibration, either flexible or rigid body, has a natural frequency that is near the aforementioned sum or difference, there will be a tendency for annoying, large, transient responses to gusts or sudden lateral control motions. In the case of resonance at the difference frequency ($\Omega = \omega_\zeta$), self-excited destructive oscillations can occur in the air or on the ground. In some cases, the SAS roll axis has coupled with the resonance and aggravated it. Thus, steps *shall* be taken either to eliminate SAS response to the mode or to make use of the SAS in suppressing it. Unless a special design effort is made, the total lag of SAS sensors, signal shaping, and actuators at the high frequencies of the transient oscillations is liable to shift phase response into a region that causes divergence rather than attenuation of the mode. It may be necessary to install separate sensors, filtered to respond only to the pertinent frequency and then phase-adjusted so that the final SAS output is at the proper phase. Both the SAS and the entire control system must respond to this frequency.

Ref. 21 discusses the theory that n-per-rev vibrations may be reduced considerably by suitably phased control inputs of the same frequency. This type of vibration suppression requires large amounts of power, and shortens the life of the control system considerably. If such suppression is to be used, the control system and SAS frequency responses must be approximately 15-20 Hz.

When dual SAS actuators are inserted in vertical linkage, the mass of the actuators may induce small control motions in response to vertical accelerations. In a specific case involving the collective pitch lever with friction lock disengaged, the weight of the pilot's arm coupled with the vertical motion of the helicopter produced a sustained oscillation. Mass balancing of the control linkage and/or the use of viscous dampers are methods of curing these oscillations.

Gust alleviation by means of control inputs responsive to gust-sensing instruments still is undeveloped. Closely allied to gust alleviation is airframe load limitation by control velocity restriction. However, the control requirements of the two tend to conflict because gust alleviation requires rapid control response. Because both programs have as their objective the reduction of airframe loads, the gust alleviation system should perform the duties of both. Ultimately, the SAS will include the functions of gust alleviation and load limiting in addition to flying-quality improvement.

6-3.3 SAS RELIABILITY

The expression for the failure rate P_x of the aggregate of components in a system, as shown in Ref. 22, is

$$P_x = n_1 P_1 + \left(\frac{n'_1}{2}\right) P_1^3 + n_2 P_2 + \left(\frac{n'_2}{2}\right) P_2^2 \text{ , hr}^{-1} \quad (6\text{-}1)$$

where

n_1 = number of nonredundant components having the failure rate P_1

n'_1 = number of redundant components having the failure rate P_1

n_2 = number of nonredundant components having the failure rate P_2

n'_2 = number of redundant components having the failure rate P_2

A mechanical SAS having fewer than a dozen parts, all of which have a very low failure rate, is ultra-reliable compared with an electro-hydraulic SAS with hundreds of parts. On the other hand, the weight penalty of providing redundancy in critical parts of the electrohydraulic system is not great. As seen in Eq. 6-1, the aggregate failure rate is highly dependent largely upon the amount of redundancy, given that the design insures that failure of one redundant component does not affect the operation of the other.

6-3.3.1 Safety

The designer must be cognizant of the influences of the inherent stability level of the helicopter and its SAS performance upon flight safety. Added stability margins can improve safety during night flying, or during limited-visibility situations caused by the presence of dust or snow clouds. During such operations, the provision of improved stability levels allows the pilot to concentrate less upon flying the helicopter and more upon other pilot duties.

The flight control system should be designed to allow the pilot to detect or diagnose a failure, disarm the failed system, and effect corrective action. This requirement may involve some form of online status-monitoring for the various control system elements.

Design compliance with the current Military Specifications does not preclude the possibility of inadvertent flight operations with one channel of a dual-channel augmentation system inoperative. In this type of failure, the difference in flying qualities is small enough to be undetectable by the pilot. Thus, the pilot may enter a flight condition in which failure of the remaining SAS channel cannot be corrected within a reasonable reaction time.

For certain critical situations, a need exists for automatic control activation. For example, electromechanical SAS links should revert automatically to a mechanical lock if hydraulic pressure is lost. This eliminates the possibility that a sloppy extensible link will create control difficulties while the pilot is attempting to cut off the failed system. Another example involves external cargo-handling or -towing operations, where it may be necessary for the load to release automatically if the applied moments exceed safe levels of controllability.

The results of the failure effect analysis (see par. 6-2.3 of this volume and Chapter 3, AMCP 706-203), including any supporting piloted simulations, should be reviewed and verified by flight test. These results then should be incorporated into flight handbooks in the form of warning notes or flight restrictions for various failure conditions.

6-3.3.2 SAS Failures

A discussion of SAS failure modes, limitation of authority, and time delay criteria may be found in par. 6-4.4, AMCP 706-201, and in par. 6-2.3 of this volume. Failures of rotor-mounted, gyroscopic SAS's are not discussed. These systems *shall* be designed so as to be at least as reliable as are the rotorcraft primary flight controls.

6-3.3.3 Fail-safe Principles

Fail-safe design, redundancy, and self-monitoring principles also are discussed in par. 6-4.3, AMCP 706-201.

6-3.3.4 Battle Damage, Vulnerability

Steps *shall* be taken to reduce SAS vulnerability in cases where loss of all stability augmentation would abort a mission. Duplication or triplication of actuators and hydraulic systems is a valid approach. SAS actuators should be designed so that it is possible to lock them in a centered position in the event of loss of hydraulic pressure. Duplication or triplication of hydraulic lines does not reduce vulnerability unless provision is made for automatically cutting off the oil supply to severed lines. Levers, bell cranks, and pushrods can be made large in size and of light-gage, low-stressed material in order to reduce vulnerability to small arms fire. Critical components not readily duplicated should be grouped and protected with armor (see par. 14-3).

6-3.4 COST

6-3.4.1 Development Cost

The cost of developing a new SAS generally is in proportion to the advance in the state-of-the-art represented by the development program. A conventional SAS for a conventional airframe can be obtained from off-the-shelf components, whereas a new concept for a rotor-located SAS, or for sensors based upon new technology, may require a large expenditure in order to bring it to production status. The new concept must promise a sufficient increase in cost-effectiveness in future production to compensate for the high cost of development.

6-3.4.2 Production Cost

SAS production cost can be reduced by adhering to the following:

1. Simplicity of design
2. Use of integrated and printed circuits
3. Commonality of circuit modules
4. Extensive use of value engineering principles.

Production cost increases may be expected with an increase in:

1. Number of system components
2. Quality or precision of components
3. Number of nonstandard parts
4. Number of parts that can be assembled incorrectly
5. Number and interdependency of adjustments to be made in final assembly
6. Number of parts that can be damaged easily in assembly
7. Degree of cleanliness required during assembly
8. Unrealistic requirements, or overemphasis on singular disciplines, such as in:
    a. Weight reduction
    b. Compactness
    c. Functional complexity
    d. Reliability
    e. Maintainability
    f. Structural integrity.

6-3.4.3 Maintenance Cost

A simple, mechanical SAS composed of infinite-life parts (as in the Bell stabilizer bar) requires maintenance only in the form of regular inspection and lubrication. In the event of battle damage or other failure, repairs can be performed by a qualified mechanic. An electrohydraulic SAS, on the other hand, may require the services of an instrument specialist, an electronic technician, and a qualified helicopter mechanic. Thus, the self-test circuits should be devised so as to indicate exactly which section is defective. Removal and replacement of plug-in modules represent field maintenance at lowest cost. Added to this cost, however, is the cost of maintaining adequate spares.

6-3.5 TECHNICAL DEVELOPMENT PLAN

For the development of a conventional (electrohydraulic) SAS, the plan outlined in MIL-C-18244 should be followed. In addition, the airframe and rotor dynamic and aerodynamic properties eventually should be included in the initial system analysis (MIL-C-18244) in order to show the possible existence of airframe cross-coupling and the need for anticross-coupling in the SAS, as well as to show the overall behavior of the SAS/airframe combination. Six degrees of freedom of the airframe, and quasinormal modes of the rotor (inplane as well as flapping motion of the blades), should be used. The resulting equations are used later in the simulation studies required by MIL-C-18244. The simulation not only will allow the pilot to evalute the system, but also will permit demonstration of the several types of failure of the SAS, and will indicate time delays permissible before starting corrective action.

In the development of unconventional SAS's, especially those involving modified rotor dynamics, several changes from the procedure in MIL-C-18244 are recommended. Unconventional systems require more initial system synthesis, or concept selection, than do conventional systems; and model studies should be undertaken as an aid. The models can range in complexity from simple mock-ups of gyro and linkage arrangements, through dynamically scaled wind tunnel models, to remote controlled flying models.

The paragraph of MIL-C-18244 dealing with model studies notes that experimental models may take the form of full-scale, engine-driven rotor and SAS assemblies, suitably mounted upon a truck bed for measurement and observation of dynamic behavior under forward-flight conditions. The maximum possible experience with and knowledge of the system should be gained before the start of testing of a man-carrying flight article.

Full-scale wind tunnel tests, although expensive, can be used to test the flight article progressively to conditions beyond the extremes of the projected flight envelope.

Further substantiation of the airworthiness of an unconventional SAS and rotor system can be obtained by operating an identical system on a tiedown test, where a given number of hours is required for each hour of actual flight testing.

The documentation and data required to establish the satisfactory fulfillment of the technical development plan are described in MIL-C-18244, substituting SAS for automatic flight control system (AFCS).

6-4 PILOT EFFORT

The helicopter designer must consider pilot effort, or control system loads, from two points of view. The first concern is the significance of control feel with regard to flying qualities. Pilots normally fly by the physical association of applied force and the maneuvering response of the aircraft. Therefore, the control feel in maneuvers plays an important role in the assessment of handling qualities. Stick positioning also is a fundamental characteristic, because it holds the helicopter in the selected trim attitude when the controls are released. MIL-H-8501 provides for stick position trim and hold by specifying breakout forces and force gradients.

The other design consideration is related to the structural integrity of the components. The components *shall* achieve specified factors of safety when subjected to loads due to pilot and copilot effort, artificial feel devices, power actuators, etc. MIL-S-8698 covers this aspect of pilot effort.

6-4.1 CRITERIA FOR POWER CONTROLS

Whenever the magnitude and linearity of control loads permit, direct mechanical control *shall* be used unless there is a valid requirement for power controls. Direct mechanical control is the simplest and most foolproof control system. However, power-operated systems may be required when the control system environment contains high control forces, feedback of vibratory forces, or mixing of control forces. The control system designer must verify a need for power-operated systems before adding their cost, weight, and complexity to the helicopter design.

6-4.1.1 Control Forces

Not all helicopters require power actuators. For example, on small, single-lifting-rotor vehicles, a system of weights may be installed in the antitorque-rotor controls. Centrifugal force acting upon the weights balances the pitch link loads, and the system is adjusted on the ground to compensate for the control forces in cruise. The pilot cannot trim the system in flight, and accepts the unbalanced forces in the pedals in hover and flight modes other than cruise. However, this is a small disadvantage in comparison with the simplicity of mechanical design. Also, it may be feasible to design a bungee spring that will counteract the steady download in collective pitch.

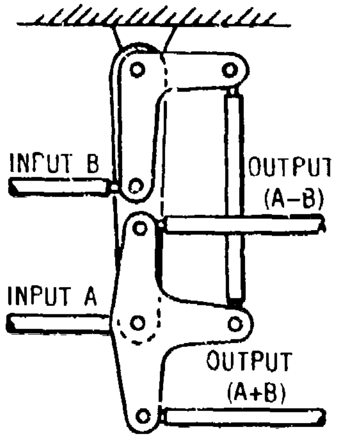
Medium- and heavy-lift helicopters generally require power actuators due to the magnitude of their pitch link loads. Pitch link loads are sensitive to rotor blade design parameters, both aerodynamic and inertial.

6-4.1.2 Vibration Feedback

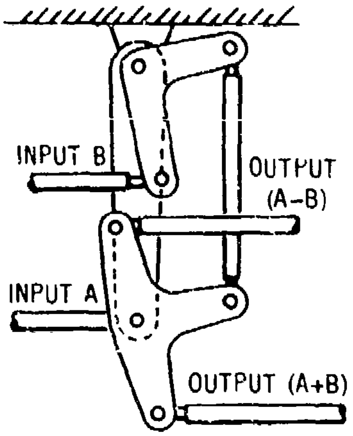
The control moment of a lifting rotor blade is a steady pitching moment with various alternating harmonic components superimposed. In the nonrotating control system, these components appear as *n*-per-rev forces due to the *n* number of pitch links passing over the attachment point where the nonrotating controls support the swashplate. The presence of these vibrations in the cyclic stick generally is intolerable to the pilot. Vibration absorbers can be used to reduce the amplitude of the vibration transmitted by means of a simple mechanical system.

6-4.1.3 Kinematic Effects

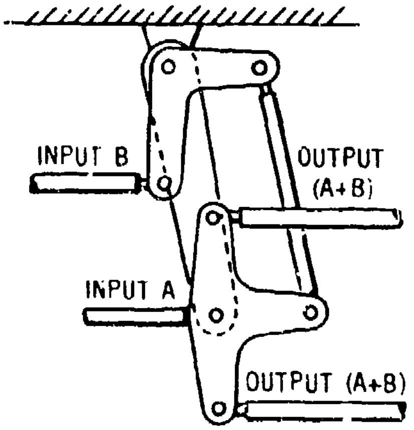
The generation of control forces and moments along and about the various axes of the helicopter is accomplished by combinations of collective and cyclic pitch on the rotor(s), as discussed in par. 3-3.3.1.3, AMCP 706-201. The motions of the cyclic stick and thrust lever (and, on some helicopters, the motion of the pedals) are transmitted through the swashplate to the rotor(s). In some installations, the control inputs are transmitted to control-mixing assemblies, where they are combined before reaching the swashplate. The degrees of rotor blade angle change in collective or cyclic pitch, per inch of control travel in the cockpit, are the dominant consideration in establishing the mechanical ratios in the mixing. Even if control forces are low and the vibratory components insignificant, there is a cross-talk of forces from one control axis to another because of the mixing. It is unlikely that the mixing assemblies, which contain components sized for stroke or travel relationships, will produce satisfactory force relationships. MIL-H-8501 sets limits upon control force cross-talk. Fig. 6-4 is a schematic diagram that illustrates the mechanical mixing of control signals. Fig. 6-5 shows a mechanical mixing assembly.



(A) IN TRIM POSITION



(B) POSITIVE VALUE OF INPUT B FROM TRIM



(C) POSITIVE VALUE OF INPUT A FROM TRIM

Figure 6-4. Control Mixing Schematic

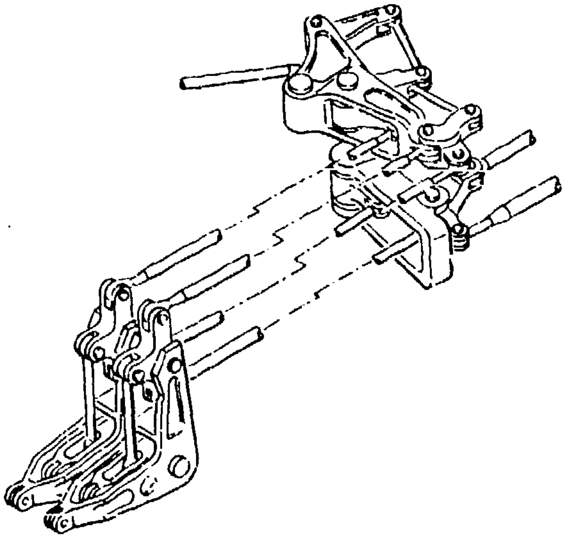


Figure 6-5. Mechanical Mixing Assembly

6-4.1.4 Control Stiffness

At high airspeeds and disk loadings, the onset of rotor stall flutter can limit the flight envelope. One of the many parameters to be considered is the compliance (stiffness) of the control system, particularly of the swashplate and its support. Hence, another justification for power actuators is based upon rotor performance. Fig. 6-6 illustrates the installation of power actuators for tandem helicopters.

6-4.2 HANDLING QUALITY SPECIFICATION

The handling quality requirements of MIL-H-8501 *shall* be specified in the detail specification if the rotorcraft under design is a pure helicopter. However, if the rotorcraft is a high-performance vehicle with fixed wings and alternate means of producing horizontal thrust, the detail specifications may specify requirements from both MIL-H-8501 and MIL-F-8785.

The requirements for control feed forces in normal helicopter operations are found in MIL-H-8501. The maximum and minimum breakouts and force gradients are defined, along with the limit forces. No gradient is specified in thrust, because a collective stick holding system — e.g., adjustable friction or a brake — generally is provided. There is no requirement for any gradient except that it be linear from trim to limit force.

MIL-H-8501 identifies the maximum control feel forces that are allowable after a failure in the power boost or power-operated system. The limit force in the failed mode is larger than, but of the same order of magnitude as, the limit load in the normal operating mode. Consequently, if hydraulic boost is required for normal operation, dual boost probably will be required for the failure mode.

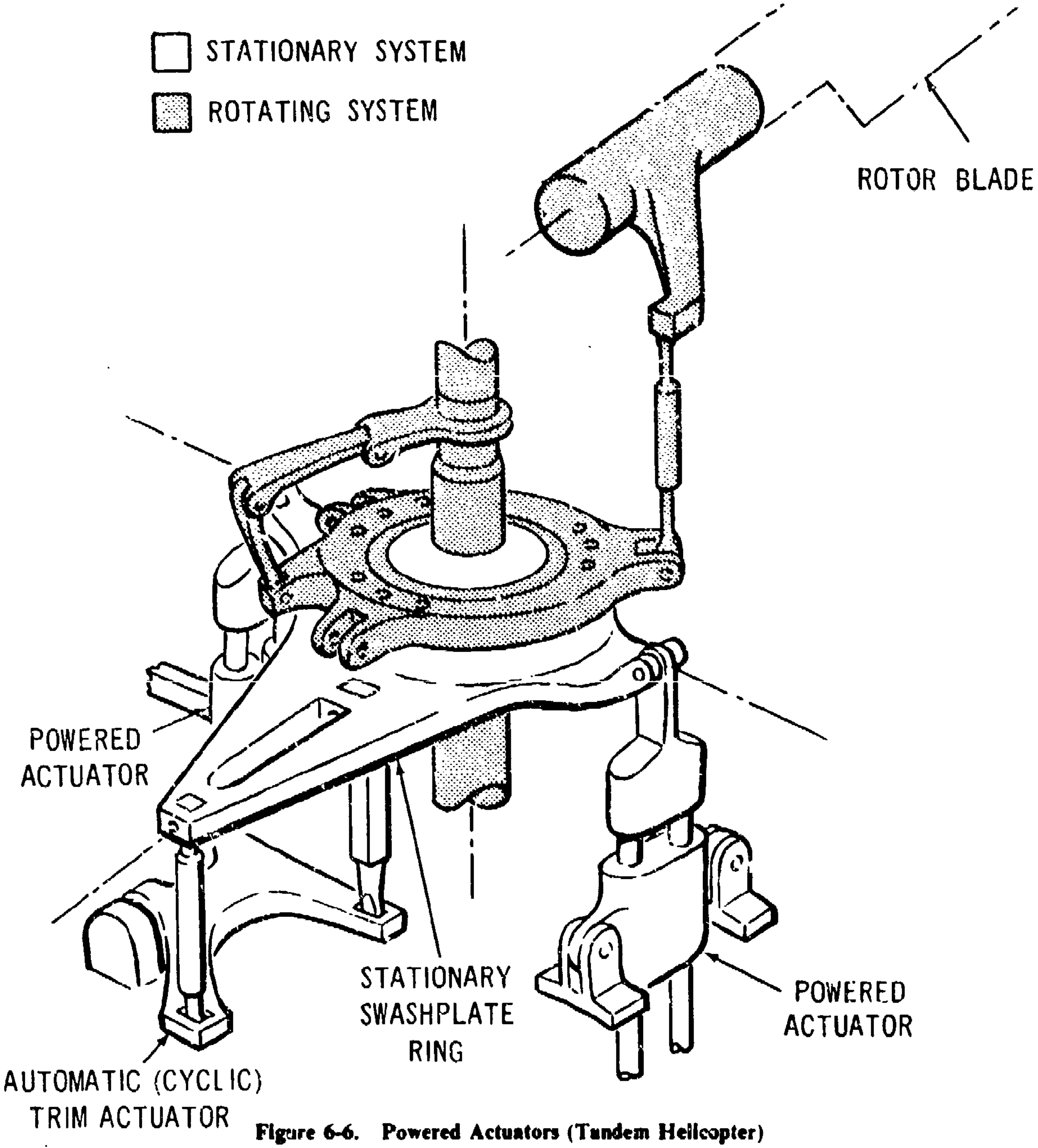

☐ STATIONARY SYSTEM
▨ ROTATING SYSTEM
ROTOR BLADE
POWERED ACTUATOR
STATIONARY SWASHPLATE RING
POWERED ACTUATOR
AUTOMATIC (CYCLIC) TRIM ACTUATOR

Figure 6-6. Powered Actuators (Tandem Helicopter)

6-4.3 HUMAN FACTORS

A prerequisite for an effective system control is a design definition of control augmentation needed as a function of total pilot workload. A force feel system may require no pilot control in order to maintain a trimmed flight condition.

6-4.3.1 Control Force Cues

The control force sytem should provide:

1. Trim position identification that will enable the pilot to feel an out-of-trim condition and to feel and identify trim when returning
2. Hold control in trim when the pilot is flying hands-off
3. An increased force cue to indicate increasing severity of maneuvering wherever it occurs. An increase in gradient with increasing airspeed is recommended. Care should be taken to avoid force cues introduced to the longitudinal control due to collective inputs. The optimal system would provide a constant relationship between longitudinal stick forces and resulting aircraft load factor during maneuvers.

The control force feel system provides an immediate and significant cue to the pilot, indicating the helicopter response to control command in any flight condition. This tightens the loop of pilot control and vehicle response, and enables the pilot to realize optimum control. A lesser performance leads to use of the feel system only as a trim hold device, and the pilot may prefer to turn it off under demanding control situations.

6-4.3.2 Developmental Test

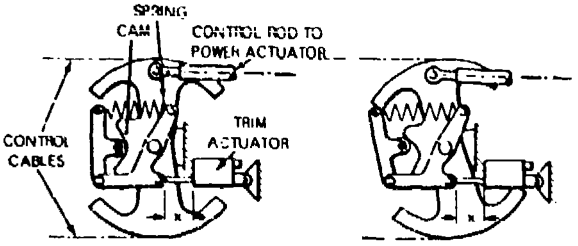
Moving-base flight simulation can be useful in developing the optimum control feel to suit the helicopter mission. In the moving-base simulator, pilots can draw upon past experience to identify desired force feel characteristics. Stick force proportional to rates of control displacement, helicopter angular rates, and to normal accelerations should be investigated so as to insure the design of an optimum system.

As the functions of the artificial feel system are increased, the complexity of the feel unit also increases. A design requirement for a specified linear gradient in the region of trim and a different linear gradient at greater excursions can result in a feel system with more than one spring. Furthermore, if a requirement exists for nonlinear force versus deflection characteristics, cams or linkages can be employed. Fig. 6-7 is a schematic diagram of an artificial feel system.

Flight safety at high speeds can be increased by reducing the occurrence of high rotor loads associated with excessive control displacement. A dynamic pressure-sensitive (q-sensitive) control force feel system produces minimum forces in hover and maximum force gradients in high-speed flight, where the sensitivity is greatest. This concept is an alternative to use of a control ratio changer in the primary control linkage. The q-feel system can be mechanical (with q-bellows), electrical, or electrohydraulic.

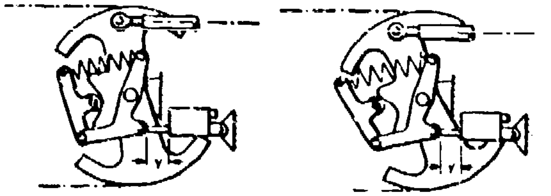
In fixed-wing aircraft, q-feel is provided for a slightly different purpose. The pilot flies the airplane by sensing, among other cues, normal acceleration and control stick forces. Response of an airplane is such that the change in normal acceleration per unit of elevator deflection increases with q. If the artificial stick force per unit of elevator deflection also is made to increase with q, then the relationship of stick force to normal acceleration can be made to approximate a constant value of stick force per g, regardless of flight speed.

Military Specifications useful in the detail design of the artificial feel system include MIL-H-8501, MIL-S-8698, MIL-F-8785, MIL-F-9490, and MIL-F-18372.



(A) TRIM CONDITION CONTROL CENTERED

CONTROL DISPLACED FROM TRIM



(B) TRIM CONDITION CONTROL DISPLACED

CONTROL DISPLACED FROM TRIM

Figure 6-7. Artificial Feel and Trim Schematic

6-4.4 AUTOMATIC CONTROL INTERFACES

Inner loop stabilization signals are summed with the pilot's commands through electro/hydromechanical actuators in series with the pilot's controls. It is important that the high-frequency, small-amplitude stabilization signals do not reach the cyclic stick in the form of forces or deflections. Thus, there is a need for a "no-back" (a device to prevent the feedback of forces) located upstream of the SAS series actuator. A stick boost also will perform this function. In addition, if the helicopter is to be equipped with an autopilot that introduces signals through actuators that move the cockpit controls in parallel with the pilot, there is a requirement for compatibility among the inertia, compliance, and damping of the primary mechanical controls and of the parallel actuator.

6-4.5 VULNERABILITY

The close support of ground operations exposes the U S Army's observation, cargo, utility, and armed helicopters to small arms and automatic weapons fire. The unprotected, single-channel flight control system is vulnerable over its entire length. There are a number of ways to reduce this vulnerability.

One method is to make the components so rugged that they can sustain a hit without losing their structural integrity. However, this is seldom feasible, especially when space and weight must be controlled rigidly.

Certain areas, such as the cockpit, will be protected with armor plate in order to safeguard the crew. The same armor can be used to shield the mechanical controls. However, it may not be feasible to run armor plate all the way to the swashplate.

A redundant control system not only helps to solve the vulnerability problem but also improves flight safety reliability. To be effective, redundant channels must be separated physically. Consideration must be given to single-channel jams and disconnects, to adequacy of control if the remaining channel goes to half gain, and to the question of whether the configuration should be active-active or active-standby.

6-4.6 RELIABILITY

The overall reliability of a flight control system depends upon the reliability of the individual components and upon their arrangement, which may be either in series or parallel. If the helicopter system specification prescribes a minimum acceptable value for flight safety reliability, this value may be so high as to require dual mechanical controls. The detail designer first must establish the single success path; then, if system reliability is inadequate (a value less than required by the helicopter system specification), he must add redundancy, beginning with the least reliable components.

The reliability of a component is a function of the historical mean time between failures (MTBF) of that component. When historical failure rates are used, similarity between the environment under which the data were taken and the environment in which the new system will perform must be assured, or appropriate adjustment of the projected rates must be made.

Another rationale for duplication is based upon failure considerations. A power actuator may provide the required reliability; but if a failure of the actuator is catastrophic, a redundant actuator is required. Further discussion of this subject is contained in pars. 6-5.2 and 9-2.

6-5 MECHANISMS

6-5.1 ROTATING SYSTEMS

The rotating controls in the main rotor sytem normally include the rotating swashplate, the pitch links, and the drive scissors. These components are shown in a typical arrangement in Fig. 6-8. Functionally, the rotating swashplate translates along the rotor shaft and tilts in any plane as dictated by control inputs. The swashplate translation and tilt are transferred to the blade pitch horn through the pitch links and, thereby, control the main rotor thrust vector. The drive scissors (a) fix the positions of the rotating controls relative to the rotor shaft and rotor blades, and (b) provide the load path for the conversion of drive shaft torque into the tangential force required to induce rotational motion in the rotating controls.

6-5.1.1 Design Factors

Structurally, the rotating system *shall* be designed to withstand the alternating (fatigue) flight loads introduced by rotor blade torsional moments and the maximum loads introduced by severe flight mameuvers or during ground operations. The fatigue loads are periodic, and alternate primarily on the basis of once-per-rotor-revolution. In other words, each time the rotor blades complete one revolution, the pitch link load completes one stress cycle. Therefore, a high-cycle fatigue evaluation is required. The primary loads are discussed in pars. 4-9 and 4-10, AMCP 706-201, and the fatigue evaluation is discussed in par. 4-11, AMCP 706-201.

In addition to the primary flight loads, special consideration *shall* be given to secondary loads. Failure to evaluate secondary loads properly may lead to service problems. Among the secondary loads that *shall* be considered are frictional moments in rods and bearings, and bending moments created by centrifugal force. A typical pitch link rod end, with a self-aligning bearing, is shown in Fig. 6-9.

Bearing motions of ±6 deg are not uncommon during each rotor revolution. The normal force (pitch link load) times the coefficient of friction produces a

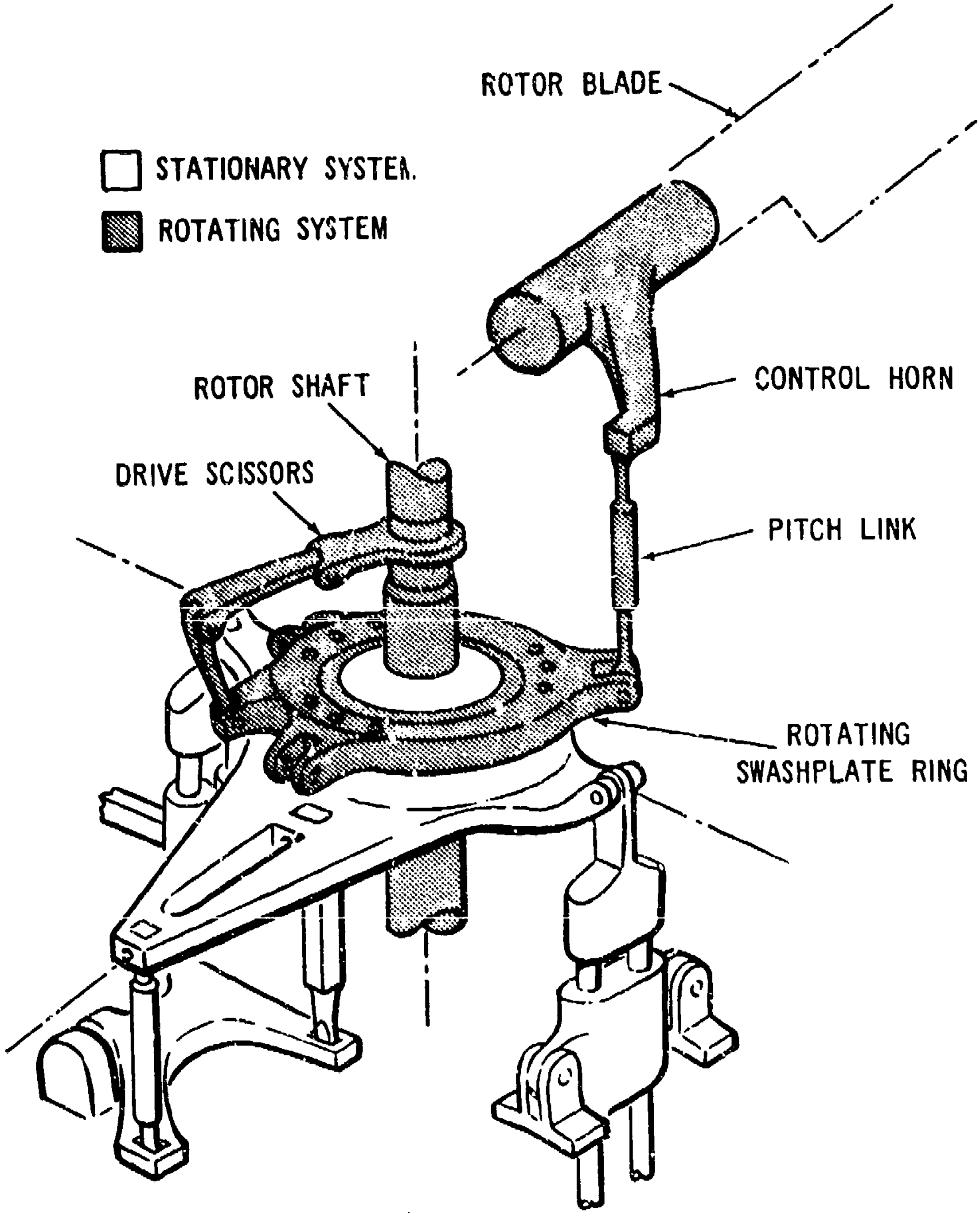

ROTOR BLADE
☐ STATIONARY SYSTEM
▨ ROTATING SYSTEM
ROTOR SHAFT
CONTROL HORN
DRIVE SCISSORS
PITCH LINK
ROTATING SWASHPLATE RING

Figure 6-8. Rotating Controls

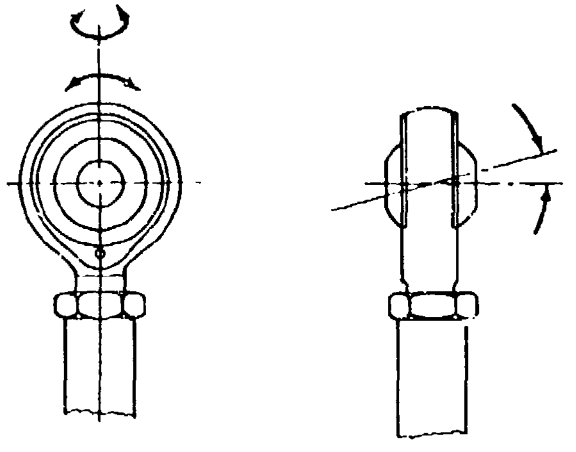
frictional force upon the spherical surface of the bearing. Rod end motion, in the presence of frictional forces, induces pitch link bending moments. If the rod end and the bearing are of different material, differential expansion due to temperature changes will alter the frictional moments. The bending stresses that result may be significant and should be evaluated at the same time as the primary loads.

A pitch link bending moment also will result from the centrifugal force acting upon the weight of the pitch link. This inertia force will produce a transverse deflection. Although this deflection may be small, its effect upon pitch link strength *shall* be evaluated from a beam-column standpoint (see Fig. 6-10).

The ultimate and limit strengths *shall* be sufficient for the maximum static loads resulting from both flight and ground operations, including loads during the blade folding if applicable. The sources of these loads are discussed in pars. 4-6, 4-7, and 4-8, AMCP 706-201.

Compensation for tolerance buildup in the rotating control system and the rotor blade usually is provided by pitch link length adjustment. Threaded rod ends are common. The adjustment provision requires close design attention. Positive locking features *shall* be provided in order to prevent any length change after system rigging. Such changes could be induced either by inflight vibrations or during routine maintenance.

The pitch links *shall* include inspection provisions so as to assure that sufficient thread engagement is present to provide structural integrity. One method is to provide an inspection hole. Fig. 6-11 illustrates a turnbuckle type of adjustment, showing the inspection holes and installed lockwire. The jam nut



NOTE: ARROWS INDICATE BEARING MOTIONS

Figure 6-9. Typical Pitch Link Rod End

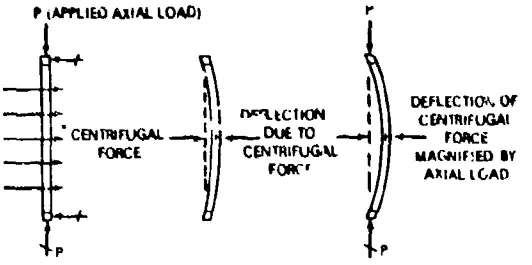

P (APPLIED AXIAL LOAD)
CENTRIFUGAL FORCE
DEFLECTION DUE TO CENTRIFUGAL FORCE
P
DEFLECTION OF CENTRIFUGAL FORCE MAGNIFIED BY AXIAL LOAD
P
P

Figure 6-10. Centrifugal Force Deflections

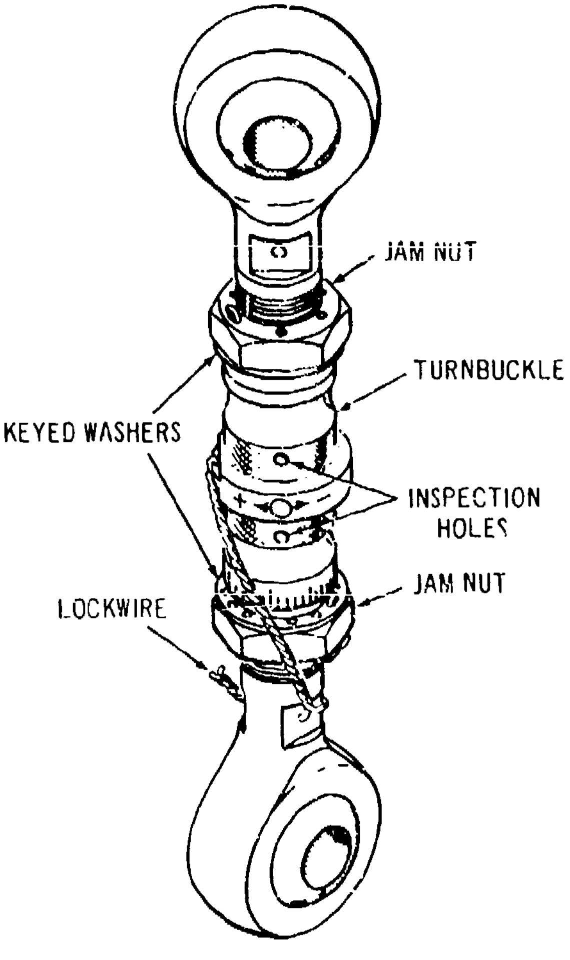


Figure 6-11. Pitch Link Adjustment Provisions

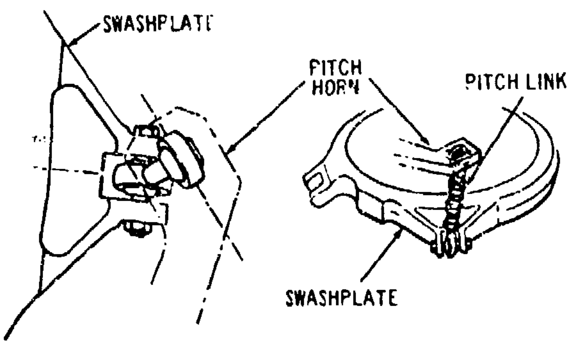


Figure 6-12. Relative Pitch Link Rod End Position

forces a keyed washer against the turnbuckle, and is attached to the washer with a cotter pin. The nut/washer combination prevents turnbuckle motion that would shorten the rod, and the lockwire prevents any rod extension. The relative position of the rod ends is maintained by an internal slot arrangement. Proper relative rod end position is important in order to assure rod end clearance within the swashplate and pitch horn lugs (Fig. 6-12).

In addition to the rotation relative to the stationary controls, relative motion occurs within the rotating controls. The drive scissors has two horizontal pivots and one universal joint to accommodate the vertical and tilting motions of the swashplate. Pitch link rod ends have self-aligning bearings to accommodate the small angular changes between the swashplate lugs and the pitch horn caused by swashplate motions. Rod-end-to-lug clearances *shall* be provided in order to prevent contact during these motions.

The bearings will wear during service exposure. Although wear may reduce frictional moments, the vibratory levels tend to increase as a result of the looseness caused by wear, and bearing replacement becomes necessary. Ease of bearing replacement is a design consideration. Bearing replacement times are established by TBO test programs and by service experience.

6-5.1.2 Test Results

As detail design progresses, it becomes possible to replace preliminary design estimates with quantitative information gained during bench and flight testing. Chapters 7, 8, and 9, AMCP 706-203, define the procedures, tests, and demonstrations involved in demonstrating proof of compliance with the design requirements. Chapter 4, AMCP 706-201, describes the procedures for fatigue-life determination. The discussion that follows defines methods for insuring that bench and flight test data are sufficiently timely and complete to be used to best advantage in the detail design of the rotating system.

6-5.1.2.1 Bench Tests

Although the fatigue analysis of the rotating controls may be thorough, the effects of the complex stress concentrations introduced by locking features and threaded connections, along with other uncertainties such as fretting, preclude an acceptable analytical fatigue strength determination. Therefore, it is essential that bench testing to determine the fatigue strength of components be coordinated properly with other elements of the design process, and that the fatigue test requirements be based upon representative — or, at least, conservative — service conditions. The factors discussed in pars. 6-5.1.2.1.1 through 6-5.1.2.1.4 influence the establishment of the test requirements. Component fatigue test requirements are discussed in detail in par. 7-4, AMCP 706-203.

6-5.1.2.1.1 Test Loads

Although it is desirable technically to duplicate all flight loads on the bench, this is not always an economic or physical possibility. When flight loads will not be the basis for bench test loading, an analytical assessment must be made in order to determine which of the secondary loads is significant. In the rotating control system rod end, frictional moments are usually significant while centrifugal forces are insignificant.

Steady loads in rotating control system components generally are low in comparison to the alternating loads. Consequently, the load range is through zero, thus increasing the relative motion of components and the possibility of fretting. Test loads should be programmed so as to insure loading through zero.

If the moment induced by rod end friction is significant, it must be included in the test. This secondary load must be phased properly with the primary load. The effects of end moments may be induced artifically by applying eccentric axial loads. Another method is to use stiff bearings and to induce bearing motion during the test. In either test, it may be necessary to evalute temperature extremes.

6-5-1.2.1.2 Instrumentation

The correlation of flight loads to bench test measurements is a primary consideration. Unless the load distribution upon the part under test can be ascertained readily from applied loads, bench test specimens should be instrumented and calibrated. Where a complex bending moment exists, a component should be instrumented with sufficient bridges to determine that distribution. The location and type of instrumentation *shall* be the same as is employed in the flight load survey.

6-5.1.2.1.3 Quantity and Selection of Specimens

A minimum of six specimens of each component is required for definition of an *S-N* curve. Where tolerance is a significant factor, specimens should be selected from those at the adverse end of the tolerance band. Dimensional tolerances of critical parts generally are tightly controlled; therefore, special selection on the basis of dimensions usually is not required. However, selectivity on the basis of more highly variable quantities, such as rod end friction, is required.

6-5.1.2.1.4 Interpretation of Data

If all significant secondary loads are accounted for during bench testing, an *S-N* curve and endurance limit can be established as a function of the primary alternating load. The endurance limit must be based upon a statistical reduction of test data so as to account for scatter. In some cases, a further reduction factor may be applied analytically in order to account for a secondary effect not included in the original test program. The preparation of an *S-N* curve from fatigue test data for a limited number of specimens is described in par. 4-11, AMCP 706-201.

6-5.1.2.2 Flight Tests

The characteristics of the alternating loading on fatigue-critical components are determined by a flight load survey. A statistically significant data sample should be obtained for each flight condition representative of helicopter usage, i.e., for each condition within the mission profile. The requirements for a flight load survey are described in detail in par. 8-2, AMCP 706-203.

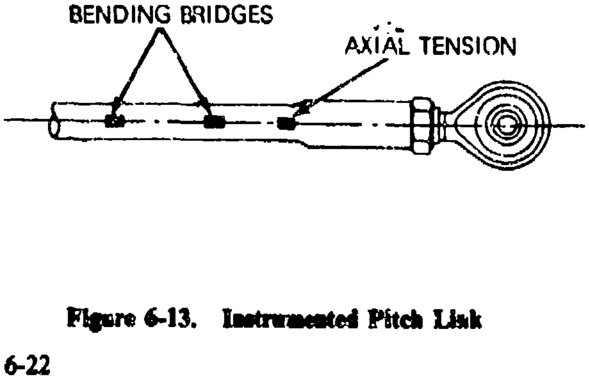


Figure 6-13. Instrumented Pitch Link

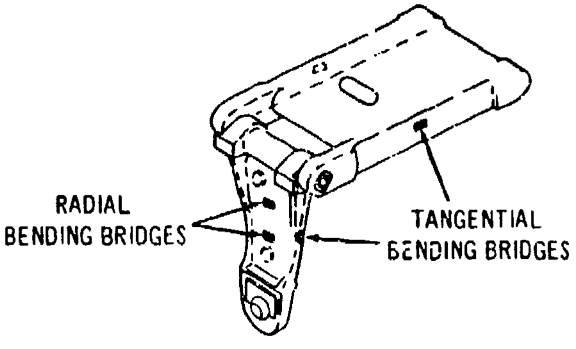


Figure 6-14. Instrumented Drive Scissors

6-5.1.2.2.1 Required Instrumentation

Pitch link axial load and drive scissors bending moment in the plane of rotation are the primary loads in the rotating control system, and must be measured. Secondary loads requiring measurement are pitch link bending and drive scissor radial bending. At least two bending bridges are required in order to determine the distribution of each of the moments.

Typical instrumentation of a pitch link and a drive scissors is shown in Figs. 6-13 and 6-14. As a rule, each of the pitch links is instrumented with a tension gage in order to determine whether or not there are any differences between the loads from the individual blades.

6-5.1.2.2.2 Flight Conditions

Flight loads *shall* be obtained for all mission profile conditions at the most adverse altitude(s) and helicopter configuration(s) within the anticipated operating regime (see par. 8-2, AMCP 706-203). Loads in the rotating control system generally are noncritical in unstalled flight. They do, however, react to the onset of moment stall and, therefore, usually establish the structural envelope for stalled conditions. Consequently, as a minimum, control loads should be measured at the conditions most conducive to stall. These are:

1. Maximum gross weight
2. Most extreme CG
3. Maximum altitude
4. Minimum rpm
5. High load factor.

6-5.2 NONROTATING SYSTEM

The location of push rods and cables must be determined early in the design of a helicopter, prior to the selection and location of other large equipment, so that it will not be necessary to route the control sytem around this equipment.

The control runs must be coordinated with each other and with the entire airframe in scaled layouts. Direct, straight-line routing improves the control system response, reduces friction and weight, and increases reliability. Other factors that must be considered during control system layout are vulnerability to small arms and automatic weapon fire, jamming by foreign objects, rigidity, strength, accessibility for inspection and service, and techniques to prevent incorrect assembly. All pertinent information should be shown clearly on the small-scale layouts in order to verify the design feasibility.

As various design options are developed, the alternative configurations should be evaluated by means of a trade-off study. The parameters in the study may include — but are not limited to — performance, reliability, cost, safety, weight, use of standard parts, logistics, maintainability, and vulnerability. In such a study, weighting factors may be assigned to the various parameters. However, because the weighting factors affect the outcome of the study, they must be assigned judiciously.

The results of the trade-off study upon the selected configuration should be evaluated carefully during a design review. The purpose of the review is to insure that the selected configurations and the applicable specifications, mock-up, and test requirements are in accordance with objectives established during the preliminary design, and that program and contract requirements for performance, reliability, cost, safety, maintainability, standardization, and ease of inspection are or will be met.

6-5.2.1 Pilot's Controls to Power Actuator

A comprehensive discussion of design standards and requirements for that portion of the helicopter flight control system between the pilot's controls and the power actuator is found in MIL-F-9490 and MIL-F-18372. Although there are minor conflicts between these specifications, the helicopter system specification generally will define the extent of their applicability.

The loads in the nonrotating control system consist of those loads present in the system at all times, operational loads due to pilot forces, and any flight loads fed back from the rotor blades. Constant loads include system preloads or rigging loads, and loads due to component weight.

The values of the control system loads required for design are given in par. 4-9.8.3, AMCP 706-201. Included are the pilot effort loads applied at each control input, together with their reaction points. Criteria are provided for dual control systems, duplicate systems, distribution of loads within a system, and power control sytems.

In addition to the requirements of the Military Specification and of Chapter 4, AMCP 706-201, the following list of requirements and general design practices are applicable to the design of helicopter control systems:

1. Push-pull rods, bellcranks, and levers:

a. Each bolt, screw, nut, pin, or other fastener whose loss could jeopardize the safe operation of the helicopter *shall* incorporate two separate locking devices. The fastener and its locking devices should not be affected adversely by environmental conditions.

b. Impedance bolts *shall* be used where loss of a bolt can cause a catastrophic failure.

c. Rod assemblies should be designed with only one adjustable end fitting. The adjustable rod end check nut should be lock-wired where the rods are subject to vibration or to high-frequency load reversals.

d. The natural frequencies of push-pull rods should be checked against the forcing frequencies of the rotor(s) in order to assure that the system is free from resonance.

e. The largest diameter and longest tube consistent with weight and strength considerations should be used in order to provide reduced vulnerability.

f. Maximum clearance in the clevis joints of the push-pull tubes *shall* be provided so as to allow for overtravel when the controls are disconnected.

2. Torque tubes and universals:

a. The natural frequencies of torque tubes should be checked against the forcing frequencies of the rotor(s) to assure that the system is free from resonance.

b. Universal joints should be used where misalignment exists between torque tubes.

c. A double universal joint assembly may be used to obtain constant angular velocity, provided that:

(1) The driving yoke of one of the joints is 90 deg offset from the driving yoke of the other.

(2) Each joint is operated at the same angle.

(3) All shafts are in the same plane.

3. Cables, pulleys, and quadrants:

a. Cables and pulleys should be used only when distinct advantages can be shown over a system using push-pull rods.

b. Cables tend to twist over each pulley. If the twist from one pulley rides onto another pulley, cable wear will result. Pulleys thus should be spaced far enough apart so that no segment of cable runs over more than one pulley during full travel.

c. Unsupported spans of 150-200 in. have operated satisfactorily. However, cable idler pulleys in long straight runs minimize friction over fairleads and grommets.

d. Close spacing of cables *shall* be avoided. Cables should not pass within 3.0 in. of structure, equipment, or other cables.

e. The angle between the centerline of the cable and the plane of the pulley should not exceed 0.5 deg.

f. To the maximum extent practicable, cables should run along, or be as close as possible to, the neutral axis of the airframe structure.

g. Friction in a cable system should be minimized by:

(1) Using a minimum number of pulleys

(2) Using the largest practicable pulley size

(3) Using the smallest cable diameter consistent with strength and rigidity requirements

(4) Designing for the smallest practicable wrap angle consistent with maintenance of good cable contact and pulley rotation.

h. The effect of changes in temperature can be a serious problem in control cable systems, due to the difference between the coefficients of thermal expansion of the aluminum airframe and the steel control cable. The problem is less severe in pulleyless cable systems because higher rigging loads are permissible.

i. Tension regulators may be installed in quadrant and pulley assemblies in order to allow for expansion and contraction of the cables without appreciable variation in rigging load.

j. Nylon-covered cables can increase cable life by damping high-frequency vibrations.

k. Cable guards *shall* be used at points of tangency of the cable to the pulley.

4. Chains. The use of chains *shall* be subject to the approval of the procuring activity.

In spite of the inherent advantages, applications of fly-by-wire techniques to helicopter control have been slow to materialize. The substitution of fly-by-wire electrical signaling systems for conventional linkages between cockpit and swashplate has a number of potential benefits. However, before any fly-by-wire primary flight control system is accepted for production, a high level of reliability must be assured.

The advantages of an electrical control system will depend upon the type and size of the helicopter in which the system is installed. For example, some of the benefits to be expected in large, heavy-lift helicopters are improved flight safety reliability and reduced vulnerability, higher fidelity of control, and reduced weight. The characteristics and capabilities of a true fly-by-wire system are compared with various alternate heavy-lift helicopter control systems in Refs. 23 and 24.

6-5.2.2 Power Actuator to the Swashplate

Design requirements and standards for that portion of the helicopter flight control system between the power actuator and the nonrotating swashplate are defined by MIL-F-9490 and MIL-F-18372.

In addition to these requirements, important considerations include fail-safe design, structural compliance, and control system dynamics.

Various fail-safe approaches are:

1. Stand-by design. Two equal-strength load paths are provided. The secondary load path is isolated until primary failure occurs. The primary load path is visible during helicopter inspections. This approach requires that each load path be designed for infinite life to insure that the components of the secondary will last between overhauls if the failure of the primary path is not detected.

2. Load-sharing design. The component is made up of two or more sections or laminations that are joined mechanically. If one element fails, the remaining elements have full load-carrying capability. If the elements are bonded, the bonding agent *shall* prevent a crack from propagating across the section. If the component is not bonded, other antifretting barriers must be employed. A section of the component consists of the maximum practicable number of laminations.

3. Crack-detection design. Design techniques could include: pressure drop, oil or fluid leak, electrical detectors, and highly penetrating dye.

Inadequate structural stiffness can affect the control system in several ways. Deflections of the airframe can introduce inputs to the control system, an effect that is minimized by routing the controls close to the neutral axis of the airframe. When the support structure is not stiff, the power actuator also can deflect under load. If the control system output is sensed by the compliant structure, a limit cycle instability can occur and ultimately may destroy the helicopter if it is allowed to proceed unchecked. The designer *shall* introduce compensating linkage or sufficient structural stiffness in order to assure that the control system is insensitive to deflections.

The primary flight controls are part of the complex servo system that determines the transient and frequency responses of the helicopter. In this system, the mechanical controls play a small but significant role. In addition to the mechanical controls, performance of the servo system depends upon:

1. Rotor dynamics and aerodynamics
2. Inherent helicopter stability

3. Power actuator dynamics
4. Automatic flight controls:
   (a) Stability augmentation system
   (b) Outer loop stabilization
   (c) Automatic trim systems.
5. Pilot in the feedback loop.

Frequently, the mechanical controls and the power actuator are analyzed together as a subsystem. Characteristics of the mechanical controls that have an effect upon the responses of the servo system are:

1. Inertia and balance of control system components
2. Damping at control stick, actuator valve, and/or control surface
3. Friction at control system joints
4. Looseness of control system joints.

Friction can cause control system hysteresis, which prevents the control stick from returning to the trim position once it is displaced. The provision of positive centering requires a preload force larger than the value of the friction force. However, excessive control breakout force around the neutral or trim position is undesirable because it results in a tendency for the pilot to overcontrol the helicopter.

Looseness, the result of excessive buildup of tolerances and wear at bearings and joints, causes backlash in the control system. The effects of backlash can range from sloppy and unsatisfactory control characteristics to pilot-induced oscillations.

Dynamic analysis conducted with high-speed digital or analog computers not only identifies required characteristics of the automatic systems, but also identifies design requirements for the mechanical system, such as:

1. Balancing of certain control components, particularly the cyclic stick and collective lever
2. Stiffening of control elements and backup structure
3. Installation of antibacklash springs to eliminate looseness
4. Additional damping at stick, actuator valves, and/or control surfaces
5. Establishment of the allowable upper limit for control system friction.

Major accidents can result from improper or inadequate maintenance of flight control systems. Specific design guidelines for maintainability of control systems include:

1. Understand the skill level of the maintenance personnel, their operating environment, and the type of errors they are likely to make
2. Replace routine maintenance with on-condition maintenance accompanied by adequate failure warning
3. Incorporate physical barriers against incorrect assembly and installation of generally similar parts. The design *shall* insure that the omission of critical fasteners either is obvious during ground runup or cannot result in catastrophic failure in flight.
4. Realize that the same maintenance error may be repeated in all paths of a redundant system
5. Human factors engineering should be applied to design for maintainability to minimize human error.

6-5.3 TRIM SYSTEMS

The force trim system is provided in order to allow the pilot to reduce the control force to zero when the helicopter is trimmed along a stabilized flight path.

MIL-H-8501 requires that, for all conditions and speeds specified, it *shall* be possible in steady-state flight to trim steady longitudinal, lateral, and directional control forces to zero. At all trim conditions, the controls *shall* exhibit positive self-centering characteristics. Stick "jump" when trim is actuated is undesirable.

Several types of control force trim systems are described in the paragraphs that follow.

6-5.3.1 Disconnect Trim

The handling quality requirements can be satisfied by a preloading spring in combination with a magnetic brake. The principal advantage of this method is simplicity. However, the magnitude of the spring force and the kinematics of the system may combine to produce an objectionable kick when the magnetic brake is released. A damper in parallel with the magnetic brake will reduce this undesirable characteristic; whenever the trim button is held, the magnetic brake is disengaged from the control linkage, and, therefore, the trim, or force feel, springs also are disengaged. In a well-designed system, this can be an advantage, as it simplifies the input of small control displacements such as those required for precise hovering control.

6-5.3.2 Continuous Trim

An alternative to the on-off system is a system in which the trim is continuous. Upon activation of the trim switch, the control forces are trimmed slowly to zero. In this system, the magnetic brakes are replaced by electromechanical actuators. This type of trim can be provided readily in helicopters that are equipped with parallel actuators for outer loop stabilization. However, the two-axis (Chinese hat) electrical trim switch on the cyclic stick grip can activate only longitudinal and lateral trim; the directional trim switch must be located elsewhere.

To avoid the trim switch limitation, it is possible to design the trim circuit so that when the pilot depresses the trim button, the trim force for any control axis that is out of trim is trimmed to zero force. If more than one control axis is out of trim, all axes would be trimmed simultaneously to zero. A detent arrangement disengages the actuator when the zero spring force has been reached.

Both the rate at which the actuator operates and the authority, or maximum value, of the feel force provided by the continuous trim system are significant in determining the acceptability of the system No specific requirements are given by MIL-H-8501.

6-5.3.3. Parallel and Series Trim

Artificial feel forces may be trimmed to zero with both parallel and series trim. Parallel trim involves the repositioning of the neutral (zero force) point of the unit; thus, a new trim position for the entire control system is created. Both the magnetic brake and the continuous trim systems are parallel. Fig. 6-7 shows the parallel trim actuator in the schematic of an artificial feel system.

Series trim involves the insertion of an extendable link in the control system between the feel unit and the power actuator, and produces control surface motion with no stick motion.

Helicopters with fixed wings and alternate means of producing thrust may require trimming of unboosted aerodynamic control surfaces. Elevators, ailerons, and rudders may be trimmed by driving a geared trim tab through electrical trim motors with mechanical override provided. In addition, the pitch control may be trimmed by adjusting the incidence angle of the stabilizer and the elevator.

Trim is provided in order to balance, or reduce to zero, the steady-state control forces that arise from changes in helicopter configuration and flight conditions. The vehicle is flown normally by the primary flight controls from one flight condition to another; after allowing time for stabilization, it is trimmed to fly hands-off. To use the trim control to change from one flight condition to another is a misuse of the trim system. Trimming the vehicle into maneuvers results in loss of the capability to return to the normal flight attitude if the controls are released, as well as in loss of the feel for the particular maneuver being accomplished.

Constant use of the trim system when it is not required will lower the system MTBF. Failures may occur at extremes of control travel or in an uncomfortable helicopter attitude. An inoperative trim system can create unusual stick forces. Runaway parallel trim produces abnormal control forces, and runaway series trim causes a change in both the control position and the force necessary to maintain trimmed flight. The magnitudes depend upon the authority of the trim system.

In addition to trimming steady-state control forces to zero, a trim system may be used for trimming of aerodynamic forces and moments (series trim). Trim at the incidence of the horizontal stabilizer may be used in single-rotor helicopters, trim of the longitudinal cyclic pitch in tandem-rotor helicopter, and trim of the wing incidence angle in rotorcraft equipped with wings. This aerodynamic trim may be programmed automatically or operated manually Fig. 6-6 shows the installation of an automatic cyclic pitch trim actuator

6-6 SYSTEM DEVELOPMENT

6-6.1 GENERAL

Design and development of the helicopter flight control system should include several forms of testing. Objectives of this testing are to improve the validity and accuracy of analytical mathematical models, to insure proper consideration of the human pilot as a controller, and to permit refined development under a full-scale environment.

6-6.2 MATHEMATICAL MODEL IMPROVEMENT

In general, the mathematical model used for analyzing the helicopter during the preliminary design phase considers first-order effects or characteristics and incorporates data or approximations based upon prior experience with similar systems, subsystems, or devices. A margin of tolerance, again based upon available experience, is applied to these results prior to their assessment with respect to specification compliance. During detail design, it is necessary to improve the accuracy and validity of the mathematical model in order to insure credible and cost-effective compliance with specifications. In this regard, wind tunnel and hardware bench tests are proper tools for engineering application.

6-6.2.1 Wind Tunnel Test

The aerodynamic forces and moments of the total helicopter and its components parts, together with their derivatives with respect to many of the variables required for stability and control studies (e.g., attitude, control deflection, and rotor thrust), can be obtained in the wind tunnel. Primary interest should focus upon the static stability derivatives M_α, N_β, and L_β; the damping derivatives M_q, N_r, L_p, and Z_w; the control derivatives M_δ, L_δ, N_δ, and Z_δ; the speed stability derivative M_u; and the flow field characteristics affecting the helicopter. These data should be gathered over the full range of helicopter configurations and for the complete flight envelope. Wind tunnel testing also can provide important information on interaxis cross-coupling effects.

Measuring the damping derivatives directly in the wind tunnel generally requires complex procedures and techniques. However, the time constants associated with the changing aerodynamic forces and moments often are small compared with vehicle response, allowing steady-state wind tunnel results to be used with sufficient accuracy in dynamic analyses. For example, wind tunnel measurements of the horizontal tail lift characteristics and the downwash existing at the tail location can be used to calculate the pitch-damping contribution of that surface, knowing the tail arm.

6-6.2.2 Hardware Bench Tests

As hardware components of the flight control system become available, they should be subjected to laboratory bench tests in order to describe accurately their performance characteristics. These results then may be used to update the data previously employed in the mathematical models. The bench testing can define items such as stiffness, frequency response, threshold level, and rate limits.

6-6.3 GROUND-BASED PILOTED FLIGHT SIMULATION

The helicopter flight control system design not only should result in compliance with the minimum requirements of the handling quality specification, but also should maximize the handling quality potential from the pilot's viewpoint with regard to the mission requirements. Therefore, detail design of complex flight control systems requires that the human pilot be inserted into the simulation by using either fixed-base or moving-base piloted flight simulators. Fixed-base simulation, however, does not provide the pilot with a realistic environment for his body sensors, and forces him to respond unrealistically or, sometimes, falsely. The correct environment consists of a proper representation of the helicopter equations of motion and a proper pilot environment, including vision, sound, touch or feel, and motion.

Use of a high-fidelity, moving-base, piloted flight simulator is a cost-effective approach to the engineering development of helicopters. The flight simulator is an engineering tool that can provide high confidence in design decision-making regarding new systems. Application of moving-base flight simulation early in the helicopter definition and developmental cycle identifies pitfalls and potential problems, and provides a means of generating good flight vehicle characteristics.

During the past few years, heavy emphasis has been placed upon efforts to identify and understand the fundamental technological and physiological factors involved in the man/machine interface, particularly with regard to nonhelicopter rotorcraft systems, where the basic vehicle configuration often is dictated by the pilot's control ability. These efforts already have produced a quantum increase in the knowledge of control system theory and criteria, stability, human factors, handling and flying qualities, and hardware design. Because nonhelicopter rotorcraft are advanced systems, their ability to comply with specifications must be substantiated prior to any significant financial expenditure. Piloted flight simulation provides the means for assessing and demonstrating the adequacy of the pilot/vehicle system prior to hardware procurement.

Piloted flight simulation should be employed in the development of any system where the pilot is involved directly in the control loop. A partial list of flight control design and development studies that can be used in piloted simulation includes:

1. Susceptibility to pilot-induced oscillation (PIO)
2. Analyses of failure mode effects
3. Definition of control harmony requirements
4. Height-velocity capability, with emphasis on human factors
5. Handling qualities in turbulent air
6. Design trade-off studies
7. Scheduling and mixing of flight control component functions
8. Optimization of stability augmentation system configuration and flight control system forces
9. Weapon delivery suitability
10. Conversion mode characteristics and requirements (for non-helicopter rotorcraft)
11. Flight test supplement (pilot familiarization, test planning, test support)
12. Autorotation entry and recovery.

Piloted simulation is effective particularly in failure mode studies. All types and combinations of failures can be presented for evaluation of transient response characteristics, profile of pilot reaction, allowable time delay prior to corrective pilot action, the need for fully automatic protection, and the resultant limitations on mission capability. Initially, such studies are conducted using a fully mathematical representation of the helicopter. As hardware components become available, they can be substituted for their corresponding mathematical models, thereby enabling refinement of the previous estimates made for items such as friction and hysteresis.

6-6.4 FLIGHT TESTS

The detail design phase of a flight control system generally extends into the early stages of helicopter flight testing. Flight testing particularly is warranted for the detail design of new flight control system concepts, or for novel applications of a given system.

Flight testing associated with the detail design effort requires that the flight control system be mechanized so that its characteristics may be altered or adjusted over a limited range. After establishing the helicopter handling qualities with the flight control system set to its nominal configuration, the engineer can vary the system configuration in an attempt to improve vehicle handling qualities. His assessment of change is based upon pilot commentary and upon recorded time histories of the important vehicle and flight control system response parameters. Such evaluations should consider the total requirements of the system over the entire flight envelope. Specific flight test requirements are discussed in Chapters 8 and 9, AMCP 706-203.

6-6.5 DESIGN REVIEW

When the configuration has been selected and the design requirements have been identified, the detail design proceeds from the one-half and full-size layouts. Stress and weight analyses are conducted simultaneously with the preparation of full-size layouts and detail, assembly, and installation drawings. Long-lead items and material are ordered in advance of the drawing release.

Detailing of the components requires a knowledge of materials, processes, and standard parts. The reader is referred to Chapters 2, 16, and 17 for guidance. Detail requirements applicable to the interfaces between the flight control system and the hydraulic system are described in Chapter 9.

A hazard analysis *shall* be performed in order to determine the design potential for incurring equipment failures or human errors that can cause accidents. Chapter 3, AMCP 706-203, details this design evaluation technique.

The critical design review, a formal technical review of the detail design, is conducted when the engineering drawings are ready for release for fabrication or procurement. The purpose of the review is to determine the total acceptability of the design, i.e., that the detail design satisfies the design requirements, and satisfies the design solution set forth in prior reviews.

Review team members representing product support engineering, flight test, reliability, maintainability, safety, human factors, stress, aerodynamics, materials, and process engineering *shall* evaluate the design as it pertains to their specialized fields.

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CHAPTER 7
ELECTRICAL SUBSYSTEM DESIGN

7-0 LIST OF SYMBOLS

i_c = charging current, A
t_c = charging time, min
VARS = volt-amperes, reactive
Δ_c = increase in battery capacity, A-hr

7-1 INTRODUCTION

7-1.1 GENERAL

The basic determinants of overall electrical system design and layout are the demands of the equipment on the helicopter for electrical power, and the physical and operational constraints imposed by the helicopter and its mission(s). The latter will include such aspects as the availability of space in the aircraft, the safety requirements imposed by the system, and the weight penalty imposed by the chosen subsystem.

The specific details of electrical subsystem characteristics and utilization are determined by MIL-STD-704. In general, however, helicopters require 28 V DC for both normal and emergency operation of such items as fuel pumps, flight instruments, panel lighting, most avionic equipment, and electrically-driven weapons; constant frequency 400-Hz AC power for some avionic equipment; and often variable-frequency AC for some heating (deicing) and frequency-insensitive loads. Thus, with suitable conversion components, the basic helicopter electrical system can be DC, variable-frequency AC (vf AC), or constant-frequency AC (cf AC).

New helicopter design trends are toward increased electrical power in general, as well as increased amounts of constant-frequency power. Many helicopter system designs provide an input speed to the generator that varies by more than ±5%.

In addition, a major consideration in electrical system selection is engine starting. If the maximum engine starting torque is less than 90 lb-ft, a 400-A, 28-V DC starter-generator powered by two CA-5, or CA-9 nickel-cadmium batteries will provide the simplest self-contained start system. However, this 90 lb-ft limit defines a small engine, and, therefore, a small helicopter.

The actual choice of electrical systems will depend upon the relative demand for DC, cf AC, or vf AC and a weight analysis of the necessary components. This system selection generally will include the decision on the existence (or not) of an on-board auxiliary power unit (APU). Above a certain total requirement for cf AC, for example, a considerable weight saving can be realized by using a CSD (constant-speed drive) input to the electrical system instead of an inverter system; or the presence of an APU may provide hydraulic or pneumatic starting and thus decrease overall electrical system weight by eliminating the need for a DC starter-generator. A transformer-rectifier would be used in this case for other DC power needs.

The electrical power system on the helicopter may thus be based upon such power sources or conversion devices as:

1. AC or DC generators driven by:
    a. The main rotor power transmission system
    b. Engine accessory drives
    c. Constant speed drives (speed controlled by hydraulic or mechanical torque converter)
    d. Constant-speed turbines (speed controlled by air or gas turbines)
2. Inverters
3. Transformer-rectifiers
4. Batteries.

The selection of the type of system as well as characteristics of the components of the electrical system are discussed further in the paragraphs that follow.

7-1.2 SYSTEM CHARACTERISTICS

The type of electrical power source generally will have been selected during preliminary design (see Chapter 7, AMCP 706-201). During the detail design phase, it is necessary to confirm this selection and to define the distribution and utilization systems.

Detail design begins with the specifications for the particular helicopter, which typically spell out:

1. The design gross weight (i.e., the weight of the primary mission payload plus the empty weight, including mission-essential equipment)

2. The maximum performance capabilities of the aircraft at its design gross weight

3. The specific primary power source(s) and the power conversion methods

4. The specific utilization equipment, which will include lights, displays, communication equipment, avionics, fire control, and additional electrically powered equipment such as hoists.

While the set of utilization equipment components is typically defined in the specifications, numerous options may still be exercised. For example, it may be left to the discretion of the designer as to whether an auxiliary power unit is to be included. The trade-off analyses involved in system selection are discussed in Chapter 7, AMCP 706-201. Approximate weights of the various electrical system components or typical weights per unit output are presented with that discussion. Only upon analysis of all secondary power requirements and systems will it be possible to determine whether or not the requirements for engine-starting, and other secondary power, would be better served by an APU or by an electrical source.

In general, duplicate primary electrical power sources will be required. The electrical utilization load will be split between the sources, being distributed on two busses such that the total load will not be more than half the capacity of the total source. Thus, in the event of failure of one source, automatic paralleling will enable the remaining source to supply all the electrical power required.

In addition, an "essential" bus must be provided. All components vital to the safe operation of the helicopter under night and instrument conditions must be connected to this bus. In the event of the complete failure of the primary source, the emergency source (battery) must provide power to this bus typically for 20 min of operation with a 10% reserve. Such a specification for the battery powered emergency bus typically will require the installation of a battery charger/analyzer in the system. This unit is designed to insure that the charge is maintained and monitored and that the power from the primary source is distributed properly during normal operation between the utilization load and battery charging.

To illustrate the level of input detail given to the designer at the outset, the system specification may delineate the electrical system for a particular helicopter — i.e., the primary AC power source *shall* be two 400-cycle, three-phase 120- to 208-V AC generators mounted on the accessory gearbox, that primary DC *shall* be supplied by two transformer-rectifiers, and that emergency power *shall* be provided by a nickel-cadmium battery with sufficient capacity to supply power for 20 min of flight with a 10% reserve. This battery load includes an inverter to provide essential AC needs. Also specified are the requirements for lighting, communications, navigation equipment, and other utilization equipment components.

Even with such characteristics predetermined, it remains for the designer to select components that meet the applicable specification and to insure that the electrical characteristics of the overall system conform to the requirements of MIL-STD-704. Substantiation of the system design *shall* include a load analysis in accordance with MIL-E-7016.

7-1.3 LOAD ANALYSIS

The information and general format required for electrical load analysis are given in MIL-E-7016; however, the requirements may be modified slightly (particularly for automated systems) to fit each program. For instance, the specification requires that the form be as indicated, but this can be modified to fit the format for automated equipment. One page can comprise the equipment list, equipment description, parts designation, and electrical ratings, while the next sheets can obtain the acutal calculations, with the pages folded such that the columns match the preceding sheet when unfolded. The paragraph that pertains to operating times can be revised to match more closely the modern generator overload times; i.e., 5 sec, 5 min, and continuous instead of 5 sec, 2 min, and 15 min. MIL-E-7016 requires that phase-to-ground identification be A-N, B-N, C-N, etc., and this can be modified to A, B, C, and D, with D being the neutral leg of a three-phase four (4) wire system. However, an explanatory note must be included; and, because delta-connected loads are rare in modern helicopters, a code also can be established for this situation and explained. MIL-E-7016 gives the formula for power factor. Usually, this information is obtained from the equipment manufacturer or by actual measurement; however, the formula in MIL-E-7016 may be used. The formula for determining single-phase and three-phase power factors PF is:

$$PF = \frac{\text{connected watts}}{(\text{connected watts})^2 + (\text{connected VARS})^2} \quad (7\text{-}1)$$

where

VARS = volt-amperes, reactive

The time intervals for the analysis can be modified, if required. For example, time periods of 5 sec in tenths of a second increments, 5 min in hundredths of a minute, and 15 min in hundredths of a minute may be used.

7-1.4 LOAD ANALYSIS PREPARATION

The load analysis, as defined by MIL-E-7016, can be written, typed, or presented as an automated printout.

From the beginning of a new helicopter design, a complete electrical load file must be kept for each piece of equipment. Ideally, a printed file card should be made, allowing space for the following information:

1. Name of equipment
2. Equipment part number
3. Rated voltage (normal operating)
4. Type of voltage (DC, AC, single- or three-phase)

5. Amperes per wire
6. Volt-amperes (normal operating)
7. Power factor (normal operating)
8. Watts (normal operating)
9. Volt-amperes (emergency conditions, if applicable)
10. Power factor (emergency conditions, if applicable)
11. Watts (emergency condition, if applicable)
12. Operating time
13. Source of the above information

Most of the preceding data can be obtained from the equipment manufacturer. In addition, the manufacturer of each piece of equipment should be required to supply a component load analysis.

After manufacturer-provided data are recorded, the remaining information can be calculated. For a nonautomated system, the calculations can be made individually and recorded on a file card. For an automated system, a computer can do the necessary calculations as a separate run, or they can be submitted from the tabulation print-out.

After the file is as complete as possible, it then is necessary to assign each component a power bus in the helicopter, an electrical system name, and a component reference item number.

7-1.5 MANUAL FORMAT

When preparing a manual load analysis, the figure examples of MIL-E-7016 can be used.

A sample power distribution system is shown in Figs. 7-1 and 7-2. The generator mounting and drive data and power source output data examples are self-explanatory.

The AC load equipment and AC power source utilization analysis charts from MIL-E-7016 can be combined on one chart for each helicopter AC power bus. See Figs. 7-3 and 7-4 for sample AC load analysis charts. The equipment components are to be arranged alphanumerically.

The typical transient analysis is required only for extreme transient loads, and, with modern generator ratings, may not be required at all

The engine starting requirements data must be shown even though the helicopter is started only on ground power and data does not appear in the "Start and Warmup" column of the load analysis chart. The information will be used to determine the ground starting power supply requirements.

If a battery is used in the helicopter, a chart must be included that shows a theoretical charging factor versus time of operation. This chart will depend upon the type and size of the battery to be used, as well as the design of the battery charger.

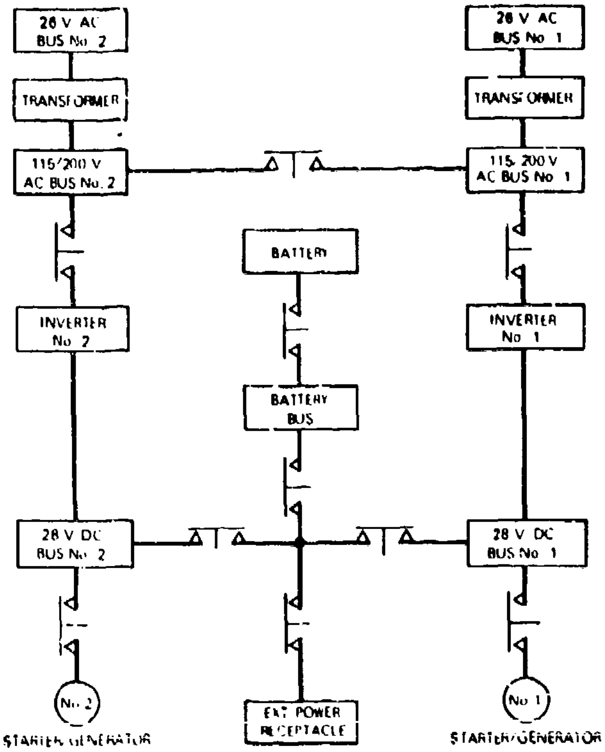


Figure 7-1. Typical DC Power Distribution System

7-1.6 AUTOMATED FORMAT

For an automated format, the same information is required. The program may be written so that calculations are done automatically as the load analysis is being processed. The information then can be checked, and bus totals given, automatically. A typical automation flow chart is shown in Fig. 7-5.

In this example, a two-card system is used. The first card contains the following information:

1. Bus
2 System
3. Item number
4. Equipment name
5. Part number
6. Volts
7. Power factor
8. Volt amperes.

The second card contains:

1. Bus
2. System
3. Item number
4. Number of units
5. Phase assignment
6. Notes.

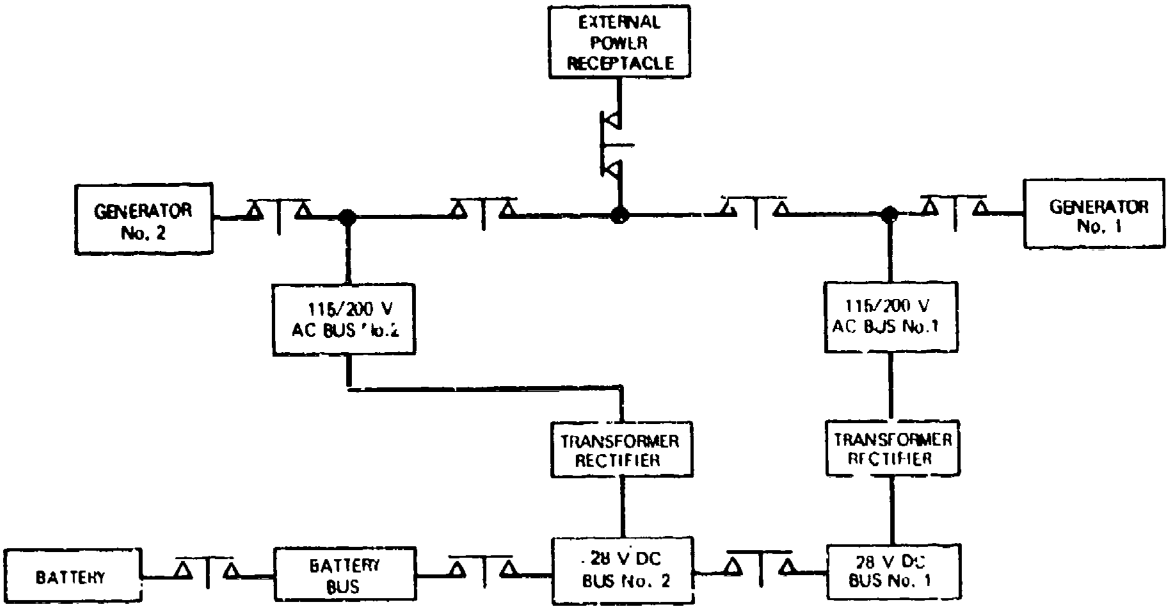


Figure 7-2. Typical AC Power Distribution System

In addition, the second card includes the 5-sec, 5-min, and 15-min averages for:

1. Start and warmup
2. Taxi
3. Takeoff and climb
4. Cruise
5. Combat cruise
6. Descent and landing
7. Emergency.

The third card shown on the flow chart is for programming purposes only.

With an automated system, changes can be made readily, new printouts requested as needed, and individual bus totals obtained at any time during the design of the helicopter.

With both the automated and manual systems, each bus must be totaled separately.

7-1.7 SUMMARY

Included in each load analysis *shall* be a summary of results, which will include a brief summary of generating, rectifying, transforming, and battery capabilities, compared with maximum, average, and emergency loads. The summary will include any special, limiting, or marginal operating conditions that may exist. The summary will be brief and concise, and indicate clearly the helicopter power system true conditions.

7-2 GENERATORS AND MOTORS

7-2.1 GENERAL

In the detail design of the electrical system, certain fundamental criteria must be followed in order to insure proper selection of power-generation equipment and motors and their applicability to electromechanical energy conversion requirements.

Before selection of the electrical rotating components, certain decisions are necessary. These decisions, which may be preliminary, will form the bases for trade-offs related to the optimization of the entire electrical system. These considerations should include the following as a minimum:

1. AC or DC system
2. AC systems — constant or variable frequency
3. Applicable power quality requirements, e.g., MIL-STD-704
4. DC systems — engine starting requirements and battery capacity restrictions
5. Electrical load analyses, including any additional load imposed upon the generator by feeder losses
6. Generators — characteristics of the prime mover; speed or speed range, torque limits, overhand moment (weight) restrictions, and vibration and shock environments
7. Rotating components — details of the installation, including envelope restrictions (length, diameter, tool clearances, removal clearances, etc.), te-

EQUIPMENT	PART No.	No. OF UNITS	OPERATING TIME, min	ELECTRICAL REQUIREMENTS PER UNIT 115/200 VOLTS									POWER FACTOR	CONNECTED LOAD	
				TOTAL VA	WATTS				VARS					WATTS	VARS
					φ1	φ2	φ3	TOTAL	φ1	φ2	φ3	TOTAL			
GUN SYSTEM	60G28593	1	▲	86		81		81		26		26	.95	85	
PNEU. PRESS. IND.	2906-22	2	30.0	3			.6	.6			2.94	2.94	.20	1.2	5.9
OXY. QTY. IND.	AERNO 64 4001	2	30.0	11.1	7.9			7.9	7.9			7.9	.71	15.8	15.8
NAV. COMPUTER	8712-23	1	30.0	90.3			85	85			30.8	30.8	.91	97.5	60.6
VENTURI HTR.	29036-32-5	1	30.0	600	600			600					1.00	600	
LEAD COMP. SET	AN/ASG-33	1	30.0	390	104	104	104	312	78	78	78	234	.80	312	234
SEAT ADJUST	DMU 806186	2	0.20	393A			315	315			236	236	.80	630	472

▲ SEE SUPPLEMENTARY DATA

ITEM No.	OPERATING CONDITIONS																							
	LOAD OR ANCHOR								START AND WARM UP								TAXI							
	WATTS	AVG WATTS			VARS	AVG VARS			WATTS	AVG WATTS			VARS	AVG VARS			WATTS	AVG WATTS			VARS	AVG VARS		
		5S	5M	15M		5S	5M	15M		5S	5M	15M		5S	5M	15M		5S	5M	15M		5S	5M	15M
A132																								
D102	1.2	1.2	1.2	1.2	5.9	5.9	5.9	5.9	1.2	1.2	1.2	1.2	5.9	5.9	5.9	5.9	1.2	1.2	1.2	1.2	5.9	5.9	5.9	5.9
D182	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8
F324	97.5	97.5	97.5	97.5	50.6	50.6	60.6	60.6	97.5	97.5	97.5	97.5	50.6	50.6	50.6	60.6	97.5	97.5	97.6	97.6	60.6	50.6	50.6	50.6
F418																								
F622																								
M101	630	630	63.0	8.4	472	472	47.2	6.3	630	630	63.0	8.4	472	472	47.2	6.3	630	630	63.0	8.4	472	472	47.2	6.3
TOTAL																								

ITEM No.	OPERATING CONDITIONS																							
	TAKEOFF AND CLIMB								CRUISE								CRUISE-COMBAT							
	WATTS	AVG WATTS			VARS	AVG VARS			WATTS	AVG WATTS			VARS	AVG VARS			WATTS	AVG WATTS			VARS	AVG VARS		
		5S	5M	15M		5S	5M	15M		5S	5M	15M		5S	5M	15M		5S	5M	15M		5S	5M	15M
A132																	81	81	20.3		26	26	6.5	
D102	1.2	1.2	1.2	1.2	5.9	5.9	5.9	5.9	1.2	1.2	1.2	1.2	5.9	5.9	5.9	5.9	1.2	1.2	1.2	1.2	5.9	5.9	5.9	5.9
D182	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8
F324	97.5	97.5	97.5	97.5	60.6	60.6	60.6	60.6	97.5	97.5	97.5	97.5	60.6	50.6	60.6	50.6	97.5	97.5	97.5	97.5	60.6	60.6	60.6	50.6
F418	600	500	600						600	600	500						600	600	500	500				
F622																	312	312	312	312	234	234	234	234
M101	630	630	63.0	8.4	472	472	47.2	6.3																
TOTAL																								

USE FOLDOUT PAGE OR MORE SHEETS FOR OTHER OPERATING CONDITIONS

Figure 7-3. Example Load Analysis AC Left Hand Main

BUS	SYSTEM	ITEM	EQUIPMENT	PART DESIGNATION	NOTES	No. OF UNITS	RATED VOLTS *	PF	VOLT AMPS	PHASE	LOAD OR ANCHOR						START AND WARM-UP					
											WATTS			VARS			WATTS			VARS		
											5S	5M	15M	5S	5M	15M	5S	5M	15M	5S	5M	15M
3	A	132	GUN SYSTEM	60G28593	1	1	115	0.86-	86.0	B												
1	D	102	PNEU. PRESS. IND.	2808-22		2	115	0.20-	3.0	C	1.2	1.2	1.2	5.9	5.9	5.9	1.2	1.2	1.2	5.9	5.9	5.9
1	D	182	PXY. QTY. INDICATOR	AERNO 64-4001		2	115	0.71-	11.1	A	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8	15.8
2	F	324	NAV. COMPUTER	8712-23		1	115	0.91-	90.3	C	97.5	97.5	97.5	50.6	50.6	50.6	97.5	97.5	97.5	50.6	50.6	50.6
2	F	418	VENTURI HEATER	29030-32-5		1	115	1.0	600	A												
3	F	522	LEAD COMPUTING SET	AN/ASG-33		1	200	0.80-	390	D												
2	M	101	SEAT ADJUST	DMU 806185	1	2	115	0.80-	393	C	630	63	8.4	472	47.2	6.3	630	63	8.4	472	47.2	6.3
			1 TOTALS																			
			2 TOTALS																			
			3 TOTALS																			

Figure 7-4. Example AC Load Analysis Format

minal blocks and/or connector restrictions, temperature-altitude environment, availability, and characteristics of air or liquid cooling

8. Adequate system growth — designer must consider future growth when selecting a power source.

7-2.2 AC GENERATORS (ALTERNATORS)

An AC power source capable of insuring a power quality equal to, or better than, that specified by MIL-STD-704 can be achieved best by utilizing the attributes of the conventional salient-pole, synchronous alternator.

The helicopter AC generator is comprised of the main generator, an exciter, and, in the majority of cases, a permanent-magnet pilot exciter sharing a common housing and shaft. Modern-day alternators are brushless; i.e., no brushes, slip rings, or commutators are employed.

7-2.2.1 Electrical Design

The main generator consists of a stator and a rotor. The stator is built of steel laminations that are uniformly slotted on the inner periphery and contain the output windings. These windings are connected in a normal three-phase, four-wire manner, and are displaced so as to minimize distortion of the output voltage waveform. The rotor or field consists of laminations punched so as to form "poles", and the number of poles and the rpm fix the output frequency. The main field winding is wound on the rotor poles and is excited with DC. This provides the magnetomotive force necessary to provide sufficient lines of force (flux) in the magnet circuit of the main generator for adequate, all-load-condition, voltage generation.

The exciter functions so as to supply DC to the main field winding. The poles are on the stator and on a disturbed winding in slots on the rotor. Both members are laminated, as in the main generator. The exciter is an AC generator, and provides DC to the main field through a bridge of rotating rectifiers.

In order to achieve a self-sufficient generator — one not dependent upon any external power source for excitation power — a permanent-magnet pilot exciter completes the electrical portion of the AC generator. This generates either polyphase or single-phase AC power, which is rectified either within the generator (stationary rectifiers mounted within the generator case) or in the voltage regulator (par. 7-4) to provide DC excitation power to the exciter. Depending upon system requirements, the magnetic pilot exciter may provide control power, protective circuitry, or operational power necessary for proper functioning of the distribution system.

7-2.2.2 Mechanical Design

AC generators are housed in either aluminum or magnesium housings. The selection of housing material is dictated by weight and/or vibration requirements. Lamination steel for the magnetic circuit is either a silicon or cobalt alloy. The latter construction results in a relatively expensive generator, but provides a weight advantage of almost 30% over a system using silicon steel punchings.

Practical generator speeds range from 6000 to 12,000 rpm for 400-Hz output. Variable-frequency machines in ratings to 120 kVA are practical to speeds of 20,000 rpm. Applications of 6000 to 12,000 rpm require grease-lubricated ball bearings with bearing lives of 5000 hr in an average helicopter en-

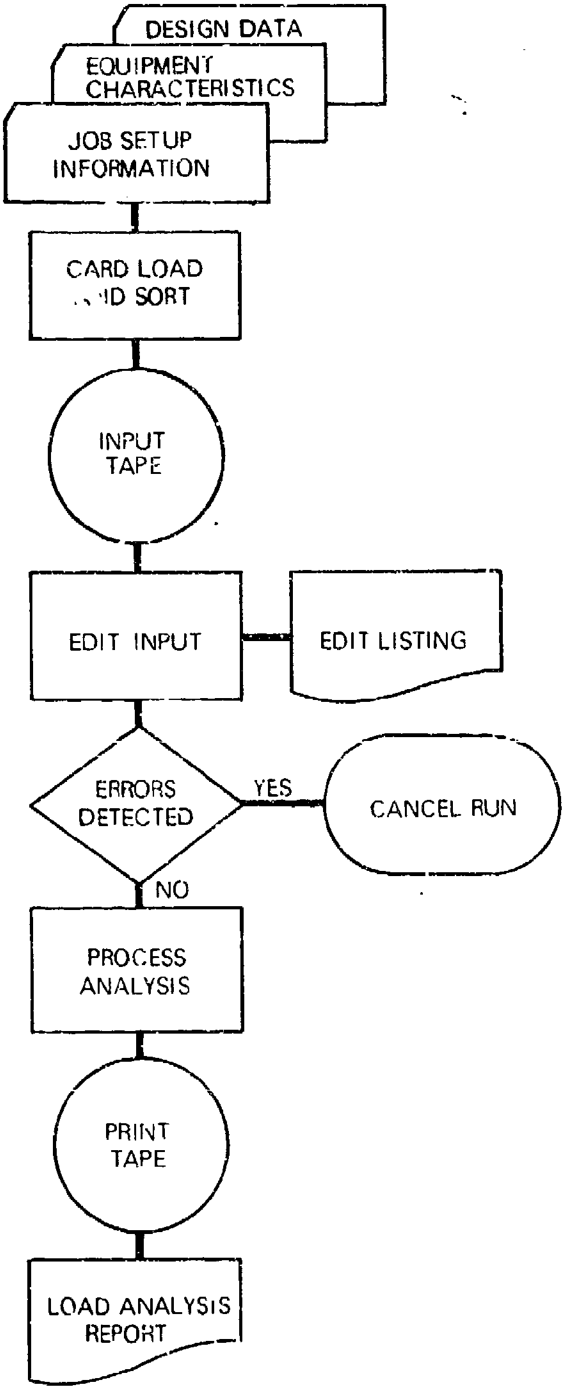


Figure 7-5. Typical Automation Flow Chart

vironment. Oil lubrication of bearings generally is necessary for generator speeds in excess of 15,000 rpm. Bearing life in excess of 10,000 hr has been achieved in helicopter generators using oil lubrication. A typical oil-lubricated bearing AC generator is shown in Fig. 7-6.

7-2.2.3 Cooling

Cooling, a primary requirement in an AC generator specification, may be accomplished by air or by liquid. Both air-cooled and oil-cooled generators are employed for helicopter applications.

For air-cooled generators, either self-cooling or blast cooling may be employed. In the case of self-cooling, an integral fan is located on the rotor of the machine, and diffusers or baffling are employed to direct the air over the hot internal surfaces. If the installation is such that ducting is provided to the fan inlet, the pressure-flow characteristics of this ducting must be considered so that all through-generator airflow requirements are met. A self-cooled generator that exhibits good performance in the laboratory may burn up on the airframe because duct restrictions were not considered. In the case of blast-cooled generators, inlet airflow and pressure-altitude characteristics of the separate forced-air supply must be defined adequately in order to allow proper use of a generator cooled in this manner. Cooling-air temperature versus altitude and ambient temperature data are vital aspects of an adequate cooling specification as discussed in MIL-G-6099. Contaminant protection of cooling air is required.

Oil-cooled generators fall into two categories, conduction-cooled and spray-cooled. Oil-cooled generators are practical with inlet oil temperatures from −65° to 330°F.

In the conduction-cooled generator, oil is circulated through closed passages in the housing and rotor shaft. Cooling is obtained by conduction of heat to the oil from the hot windings. The bearings use the oil for lubrication as well as for cooling, and rotating seals are required. The weight of the conduction-oil-cooled generator is comparable to that of the air-cooled generator.

In the spray-cooled generator, the cooling oil, in effect, is sprayed directly on the windings. This results in an improvement in heat transfer, along with a weight reduction of approximately 15% compared to the air-cooled or conduction-oil-cooled generator. To date, all spray-oil-cooled generators have been applied to 400-Hz systems, and operate at 12,000 rpm. For comparison assume a 90-kVA rating; a modern air-cooled generator using magnesium housing and cobalt alloys weighs approximately 90 lb. A spray-cooled generator with the same rating weighs 55 lb. A generator weight of approximately 0.5 lb per kVA is achievable with spray cooling. If spray cooling is used, it is necessary to scavenge the generator cavity, i.e., to remove excess oil resulting from spraying of the windings.

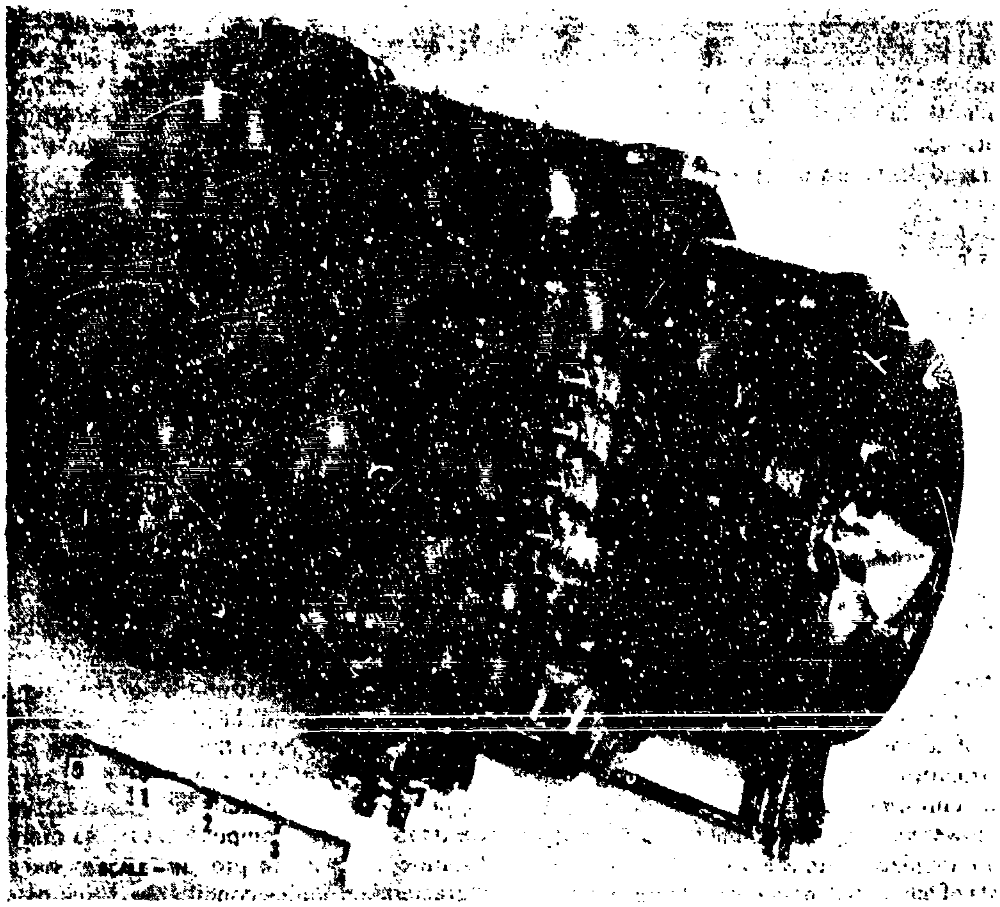


Figure 7-6. Typical AC Generator with Oil-lubricated Bearings

7-2.2.4 Application Checklist

The criteria that follow should be considered in the design of all AC generators. This is a minimum list of characteristics that must be defined and adapted to a given application:

1. Rating:
   a. kVA at required power factors
   b. Voltage at terminals
   c. Phases
   d. Frequency
2. Speed, rpm
3. Maximum weight
4. Envelope, diameter, and length
5. Mounting details
6. Cooling requirements
7. Applicable Military Specifications regarding generator and/or system performance, electromagnetic interference, vibration, etc.
8. Minimum efficiency
9. Overloads and time at each overload
10. Short-circuit current capacity and time at short-circuit
11. Waveform
12. Performance requirements under unbalanced load conditions.

7-2.2.5 Variable-frequency AC Generators

Variable-frequency generators are practical in ratings to 120 kVA at speeds to 20,000 rpm. The previous discussions relative to mechanical design, cooling, and application checklist generally are applicable also to the variable-frequency generator.

There are two significant performance characteristics peculiar to the variable-frequency generator that should be considered prior to its application:

1. Voltage transient performance at high speed
2. Voltage regulation problems over a wide speed range.

Voltage transient performance at high speed for a wide-speed-range generator (e.g., 1.5:1) can result in severe system problems. This is because the maximum voltage attainable from the generator at the high speed is the speed range times the voltage attainable at the low speed. Upon application of load, severe dips in system voltage could be experienced.

With regard to voltage regulation, when a speed range approaches or exceeds approximately 2.5:1, high-speed instability can result. Because the regulator is called upon to adjust from an overload at low speed to no load or leading power factor loads at high speeds, exciter field current variation may be extreme. Such a field current variation could be in the order of 15:1.

7-2.3 STARTER/GENERATORS, DC GENERATORS AND STARTERS

State-of-the art DC systems for helicopter applications are designed for operation at a nominal 28 V, with power quality defined in accordance with the requirements of MIL-STD-704. For the majority of applications, advantage is taken of the volumetric efficiency and lightweight properties of the DC starter/generator. Nevertheless, there exist many applications that, for various reasons, employ both a DC generator and DC starter.

7-2.3.1 Starter/Generators

The construction of the DC starter/generator is comprised of a rotating armature and stationary field. The armature is constructed of a stack of steel laminations uniformly slotted on the outer periphery; the power windings are connected to the commutator and are placed into the slots. The stationary field consists of laminated main poles, interpoles, and a solid steel field ring to which the poles are attached on the inner periphery.

On the main poles are wound the main field windings, connected either in parallel (shunt) or in series with the armature winding. The interpole coils are connected in series with the armature. A fourth winding — distributed in slots, placed in the pole faces, and connected in series to the armature — serves to support the main field and to overcome the demagnetizing effects of an armature reaction to the magnetic field set up by load currents flowing in the armature winding. This winding is termed the compensating winding, and generally is employed with generator ratings of 200 A or more.

The starter/generator normally requires a fifth winding consisting of a single turn in series with the armature and wound on the main poles. This winding, during starter operation, aids in increasing the torque output per ampere of input current. Some manufacturers leave this winding connected during generator operation as a differential compound winding that aids in the regulation over the load and speed range.

The discussion of the mechanical construction of the AC generator is applicable generally to the DC starter/generator also. In order to achieve the lightest possible weight without sacrificing mechanical integrity, both aluminum and magnesium are used for housing materials, the choice being related directly to the mechanical environment requirements.

As in the AC generator, cobalt alloy laminations are employed where weight is of significant importance. The weight advantage of a DC starter/generator employing this high-permeability material, compared to a unit employing silicon steel punchings, is of the order of 20%.

The starter/generator normally operates from stand-still to speeds of 6000 rpm in relation to starter mode operating speeds. For generator operation, it ordinarily covers a speed range of approximately 2:1, with 3000 rpm the usual minimum speed, and seldom is applied where maximum speeds exceed 12,000 rpm.

State-of-the-art DC starter/generators generally employ grease-lubricated ball bearings. For helicopter usage, the bearing life generally falls between 1000 and 3000 hr.

The brushes that ride on the commutator and conduct the current from the power source — in the case of the starter — and to the load — in the case of the generator — are made normally of carbon and copper. Because of altitude requirements, the brushes are treated with a compound (such as molybdenum disulfide) in order to provide the necessary filming characteristics under conditions of low oxygen and moisture. For starter/generators, brush life is limited to 500-1000 hr, depending upon the severity of the start. For those applications requiring generator operation only, brush lives of up to 2000 hr are possible.

Air cooling of the DC starter/generator and generator is standard practice. Contaminant protection of this cooling air is required. This cooling may be accomplished by integral fan (self-cooling), by blast cooling, or by a combination of the two. Precautions are necessary in order to define the cooling conditions adequately. Because of the problem of providing adequate heat transfer from the brushes and commutator to the oil, oil cooling seldom is employed.

Following is a checklist of minimum information necessary in order to define adequately a starter/generator for a given application:

1. Engine type and manufacturer
2. Intended installation
3. Envelope requirements (diameter and length)
4. Maximum allowable weight
5. Maximum allowable overhand moment
6. Engine mounting pad details

7. Applicable specifications, if any
8. Type of cooling — blast, self, or other
   a. If blast cooling, pressure available
   b. Temperature of air
9. Ambient temperature range
10. Altitude requirements
11. Direction of rotation facing engine pad
12. Engine to starter/generator pad gear ratio
13. Power supply for starting:
   a. Battery, type and voltage
   b. Ground power unit, type and voltage
14. Engine torque versus speed curves for standard conditions and −65°F (or the lowest applicable temperature), plus a notation of whether or not these curves include accessories and gearing
15. Engine light-off speed if not shown on curves
16. Starter cutoff speed if not shown on curves
17. Maximum allowable time to light-off speed
18. Maximum allowable time to cutoff speed
19. Starter/generator pad rpm at engine idle
20. Starter/generator pad rpm at minimum cruising speed
21. Starter/generator pad rpm at maximum engine speed
22. Required generator output and voltage under all speeds in range of regulation
23. Percentage of maximum generator output used at engine cruising speed
24. Voltage regulator type and applicable specification.

For engine starting, either a ground power supply or aircraft battery is used. Ground power supplies generally are of the constant-current type, and provide the best power source available for engine starting. In the majority of helicopter applications, where starter/generators are used, aircraft batteries are employed for starting. The batteries are rated 24 V and are either silver-zinc or nickel-cadmium (par. 7-3). If multiple batteries are used, they may be connected parallel or in series to provide the desired starting characteristics, but consideration must be given to the applied torque vs generator shaft shear section and the engine gearing limitations.

A typical starter/generator used in helicopter applications is shown in Fig. 7-7.

7-2.3.2 DC Generators

The helicopter DC generator is identical electrically and mechanically to the DC starter/generator, with the exception that, generally, no series turn is employed on the main field winding.

The preceding discussion relative to mechanical

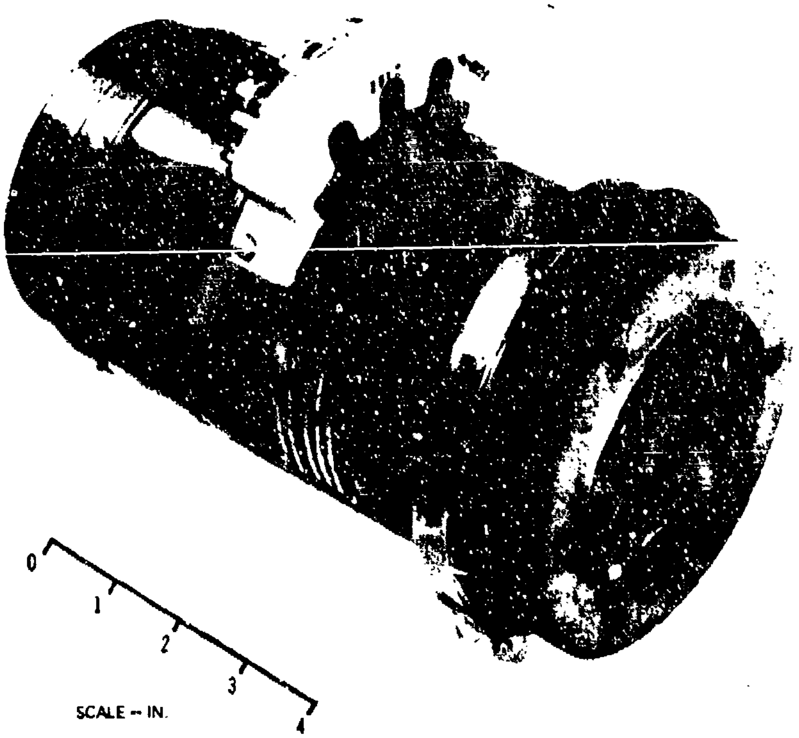


Figure 7-7. DC Starter/Generator

design, speed range, bearings, brushes, and cooling also are applicable here. It is suggested that reference be made to MIL-G-6162 as a guide for specification preparation.

A typical blast-cooled DC generator is shown in Fig. 7-8.

7-2.3.3 DC Starters

With the advent of the DC starter/generator, the DC starter now has limited application. In the past, the majority of applications were found on helicopters employing reciprocating engines, and, therefore, these starters were designed for cranking service rather than for the type of starting service required by the turbine engine. For cranking, the starter operates at a nearly fixed speed until engine light-off. Present-day starters are designed for starting the turbine engine, and the starter operates over a speed range of 0-20,000 rpm.

The DC starter/motor is quite similar, electromechanically, to both the DC generator and the DC starter/generator. Ordinarily, often three windings are used, as opposed to the five windings often employed in generators. The series connection of the main field with the armature and interpole winding provides the highest torque per ampere of current, and lends itself to use with battery power supplies.

As in the starter/generator and generator, starter/motors employ grease-lubricated ball bearings. Brushes usually are of the low-contact drop type, and contain a high percentage of metal so as to minimize voltage drop at the high currents required by the starting cycle. Because of the short duty cycle requirements, an integral fan provides required cooling.

The DC starter/motor serves no function after starting the engine; therefore, design provisions to disconnect it from the engine accessory drive must be made. This usually is accomplished by a mechanically or electrically operated jaw engaging and disengaging mechanism.

A turbine engine starter is shown in Fig. 7-9.

7-2.3.4 Boost Starting System

Modern military practices dictate the use of turbine-powered helicopters for a variety of reasons, one of which is their ability to use completely unprepared terrain for takeoff and landing. Therefore, deployment of helicopters away from airfield support functions is common. Deployment in hot or cold temperature environments having no APU or ground power facilities requires that batteries not lose their gas turbine engine-start capabilities.

The effect of cold temperature on the energy release rate of storage batteries is a matter of common practical knowledge. The engine-starting capability of a fully charged battery in a 70°F environment is reduced to 27% after the battery has been stabilized in a commonly experienced −25°F environment. Experience has shown that the probability of a successful 0°F start capability decreases rapidly as the level of the battery charge decreases.

Gas turbine starting problems are not restricted solely to the low end of the temperature spectrum. A less obvious, but very real, problem is encountered at the high end of the temperature range. The hot start, in which the temperature within the turbine exceeds the safe operating range for turbine materials, is encountered all too frequently in high ambient temperatures during the starting sequence of gas turbine-powered helicopters. The hot start can be caused by a number of conditions. A battery that is not charged sufficiently and improper fuel control adjustments are two conditions that can lead to critical hot-start problems. Another cause of hot starts is the time-lapse needed to accelerate the engine from light-off to idle speed. The net result of hot starts, whatever the cause, is premature engine failure. Even though a hot start might not result in an immediate and catastrophic engine failure, it will shorten the times between engine overhauls, thereby increasing helicopter operating costs.

The characteristics of electrical starting systems (and starter/generator systems) for turbine engines generally are such that maximum torque is delivered from the starter/motor to the turbine upon the initiation of the starting sequence. Starter/motor output torque decreases approximately as a straight-line function at increased turbine speeds, with the starter/motor torque reaching a very low value at starter/motor cutoff. The starter/motor is designed to have a stall torque capability exceeding the torque limit of the engine accessory drive system.

Adding a boost feature to the electrical starting system provides an additional supply of starting energy for extraordinary starting situations. Conceptually, the boost power is obtained from a combination of gas generator and gas motor, and by coupling the gas motor with the electrical starter/motor. This boost feature is a supplement to, and in no way is intended as an alternative for, the electrical starting system. Even though the added feature is used only occasionally, the components added to the electrical starter, or starter/generator system, nevertheless must be taken into account, and their effect must be established upon such elements as overhand moment, vibration characteristics, operation of the system in a generating mode, and the overall physical

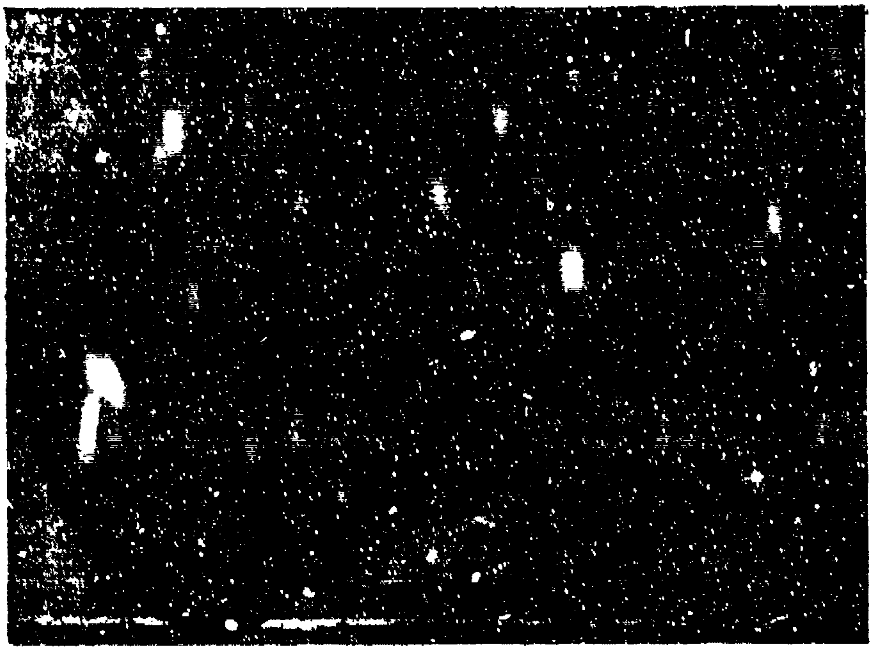


Figure 7-8. Blast-cooled DC Generator

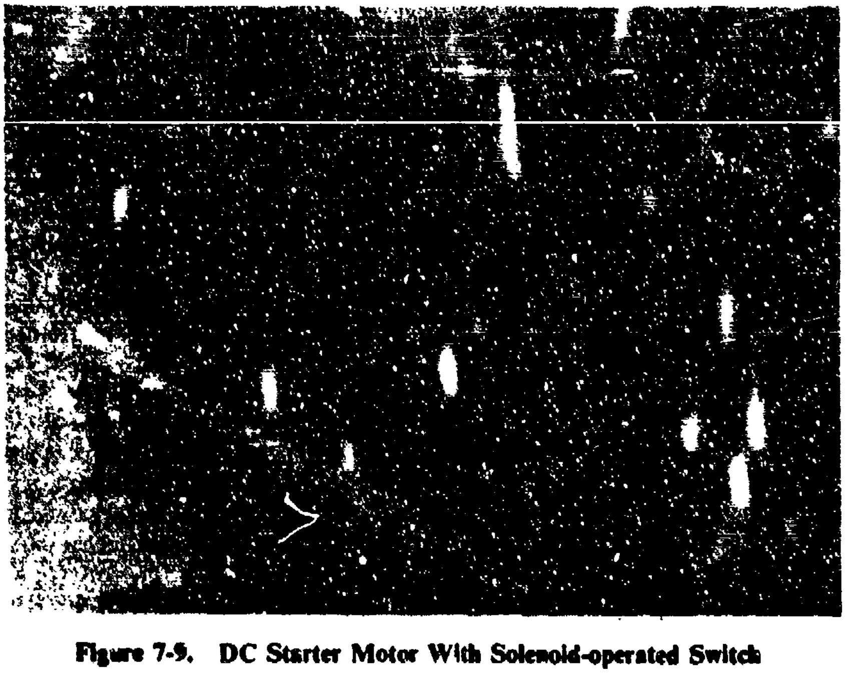


Figure 7-9. DC Starter Motor With Solenoid-operated Switch

Figure 7-10. Prototype Cartridge-boosted Electrical Starter System

constraints of specific engine installation. Prototype hardware is shown in Fig. 7-10.

7-2.4 ELECTRICAL MOTORS

Electrical motors for use in helicopter electrical systems may be either AC or DC, depending upon the primary power source selected. AC motors in use are almost universally of the squirrel-cage induction type. They may be three-phase or single-phase, with the number of phases to be used being governed principally by motor size and by the characteristics of typical load requirements.

The squirrel-cage induction motor comprises a laminated stator — identical in configuration to that employed in the AC generator — and a laminated rotor. The rotor has slots on the outer periphery containing either copper or cast aluminum bars — short-circuited on both ends — with end rings of the same material so as to form, ultimately, what resembles a squirrel cage. The input winding is contained in the stator slots and may be wound three-phase or two-phase.

Housing materials may be either aluminum or magnesium, depending upon weight and environmental considerations; and bearings are almost universally of the grease-lubricated ball variety. For the specification of the AC motor, the best guide is contained in MIL-M-7969.

The factors involved in construction of the DC generator generally apply also to the DC motor, with the exception that seldom, if ever, are pole face compensating windings employed. Depending upon the load, the DC motors may be used in series, shunt, or compound winding configurations. For very small motors, permanent-magnet fields are employed in place of the shunt or series field windings. MIL-M-8609 is recommended as a guide for specification preparation.

In general, the electrical motor may be termed a torque device, inasmuch as a certain volume of iron and copper is required to produce a given torque. The size of the motor is dictated by the torque requirements. For helicopter applications, where size and weight are at a premium, it is normal to operate motors at high speeds in order to obtain a high power output (power equals the product of torque and speed) per unit volume. This, fundamentally, is why the majority of aircraft AC power systems operate on 400 Hz rather than the 60-Hz power usually employed for industrial and commercial uses. High-speed operation poses certain problems in bearing and/or brush life, which, when coupled with the fact that the airframe itself has a relatively short life, result in overhaul rates measured in hundreds of hours, rather than in the 10-20-year life rates considered normal in industrial and commercial ground applications.

High-speed, high-power-per-unit-volume operation points up the design parameter that usually determines the size of the helicopter electrical motor. Increased losses follow increased power output, and the size of the motor must be adequate to dissipate these losses without exceeding motor material temperature limits. It follows that the availability of effective cooling directly affects motor size. High altitudes, with low air density, will decrease the cooling available from a fan. High ambient temperatures reduce the heat transfer and add to the motor total temperature. Also, there is a significant amount of heat generated in the rotating member of the motor during acceleration from standstill; and if repeated starts are made, motor temperatures rise significantly. Another significant item is the effect of voltage and/or frequency variation on motor life and size. These items must be accounted for in the design so that, under the worst conditions, the motor continues to produce the required speed at the specified torque. This means that, for all other conditions, the motor will be operating at high-speed, high-power output with higher losses. In brief, the greater the variation of input power, the larger the motor that is required.

The electrical motor also is a torque device in the sense that speed and power input, etc., are direct results of the torque imposed by the load. This leads to use of motor speed torque curves. These curves are, simply, the equilibrium operating points for the motor. The speed, current, power input, etc., are those values that occur when the motor operates at a given torque.

7-2.5 ELECTRICAL SYSTEM CONVERSION

The typical helicopter requires both AC and DC power. Therefore, the ability to convert one to the other as needed also is required. Various types of devices are available by which this conversion can be effected.

7-2.5.1 AC to DC Converters

Devices for converting AC into DC are of two basic types: rotary and static. Rotary systems may be either AC-driven motor-generators or synchronous converters. The latter are essentially DC generators in which slip rings have been connected to the armature winding by equidistant taps. The synchronous converter, in effect, combines the functions of an AC drive motor input with those of a DC-generator output, although with less flexibility in voltage and power-factor control than the motor-generator combination. Converters typically are cheaper, more efficient, and more compact than corresponding motor-generators. It is important, however, that they operate as near to unity power-factor as possible since their "rating", i.e., relative output, decreases rapidly with decrease in power-factor. Table 7-1 displays this relative output relationship for various power factors and number of phases. Converters also must be synchronized with the input AC supply.

The relationships between the AC and DC voltages and currents are functions both of the power-factor and the number of phases (hence, number of slip rings). Converters may be single-phase, in which case there are two slip-rings and two slip-ring taps per pole pair; three phase, in which there are three slip-rings and three taps per pole-pair; and so on. However, because of the sensitivity of the output to the total number of phases, converters usually are operated with six phases. With a sine-wave input voltage, the DC voltage is the peak of the diametrical AC voltage, the latter being the voltage between any two diametrically opposed taps.

At unity power factor and a typical 95% efficiency, the DC and AC currents are equal with three slip rings; with six slip rings, the DC current is twice the AC current.

Static conversion from AC to DC is accomplished by transformer-rectifier units (TRU). The rectifier concept essentially acts to block conduction during one-half of the reversing alternating current. Thus, current is unidirectional, but only during one-half of the sine-wave cycle. By combining two units and a center tap from a transformer, rectification throughout the full sine-wave can be achieved. A smoothing inductance connected in series with the load in such a device effectively smooths out the wave peaks to a relatively small pulsating ripple.

In practice, most rectifiers employ bridge circuits to achieve such full-wave rectification; however, the active components of which may be diodes or thyristors (silicone-controlled rectifiers). The output results are a series of square current pulses, which together produce a continuous current output. Rip-

TABLE 7-1. OUTPUTS OF CONVERTERS RELATIVE TO CONTINUOUS-CURRENT GENERATOR

POWER-FACTOR, %	CONTINUOUS-CURRENT GENERATOR	SINGLE-PHASE CONVERTER	THREE-PHASE CONVERTER	FOUR-PHASE CONVERTER	SIX-PHASE CONVERTER
100	100	85	132	161	194
95.5	100	78	120	145	170
90	100	74	109	128	145

ple voltages in the DC output can be avoided by the use of polyphase input supplies.

MIL-STD-704 specifies that utilization equipment requiring an AC input of 500 volt-amperes (VA) or more and a 28 V DC output of 5 A or less *shall* use static conversion, unless it is designed specifically for use with DC generators.

7-2.5.2 DC to AC Converters

Devices for converting DC into AC also fails into two fundamental classes: rotary and static. The rotary class are basically the same types of devices used in AC-DC conversion — they may be either motor-generator sets or synchronous converters; the synchronous converter having the capability of operating on DC and converting into AC. In this condition they are said to be operating inverted and therefore are known commonly as inverters. In general, what has been said regarding motor-generators versus synchronous converters remains true here — the inverters are more efficient, cheaper, and more compact than comparable motor-generator sets. In this DC-AC mode, however, some suitable electrical or mechanical speed control must be used since converters tend to "run away" in this condition. (The highly inductive load weakens the field through armature reaction and allows the speed to increase.)

For most ordinary, i.e., relatively low-load, applications the most common DC-AC conversion device is the static inverter. This is a circuit that alternately connects the output lines to opposite side of the DC supply typically via the use of such solid-state components as thyristors. Such semiconductor inverters, using both thyristors and transistors in three-phase, full-wave bridge circuits, are in conventional aircraft use. The choice between rotary and static inverter in any specific case must depend upon such parameters as the available input sources, the required output characteristics of the system, and the relative costs and weights.

7-3 BATTERIES

In general, battery selection is based upon battery characteristics, electrical characteristics of generators and associated controls, utilization loads, and certain assumptions in aircraft operations.

7-3.1 BATTERY CHARACTERISTICS

Nickel-cadmium and silver-zinc storage batteries presently are used in aircraft electrical systems. Each electrochemical system has particular service characteristics. Nickel-cadmium exhibits excellent cycle life and output over a wide range of discharge rates, and is preferred to other systems for starting turbine engines. Silver-zinc gives the highest electrical output per unit weight and volume, but is the most expensive of batteries and has the shortest cycle life. Lead-acid is the oldest of the systems, and is used in helicopter design not requiring main engine starts. It is well-adapted to most conventional electrical circuits requiring moderate discharge rates. Typical comparative characteristics are listed in Table 7-2.

7-3.2 GENERATOR CONTROL BATTERY CHARGING

Engine-driven generators are subject to variations in speed in the approximate ratio of 3:1. Thus, a voltage regulator must be provided in order to maintain constant voltage at high engine speeds. The generator is dropped off the bus by the reverse current relay at all lower engine speeds, and the battery must assume all utilization loads at engine speeds below the cutoff value. Various generator and voltage regulator combinations have been designed that maintain system voltage down to idling speed.

If the generator is current limited, the manner in which the battery is charged will depend upon the utilization load current during a specific time interval, and upon the state o charge of the battery in that period.

TABLE 7-2. TYPICAL CHARACTERISTICS OF 24 V, 34 AH BATTERY SYSTEMS

TYPE	WEIGHT, lb	W-hr/lb AT 2 hr RATE	CURRENT CAPABILITY	W-hr/lb AT 200 A, 80°F	W-hr/lb AT 200 A, 0°F	RETENTION OF CHARGE 14 DAYS AT 80°F	No. OF CELLS	AVERAGE CELL VOLTAGE
NICKEL-CADMIUM	75	10-13	50 x CAPACITY	9	7	93%	19	1.25
LEAD-ACID	78	10-13	20 x CAPACITY	5	3	90%	12	1.9
SILVER-ZINC	34	25-30	6 x CAPACITY	17	16	88%	17	1.4

When the generator output current does not reach the current-regulator limit, the battery will charge at constant voltage. The charging current will be determined by the state of charge of the battery during the time intervals considered. When the generator output current tends to exceed the current-regulation limit, the regulator automatically reduces the generator voltage so as to limit the current to the regulated value. Under these conditions, the battery will be charged at a constant current equal to the difference between the regulated current and the utilization-load current. The increase in battery state-of-charge Δ_c in this case can be computed by the following formula:

$$\Delta_c = \frac{0.8 I_c t_c}{60}\text{, A-hr} \qquad (7\text{-}2)$$

where

I_c = charging current, A
t_c = charging time, min
0.8 = charging efficiency

When total system loading — including battery charging — exceeds the continuous rating of the generator, a current regulator limits the output current to a safe value. This means that the battery must supply the difference between the utilization load current and the maximum generator current.

In any case, when the system is operating, the battery is always either charging or discharging at some rate determined by the demands of the overall utilization system and the output of the generator. The actual conditions of the charging state at any time are controlled by the generator voltage and current regulator.

Battery temperature, especially in nickel-cadmium batteries, should be monitored to prevent overheating, which reduces capacity. A combination of high battery temperature (i.e., in excess of 150°F) and overcharging at constant voltage can result in a condition called "thermal runaway". This is an uncontrollable rise in battery temperature that ultimately will destroy the battery. As the temperature increases, the effective internal resistance decreases, permitting ever-higher currents to be drawn from the constant-voltage source. This in turn decreases the resistance still further, in an ever-increasing spiral.

In general, the over-all condition of the battery should be monitored during charging. Some installations incorporate control systems (battery conditioner/analyzers) which monitor temperature and state-of-charge, constantly analyze the general battery status, and cut the charging process on and off as conditions dictate.

Such systems typically use other approaches to battery-charging, some of which are itemized in Table 7-3, along with their principal operational characteristics.

7-3.3 UTILIZATION LOAD ANALYSIS

The utilization load assignments should be based upon the most demanding conditions likely to be encountered during operation. For example, it should be assumed that the aircraft is operating at night, with landing lights used during takeoff, climb, and landing. Approximate data concerning the duration of each load should be known or assumed. To obtain conservative results from the load analysis, the intermittent load peaks usually are considered to occur concurrently. Despite the short duration, heavy loading, such as during engine starting, can reduce the battery capacity sharply and should be considered.

The load analysis is prepared for an arbitrary set of operating conditions. Too severe a set of conditions would overburden the power system during most operations. An overly optimistic choice of conditions would limit the usefulness of the aircraft. The

TABLE 7-3. ALTERNATIVE CHARGING METHODS

METHOD	PRINCIPLE OF OPERATION	EFFECT ON BATTERY PERFORMANCE		CHARGER CONSIDERATIONS	SYSTEM CONSIDERATIONS
		MAINTENANCE	OPERATIONAL CAPABILITY		
TRUE CONSTANT POTENTIAL	BATTERY CHARGED AT CONSTANT VOLTAGE, INITIAL CHARGE CURRENT LIMITED ONLY BY SOURCE AND LINE IMPEDANCE	MAX ELECTROLYTE LOSS EACH CYCLE	FAST RECHARGE POSSIBLE	SIMPLE DESIGN	DOES NOT REQUIRE CELLS MATCHED CLOSELY
CONSTANT POTENTIAL/ CONSTANT CURRENT	BATTERY CHARGED AT TEMPERATURE-COMPENSATED VOLTAGE, CURRENT LIMITED INITALLY BY BATTERY CONSIDERATIONS	ELECTROLYTE LOSS EACH CYCLE	FACILITATES QUICK RECHARGE	REQUIRES BATTERY TEMPERATURE SENSOR REQUIRES WIDE RANGE OF POWER HANDLING CAPABILITIES REQUIRES GOOD MATCHING OF CHARGER VOLTAGE TO BATTERY CHARACTERISTICS TO AVOID THERMAL RUNAWAY OF NICKEL CADMIUM CELLS	DOES NOT REQUIRE CELLS MATCHED CLOSELY
MULTI-LEVEL CONSTANT CURRENT	BATTERY CHARGE REDUCED AS STATE-OF-CHARGE INCREASES	ELECTROLYTE LOSS EACH CYCLE	REQUIRES LONGER RECHARGE TIME THAN CONSTANT POTENTIAL	REQUIRES BATTERY TEMPERATURE SENSOR REDUCED POWER CAPABILITY AS OPPOSED TO CONSTANT POTENTIAL METHOD DIFFICULT TO DETERMINE OPTIMUM CHARGE TERMINATION CONDITION	REQUIRES CELLS MATCHED CLOSELY FOR CAPACITY
PROGRAMMED PEAK CHARGE (PPC)	CHARGING CONTROLLED OR PROGRAMMED SO THAT BATTERY RETURNS TO FULLY CHARGED CONDITION ONLY PERIODICALLY. FREQUENCY DETERMINED BY ACCURACY OF STATE-OF-CHARGE SENSING DEVICE, WHICH MUST BE RESET PERIODICALLY TO NULL OUT ERRORS	ELECTROLYTE LOSS ONLY WHEN BATTERY RETURNS TO FULLY CHARGED CONDITION	FAST RECHARGE CYCLE POSSIBLE WHEN BATTERY NOT FULLY CHARGED	REQUIRES BATTERY TEMPERATURE SENSOR REQUIRES INTEGRATING DEVICE FOR A-hr INPUT-OUTPUT MONITORING REQUIRES COMPENSATION FOR CHARGE EFFICIENCY AND POSSIBLY STANDBY LOSSES	REQUIRES CELLS MATCHED CLOSELY FOR CAPACITY
PULSE CHARGING	CHARGER PROVIDES REVERSE CURRENT PULSE AFTER EACH CHARGING PULSE TO CAUSE DEPOLARIZATION OF BATTERY PLATES TO ALLOW BETTER ABSORPTION OF CHARGING CURRENT	ELECTROLYTE LOSS EACH CYCLE	SHOULD REDUCE CHARGE TIME SINCE HIGHER CHARGE RATES CAN BE USED AS COMPARED TO CONSTANT POTENTIAL	REQUIRES HIGH POWER CAPABILITY	REQUIRES CELLS MATCHED CLOSELY FOR CAPACITY POSSIBLE EMI PROBLEM POSSIBLY REQUIRES BATTERY REDESIGN FOR MOST EFFECTIVE UTILIZATION
PULSED CONSTANT	CHARGE CONSISTS OF CURRENT PULSES WHOSE PEAK VALUES MAY BE AS HIGH AS 500 A, WHILE CONTROLLING THE DUTY CYCLE TO OBTAIN THE DESIRED AVERAGE VALUE OF CURRENT	ELECTROLYTE LOSS EACH CYCLE	SHOULD REDUCE CHARGE TIME AS COMPARED TO CONSTANT POTENTIAL	REQUIRES HIGH POWER CAPABILITY	REQUIRES CELLS MATCHED CLOSELY FOR CAPACITY POSSIBLE EMI PROBLEM

choice is, necessarily, a compromise. For most aircraft, the following is suggested:

1. Battery operating temperature, 0°F
2. Duration of flight
3. Night operation.

See Fig. 7-11 for a sample set of utilization loads based on a typical mission profile.

7-3.4 HEAVY CURRENT STARTING REQUIREMENTS

Some aircraft engines have starting characteristics that place severe loads upon batteries, particularly in extreme low-temperature environments, or where the battery is not used for long periods. Under such conditions, not only is it more difficult to start the engine, but the battery itself is less active electrochemically, causing the internal resistance to increase greatly.

Additional energy must be incorporated in order to insure a reliable system. This can be accomplished by the use of parallel-series connected batteries, so as to provide a marked increase in the available voltage

EQUIPMENT	No. OF UNITS	AMPERES PER UNIT	OPERATING TIME, Min	CURRENT, A													
				STARTING	TAXIING				TAKEOFF AND CLIMB			CRUISE			LANDING		
				0.5 Min	0.5 Min	1.5 Min	3.0 Min	10 Min	0.5 Min	2.0 Min	15.0 Min	0.5 Min	2.0 Min	60.0 Min	0.5 Min	1.5 Min	3.0 Min
STARTER	1	150	0.5	150													
RELAY-BATTERY	1	0.7	**C	0.7	0.7	0.7	0.7	0.7	0.7	0.7	0.7	0.7	0.7	0.7	0.7	0.7	0.7
INDICATOR LIGHTS	1	0.5	C	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5
INSTRUMENTS	1	0.5	C	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5	0.5
POSITION LIGHTS	3	1.7	C		5.0	5.0	5.0	5.0	5.0	5.0	5.0	5.0	5.0	5.0	5.0	5.0	5.0
INSTRUMENT LIGHTS	10	0.2	C	0.2	0.2	0.2	0.2	0.2	0.2	0.2	0.2	0.2	0.2	0.2	0.2	0.2	0.2
LANDING LIGHTS	1	8.4	5.0		8.4	8.4	8.4		8.4	8.4					8.4	8.4	8.4
MISC. ELECTRICAL CHECK OUT	1	20.0	0.5		20.0				20.0			20.0			20.0		
RADIO RECEIVER	1	4.5	C		4.5	4.5	4.5	4.5	4.5	4.5	4.5	4.5	4.5	4.5	4.5	4.5	4.5
RADIO TRANSMITTER	1	10.5	2.0		10.5	10.5			10.5	10.5		10.5	10.5		10.5	10.5	
TOTAL UTILIZATION LOAD, I_L				152	50.3	30.3	19.6	11.4	50.3	30.3	11.4	41.9	21.9	11.4	50.3	30.3	19.8

**C SIGNIFIES CONTINUOUS

Figure 7-11. Sample Set of Utilization Loads

and current, resulting in a higher power output and greater torque at the starter. The batteries are connected initially in parallel and then switched to series after a predetermined time delay for completion of the start.

7-3.5 MAINTENANCE

Battery maintenance can pose a serious operational problem. Because of the necessity for periodic addition of water to the cell electrolyte, the battery generally is removed from the aircraft on a scheduled basis, and operational delays thus are encountered.

Operating water loss results from the two natural functions of evaporation and electrolytic dissociation. Except at extremely high temperatures, water loss by evaporation can be considered unimportant. Dissociation occurs at a relatively constant rate, and is a function of voltage, current, and temperature.

Higher voltage will increase the overcharge current and gassing rate (Fig. 7-12). Higher temperatures will increase losses by evaporation and will lower the potential at which electrolysis occurs. Higher temperatures also will increase the overcharge current when charging at a constant voltage.

The requirements for battery maintenance must be considered in locating the battery compartment. The location should provide for easy access to remove and install batteries under operational conditions. Battery installation is described in par. 7-7.8.

7-4 VOLTAGE REGULATION AND REVERSE CURRENT RELAY

7-4.1 DC VOLTAGE REGULATION

DC generating systems in most helicopters consist of a starter/generator with a series-field starter, voltage regulator, reverse-current relay, overvoltage relay, field relay, starter relay, and start-control relay. One or more of these components may be supplied to the contractor as Government-furnished equipment (GFE), and thus will establish some areas of the design.

Military Specifications for a generating system — using carbon pile regulators and reverse current relays — include MIL-C-5026, MIL-R-6106, MIL-G-6162, MIL-R-6809, MIL-R-9221, MIL-R-25078, and MIL-R-26126.

7-4.1.1 Voltage Regulator

A voltage regulator designed to incorporate all but the line contactor functions, and having additional functions such as feeder fault protection and field weakening for shunt starters, is available. The Military Specification for voltage regulators of the

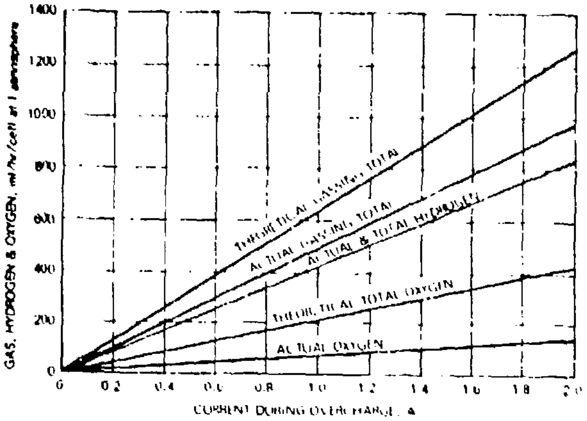


Figure 7-12. Gases Emitted from Nickel-Cadmium Sintered Plate Cell During Overcharge at 70°-75°F

static type is MIL-R-23761. However, as this specification deals only with voltage regulation and paralleling, the system designer must consider the functioning of voltage regulation, generator paralleling, field weakening for shunt starting, line contactor control, engine start control, and protection against reverse current, overvoltage, overexcitation, startup into shorted bus, and feeder fault.

It is recommended that, whenever possible, the static type voltage regulator be used. The regulator procurement specification should include all of the foregoing functions. This will economize on weight and installation time since separate components will not be required.

The switching action of some static voltage regulators has caused application problems. Switching frequencies that are kept constant, and at values above 1000 Hz, generally will be above any engine or generator resonant frequencies. This switching action also can produce some radio frequency noise; but if proper switching speeds and filtering are used, radio noise can be held to a minimum. Locating the regulators close to the generator also will serve in keeping down radiated and conducted interferences.

The use of shunt starters with field weakening is a recent approach to turbine engine starting. The regulators sense the voltage on the starter/generator at the equalizer terminal, and use this variable-current voltage by varying the shunt field current in the starter/generator so as to provide a predetermined armature current. Starter/generators with interpole windings can develop a shunt field current that can result in no-load, overspeed self-destruction. In case of shaft failure, a means must be provided to limit the no-load speed. Some manufacturers provide, as an integral part of the starter/generator, a tachometer generator that enables the regulator to sense speed, thereby terminating the start at a predetermined speed. The primary advantage of shunt starting is weight economy.

A line contactor, designed in accordance with MIL-R-6106 and with the proper rating, can be used to connect the generator to the bus. This contactor also would be used for the starter armature current during starting, and by the reverse current overvoltage, overexcitation, startup into short, and feeder fault functions in disconnecting the generator from the bus.

A relay in the regulator should be used to de-excite the generator in the event of overvoltage, overexcitation, startup into short, and feeder fault conditions.

7-4.1.2 Reverse Current Relays

The reverse current (cutout) relay is designed to connect and disconnect a generator automatically from the bus in a 28 V DC system. The reverse current relay will close when the generator is producing 18 to 28 V and is at least 0.5 V above the bus potential. Depending upon the unit-rating, when the generator voltage drops below bus voltage, the relay will open with a given reverse current. These units are available in 100-, 300-, and 600-A continuous ratings. Depending upon generator capacity, reverse current relays *shall* be sized to match the maximum continuous generator output.

7-4.1.3 Overvoltage Relays

Overvoltage relays are used to remove the generator from the bus by tripping the field relay if the generator voltage exceeds a specified limit.

7-4.2 AC VOLTAGE REGULATION

MIL-G-21480 is a representative Military Specification for AC systems. Highly reliable control units — which provide voltage regulation, field relay control, contactor control, and overvoltage, undervoltage, feeder fault, and underfrequency protection — are available in solid-state versions.

AC generator manufacturers design and build static AC voltage regulators to match their generators. The designer must consider the electromagnetic interference requirement, regulator operation environmental conditions, and the qualification data before choosing a regulator.

7-5 OVERLOAD PROTECTION

7-5.1 GENERAL

The primary objectives of overload protection are to limit malfunction automatically to a single circuit, and to minimize the danger of smoke and fire not only in the components, but also in the wiring.

Overload protection of the equipment should be considered separately from circuit overload protection. In order to obtain maximum safe use of the equipment, any protection required *shall* be integral. If the equipment is not required in order to maintain controlled flight, and maximum equipment use is not necessary, the equipment and circuit protection may be accomplished by the same device, provided that this dual function does not conflict with the basic requirement of protecting the wiring bringing power to the equipment.

The primary intent of circuit protection is to protect the interconnect wiring and the equipment. All wires measuring more than 1 ft from the bus to the load *shall* be provided with some form of circuit protection. Proper selection of the protective device should result in the lowest rating that will not open the circuit inadvertently.

A circuit-protection device should be used at any point in the circuit where the wire size changes, unless the upstream protection provides for the smallest wire. Where more than one circuit is fed from a single circuit-protection device, the protection should be sized to provide adequate protection for the individual circuit. The circuit protection should be located as close to the power source as is practicable in order to minimize unprotected wiring.

7-5.2 OVERLOAD PROTECTION DEVICES

Overload protection devices fall into three categories: circuit breakers, including remote circuit breakers; current sensors; and fuses.

7-5.2.1 Circuit Breakers

Circuit breakers may be actuated either thermally or magnetically. Both types are covered by MIL-C-5809.

7-5.2.1.1 Thermal Circuit Breakers

The actuation of thermal circuit breakers is dependent upon a temperature increase in the sensing element which is produced principally from the load current heating. The thermal element will be affected by external heating or cooling, and must be derated or uprated from calibration temperature to allow for fluctuations in ambient temperature. The majority of the circuit breakers used at the present time are of the thermal type.

7-5.2.1.2 Magnetic Circuit Breakers

Magnetic circuit breakers use a trip mechanism that responds to a magnetic effect rather than to the heating effect of the current carried by the breaker. Magnetic circuit breakers normally incorporate time delay so as to avoid nuisance tripping from current surges of short duration. Although the magnetic circuit breakers are less affected by adverse environment, they are not used to the extent that thermal circuit breakers are because the trip characteristics of magnetic circuit breakers may be affected by their mounting position and vibration.

7-5.2.2 Remote Control Circuit Breakers

A remote control circuit breaker consists of a contactor whose solenoid circuit is controlled by a current-sensitive element, plus a manual-switching and trip-indicating device. The latter unit often consists of a manually operated circuit breaker arranged so as to trip whenever the remote sensor trips. The remote circuit breaker can be utilized best for bus feeders and wiring connected to a single load. Although an approved remote control circuit breaker is not available, Military Specification MIL-C-83383 is being developed for a family of remote control circuit breakers.

7-5.2.3 Current Sensors

A current sensor is used in conjunction with a contactor and a manual-switching or trip-indicating device in order to obtain the actuation of a remote control circuit breaker. The sensor current-sensitive element controls the solenoid of the contactor. The trip-indicating device often consists of a manually operated circuit breaker arranged so as to trip whenever the current limit of the sensor is exceeded. When a circuit breaker is used with a current sensor as a trip-indicating device, the lowest possible rating should be used in order to obtain an immediate indication of when the sensor has tripped. The current sensor can be utilized best when there is a need to control a high-current load, such as in a motor with a low-current control circuit, and to keep the high-current loads to a minimum length.

7-5.2.4 Fuses

A fuse relies upon the melting of the current-carrying element in order to open the circuit when an overload occurs. The four basic fuse types are: normal, time delay, very fast-acting, and current-limiting.

Each type of fuse is available in a variety of characteristics so as to meet various circuit requirements. For a complete listing of characteristics, refer to MIL-F-23419 and MIL-F-5372.

7-5.3 OVERLOAD PROTECTION APPLICATION

Circuit breakers are preferred to fuses. A fuse must be replaced once its current limit has been exceeded, and replacement with an improper size or type is possible. Circuit breakers should be grouped in order of function or usage, and should be labeled by function for rapid identification. They should be located in a protective panel, or covered so as to eliminate the possibility of hazard to personnel or contamination by foreign objects. The placement of circuit breakers in the crew area should be avoided. Only those necessary in order to maintain safe flight should be accessible to the flight crew, as any malfunction must be corrected prior to reinstating the circuit.

The installation requirements for fuses and circuit breakers are detailed in MIL-E-7080.

7-6 ELECTROMAGNETIC INTERFERENCE (EMI/EMC)

7-6.1 GENERAL

Electromagnetic compatibility (EMC) describes the ability of aircraft electronic/electrical equipment to perform in its intended operational environments without suffering or causing unacceptable degradation as a result of unintentional electromagnetic radiation or response, i.e., electromagnetic interference (EMI).

EMI is generated by a varying electrical or magnetic field. As a result, almost any device carrying electrical current is a possible source of interference. Likewise, within a weapon system, each subsystem is a potential victim of a generated interference. In the course of EMC qualification of a weapon system, electrical equipment victim response to interference sources is defined and evaluated. The solution is to control the EMI by reducing the magnitude of interference, isolating the source, or designing the receptor to be less susceptible to the EMI.

To achieve a compatible weapon system, the entire environment, from intercircuit and intersystem to intrasystem, must be considered by following interference specifications and state-of-the-art engineering designs. The same results can be achieved by several means; and the best solution depends upon the judgment of the cognizant engineer, and upon the budget and time allowance of the particular application.

This paragraph outlines the design procedures for the determination of acceptable EMI levels. In addition, the identification of sources of interference and possible methods of compatibility correction or alleviation are discussed.

7-6.2 ACCEPTABILITY REQUIREMENTS

Unacceptable equipment responses to EMI levels are exhibited as aural, video, or equipment malfunctions. In some cases, negative aural response can be acceptable if testing indicates that it does not affect overall mission capability or flight safety.

EMC tests are required to demonstrate control of the electronic interference environment. The detailed requirements for these tests *shall* be specified in the contractor's control and test plan. See par. 9-11, AMCP 706-203, for a discussion of the helicopter system EMC demonstration requirements.

In testing certain equipment — for example, ordnance — for undesirable response, it is necessary to insure that the system functions within a wide safety margin. Military requirements state that an interference signal impressed upon the most critical point of a subsystem must be at least 6 dB (20 dB for explosives) below the level that would cause an undesirable response. Items of equipment that directly affect flight safety, or that cause or lead to a mission abort or to failure to accomplish a mission, are determining factors for the safety margin tests as indicated in MIL-E-6051.

7-6.3 INTERFERENCE SPECIFICATIONS

Military Specifications require that sufficient tests be made of equipment or weapon systems to insure that they are compatible.

Specifications and standards applicable to the design requirements and test procedures necessary to control the electronic interference environment of a helicopter are MIL-B-5087, MIL-E-6051, MIL-I-16165, MIL-STD-454, MIL-STD-461, and MIL-STD-462. In general, the most current specification in force will be the controlling factor for EMC qualification.

7-6.4 INTERFERENCE SOURCES

Electromagnetic interference originates from either natural or manmade sources. Natural sources include atmospheric, precipitation, corona, and lightning discharge noise. Natural EMI varies randomly with time, geographical area of operations, and seasonal conditions. This type of interference generally affects a broad frequency range in the low-frequency band.

Manmade sources of EMI are either broadband or narrowband generators, and they must be evaluated and handled separately.

Broadband interference distributes energy over a wide frequency spectrum, and can be either random or constant in time and amplitude. Typical broadband generators of EMI are motors, switches, power distribution lines, ground currents, pulse circuits, transistors, and capacitors.

Narrowband interference is produced by an oscillatory circuit that contains energy only at the frequency of oscillation or its multiples. The output harmonics of a communication transmitter or its internal oscillators are typical of narrowband EMI. Spurious outputs of a transmitter or receiver can cover a wide range of frequencies and exhibit the characteristics of broadband noise; however, the energy distribution is defined sharply.

Inherent interferences unique to the helicopter can arise from sources such as the rotating members of the engine, drive shaft, and main and tail rotors.

In small- and medium-sized helicopters, radio/radar operation frequently is hampered seriously by a phenomenon called rotor modulation which creates problems especially in VOR/ILS, ADF, and some communication systems. Rotor modulation interferences arise due to the chopping or reflection of the RF signal by the main rotor. The rotor speed and the number of rotor blades combine to pass a given point resulting in the modulation of the arriving RF signal. These distortion perturbations (amplitudes, cancellations, or harmonics) can set up interference patterns that create navigation system noise, error, and needle oscillation. The interference can become critical when integrated flight control systems are used, resulting in helicopter oscillation.

The expanding use of helicopters in a variety of environments has resulted in interference sources not considered previously. These interference effects can downgrade seriously, or even prevent, a particular mission capability. Some interference problems arise from atmospheric field charging potentials, precipitation charging, corona discharge phenomenon (electrons accelerated by a strong electrical field around a sharp point), or triboelectric charging potentials (frictional charging as a result of dissimilar material contact).

Of these sources, probably the most noticeable effect for EMC qualification will be produced by the triboelectric charging of helicopter rotating members (engine, transmission, drive shaft, and rotors).

7-6.5 INTERFERENCE SUPPRESSION

EMI within a subsystem may be divided into four categories:

1. Device signal interference emissions
2. Device susceptibility to such signals
3. Transmission path of interfering signals (solid or wave)
4. Interference time coincidence, i.e., signal presentation during times of receptor susceptibility.

The complexity of the subsystem, and the number and magnitude of the internal interference sources, determine the choice of protective design approaches. Basic approaches to interference reduction within the helicopter or subsystem include:

1. Design of inherently interference-free components
2. Equipment isolation
3. Cable routing
4. Source suppression
5. Signal point containment and suppression.

7-6.5.1 Interference-free Components

All electrical systems *shall* meet the limits imposed by the applicable equipment specification, such as MIL-STD-461. These specifications primarily are concerned with radiation, and with susceptibility to radiation- or conduction-propagated broadband and narrowband interference. Compliance with these specifications represents maximum state-of-the-art interference control. However, the specifications are broad and do not necessarily solve the interference problems arising in all systems. If individual borderline component interference sources are not eliminated, compliance with specification limits does not insure that EMC problems will not develop when the total system degrades from specification limits.

7-6.5.2 Equipment Isolation and Cable Routing

Many EMC problems are solved by positioning electronic equipment or routing cables such that they pick up or radiate minimal interference. Location and orientation are two important parameters in preventive isolation. Because electromagnetic radiation attenuates with distance, antenna location and orientation can prevent or reduce EMI.

Simple shielding of cables is not always effective, due to the magnitude of interfering signals. In such instances, isolation of equipment cables is necessary. Separation of high-level from low-level cables may be required, depending upon design and space allowances. Signal wires and primary power cables may require separate routing even when terminating at a single connector.

If interference is a result of equipment location or cable routing, the following areas should be investigated:

1. Power and control wiring run separately from signal-carrying wires
2. Audio frequency wires run separately from wires of higher frequency
3. Provisions made for the right-angle crossing of sensitive circuit cables

4. Proper wire types used
5. Maximum spatial separation of antennas or interference-producing cables
6. Grouping of noninterfering equipment away from known interference sources.

7-6.5.3 Source Suppression and Susceptibility Reduction

After using physical isolation and cable routing to the maximum extent, additional techniques for EMI source and susceptibility reduction include:

1. Grounding and bonding
2. Cable and equipment shielding
3. Filtering.

Source suppression is the application of appropriate bypassing, decoupling, or filtering at the source of interference or at a point of maximum susceptibility.

7-6.5.3.1 Grounding and Bonding

A fundamental requirement for helicopters is the establishment of a well-bonded, low-impedance ground plane extending to all extremities. A unipotential ground plane prevents EMC problems resulting from unequal ground potentials and ground loop currents, and reduces the possibility of equipment transmitting or receiving undesired energy while insuring that shield and filter applications are effective.

Bonding refers to the method in which various subsystems or structures are connected or integrated electrically and mechanically. Bonding avoids the development of electrical potentials between adjacent metallic parts, and provides homogenous flow of radio frequency currents between subsystems and structures. MIL-B-5087 provides detail requirements for all bonding aspects of airborne systems.

7-6.5.3.2 Shielding

A major area of practical EMI suppression involves the application of component or cable shielding. Effective use of shielding requires investigation of the interference signals, and of the nature of metallic shielding. The question of whether the source or receptor is prevented from radiating or receiving undesired signals deserves equal attention.

Metallic shielding is dependent upon the interfering signal component, e.g., the electrical or magnetic field. The lowest frequency for which a desired shielding is required normally determines the type of shielding material.

High-permeability materials can be used to improve shielding effectiveness for low-frequency, low-impedance magnetic fields. Aluminum, copper, or ferrous materials will provide shielding above audio frequencies (electrical fields).

Shielding used to contain interference is dependent primarily upon the attenuation (absorption) properties of the shield. Reflection loss becomes an important consideration for exclusion of interfering signals.

Discontinuities in a shielded enclosure can provide an entry/exit path for EMI radiation. Ventilation openings, panel meters, access covers, dial shafts, or switches are possible EMI containment problem areas.

Interference coupling of electronic subsystems can be reduced by careful selection of interconnecting cables. Types of interconnecting cables available to the designer include unshielded wire, twisted pair, shielded wire (single or double), twisted shielded pair, and coaxial (single or multiple shield).

The selection of interconnecting cables to reduce interference coupling and audio crosstalk will be dependent upon physical isolation of the operating frequency range, and the power and susceptibility levels. In general, a shielded wire provides protection against electrical fields, while the twisted pair reduces susceptibility to magnetic fields.

To achieve maximum EMI shielding from enclosures and shielded cables, it is necessary to terminate them effectively to the helicopter unipotential ground plane. Both multipoint and single-point ground systems provide certain design features.

Single-point grounding (floating shield) may provide the best approach where the possibility of interference coupling with sensitive low-frequency circuits is a matter of concern. When a shielded cable, in a sensitive circuit, is grounded at both ends for the return circuits, power frequencies in the ground plane can induce audio frequency interference in the signal wires.

When electronic and electrical equipment is distributed over large areas, experience has shown that multipoint grounding is superior for RF frequencies. Multipoint grounding involves shield grounding at both ends of all cables, and at all immediate points where the cable runs through equipment.

Application of proper shielding techniques for interference alleviation should be performed in the following areas:

1. The radiation source or sensitive component should be installed in a properly bonded metallic housing with limited openings.
2. The magnetic field should be directed away from sensitive components or wiring by use of low-reluctance, high-permeability material.
3. Twisted, shielded, or shielded and twisted cable

should be used for AC and DC power circuits in order to prevent coupling of super-imposed EMI noise and transients.

4. Two conductor-twisted and shielded cables should be used for DC signal, control, and audio circuits. Single-point grounding is required.

5. Single- or multiple-shield coaxial cable should be used for RF circuits. Multipoint grounding is required.

6. Continuity of shielded enclosures is necessary.

7. Shields should be routed through connectors.

8. Minimum-length ground returns should be used, and shield insulation from structural members should be insured.

7-6.5.3.3 Filters

Filters are used at the outputs of EMI generating sources in order to prevent EMI signal (broadband or narrowband) interference coupling paths. Types of filters utilized for EMI containment and attenuation include low-pass, high-pass, and band-pass filters, as well as bypass and feedthrough capacitors.

Basic filter parameters include capacitance, inductance, and resistance. Each parameter accomplishes filtering action by a different method; i.e., capacitance by short-circuiting, inductance by open-circuiting, and resistance by dissipation. Filters should suppress only the interfering signals. However, the filter may have an effect upon desired currents necessary to the operation of the equipment. Therefore, an understanding of insertion loss is important to filter applications.

In the application of bypass capacitors, the lead length from the capacitor to ground becomes an important factor. Self resonance nullifies the effectiveness of the filter for signals at frequencies equal to, or greater than, the resonant frequency.

Filter containment of EMI can be effective only if the source can be shielded and isolated from other internal circuitry, thus preventing the interference from being coupled into other wiring or circuitry within a subsystem. Such coupling may conduct spurious energy to external wiring, or radiate directly from other parts of the unit. Proper bonding must be used in order to prevent interference currents in the ground circuit from shunting the filter element.

7-7 ELECTRICAL SYSTEM INSTALLATION

7-7.1 GENERAL

Electrical system installation refers to the installation of electrical and electronic equipment (equipment installation) and wire bundles (electrical installation) in the airframe. This includes electronic components, electrical relays, electrical power generators, wires, coaxial cables, junction boxes, test connectors, etc., but does not include aircrew control panels and instrument panels. Electrical system installation should be in accordance with MIL-E-25499, MIL-E-7080, and as described subsequently.

7-7.2 EQUIPMENT INSTALLATION

During the design of equipment installations, maintainability, reliability, and producibility must be considered from design concept to the production hardware phase. Close attention should be given to the servicing problems that might arise with each particular installation. It is not likely that all electronic components can be made immediately accessible. The service reliability of each must be considered during design of the installation. Factors such as electronic alignment after installation and accesibility to test points must be considered. If equipment is installed in rows, front row components must be capable of being removed quickly to provide accessibility to rear mounted components.

Equipment-mounting hardware should consist of not less than Number 10 screws, except where vibration isolators are used, in which case the box mounting screws should be no smaller than Number 10, with the isolator multiple mounting screws no smaller than Number 8. Care must be taken to insure that mounting screws are not hidden behind flanges and protruding portions of neighboring boxes. For easy accessibility, the straight-in approach should be provided for all mounting hardware.

Equipment installations involving the placement of electrical receptacles facing bulkheads or other obstructions must allow sufficient room for installation of the wire bundle with a bend radius in accordance with MIL-W-5088, as well as room to engage and disengage electrical connectors without damaging the wires. If possible, electrical terminals on boxes should permit the use of a ratchet-drive socket wrench for wiring installation and removal.

Junction boxes must be designed so as to facilitate maintenance and troubleshooting. Access to internal components must be such as to permit easy replacement. The locations of internal components must be identified by permanently attached decals. Foreign-object protective covers must be provided on all junction boxes. On all nonsealed boxes, drain holes must be incorporated at the lowest point. Wiring must be installed neatly, and numbered or color-coded for ease of maintenance.

Relays, resistors, small transformers, etc., must be

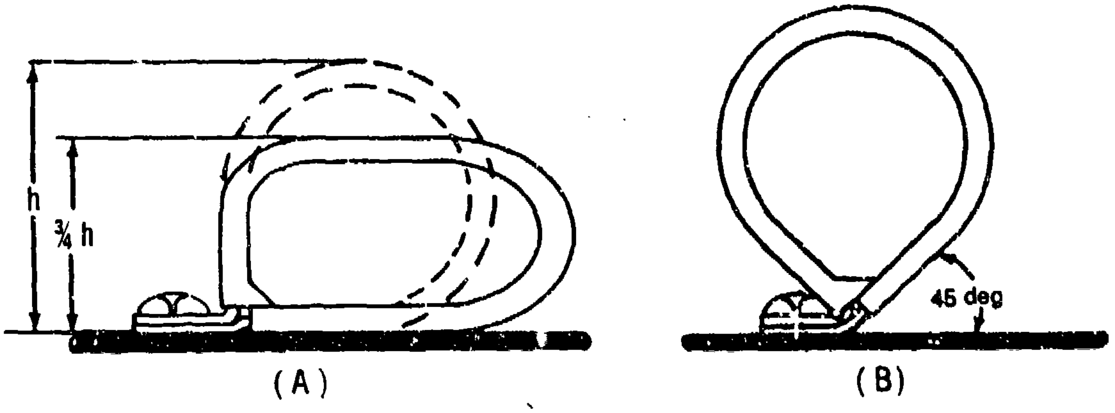


Figure 7-13. Permissible Clamp Deformation

grouped functionally in panels similar to relay panels. All components must be located and identified by means of a decal permanently attached to the panel. Power contactors must be installed so that the contactor case or box is isolated from the airframe structure.

All junction boxes and panels should have power and chassis grounds emanating from at least one of the electrical connectors.

7-7.3 ELECTRICAL WIRE BUNDLES

Basically, there are two types of wire harnesses allowable:

1. Open-wire bundles, where individual wires are tied in bundles and routed through the airframe
2. High-density bundles, where an abrasion-resistant covering is braided, extruded, etc., over the entire bundle.

In either case the wire best suited for the particular application must be used; and, when open-wire bundles are used, the wires *shall* have markings in accordance with MIL-W-5088 and the bundles *shall* be tied at 3- to 8-in. intervals. Lacing *shall* be compatible with the operating environment of the helicopter.

Where high density bundles are used, the bundles must be taped at 8-in. intervals with a thin layer of Teflon tape. An outer abrasion-resistant covering must be braided or extruded over the wire bundle.

Tape is not acceptable as an abrasion-resistant covering except on repair areas or at the ends of a bundle. Tape must never be used as primary insulation. Repairs to high-density bundles should be made by routing a wire external to the abrasion-resistant covering. The external wire must have an abrasion resistant covering. Splices are to be coverd with an abrasion-resistant material, such as Teflon tubing, but should not be covered by the braid or extruded outer jacket of the bundle.

The primary wire bundle clamps should be of an environmentally compatible type. Nylon clamps are permissible in low-temperature, low-vibration, easily accessible areas. Plastic clamps are not to be used for wire bundle support in areas where a clamp failure could allow the wire bundle to chafe on sharp edges or to interfere with controls. The preferred orientation of all wire bundle clamps is with the bell (loop) down. The bell should not be turned upward if the wire bundle weight threatens to deform the clamp.

Clamps of the MS 21919 type may be deformed in order to meet special installation problems by flattening the clamp bell, as in Fig. 7-13, to a height no less than 3/4 of the original bell height. The mounting ears may be bent, but not more than 45 deg, as shown in Fig. 7-13.

7-7.4 TERMINAL STRIP INSTALLATION

Terminal strips should be MS 27212 or MIL-T-81714 with MS 18029 covers. Terminal strips *shall* be installed as shown in Fig. 7-14, with the mounting holes isolated, for example, by filling with MIL-A-46146 Type I sealant to prevent short circuits to ground. MS 25227 insulating strips may be used in lieu of potting; however, an additional nut must be installed between the insulating strip and the bottom terminal so that there is no resilient material in compression with the terminals.

A maximum of four terminals *shall* be used on one stud. MS 25266 bus bars may be used between studs to interconnect terminals. When terminals are exposed to the weather — such as in wheel wells — terminals and studs *shall* be brushed with phenolic resin varnish.

The wire bundle *shall* be tied to a terminal at each breakout. There *shall* be at least one wire identi-

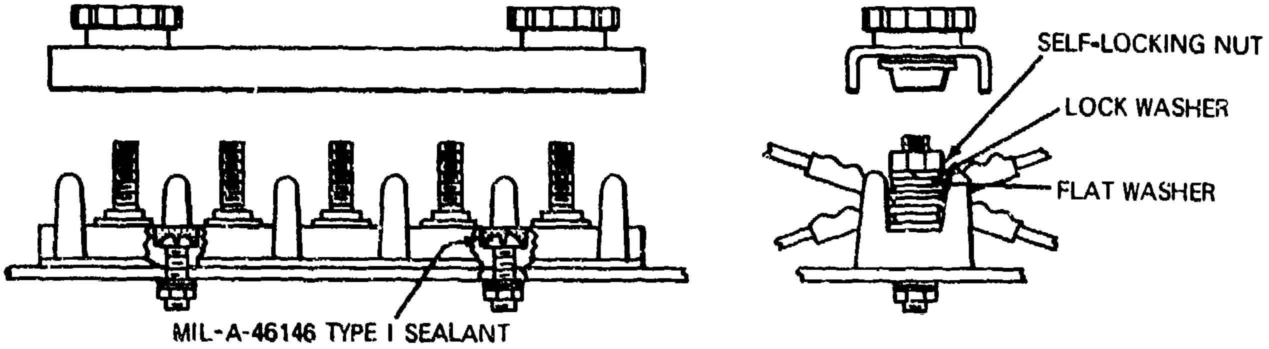


Figure 7-14. Terminal Strip Installation

fication number visible on each wire without cutting ties. Lacing (or tying) *shall* be done with single ties. Continuous lacing *shall* be permitted only in junction boxes and panels.

The end studs used for attaching the MS 18029 terminal covers cannot be used for electrical purposes. If two electrical terminals with mounting hardware are placed on the end studs, the self-locking feature will not engage in the terminal cover nuts.

See par. 7-8.2.1 for a further discussion of terminal blocks.

7-7.5 ENGINE COMPARTMENT WIRING

The two major installation hazards encountered in engine compartment environments are heat and vibration. Special attention should be paid to the high-vibration environments of engine enclosures. Wire gage *shall* be a minimum of 20 in order to reduce strand fatigue breakage. Wire bundle clamps *shall* be spaced in close proximity so as to prevent wire vibration between clamps and possible resultant breakage. Crimp-type contacts *shall* be used in order to eliminate strand vibration breakage due to solder capillary action.

Wire bundles in low-temperature areas (200°C or lower) of the engine compartment may be in accordance with par. 7-7.3; in higher-temperature areas and on the engine itself, open wire bundles of wire rated at 260°C *shall* be used.

Particular care *shall* be taken to route all wire bundles away from sharp edges, and around equipment in the engine area to allow extra room for vibration and for structural expansion and contraction due to ambient temperatures and engine thrust. Wire bundles *shall* be routed and clamped well out of the way for engine change, and design *shall* take into consideration the use of any necessary installation/removal ground-handling tools. Fire detector elements *shall* be routed, and securely clamped into position, to eliminate crush possibilities during engine change.

7-7-6 DOOR HINGE WIRE BUNDLE ROUTING

Electrical components mounted on access doors will require routing the wire bundles over the door hinges. The wire bundles *shall* be routed so that they twist instead of bend, i.e., the bundle *shall* be routed parallel to the hinge for a distance sufficient to allow the bundle to twist. Consideration should be given to using Teflon-cushioned clamps at the twist points to provide added bundle mobility. Added abrasion resistance at the hinge, in the form of vinyl or Teflon tubing may be required.

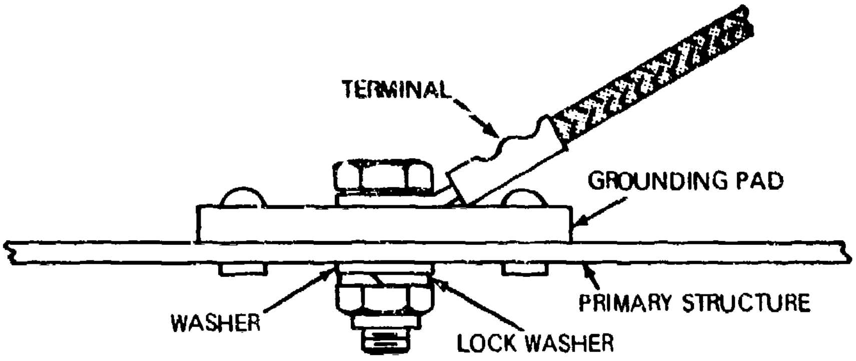
Wire bundles that are exposed to weather and abrasion when doors are opened during flight, or during ground servicing, *shall* be protected by extra covering (such as braiding or tubing). Weather-exposed braiding *shall* extend into the connector back shell clamp, but, because of the water-wicking properties of the braid, should not extend into potting or connector waterproofing.

7-7.7 WIRING TO MOVING COMPONENTS

Special attention is required when it is necessary to route wiring bundles to components such as actuators, missile launchers, or electronic components that move during use or storage. These bundles usually flex a number of times and are critical in their operation.

The installation should be designed as follows:

1. The wire bundle *shall* be clamped firmly to the moving component so that no movement of the wire takes place at the connector or terminal.
2. The wire bundle *shall not* be under tension at any point in the movement of the equipment.
3. The wire bundle *shall* be clamped firmly to the fixed structure at a position where if there is any motion, the wires will twist and not bend.
4. The attach point of the fixed structure must be, whenever possible, at the center of the arc formed by the moving equipment.
5. If the fixed point cannot be at the center of the



NOTE: BOND ALL PARTS PER MIL-B-5087

Figure 7-15. Typical Connection to Grounding Pad

moving arc, a loop must be made to take up the slack in the wiring. This loop must be of sufficient length to insure that the wire bundle is never under noticeable tension. This loop must be self-supporting and self-forming. The self-supporting feature can be assisted by a preformed spring steel wire woven in, or attached to, the wire bundle.

6. Attention *shall* be paid to chafing of the wiring. Added protection, such as vinyl or Teflon tubing, may be required on the slack wire bundle. Vinyl sleeving is not to be used as a substitute for good engineering. Protective tubing should not ride on sharp edges of structure.

7-7.8 BATTERY INSTALLATION

Batteries *shall* be installed so that they are readily accessible from the outside of the helicopter. The aircraft connector *shall* be of the quick-disconnect MS 25182 type in accordance with MIL-C-18148, and *shall* be accessible without moving any equipment or reaching around any obstruction.

The battery compartment must be located in such an area that battery gas and fumes will not enter the cockpit or cabin. The battery compartment *shall* be painted with a material resistant to the electrolyte used in the battery. There *shall* be no oxygen, hydraulic, or flammable lines in the battery compartment.

The battery cables *shall* be clamped and protected against chafing during installation and removal of the battery. The battery ground cable *shall* be attached to primary structure that is heavy enough to carry short-circuit current without damage. A grounding pad, as shown in Fig. 7-15, may be used to increase electrical current capacity.

7-8 COMPONENTS

7-8.1 WIRE

The choice of wire should take into consideration not only the electrical requirements of the wire, but also the environment in which the wire must operate. The electrical requirements can be satisfied by the wire current capability; however, the environmental requirement may be compatible with the wire defined in only one appropriate specification. Environmental compatibility will vary depending upon the type of insulating material used. The designer *shall* assure that the finished diameter of the wire selected is compatible with the wire sealing ranges of the connector used and compatible with the connector insertion/extraction tool.

7-8.1.1 Wire Insulating Materials

7-8.1.1.1 Polyethylene

Polyethylene is a commonly used dielectrical material. It is excellent for high-frequency applications. However, because of its physical properties, it has definite limitations as an insulating material. Polyethylene possesses low abrasion resistance; the maximum safe operating temperature is only 80°C, and it will burn freely in the presence of an open flame.

7-8.1.1.2 Polyvinylchloride

Polyvinylchloride (PVC) has physical properties that surpass those of the basic polyethylene. It possesses greater abrasion resistance, higher operating temperature limitations, and increased resistance to flame. However, the molecular imbalance of PVC precludes its use at high frequencies, although it is excellent in low-frequency applications where resistance to moisture, flame, oil, and many acids and alkalines is important.

7-8.1.1.3 Fluorinated Ethylene Propylene

Fluorinated ethylene propylene (FEP) demonstrates excellent electrical stability over a temperature range of −65° to +200°C, and is suitable for ultra-high-frequency applications.

7-8.1.1.4 Polychlorotrifluoroethylene

Polychlorotrifluoroethylene, more commonly known as KEL-F, combines many of the advantages of Teflon with a superior resistance to abrasion, thus enabling it to be used as a thin-walled insulation without any outer covering or mechanical protection. This material is rated for continuous operation through the temperature range of −65° to +150°C.

7-8.1.1.5 Polyhexamethylene-adipamide

Polyhexamethylene-adipamide is a readily extrudable polyimide, better known as by its family name of nylon. Because of its relatively poor electrical characteristics, it rarely is used as a primary insulation on wire. However, it makes an excellent outer coverings when applied over vinyl insulation. Extruded nylon jackets are tough and resistant to abrasion and oil, and have a tendency to increase the temperature stability of the primary insulation.

7-8.1.1.6 Tetrafluoroethylene

Tetrafluoroethylene (TFE), better known as Teflon, is an excellent electrical balance and, therefore, is well suited for high-frequency applications. TFE offers exceptional electrical, chemical, and thermal properties not available in any other wire insulation material. TFE insulation is rated for continuous operation at 200°C, but remains flexible at cryogenic temperatures.

7-8.1.1.7 Dimethyl-siloxane Polymer

Better known as silicone rubber, dimethyl-siloxane polymer is finding widespread application as a wire insulation because of its good high-temperature characteristics and low-temperature flexibility. It will withstand 200°C continuously, and can withstand as much as 300°C for short intervals. However, in the presence of flame, silicone rubber will burn to a non-conductive ash, which, if held in place, could function as an emergency insulator. Its abrasion resistance is improved greatly by the addition of a saturated glass braid. Unlike vinyls, polyethylene, and nylon, silicone rubber is a thermosetting plastic.

7-8.1.2 Military Wire Specifications

The Military Specifications for aircraft wire are too numerous to cover in detail. However, a brief description of some of the more commonly used types of wire and of the specifications defining them is given to assist in selecting the specification that satisfies the general requirements.

MIL-W-5086 covers PVC-insulated, single-conductor hookup and interconnecting electrical wires made with tin-coated or silver-coated conductors of copper or copper alloy. PVC insulation may be used alone or in combination with outer insulating or protective materials. It is a good general purpose wire, and is available in voltage ratings from 600 to 3000 V and a temperature range of −55° to +110°C. The wire construction of this specification contains nylon jackets for increased mechanical toughness and resistance to fuels, solvents, and hydraulic fluids.

MIL-C-7078 covers single-conductor and multi-conductor shielded wire. The basic wire in this specification is MIL-W-5086 and MIL-W-81381.

MIL-W-16878 covers wire designed for internal wiring of meters, panels, and electrical and electronic equipment, and requires that such wire have minimum size and weight consistent with service requirements. The temperature rating of wire included in this specification ranges to 260°C, with potential ratings of 250 to 3000 V. This wire is primarily a hookup wire, but it may be used for wiring electronic equipment in protected areas of the aircraft.

MIL-C-22759 covers fluorocarbon-insulated, single-conductor electric wire made with tin-coated, silver-coated, or nickel-coated conductors of copper or copper alloy. The fluorocarbon insulation of these wires may be polytetrafluoroethylene, fluorinated ethylene propylene (FEP), or polyvinylidene fluoride. The fluorocarbon may be used alone, on in combination with other insulation materials. This wire is available in a temperature range of 200°C to 260°C, and voltage ratings of 600 to 1000 V.

MIL-W-7072 covers low-tension, insulated, single-conductor, aluminum wire for aircraft electrical power distribution systems. Aluminum wire usually is used where an appreciable weight saving can be realized.

MIL-W-81044 covers a variety of construction suitable for airframe and electronic hook-up wire, including light, medium, and heavy wall insulation thickness and tin- and silverplated-copper conductors. These wires are rated to 500 V over a temperature range of −55° to +150°C. The insulation consists of crosslinked polyvinylidene fluoride. Improved thermal stability is realized through molecular crosslinking of both materials by the high-energy electronic beam process. These constructions provide significant space and weight savings while retaining excellent abrasion resistance.

MIL-W-25038 covers single wire for electrical use under short-time emergency conditions involving exposure to flame and temperatures of up to 2000°F. This wire is intended for use in circuits where it is necessary to maintain the electrical integrity of the insulated conductor for 5 min in a 2000°F flame with the operating potential not exceeding 125 V.

7-8.2 FITTINGS

Fittings cover a broad area, and include any fixture attaching to a wire. Two basic fittings are terminal strips and connectors.

7-8.2.1 Terminal Strips

Terminal strips are used where there is a requirement for a junction of two or more wires. Terminal strips also may be used as disconnects in applications where it is impractical to use a connector, or to simplify assembly and maintenance procedures.

The standard terminal strip is the MS 27212 or MIL-T-81714 which consists of a series of threaded studs retained in a plastic insulating strip. Each terminal stud will accommodate a maximum of four terminals; however, a bus bar may be used between studs in order to allow for more than four wires having a common junction.

The new NAS standard terminal strip, which consists of series of modules retained between mounting rails, offers many advantages over the old style MS terminal strip. MIL-T-81714 covers environmental feedthrough and nonfeedthrough terminal strips. For new designs qualified parts *shall* be in accordance with MIL-T-81714. This type of unit is similar to an electrical connector in concept in that it uses a crimp pin, and an insertion-extraction tool for installing the wires.

Each terminal strip requirement must be evaluated individually in order to determine which of the types can be used best.

7-8.2.2 Connectors

The ideal situation, as far as reliability is concerned, is to have continuous conductors throughout the entire circuit. However, this usually is not possible; interconnects must be added to facilitate assembly and maintenance. The designer must select the connector that best combines high-performance factors with capabilities for meeting environmental requirements. Thus, the selection of a connector for a specific application will involve a compromise.

MIL-C-5015 covers circular electrical connectors with solder or removable crimp contacts, and accessories such as protective covers, storage receptacles, strain relief clamps, and potting molds. These connectors are for use in electronic, electrical power, and control circuits. They have threaded couplings, and may require safety wiring in order to eliminate inadvertent decoupling in high-vibration areas.

MIL-C-26482 covers environmental-resistance, quick-disconnect, miniature electrical connectors with solder or removable crimp contacts and accessories. These connectors have bayonet couplings and do not require safety wire.

MIL-C-83723 covers an environmental-resisting family of miniature, circular, electrical connectors. These connectors may have threaded or bayonet couplings.

MIL-C-28748 covers rectangular rack and panel and electrical connectors with nonremovable solder contacts and removable crimp contacts.

MIL-C-39012 covers the general requirements for radio frequency connectors used with flexible coaxial RF cable.

The designer *shall* make every effort to select only connectors that provide common termination methods; i.e., common contacts, common back hardware, and common assembly methods and tools — using MIL-STD-1353 as a guide.

7-9 LIGHTNING AND STATIC ELECTRICITY

7-9.1 GENERAL

The proper functioning of electronic systems is taking on increased importance in mission effectiveness and flight safety with the development of electronically controlled, automatic flight and engine controls. Thus, the common occurrence of total electrical system failure from lightning strikes is no longer acceptable and a higher degree of static electricity and lightning protection must be provided for the helicopter in order to assure reliable, safe, and effective operation over its operational lifetime.

One lightning strike can be expected to occur on a helicopter approximately every 2500 flight hr (Ref.1), depending upon aircraft zone of operation, mission, normal flight altitudes, susceptibility, etc. Minor to serious structural damage can result in cases where protection is not provided.

New materials, such as polyurethane paints, have many advantages relative to corrosion protection;

but their excellent dielectric characteristics also can introduce serious static electricity problems. The high dielectric strength of the painted surface permits the buildup of 5000 to 50,000 V from friction charging of the surface, which may be followed by puncture of the base metal and accompanied by an energy release in tens of joules. This can cause precipitation static or streamer radio interference, and — if the paint is covering an electrical component, such as an engine inlet heating grid — also can result in a short circuit of the element. This often is followed by burnup, as a result of energizing of the initial spark by the power system, with resultant major damage.

Possible internal problems with high-quality dielectrics include the charging of fluid lines from the liquid flow and the charging of painted internal fuel tank walls from spray electrification or sloshing.

7-9.2 LIGHTNING PROTECTION FOR ELECTRONIC SUBSYSTEMS

The designer of lightning protection for helicopter electronic subsystems should make maximum use of the metallic frame and skin for shielding purposes. Specific lightning protection, or lightning-resistant designs, should be provided at the major lightning entry points. These include main rotor and tail rotor blades, antennas, navigation lights, pitot-static tubes, active electrical discharger probe heads, and any other electrical components exposed on the exterior of the helicopter. In addition, because of the generally reduced shielding of helicopter frames and skins (compared with fixed-wing aircraft), greater considerations must be given to magnetic and electrical field penetrations into the vehicle interior.

Where all other factors are roughly equal, it is advisable to use mechanical primary flight controls as engine and rotor controls and to use the electronic systems primarily for trim or management controls. Electronic surge suppressors of various types, such as gas or zener diodes and simple capacitors, may be used on critical circuits for suppressing the residual voltage surge (which can penetrate despite the external lightning protection design), particularly if the electronic systems require very-low-voltage protection.

In summary, the preventive design approaches are:

1. Principal lightning protection efforts should be directed toward blocking electromagnetic energy entrance through electromagnetic windows such as navigation lights and antennas.
2. Use of electronic systems for primary flight controls should be avoided. Use should extend only to trim or management.
3. Surge suppressors should be used where required, either because of large surge voltages that cannot be reduced at the entry point or for low-signal-level circuits that require low-level protective devices.
4. Simple lightning test facilities should be used to permit quick evaluation of component performance.

Untested lightning protection designs often have proved to be not only ineffective, but sometimes more dangerous than the components they were intended to protect.

Lightning protection through geometrical configuration control of external components, such as antennas and navigation lights, has proven to be one of the most effective methods of preventing lightning penetration into the aircraft. For example, tests of navigation and collision light designs have shown that a 1-in. change in a cover screw position can reduce the resultant lightning damage from total destruction of the element, with major energy penetrations into the vehicle interior, to negligible physical damage resulting in voltage pulse amplitude reductions to a few hundred volts. Thus, geometrical control of all external components for lightning protection purposes generally is the most economical approach, in terms of weight and cost. Typical entry points requiring protection design effort are shown in Fig. 7-16.

Earlier HF and UHF antennas of the voltage-fed type constituted one of the principal electromagnetic windows through which lightning energy could enter the vehicle interior. To offset a possible total electrical system loss, these units often can be replaced with shunt-fed antennas, which are inherently grounded designs in which the lightning energy essentially is channeled into the external vehicle skin, with only residual high-voltage, low-energy pulses entering the electronic systems. HF lightning arresters are available commercially for HF antennas, and their effectiveness in preventing both structural and radio equipment damage has been demonstrated in their use on commercial jet airliners during millions of flight hours.

Other external components, such as pitot-static heads and active discharger probe heads, require typical electronic system protection approaches. The pitot-static heads can be protected effectively by conventional electrical system protective devices such as zener diodes or gas diodes; however, the high-voltage active discharge probe heads require more extensive protection development because of high operating voltage levels.

For electrical surge suppression, many types of devices are available commercially — including zener diodes, gas tubes, simple capacitors, spark gaps, and

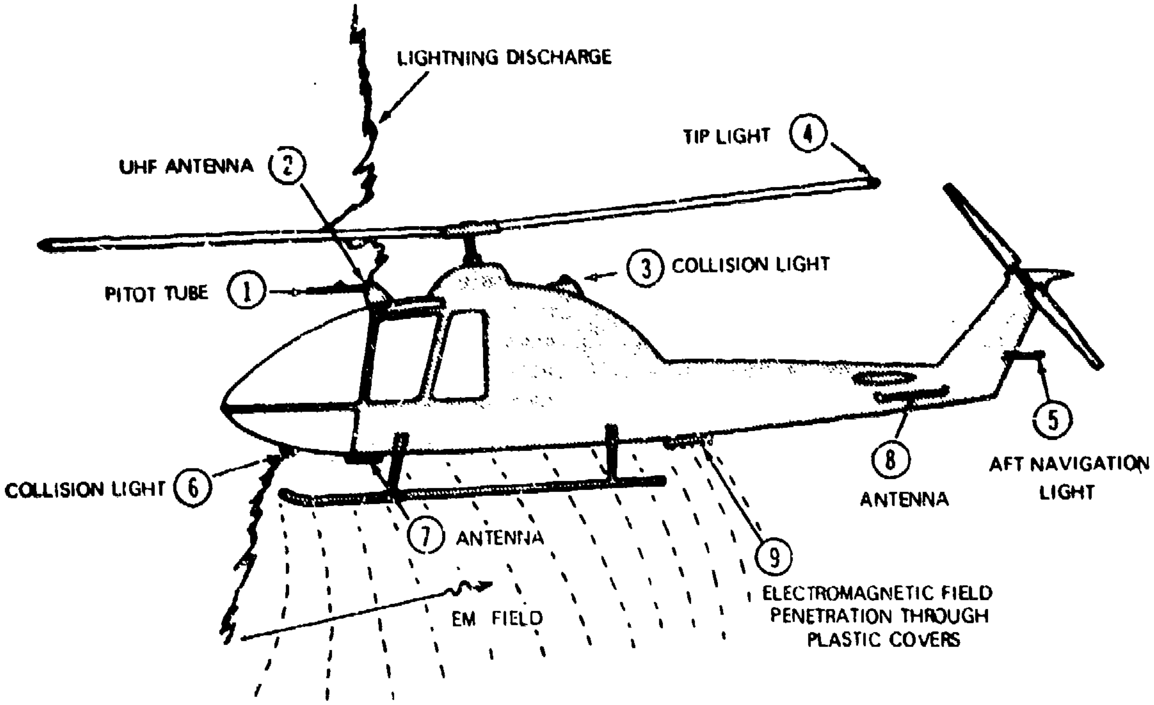


Figure 7-16. Typical Lightning Electrical Circuit Entry Points

silicon controlled rectifiers (SCR). The principal problem in their application is the selection of the right device, or combination of devices, for the particular equipment being protected. As an example, for antenna front ends, semiconductor devices have a major shortcoming, they introduce cross-modulation through their inherently nonlinear transfer characteristics. Simple gas tubes present only a light additional capacitive load on the front end, and thus provide more suitable protection for this application. For other types of components — such as electronic control systems, where nonlinearity may not be as important as is obtaining sufficient low-voltage protection levels — zener diode protection devices may be more suitable.

7-9.3 STATIC ELECTRICITY

The requirements for control precipitation charging are much more severe for helicopters than for fixed-wing aircraft because of the cargo-handling requirements. Potentials that would be acceptable on fixed-wing aircraft — 20,000 to 30,000 V, which is well below the radio noise threshold of the vehicle — can represent a serious shock hazard to ground personnel unloading cargo from helicopters, and possibly can cause ignition of ordnance or fuels. Several general approaches have been suggested and carried to various degrees of development, including use of active dischargers in which the electrical field from the aircraft is measured and an opposite charge is applied to the vehicle, the use of passive wick-type discharge devices at the blade tips, and the use of conductors hanging from the helicopter to the ground to discharge the vehicle before the ground crew contacts the load.

The active dischargers suffer from several disadvantages, including indicating a charge in external electrical crossfields when using single-head field meters when no charge actually is present on the helicopter and thereby charging the vehicle with the protection device. This can be prevented by using dual field meters, one above and one below the vehicle. However, space-charge shielding of the field meter sensing head can occur from a recirculating charge during hover. It generally is acknowledged that the use of active dischargers, in spite of the shortcomings, is advisable, particularly when ground handling is frequent.

The passive wick dischargers located on the blade tips have the advantage of simplicity, but suffer from the fact that substantial potential is required on the vehicle before they begin discharging, i.e., they do not bring the vehicle potential down to zero. This still permits sufficient potential to give shocks to the ground-handling crew. The technique of using a conducting cord from the vehicle to the ground, and permitting it to contact the ground before the ground crew handles the load, has the disadvantage of the cord being whipped by helicopter downwash, and will not necessarily hold the vehicle potential down continuously while the ground crew is in contact with the load.

The other major problem with external static electricity on helicopters is radio interference. The complexity of the problem is caused by: the variety of charge-generating mechanisms, of noise-generating mechanisms, and of coupling modes into the communication systems; the difficulty in separating the effects from internally generated equipment interference; and the differences of effects upon different types of equipment.

The basic method of controlling radio interference includes:

1. Avoidance of all electrically floating external sections on the aircraft
2. Use of some type of active or passive dischargers in order to reduce the potentials on the vehicle under friction electrification conditions
3. Location of antennas in areas where the DC electrical fields are minimized under thunderstorm crossfield conditions
4. Use of radio-interference-resistant antennas
5. Coating of all external dielectric surfaces subject to particle impingement with resistive paints so as to prevent streamer interference, particularly over plastic sections where the interference coupling is most severe.

In addition to the external problem, which is complicated by the difficulty of proper identification of the interference source, internal static electricity problems involve the fact that helicopters often are engineered by designers who possess little knowledge of the hazards posed by electrical interference of fuel systems. As an example, plastic tubing often is considered for fuel jettison tubes. Friction electrification of the plastic surfaces of these tubes can ignite the fuel vapors, particularly when the fuel tanks and jettison tubes are nearly empty. As a solution to this problem, it has been suggested that all dielectrics with a resistivity of higher than 10^9 ohm-cm be carefully considered for aircraft use. Thus, the use of such materials would be permitted, but freedom from static electricity hazards would have to be assured for each specific installation.

7-9.4 LIGHTNING AND STATIC ELECTRICITY SPECIFICATIONS

There are a number of Military Specifications containing references to surges and protections. MIL-STD-704 defines the acceptable limits of transients on electrical power systems. MIL-A-9094 specifies the requirements for aircraft lightning arresters for HF antennas, and it probably will be extended to include all surge penetration into vehicles. MIL-E-6051 is the electromagnetic compatibility specification, and refers to permissible EM pulse limits. MIL-B-5087 is the standard military bonding specification and covers test current waveforms, bonding jumper sizes, protection of canopies, and lightning-induced surge penetration limits. There are other specifications with reference to lightning, but those listed herein are the principal ones with specific data on waveforms, test arrangements, and requirements.

REFERENCE

1. *Rotary Wing Aircraft Susceptibility*, DN 74A, AFSC DH 1-4, 10 January 1972.

CHAPTER 8
AVIONIC SUBSYSTEMS DESIGN

8-1 INTRODUCTION

8-1.1 GENERAL

Avionics (aviation electronics) is defined as the application of electronic techniques to accomplish such functions as communication, navigation, flight control, identification, sensing, surveillance, and target designation. The avionic subsystems will be defined by the detail specification. This chapter will discuss design requirements to interface these subsystems with the helicopter.

From an operational viewpoint, the helicopter avionic complement can be subdivided into (1) the basic helicopter configuration, and (2) the special-mission equipment.

The basic helicopter configuration as discussed in this handbook is limited to the space, weight, and power requirements of the minimum electronics necessary in order to provide the basic mission capability for a specific class of helicopter. The helicopter classes include light observation, utility, tactical, medium and heavy transport, and external heavy lift transport.

Special-mission equipment is defined as the additional electronics — beyond the basic communication, navigation, and identification functions — required to accomplish specific missions such as IFR flight, night operation under reduced visibility conditions, target detection and recognition, target designation, and integrated fire control, such as is found in gunships and tactical aircraft weapon systems.

Avionic procurement, installation, and qualification, along with bench, preflight and flight test requirements, are defined by Military Specifications such as MIL-STD-454, MIL-STD-461, MIL-STD-462, MIL-STD-704, MIL-B-5087, MIL-W-5088, MIL-E-5400, MIL-E-6051, and MIL-I-8700.

The first step in avionic system design is to determine the proper location for each individual system. Because avionic systems are made up of several subsystems and components, it is mandatory that the total helicopter system and its environmental capability be known. Every avionic system component has temperature and vibration limitations. Before any placement or location is determined, the inter/intra-system compatibility of the location must be determined to insure that heat and vibration will not have a detrimental effect upon the performance of the equipment. In addition, electromagnetic compatibility/interference (EMC/EMI) must be considered.

In general, the following design sequencing must occur.

1. Determine the avionic requirements.
2. Determine the avionic characteristics.
3. Construct a block diagram of the interface to the electrical system.
4. Develop a basic layout of the system in the aircraft for mock-up purposes.
5. Develop a schematic wiring diagram for the system.
6. Develop an interconnect diagram.
7. Develop a parts list.
8. Develop a wire list.
9. Develop an electrical load analysis.
10. Complete a preliminary EMC/EMI analysis plan for the system.

8-1.2 ELECTROMAGNETIC COMPATIBILITY PROGRAM

Interference generated by items of electrical/electronic equipment installed in close proximity, as in a typical helicopter system, easily can result in an intolerable interference level that could reduce seriously the usefulness of airborne equipment, or might even render it inoperative. As defined in par. 9-11.2, AMCP 706-203, the prime contractor *shall* establish an overall integrated EMI compatibility program for the helicopter.

EMC is achieved by application of an optimum combination of managerial and technical resources from the earliest design stage through the final product or operational feasibility demonstration stage. Accordingly, an EMC program *shall* be established that will:

1. Insure the efficient integration of engineering, management, and quality assurance tasks as they relate to EMC.
2. Insure the efficient integration of EMC with all other systems and subsystems.

The first requirement for achieving EMC in an avionic system is that all major components and subsystems be designed, constructed, and tested in compliance with MIL-STD-461.

The second requirement is compliance with MIL-E-6051 as an operating helicopter system, with all avionics and other equipment installed and performing their normal functions.

8-1.3 DESIGN CONSIDERATIONS

The design considerations that follow are applicable to EMI and should be used to assist in keeping EMI to a minimum.

The first design consideration involves the creation of a good, basic ground plane. This is normally the avionic component chassis or the airframe structure for the avionic system installation. An ideal ground plane will provide a zero-potential, zero-impedance reference base for all circuits, and a sink or trap for all undesired signals that can become interference sources.

A second design consideration, particularly at the lower communication frequencies, is the requirement for single-point grounding so as to avoid ground loops. The large, circulating currents in ground loops are potential causes of interference.

A third design consideration concerns shielding practices for major components and for the total aircraft installation.

A fourth design consideration calls for isolating, as far as possible, the power-carrying wires and cables from the high-impedance, low-level signal wiring. The basic principle is to categorize conductors on the basis of whether their primary leakage field components are magnetic or electrostatic. All conductors carrying power or signal energy have associated with them an external or leakage field that can induce unwanted signals or noise in nearby conductors by inductive or capacitive coupling. To minimize these undesirable field components, various techniques are used — such as electrostatic and magnetic shielding, space separation, twisting of wire pairs, crossover wiring methods, use of field-absorbing materials, and sophisticated neutralization methods.

A fifth design consideration is to provide adequate bonding and grounding for all electrical and electronic equipment, and for parts of the vehicle structure that can contribute to the generation of electrical noise. All electrical and avionic equipment, subsystems, and systems that produce electromagnetic energy *shall* be installed to provide a continuous low-impedance path from the equipment enclosure to the aircraft structure. The designer must demonstrate that the proposed bonding methods result in a DC resistance as specified for the various classes of bonding in MIL-B-5087. The design *shall* minimize the long-term effects of operational vibration, the effects of corrosion between adjacent surfaces and of galvanic action, the dielectric breakdown of insulating finishes, and the undesirable effects of intermittent electrical contact. Bonding *shall* be accomplished by direct metal-to-metal contact wherever practicable. A bonding jumper *shall* be used where direct metal-to-metal contact is impracticable. Such jumpers *shall* be of the standard types as specified in MIL-B-5087, or other appropriate types, and *shall* be kept as short and direct as possible. Where practicable, the jumper *shall* not exceed 3 in. in length. Surface preparation for bonds and grounds *shall* be accomplished by removing all anodic film, grease, paint and lacquer, or other high-resistance materials from the immediate area of contact. Direct-to-basic structure bonding *shall* used wherever possible. For vehicles with metallic skin, the skin *shall* be designed so that a uniform, low-impedance skin is produced through inherent RF bonding during construction. RF bonding must be accomplished between all structural components. Hatches, access doors, and similar components not in proximity to interference sources or wiring *shall* be either bonded to or permanently insulated from the vehicle skin except for the protective static drain band. It is highly desirable, during the design phase, to confer regularly with airframe designers so as to resolve compatibility problems. For guidelines to analysis and design, the design engineer should consult MIL-B-5087, AFSC DH 1-4, Ref. 1, and the NAVSHIPS documents referenced in MIL-STD-461.

A sixth design consideration for minimizing EMI is to separate and isolate pulse devices and equipment from other devices that are highly susceptible to EMI. This is accomplished by attempting to separate such items as pulsed radars, interrogators, transponders, and HF transmitters from computers, data processors, and susceptible receivers. This is not always possible, inasmuch as the physical locations of some devices are dictated by mission requirements. However, the designer should strive to achieve as much physical and electrical isolation as is practicable.

A final design consideration involves the use of double-shielded coaxial cables. Other cables or wires requiring shields *shall* have a minimum of 90% coverage. Connectors used with shield cables *shall* be provided with black shells for fastening cable shields.

8-1.4 ENVIRONMENTAL ASPECTS

Environmental considerations are pertinent to the design of the basic avionic system, and to the airframe-system interface. Susceptibility to rotor modulation must be considered. The very high frequency omni-directional range (VOR), instrument landing system (ILS) localizer and glidescope, VHF-FM homer, and other equipment have been affected adversely by near-frequency rotor modulation. As rotor blades pass over the aircraft, a modulation of the incoming wavefront is set up, with pronounced results. In addition, the modulation is in a nonsinusoidal manner and the harmonic content is high. The variable and reference modulation in a VOR is 30 Hz and the localizer and glidescope frequencies are 90 and 150 Hz; these are all convenient harmonics. This, coupled with the fact that helicopter rotor speed often is such as to give harmonics of 30, 90, and 150 Hz creates problems for avionic system designers. Techniques have been developed, as discussed in par. 8-3.2, for phase inversion and cancellation of the modulation. This technique shows promise of solving the problem. However, the characteristics of certain existing ground facilities are such that when this technique is used one error simply is exchanged for another. In any case, it is essential that manufacturers of equipment for helicopters incorporate very narrow band filters into their equipment, and procurement should be based upon this criterion. In addition, antennas must be decoupled from the main rotor insofar as is possible.

In general, helicopter avionics do not have to be designed to withstand the altitude extremes that fixed-wing avionic systems do.

Dust and sand are more of a problem to helicopter avionic equipment than to fixed-wing equipment. Dirt can pack up in voltage regulators, rotating equipment, relays, switches, and other critical devices and cause malfunctions. If these components cannot be hermetically sealed, they must have shrouds or other protective covering, and they will require additional maintenance.

Helicopter vibration must be considered based upon the installation of all equipment. The addition of equipment, particularly in the instrument panel, affects the frequencies and amplitudes of vibration. With the advent of solid-state components, vibration problems have been reduced, but most internal components still are vulnerable to vibration fatigue. While avionic manufacturers qualify their products to a specification, the applicable specification cannot duplicate absolutely the situations encountered in the actual installation. Therefore, it is desirable to design for a minimum level of vibration.

Temperature ranges may be severe, depending upon ambient conditions. Because of the large areas of transparent windshield or canopy, solar heat can become a problem when the doors are closed and the helicopter is on the ground. Avionic packages are good heat sinks; they will absorb a great deal of heat and will not dissipate it for some time after becoming airborne. Some ground temperature conditions are worse than conditions in flight.

A temperature survey is an important design consideration for avionic equipment installation. Data based upon heat rise can be developed for various critical ambient temperature situations. It must be remembered that ram air is not available on the ground or when the helicopter is hovering; therefore, either auxiliary air from outside the aircraft or engine bleed air must be used to meet cooling requirements.

If a particular helicopter is to be a multiuse aircraft, requirements applicable to the various uses must be considered. If a single airframe is to be used for two or more different types of missions, the avionic configuration design must accomplish all requirements economically.

Environmental test requirements should be formulated during the initial helicopter planning phase and should consider all environmental conditions to be encountered.

8-2 COMMUNICATION EQUIPMENT

8-2.1 GENERAL

Army helicopters contain many combinations of communication equipment. Because of the rapid development of new devices and the changes in nomenclature, no specific radios are referenced in this chapter.

The types of communication equipment currently in use include high frequency, HF (3-30 MHz) very high frequency, VHF (FM) (30-75.9 MHz); VHF (AM) (118-150 MHz); ultra high frequency, UHF (AM) (225-400 MHz); and millimeter wave. A typical communication block diagram is shown in Fig. 8-1.

Guidelines for the radio installation include:

1. The transmitter should be mounted in close proximity to the antenna in order to preclude line losses.

2. The control head should be mounted to provide ease of access for the flight crew.

3. Routing of audio wires should be such as to prevent crosstalk and feedback.

4. Power leads should be of sufficient size to permit full generator/battery voltage to appear at the radio under transmit conditions.

5. Components should be mounted in an area where sufficient cooling will be available.

6. A low-vibration area should be provided for mounting. The vibration limitations for communication equipment are identified normally in the applicable equipment installation specification (SCL-I-OOXX).

Each of these guidelines contributes to the reliability of the overall system, and, therefore, is essential for helicopter mission accomplishment.

Transmission lines, usually coaxial cables, are used to carry the transmitted signal to the antenna and the received signal from the antenna to the receiver. Low-loss cable, such as RG-214/U, should be considered for lengthy runs where excessive loss could occur. Newer cables are being developed, and appear promising. Commercial cables, even if not yet approved by Army qualification tests, should be proposed by the contractor if their use assists in maintaining efficiency and low cost. AMCP 706-125 (Ref. 2) should be consulted for further information on transmission lines.

Antenna considerations for communication equipment are presented in par. 8-5.

8-2.2 MICROPHONE-HEADSET

A helicopter microphone must be of the noise-cancelling type; Army and civilian experience has shown that a dynamic microphone is the most effective. In the noise-cancelling microphone, ambient acoustic noise enters both sides of the microphone with equal intensity and at the same phase relationship. Unfortunately, the face, lips, teeth, and protective helmet have a major effect upon the noise-cancelling characteristics. In the case of a helicopter with a high ambient acoustic noise level, it may be necessary to conduct a power spectral density measurement of the noise level in the microphone area, using standard microphones, and then to develop a filter that attenuates unwanted noise while permitting a voice to pass through the microphone amplifier.

The headset also must be of a dynamic type, and it is highly desirable that it have minimum high-level distortion. The ear muffs should be large enough to exclude extraneous noise while providing operator comfort on long flights.

The microphone-headset, if included as part of the helmet, also should possess the foregoing characteristics.

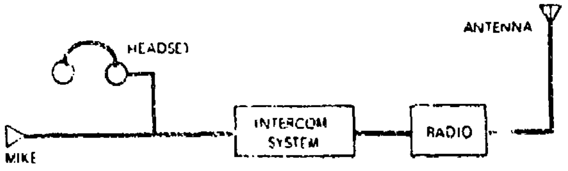


Figure 8-1. Block Diagram of Classical Communication System

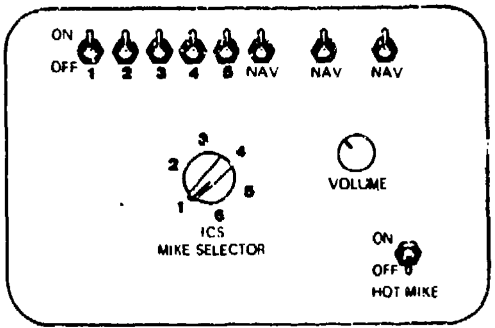


Figure 8-2. Typical Intercommunication Selector Box

8-2.3 INTERCOMMUNICATION SELECTOR BOX

The microphone-headset plugs into an intercommunication selector box (ICS), which will have a number of microphone and headset selector switches. Usually, the selector unit has an integral volume control and a "hot" microphone switch. A typical unit is shown in Fig. 8-2.

The ICS control unit selects — for each crew member — the radio for which the headset is selected, the intercom functions, and the emergency radio. (The emergency radio function normally is unswitched audio which cannot be disabled.) Often, other functions are routed through the ICS control box — such as engine failure warning, landing gear warning, rotor brake-on warning, and other audio warning signals. The ICS control box is the main switchboard for the flight crew. It usually is arranged in a standard configuration so that a crew member can transfer from one type of aircraft to another without confusion.

It is imperative that each crew member have ready access to his ICS control unit. Cockpit and crew station arrangement is described in par. 13-2.1.

8-3 NAVIGATIONAL EQUIPMENT

8-3.1 GENERAL

The categories of navigational equipment are:

1. Terminal maneuvering
2. En route navigation
3. Interdiction
4. Low-light-level navigation
5. Station-keeping.

Because the usefulness of VFR aircraft has been limited in recent combat operations, future helicopter requirements will include all-weather operation with stability augmentation systems.

The location and installation of the antennas required for the navigational equipment discussed are further defined in par. 8-5.

Navigation displays are discussed, along with other flight instruments, in Chapter 10.

8-3.2 TERMINAL MANEUVERING EQUIPMENT

In addition to the basic flight instruments, a helicopter may have VOR, ILS, marker beacon, and, preferably, a radar altimeter.

The helicopter VOR Antenna Array is mounted for horizontal polarization in an area that provides maximum performance and minimum rotor modulation. The basic ground signal is both frequency- and amplitude-modulated at 30 Hz, and the aircraft VOR receiver measures the phase difference between these two signals in order to obtain the angular displacement to the station; accuracy of the VOR is about ±2 deg. Because the main rotor blade often rotates at an angular rate that is some subharmonic of 30 Hz, helicopter manufacturers have experienced a great deal of difficulty with rotor modulation of the VOR. Usually the rotor modulation can be seen in the oscillation of the course needle. In extreme cases, the course information will become unusable.

There are some experimental techniques available by which this rotor modulation can be eliminated. Essentially, these depend upon detecting the rotor amplitude-modulation on the subcarrier, inverting it, and summing it with the modulation on the VOR receiver variable phase channel — in effect, cancelling it out. While this principle has been demonstrated, it also has been found to introduce an additional error into the system. The small variation in wheel-tooth symmetry in the spinning tone wheel typically used to generage the subcarriers in VOR systems results in an amplitude modulation of the subcarrier. When the modulation of the subcarrier is summed with the variable phase signal, the rotor component is cancelled but these asymmetries impose a new AM component on the tone. Thus, the principle cannot yet be applied to operational equipment (Ref. 3).

An instrument landing system (ILS) is used to guide the aircraft to the ground. The ILS is composed of a localizer, which operates between 108 and 112 MHz and uses the same antenna as the VOR, and a glidescope, which operates between 329 and 335 MHz. When a localizer frequency is dialed, the glidescope is channeled automatically to the proper frequency. Both the localizer and the glidescope use 90 and 150 Hz to provide right-left or up-down signals. Again, rotor modulation has been a major problem in both these devices.

An additional radio for use with the VOR and ILS is the marker beacon receiver. The receiver operates at 75 MHz and is used to locate points along the ILS path.

8-3.3 EN ROUTE NAVIGATION EQUIPMENT

Navigational equipment used in en route flying may include ADF, DME, TACAN, LORAN, compass, Doppler radar, and inertial navigation systems. The mission requirements dictate the degree of sophistication required.

8-3.3.1 Automatic Direction Finder (ADF)

This is a refinement of the old radio direction finder, and employs both a sense antenna and a loop antenna on board the aircraft. The signals from these two antennas are added vectorially and a cardioid pattern results. By means of circuitry within the radio, the system always seeks the null of the cardioid and, by means of proper calibration and instrumentation, a pointer shows the relative bearing to the station to which the radio is tuned.

The equipment designer must be certain that the receiver antennas are selective, so that extraneous signals will not affect the operation of the unit. Attempts have been made to install wideband amplifiers in the loop and sense antenna circuits in order to try to improve the efficiency of a short antenna or a long cable run from antenna to radio; however, such attempts usually have resulted in additional design problems.

The sense antenna should be as far from the main rotor as possible in order to prevent interference from triboelectric noise, and should be close to the electrical CG of the helicopter for good reversal characteristics. The location of the sense antenna determines the reversal characteristics of the system. Empirically, it has been shown that if the sense antenna is located forward on the aircraft belly, an early reversal (and, possibly, multiples) will occur; if it is mounted aft on the belly, a late or post-reversal will result. If it is mounted forward on the top, a later reversal will occur; and if mounted aft on the top, an early reversal will result.

The loop antenna should be mounted so that it is symmetrical in the longitudinal axis for symmetry in calibration. Because the loop is affected by large masses, consideration should be given to the location of deployable or disposable stores when placing the loop.

The ADF never is to be considered as precision equipment. It is versatile, and its angular error varies inversely with distance and increases with atmospheric noise level. It is vulnerable to countermeasures.

8-3.3.2 Distance-measuring Equipment (DME)

Distance-measuring equipment has been in use for some years and is very accurate. It consists of an airborne transponder that sends out a signal that triggers a ground transponder into sending back another signal to the aircraft on a slightly different frequency. The airborne transponder measures the total elapsed time, divides by two, and converts this figure into miles. The distance then is presented to the pilot by means of a dial instrument or a digital display. The antenna should be isolated from other antennas as much as possible due to the pulsed characteristic of the output.

8-3.3.3 Tactical Air Navigation (TACAN)

TACAN (Tactical Air Navigation) is a military system that combines DME and a form of VOR (Station Bearing) so as to give the pilot a continuous position fix with respect to a single station, in terms of distance and bearing to the station. Each ground TACAN beacon consists of a transmitter and an antenna. The transmitter operates in the UHF band, between 962 and 1213 MHz. The airborne receiver-transmitter sends out a chain of interrogation pulses and decodes the reply from the ground station

TACAN is a line-of-sight system, and there is almost no chance of interference from stations beyond the radio horizon.

8-3.3.4 Long-range Navigation (LORAN)

Hyperbolic navigation is achieved when synchronized signals having a known velocity of propagation are transmitted from at least three known points, and the relative times of arrival of these signals are measured and interpreted. Standard LORAN is a hyperbolic navigation system that was developed primarily for long-range navigation over water. It operates on one of several frequencies between 1700 and 2000 Hz, and its propagation characteristics are determined primarily by soil conductivity and ionospheric conditions.

The long pulse length requires the use of careful matching techniques in order to achieve reasonable precision The chief disadvantages of LORAN include the impossibility of instantaneous fixing without dual installations, the presence at night of long trains of pulses reflected from the ionosphere, and the fact that ionospheric transmission is not homogenous, so that the shapes of the sky-wave pulses often are distorted and difficult to match.

LORAN C/D is the latest model of this type of airborne equipment. The accuracy is extremely good when the set is operating within the range of highest accuracy of the transmitting stations. The LORAN equipment operates in the HF region, so a relatively long, wire antenna is desirable. Except for reversal characteristics, the same considerations should be used in the placement of this antenna as are used for the ADF sense antenna.

8-3.3.5 Compasses

There usually are two types of compass systems abroad a helicopter. The most precise is the slaved directional gyro (DG), that has two modes of operation — free directional gyro and slaved compass. In the free mode, it acts only as a directional gyro; in the slaved mode, signals from a flux valve slave it to magnetic north. Particular care must be used in the location of the flux valve; it must be as far as practicable from any ferrous material, and any DC wiring in close proximity must be two-wire twisted.

The stand-by magnetic compass usually is located above the instrument panel. Compensation is integral to the unit. If light wires are installed for this compass, they must be two-wire twisted.

Intermittent fields should be avoided for all compass systems.

8-3.3.6 Doppler Navigation Systems

Doppler systems measure velocity only, by the well-known Doppler effect in which radiation from a source in motion relative to the viewer is displaced in frequency. In practice, this means comparing the frequency of the returned echo with a stable reference frequency; the difference between the two is a direct measure of the relative velocity. Accuracy thus depends upon the echo quality. Echo quality from water, for example, often is poor.

Doppler systems determine location relative to the point of flight origin by integration of measured velocity vectors. Doppler accuracy represents an improvement over airspeed-clock-compass dead reckoning because the velocity vectors measured are relative to the ground. Generally, the vectors in the direction of flight and normal to it (x and y) are measured. The system accuracy is expressed as a percentage of the distance travelled, as opposed to inertial systems whose accuracy is relative to the time of flight.

Typical performance accuracy of Doppler systems for ground speed is 0.11% (rms) over 10 nmi, and 0.06 deg (rms) for average drift. Reliability of actual installations has typically been 1000 to 1200 hr MTBF. Further information on Doppler systems can be found in Ref. 4.

8-3.3.7 Inertial Navigation Systems

Inertial navigation systems (INS) have been developed primarily for use on fixed-wing military and commercial aircraft. Because of the requirement that an inertial platform be precise without ground station correction, this device is a valuable navigational standard in forward areas. Countermeasures are virtually nonexistent.

INS operation is governed by two basic physical principles — the gravitational pull of the earth and the gyroscopic principle. Essentially, the INS is made up of a stable platform, a computer, a memory, and a presentation. Many outputs can be derived from such a unit — e.g., true north, velocity and direction, crab angle, and all autopilot signals necessary to predetermine flight track.

The only requirement necessary for installation of inertial navigation equipment is the provision of a precise longitudinal axis for reference. Normal EMI/EMC precautions also must be taken.

8-3.4 INTERDICTION EQUIPMENT

Because of security considerations, this discussion necessarily is limited. Generally, a specific electronic countermeasure (ECM) device will give quadrature information (general location of the equipment under surveillance), frequency, pulse width (if pulsed), duty cycle, peak and average power, and repetition rate. The ECM also is designed to provide other types of information if required. ECM is desirable as interdiction equipment, not only to provide the helicopter crew with information, but to telemeter the information back to secondary forward analysis areas.

In addition to ECM, it may be desirable to include optical or laser range-finders, ranging gunlaying radar, or other devices to aid the interdiction aircraft in performing its mission and to pinpoint targets for forward ground artillery. Communications, usually secure, will form a part of the system.

While not actually a part of the interdiction equipment, the electrical characteristics of the helicopter must be considered to be a part of the mission. Acoustic noise, radar reflectivity, and infrared (IR) signature must be minimized. In addition, the aircraft mission will specify radio frequency transmission usage.

8-3.5 LOW-LIGHT-LEVEL NAVIGATIONAL EQUIPMENT

This equipment is unique, and is required only for specific missions. There are three basic types of such equipment. The first type is low-light-level television. In is simplest form it is nothing more than a closed-circuit TV employing a camera that is sensitive particularly to low light levels.

The second type is an adaptation of the photomultiplier or "snooperscope" device used during World War II. Optical stabilization and intensification techniques have been refined, and the improved system has some unique advantages.

The third, and most promising, system is the infrared (IR) detection type. It consists of a sensor-scanner, a signal conditioner, a power supply, and a video display. The sensor responds to a selected spectrum in the IR region, and operates in total darkness. It is extremely sensitive and can discriminate between slight temperature differences. The system can be designed to include very accurate definition, and the resultant display on a dark night can duplicate a daylight TV picture.

These low-light-level devices are used in activities that call for radio silence, acoustic silence, low radar reflectivity, and low IR signature. Flight techniques also are important.

8-3.6 STATION-KEEPING EQUIPMENT

This equipment generally is used for maintaining a position directly over a point on the earth and for formation flying. The systems usually are employed on larger, load-carrying helicopters requiring precision positioning for pickups and drops.

In applications requiring loading and unloading from a hover, Doppler radar is the most readily available and precise type of station-keeping equipment. Most conventional Doppler navigational systems are used as an adjunct to en route navigation; the frequencies are relatively high and accurate. At zero velocity, the Doppler shift will be zero, and determination of movement is difficult; however, recent developments by avionic manufacturers effectively have permitted zero error during hover. This type of system is recommended for station-keeping for loading situations.

Doppler systems require specific antenna locations. Generally, three- or four-beam patterns are used. Some manufacturers incorporate all functions into one antenna. Certain types of equipment require that the antenna be gimbaled for stabilization purposes. Provisions for antenna location must be made during the early design phase in order to maximize efficient use of the area in the fuselage belly.

For station-keeping during formation flying, many techniques have been used in the past and are satisfactory under both VFR and IFR conditions. Cost-effectiveness decisions will determine equipment selection. Radar, together with beacons, LORAN, or special IR, may be considered.

8-4 FIRE CONTROL EQUIPMENT

8-4.1 GENERAL

The airborne fire control system selectively performs the tasks of (1) establishing that the weapon is aligned properly to hit the target, and (2) driving and holding the weapon platform to a commanded position.

The major elements of any fire control system consist of sight, sensors, and computer. The weapon controls are a part of the fire control equipment, and their functions are to activate the gun or missile, regulate gun firing rate, select the weapon, regulate the ammunition feed system, inventory the ammunition supply, etc.

The complexity and sophistication of an avionic fire control system will vary according to the degree of accuracy required, and the type and flexibility of the armament subsystem. The armament subsystem may be an integral subsystem of the helicopter, or it may be a modular component that can be snapped on or off to suit the particular mission requirements. Thus, the designer must establish fire control design requirements commensurate with required aircraft missions.

Because the kinds of missions to be performed are likely to be broad in scope (ranging, perhaps, from close tactical support to rescue), the fire control requirements likewise will be varied. Mission analysis will determine the fire control functions to be performed, and a careful selection of multiuse armament subsystem equipment will reduce weight and space requirements. Armament subsystems available for helicopters include flexible turreted guns, fixed guns, rockets, and missiles. The interface characteristics of their supporting functions are as different as the armament systems themselves. Consequently, detailed integration design specifications for each type of armament subsytem *shall* be issued so as to insure effective weapon delivery.

8-4.2 INSTALLATION

Adequate provisions for installation of the elements of the respective fire control systems should be incorporated into the helicopter to insure proper matching or harmonization of such systems with armament. The fire control system should be installed as specified in the helicopter specification governing control of guns, rockets, and guided missiles. The helicopter manufacturer is responsible for the shock mounting of all fire control equipment installed. Vibration-isolating mounts should be incorporated so that equipment will not be affected adversely by vibrations in the helicopter. Testing *shall* be in accordance with MIL-STD-810.

8-4.3 SIGHTING STATION

The sighting station provides the means by which the weapon operator establishes the azimuth and depression coordinates of the target relative to the aircraft position. For flexible weapons, the sighting station includes the operating controls by which the weapon is aimed and fired. An optical sight, either direct-viewing or periscopic, generally is used for daylight operations, and may provide selectable degrees of optical magnification.

The sighting station also may provide target range and image intensification sensors. Target range equipment can include lasers, radar, or stadiametric ranging devices. Image intensifiers include low-light-level television, electronic image amplifiers, and optical telescopes.

Direct viewing weapon sights have been a major source of fatal and serious head trauma during crashes in US military aircraft during World War II, Korea, and Vietnam. It is essential that all sighting devices be designed to eliminate their potential as injury producers. Factors to be taken into consideration in order to delethalize sighting devices are:

1. Ability to instantly remove, jettison, or stow during emergency
2. Not to create additional hazard(s) to other crew personnel in the event of emergency
3. Adequate stowed tiedown strength to prevent sight from rebounding during impact
4. Not to represent lethal missile hazard in the event of crash
5. Review of Ref. 5.

8-4.4 SENSORS

Fire control system sensors provide the information necessary for solving the fire control problem and directing the weapon(s) at the target. Sensor types include those that measure target and aircraft motion or position, and those that assist in target detection. Externally mounted sensors should be housed in aerodynamic fairings wherever possible, and protection should be provided against such environmental conditions as handling and accidental ground maintenance damage. Sensors that produce electro-optical or electromagnetic energy should be located so that neither direct nor reflected energy enters the crew compartment. Furthermore, the mounting provisions for such sensors should permit attachment of ground operation warning devices to alert ground crews to potential radiation hazards.

The mechanical interface between the sensor and the helicopter should be designed for adequate strength, ease of maintenance, and accurate alignment with the helicopter datum plane. Sensors projecting from helicopter mold lines should be located so that they will not interfere with aircrew entry and exit. Aerodynamic sensors include those for the pitot tube, angle-of-attack indicator, and air data computer. They should be located as far forward on the aircraft, and as far from the fuselage or appendages, as is practicable to minimize aerodynamic interference and local flow variations due to the influence of the main rotor(s). Sensor accuracy levels should be selected for compatibility with fire control accuracy requirements.

Sensors located internal to the helicopter *shall* be protected against the vibration shock and power load environments associated with the specific helicopter design. In general, any sensing equipment located within the crew compartment must be capable of withstanding crash load factors without detaching from its mountings. Vibration isolation should be provided in accordance with MIL-E-5400 and MIL-STD-810. Power requirements (including number and size of electrical wires), the necessity for shielding, and/or the use of nonstandard electrical connectors should be considered in the design of the internal mounting structure. Mounting provisions for all sensor units should provide for easy removal of fasteners and connectors, and for structural clearance adequate to permit rapid removal and/or repair during maintenance operations.

Sensors using electro-optical or electromagnetic energy should be located in regions where they will experience minimum electrical interference from other aircraft equipment. This principle applies to both the internal and the external (transmitting) portions of the equipment. The electrical power requirements of these equipment types can be significant. In order to minimize transmission losses and electrical interference, it is necessary to: (1) locate the sensor unit power supply in close proximity to the helicopter power source, and/or (2) minimize the separation between the sensor unit and the point of aircraft transmission. In addition, care should be taken to avoid potential hazards to the aircrew during operation of the equipment.

8-4.5 COMPUTERS

The airborne fire control computer is a special-purpose device that accepts quantitative information, arranges it, performs a mathematical calculation, and provides qualitative output information. This definition describes a simple electrical computing circuit as well as a digital computer. The specific requirements for the computer are established by the degree of fire control accuracy desired. In addition to supporting fire control computation, the airborne computer may be employed to assist in flight control, navigation, and communication tasks.

The specific design requirements of the computer system *shall* be in accordance with the governing design requirements of the fire control system for sharing helicopter navigation-avionic computation functions.

Power requirements *shall* be as specified by computer design requirements. Computer vibration isolation will be required as specified in the design requirements. In meeting computer access requirements, consideration must be given to removal, safety, replacement, and component inspection.

8-4.6 FIRE CONTROL ACCURACY

Guns and rockets should have adequate structural support in order to minimize helicopter structural deflections during firing or launching. Optical sights should be located so as to avoid the sighting aberrations of canopy distortions. Sights should be installed upon rigid mounts, but without inducing undue sight-line vibrational distortion. Locations of sights and armament should be such as to avoid the probability of excessive parallax errors in the fire control computation. The sight and armament subsystems should be installed with suitable adjustment and lock devices for proper boresighting and harmonization. Electronic equipment should be protected from the noise generated by helicopter power equipment (and by any other components likely to generate electronic noise). Data sensors and ballistic computers should be chosen so as to provide component accuracy characteristics consistent with the system accuracy requirements of the governing helicopter specification.

8-4.6.1 Inertial Stabilization

For some applications, stabilization of the sight reticle is used in order to remove helicopter motion dynamics as a source of sighting error. Stabilization is also incorporated in automatic target tracking equipment. This equipment typically is designed as part of the sighting station, but remote auxiliary component location may be required. Auxiliary components should be installed in accordance with design practices specified by the sight supplier.

8-4.6.2 Fire Control Datum Plane

A physical reference surface should be established so as to relate sight and armament equipment for basic alignment and harmonization of the fire control system with the aircraft structure.

In a fixed weapon system installation, the weapon firing line generally is aligned so as to be parallel to the aircraft datum line. The designer should consider weapon characteristics, such as tangential projectile throw and/or barrel cant, during the alignment process. If the system is radar-directed, the radar line must be aligned parallel to aircraft datum.

For flexible fire control systems — either radar- or optically-directed — provisions *shall* be made to align and test the line of sight, the tracking line (turret minus weapon), and the weapon firing line parallel to the aircraft datum line. Provisions must be included for testing and harmonizing the fire control coordinate system with the aircraft coordinate system. The precision of measurement is dependent upon the overall fire control system requirements and the associated error allocation.

8-4.6.3 Harmonization

In order to provide the greatest firing accuracy, the major components must be adjusted carefully. This is true especially of the sight and its accessories, the projectile-launching equipment, and the aircraft itself. This harmonization involves the orientation of three reference lines in the aircraft: the aircraft datum line (flight path), the sight line, and the armament line. Lugs aligned with the aircraft longitudinal axis should be installed to provide a surface for leveling the equipment and for establishing reference lines.

Harmonization of fixed weapon installations generally can be accomplished by mechanical adjustment of the weapon. For flexible installations, mechanical and electrical adjustments are required in order to assure coincidence and alignment of sight line and weapon line.

Harmonization techniques can be parallel, point, or pattern; and the technique best suited to the specific weapon type should be selected.

In parallel harmonization all armaments, plus the sight line, are aligned parallel to the armament datum plane. Point harmonization typically is used when the armament is installed well outboard from the centerline of the aircraft or the sight location. In this case, all armament is adjusted so that the firing lines intersect at a point ahead of the aircraft, thus concentrating the fire of the weapons upon a single, small area at a selected range. Pattern harmonization is similar to parallel harmonization, except that the parallel armament line is elevated above the sight line by gravity drop or velocity hump corrections. All harmonization techniques require the use of a special target (or harmonization board) as support equipment.

8-4.7 COMPONENT LOCATION

The major points to be considered in fire control component installation include vibration and shock isolation, cooling and heating, radio noise interference, accessibility, electrical shock hazards, and crash safety. All weapon system components should be mounted so that the entire travel of the shock mounts is possible in all directions without interference between components. For small components, the shock mounts should be adequate to support the weight of both the component and its cable connections. The components *shall* be located where they will receive an ample supply of circulating air, particularly when airborne.

Heating may be necessary for some components of the fire control system under extremely low-temperature conditions.

Radio noise interference should be reduced by grounding the cases of all components securely and filtering power supplies, and by using shielded cables for pulse-carrying applications.

Electrical shock hazards will be reduced greatly by secure grounding of all components.

Ease of installation, alignment, and troubleshooting should be considered during equipment design so that connectors can be disconnected readily, even under adverse conditions. Components should not block access to other components. Serviceability of equipment should be enhanced by locating adjustments and test points on a single, accessible surface. Where this is impossible, the use of slideout racks to permit removal of equipment from shock mounts should be considered. If the component must be removed from the aircraft for adjustment, sufficient cable length should be available to allow the component to be removed and placed on a service rack without disconnecting its cables.

Great care should be taken to insure that avionic subsystem components cannot enter crew spaces as lethal missiles in the event of a crash. Crashworthy component tiedown strength and/or crashworthy barriers should be provided in order to overcome the lethal potential of avionic components.

8-5 ANTENNAS

8-5.1 GENERAL

The communication and navigation equipment discussed in pars. 8-2 and 8-3 requires a variety of antennas, ranging from those in the low-to-microwave frequency spectrum to those having horizontal and vertical polarization and those with different radiation patterns. Antennas are susceptible to rotor-induced modulation, triboelectric charging, and noise generated by corona discharge. The combination of electrical requirements and problems created by the platform presents the antenna designer with difficult design requirements. Safety problems are important. It is essential that antennas capable of emitting potentially harmful or fatal radiations be marked with appropriate warning labels to preclude fatal or serious injury to aircrew or maintenance personnel during normal operation or routine maintenance.

8-5.2 ANTENNA DEVELOPMENT

The development of a helicopter antenna is based upon the requirements of the associated equipment and the mission for which the helicopter is intended. Some of the considerations are:

1. Frequency range
2. Radiation characteristics
3. Polarization
4. Efficiency
5. Voltage standing wave ratio (VSWR)
6. Noise
7. Environment
8. Structure.

The frequency range of the antenna determines its basic dimensions. Army helicopters use frequencies from 150 kHz up to the visible range. Without consideration of power output, the high-frequency region extends to 30 MHz. Depending upon the time of day, the time of year, the sunspot cycle, and the vagaries of the ionosphere and its various layers, the HF band is considered to be the best for long-distance communication.

With the perfection of solid-state, single-sideband (SSB) equipment, high-frequency communication is becoming more prevalent. However, mission requirements will establish usage criteria.

For VHF, 30-300 MHz and up is considered to be line-of-sight communication or propagation, depending upon radiated power and receiver sensitivity, and upon the points of radiation and reception. For example, a reasonably clear area provides true line-of-sight communication, while multiple layers of vegetation, such as are encountered in Southeast Asia, require much more power in order to effect through-vegetation transmission of electromagnetic impulses.

For operation at frequencies in the LF and HF ranges, antennas are quite long—e.g., one-quarter wavelength at 1 MHz is 246 ft. Because the dimensions are so large, it is standard practice to have the antenna system include an antenna tuner as a coupler. The coupler automatically matches the impedance of the electrically short antenna to that of the transmission line. This method of loading wire antennas becomes less efficient as the ratio of antenna length to the wavelength of operation becomes smaller.

For frequencies at or above VHF, the size of the antenna is less of a problem. As the electrical length increases, the instantaneous bandwidth of the antenna also increases, and the result is operation over a wider bandwidth without tuning. Whereas the wire antenna must be tuned each time the frequncy is changed, the VHF and UHF antennas are fixed-tuned, and are capable of efficient operation, with low VSWR, over a band of frequencies. Because of the wide instantaneous bandwidth, the antenna also can be used simultaneously by different equipments tuned to different frequencies. Diplexers and hybrid devices are used to provide isolation between equipments using the same antenna.

Radiation patterns of the antenna indicate where energy is being radiated, or, conversely, from which direction it can be received. Communication and direction-finding equipment generally requires omnidirectional radiation in the azimuthal plane, with the maximum amount on the horizon in the vertical plane. Navigational equipment requires radiation in specific directions. (Because of the physical geometry of the airframe, truly omnidirectional patterns never are obtained.)

The airframe directly influences the radiation by its shadowing and re-radiation effects. The airframe can radiate energy coupled to it at frequencies where its dimensions are an appreciable part of a wavelength. At higher frequencies, the airframe blocks and shadows radiation in certain directions.

The relative positions of the antenna and the rotor also affect the radiation patterns. The effect of the rotor is to modulate the radiation pattern at a frequency determined by the number of blade passages per second over the antenna location. The carrier frequency, along with each sideband, will be modulated by this frequency. As discussed in par. 8-1.4, this modulation interferes with the performance of equipment that makes use of information contained in modulation components close to the same frequency. Rotor passage near an antenna also can affect the impedance of the element, which results in a modulated signal.

From empirical data, it appears that an antenna can have a peak-to-valley variation of about 6 dB and sharp nulls of 30 dB without experiencing overall degradation of performance. For the sake of economy and practicality, it is imperative that the best antenna possible be provided. Mission requirements will determine the selection and use of radio type(s) and associated antennas. For instance, the VOR pattern is optimum in a forward direction, while tactical communications, IFF, and other primary radio aids should be as omnidirectional as practicable.

The polarization of the helicopter antenna must correspond to that of the antenna at the other end of the communication link. This requirement does not apply to HF antennas because of rotation of polarization by the ionosphere. Cross-polarized signals can be radiated from linear antenna elements as a result of reflections and currents in the airframe. The cross-polarized component represents an inefficiency or a power loss, and is a consideration when selecting the antenna location.

The effects of triboelectric (friction) charging influence the selection of antenna location. The helicopter airframe and nearby antennas are charged electrostatically by the rotor downwash, and voltages high enough to produce corona can result. The corona will occur on sharp points, or on points of high electrical stress concentration. If these areas are on antennas, or are electromagnetically coupled to antennas, the broadband noise generated by the corona will be introduced into the receiver.

Antenna location must insure that the antennas are decoupled from each other; this is practicable for antennas operating in the same frequency range. Mutual coupling between antennas affects their impedance and the radiation patterns of individual elements. High voltages can be introduced in passive circuits if excessive mutual coupling exists between transmitting and receiving antennas.

Development of helicopter antennas depends upon the results of model measurements, as contained in MIL-A-25730. When employed with discretion, the use of scale models for antenna development provides the ability to predict the suitability or unsuitability of an antenna location. Usually, the model technique is used on large prototype aircraft with the model scaled down to 1/20th to facilitate manipulation. The scale factor is variable, and any scale factor may be used provided that the resultant frequencies (which must be scaled upward by the same scale factor) are easily obtainable.

Frequencies in the gigahertz region are not easily scaled. In addition, frequencies in the HF region often are not scaled. On a 1/20-scale model at 15 MHz, for example, the model frequency must be 300 MHz and these two frequencies inherently propagate differently. Theory is quite logical for scale models, but care must be taken to scale every detail affecting antenna performance. For example, scaling wire size in order to maintain identical current distributions must be considered. The model must be isolated from its surroundings, and consideration must be given to ground reflections and to radiation from connecting cables.

The model measurements are made in the scaled dimensions and for orthogonal polarization. The radiation patterns are measured over the scaled frequency range of operation, for varying positions of the rotor and for different antenna locations. The optimum location of the antenna with respect to radiation patterns will result from these measurements.

Measurement of impedance and mutual impedance is accomplished most easily on the helicopter although a full-scale mock-up, containing those portions of the helicopter within several wavelengths of the antenna, gives accurate results. All model measurements must be verified on full-scale aircraft. This technique is analogous to aerodynamic model testing in a wind tunnel.

The present range requirement for VHF-FM communication having a 10-W power output to a nonmatching antenna is 40 mi. This is a reasonable range when tested in the optimum condition.

When an antenna is to be used over a wide frequency range, i.e., VHF-FM 30-76 MHz, it must be broadband if efficient operation is to result. If the transmitter/receiver mismatch is not too great, reasonable efficiency will result. VSWR, the Voltage Standing Wave Ratio, is the criterion for acceptance of antenna matching; it is determined by the ratio of forward to reflected power. The Radio Technical Commission for Aeronautics (RTCA) has chosen a VSWR of 5:1 as an applicable standard. However, a more stringent ratio is required in many cases. Communication antennas should have a VSWR of 2.6:1 or less. TACAN and transponder antennas should have a VSWR of 2.5:1 or less. VSWR is a complex ratio of incident power to reflected power as given by Eq. 8-1.

$$\text{VSWR} \geq 5 = \frac{1 + \dfrac{\text{Reflected Power}}{\text{Forward Power}}}{1 - \dfrac{\text{Reflected Power}}{\text{Forward Power}}} \qquad (8\text{-}1)$$

8-5.3 LOCATION AND INSTALLATION OF ANTENNAS

In most cases, the [illegible]veral types of antennas that can be used for any given item of communication or navigational equipment, and the final choice is dependent upon the requirements of the specific installation. For additional information governing antenna installation and location, see MIL-STD-877.

During the initial design phase, incorporation of system zero-drag or flush-mounted antennas must be considered. It is extremely important to submerge the antennas, not only for increased flight efficiency, but also to minimize maintenance problems.

In most cases space limitations prevent the location of antennas far away from one another. Specifications usually require a three-eighths wavelength separation, but this is unrealistic; therefore, separation to the greatest degree possible should be made. An alternative solution is the multiusage of a single antenna, with passive devices added to aid in multiplexing. An excellent example of this usage is a zero-drag, wide-band antenna for both UHF and VHF frequencies. With the assistance of a diplexer this single, simple antenna is used to receive and transmit simultaneously in the UHF and VHF communication bands. The same philosophy could be used for lower frequency VHF (30-75) and ADF sense antennas.

8-5.3.1 Communication Antenna Considerations

Together with EMI/EMC considerations, antennas pose the most difficult problem for the helicopter avionic engineer. A typical helicopter with standard communication antennas is shown in Fig. 8-3, which depicts a simple, operational combat scout or observation helicopter (none of the navigational antennas are shown).

Antenna functions are affected by antenna location, and the helicopter in Fig. 8-3 illustrates typical problems. For example, due to the HF antenna location, a hard landing could affect operational characteristics. This antenna also is beaten by the main rotor downwash, and breakage could cause the antenna to become wrapped up in the tail rotor. In addition, an electrical impedance problem results from the main rotor blades passing over the antenna and causing rotor modulation. The VHF-FM #1 antenna, used for tactical communications, is in the exhaust, which could cause physical degradation, and the bulk of the helicopter is forward of the antenna, resulting in partial antenna shadowing. The VHF antenna, shown in the belly, may be relatively clear of many problems, but the landing gear would give reflective properties and consequent nulls in the antenna pattern. The UHF antenna, shown forward and above the cab top, is vulnerable to triboelectric noise and to rotor modulation. The VHF-FM #2 antenna is in front of the aircraft and in the field of view of the flight crew, which could be distracting. It also may be vulnerable to rotor modulation and triboelectric noise.

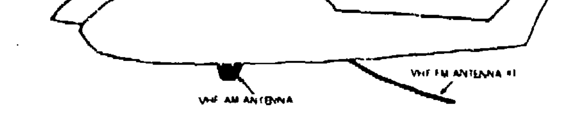


Figure 8-3. Typical Communication Antenna Layout

Effects of rotor modulation can be controlled effectively by installing notch filter equipment in the primary area of interest. In the case of communication receivers, band stop filters are used to eliminate unwanted modulation frequencies in the audio band.

8-5.3.2 Low Frequency (LF)

The primary use of the low-frequency spectrum is for automatic direction-finding. The ADF system uses a loop antenna having a figure-of-light radiation pattern, plus an omnidirectional whip (sense) antenna. The sense antenna output is combined with that of the loop antenna to produce a cardioid pattern, thereby eliminating the directional ambiguity of the loop. The location of the loop antenna is restricted by two considerations:

1. The cable between the loop and the receiver input is part of the receiver input circuitry and is of fixed length.
2. The loop must be located in a position of minimum pattern distortion.

The magnetic field lines that induce a current in the loop are distorted by the airframe, thereby causing an apparent bearing error. Although small errors can be compensated in the equipment, the design engineer must determine the best location for each installation. The sense antenna should be positioned in an area of minimum electrical field distortion to maintain accurate ADF performance as the helicopter flies over or near the ground station in what is called the "confusion" zone. The size of the confusion zone, in which the ADF indication can vary as much as 180 deg, depends upon the characteristics of the sense antenna and upon maintaining a minimum signal input level to the receiver.

8-5.3.3 High Frequency (HF)

HF, employed for long-range communication, uses wire antennas. The wire can be fixed between two points on the helicopter, or a trailing wire can be used. The use of antenna couplers is required with this type of antenna. A major problem with wire antennas is the possibility that they will become tangled in the rotors.

Radiation patterns, which, ideally, would be omnidirectional, are dependent upon location. They can be shadowed by the airframe, which itself can radiate and cause distortion. The wire antenna usually will have nulls at its end directions. The antenna wire is coated with polyethylene in order to prevent corona, and the supports must be designed to withstand transmitting level voltages.

LORAN also utilizes the wire antenna. Corona and voltage breakdowns are not problems, but omnidirectional coverage still is a requirement.

8-5.3.4 Very High Frequency (VHF)

The marker beacon receiver operates at 75 MHz. The antenna radiation pattern must be downward-looking, and must be polarized parallel to the axis of the helicopter. There are several antennas that will meet the radiation requirements. One is a balanced antenna mounted under the airframe. Other suitable designs are loaded half loops and flush-mounted cavity elements mounted in the same location.

The glidescope receiver operates in the frequency range of 329-335 MHz. The glidescope antenna must be designed for reception of horizontal polarized signals with minimum reception of vertically polarized signals. The antenna must be located forward on the aircraft for proper glideslope reception.

The VOR operates in the frequency range of 112-118 MHz and requires a horizontally polarized, omnidirectional antenna. The ILS receiver operates in the frequency range of 108-112 MHz and uses the same antenna as does the VOR. The location of the VOR antenna is critical due to rotor modulation effects. The VOR determines the phase difference between two received signals modulated with 30 Hz, which is about the third harmonic of rotor-induced modulation; if interference is too great, the VOR is inoperable. A loop mounted on the underside of the airframe and configured with a vertical axis is suitable for the VOR antenna. Location of any antenna on the underside can result in airframe shadowing in some direction. The ram's horn (a modified dipole), stacked dipoles, folded dipoles, and flush-mounted cavities also all are suitable designs.

Communication equipment operates in the frequency ranges of 30-76 MHz and 118-150 MHz. Some form of monopole is used most often for communication. The position of these antennas is determined by the necessity of obtaining omnidirectional radiation patterns. The antennas must be designed to prevent corona discharge, and should be decoupled from triboelectric discharges that would introduce noise into the receiver. The difficulty of obtaining omnidirectional coverage sometimes can be overcome by using two antennas. If the radiation patterns of the two antennas are complementary and the antennas are isolated from each other, they can be driven in parallel with appropriate impedence matching.

8-5.3.5 Ultra High Frequency (UHF)

UHF communications operate in the frequency range of 225-400 MHz. Monopole-derived configurations are used as both UHF and flush-mounted types, such as the annular slot. The general requirements for VHF location and installation pertain to UHF as well.

8-5.3.6 Special Purpose

There are other types of equipment that require antennas, but these are limited in use. The considerations for location and installation of these antennas depend upon individual system requirements. Doppler radar antennas, for example, always are mounted flat in the bottom of the aircraft. In general, the same restrictions discussed heretofore apply to all types of antennas.

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1. USAECOM Manual, *Interference Reduction Guide for Design Engineers*, Vols. I and II, AD 619 666.
2. AMCP 706-125, Engineering Design Handbook, *Electrical Wire and Cable*.
3. John G. Mast, *Elimination of VOR Signal Rotor Modulation*, Paper 924, *31st Annual National Forum of the American Helicopter Society*, May 1955.
4. Myron Kayton and Walter Fry, *Avionics Navigation Systems*, John Wiley and Sons, NY, 1969.
5. Turnbow et al., *Crash Survival Design Guide*, TR 71-22, USAAMRDL Fort Eustis, VA, Revised October 1971.

CHAPTER 9
HYDRAULIC AND PNEUMATIC SUBSYSTEMS DESIGN

9-0 LIST OF SYMBOLS

g = acceleration due to gravity, ft/sec^2
H = true pressure altitude, ft
H_m = measured pressure altitude, ft
dH/dt = change in pressure altitude with time at standard sea level conditions, ft/sec
ΔH = height difference, ft
ΔH_c = altimeter position error correction, ft
P = static pressure, psi
P_m = measured static pressure, psi
P_t = total pressure, psi
P_{tm} = measured pitot pressure, psi
ΔP = pressure difference, psi
p = true atmospheric pressure, psi
q_c = true impact pressure, psi
q_{cm} = measure impact pressure, psi
V = true airspeed, mph
V_c = calibrated airspeed, kt
V_{cm} = measured calibrated airspeed, kt
ΔV_c = airspeed position error correction, kt
ρ_{sl} = density at standard sea level conditions, slug/ft^3

9-1 INTRODUCTION

Chapter 9, AMCP 706-201, describes the many design trade-offs necessary in the final selection of secondary power subsystems. This chapter deals with the detail design of the hydraulic and pneumatic subsystems.

Hydraulic applications primarily include flight control and utility functions. Flight control functions include servo control of cyclic pitch, collective pitch, and directional surfaces. Utility functions may include part or all of the following:

1. Personnel/cargo hoists
2. Cargo hooks
3. Loading ramps
4. Doors
5. Landing gear
6. Gun turrets and drives
7. Rotor braking
8. Wheel braking and steering
9. Engine starting
10. Fluid dampers.

Pneumatic applications, while not as widely used in helicopters as are hydraulics, may include such functions as engine starting, auxiliary utility systems, and emergency backups. Pneumatic power also may be used for auxiliary power unit (APU) starting.

9-2 HYDRAULIC SUBSYSTEMS

9-2.1 FLIGHT CONTROL POWER SYSTEMS

Helicopter flight control systems may vary in complexity from the relatively simple power boost system with manual reversion to multiredundant systems where each system is designed to provide the full power required to operate the flight control functions throughout the vehicle performance envelope. The multiredundant system is discussed in this chapter because it contains the basic elements of all types of systems. This type system *shall* be employed unless the aircraft can be controlled without boost. A schematic of a typical central hydraulic system is presented in Fig. 9-1. The system shown contains its own fluid power generation, fluid transmission, and fluid supply components.

9-2.1.1 Central Hydraulic System

Hydraulic power is generated by variable-displacement pumps that are compensated for a nominal system design pressure. Fluid may be supplied to the pumps from gas-pressurized or bootstrap reservoirs. The pumps are driven by an accessory gearbox, which, in turn, is driven by the transmission when the rotor(s) is turning and by an

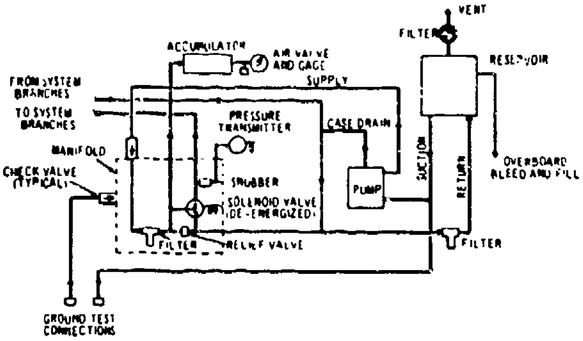


Figure 9-1. Central Hydraulic System

APU during ground checkout. Central system components include:

1. System manifold, a package containing system filters, pressure transmitters (with associated snubber and fuse), system relief valves, ground test connections, and return line check valves
2. System accumulator, a gas-precharged, piston unit with associated servicing valve and pressure gage
3. System reservoir, including return and pump suction line fittings, bleed and fill provisions, overboard vent, and level indicator. The reservoir also incorporates reservoir level-sensing, with associated subsystem isolation valves.

9-2.1.2 Flight Control Subsystems

A typical flight control subsystem consists of (1) a boost-actuating system, (2) a stability augmentation system (SAS), and (3) a stick boost hydraulic system. The pilot's control movements, transmitted through a system of bell cranks, rods, and levers, are mixed to provide the correct lateral, cyclic, and pitch motions through hydraulically powered actuators as shown in Fig. 9-2. If dual actuators are used, each half of the actuator is powered by a separate system.

A typical dual reversed SAS is illustrated in Fig. 9-3. The SAS actuator inputs to the boost actuators affect the movement of the rotor blades without feedback forces to the pilot controls. The SAS actuators must be capable of being engaged or disengaged by the pilot, and must incorporate an automatic lockout feature that operates in the event of hydraulic power failure.

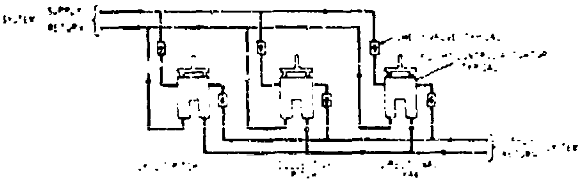


Figure 9-2. Dual System Hydraulic-powered Flight Control Actuators

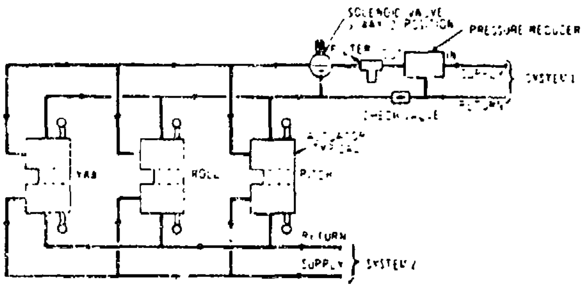

SOLENOID VALVE
PRESSURE REDUCER
IN
SUPPLY
SYSTEM 1
RETURN
CHECK VALVE
ACTUATOR TYPICAL
YAW
ROLL
PITCH
RETURN
SUPPLY
SYSTEM 2

Figure 9-3. Dual-powered Stability Augmentation System

The stick boost system provides the pilot with low-force stick movement capability. As shown in Fig. 9-4, an actuator is provided in each axis to overcome friction and inertia loads. These actuators function to react inputs from the SAS so that they cannot be felt through the pilot controls.

9-2.2 UTILITY HYDRAULIC SYSTEMS

The utility system may be powered in essentially the same manner as is the flight control system, using accessory gearbox-driven pumps when the rotor is turning or the APU for ground operation. The central portion of the system will contain basically the same components as does the flight control system. If system demand during peak load phases is relatively high, additional pumps in parallel may be necessary. To minimize the weight and size of subsystem circuits that do not have high pressure and flow requirements, pressure reducers should be considered.

9-2.2.1 Engine-starting Subsystems

There are two basic types of hydraulic engine-starting systems. One uses a limited amount of stored energy that is available in an accumulator, while the other considers the maximum power output available from an auxiliary power supply.

Because energy is limited in the first type of system, the design goal is to complete the start in the shortest possible time. As shown in Fig. 9-5, engine starting is

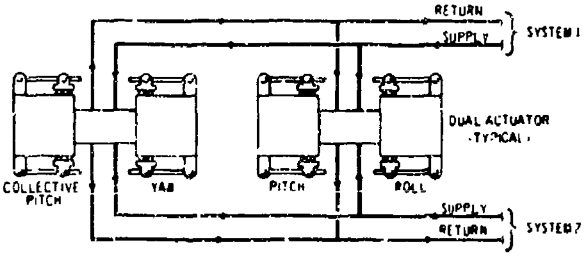

RETURN
SUPPLY
SYSTEM 1
DUAL ACTUATOR (TYPICAL)
COLLECTIVE PITCH
YAW
PITCH
ROLL
SUPPLY
RETURN
SYSTEM 2

Figure 9-4. Dual-powered Stick Boost Hydraulic System

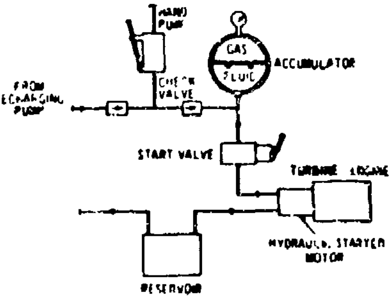

HAND PUMP
ACCUMULATOR
GAS
FLUID
FROM RECHARGING PUMP
CHECK VALVE
START VALVE
TURBINE ENGINE
HYDRAULIC STARTER MOTOR
RESERVOIR

Figure 9-5. Hydraulic Starting; Energy-limited System

accomplished by releasing stored energy from a charged accumulator. This energy drives a positive-displacement starter mounted on the engine. The starter may consist of a simple fixed-displacement motor, and is sized so that no special controls are needed. Weight can be minimized by using the highest pressure and largest displacement acceptable to the motor. These two factors are limited by the torque capacity of the mounting pad and the fuel control acceleration capabilities of the engine. This type of starting system is most appropriate for starting small engines (50-150 hp), where the accumulator size and charging time are not excessive.

The power-limited type of starter usually consists of a self-sufficient system that uses a small turbine engine as an APU (Fig. 9-6). Hydraulic fluid is stored in an accumulator that has been pressurized either by hand pump or by a previous operation of the hydraulic system. The pressure released by energizing a solenoid-operated starter valve drives a fixed-displacement starter pump, which acts as a motor to start the APU. When the APU starts, the starter pump converts to a pumping mode so as to drive the main engine starter. This starter senses the proper flow or pressure, and its displacement is varied automatically so as to accelerate the engine to idle speed. The power limit of the APU is not exceeded because of this variable-displacement feature. In power-limited APU systems, the starter/pump is sized for the maximum output capability of the auxiliary engine because the starting requirements are lower than are the pumping requirements.

The system shown in Fig. 9-7 uses an energy-limited, dual purpose, starter-pump system on the APU, and a power-limited, variable-displacement starter on the main engine. As the APU is started, the starter/pump drives a fixed-displacement motor mounted on the accessory gearbox. The gearbox motor drives all accessories, including the utility pump(s), which in turn provides the power to drive the main engine starters. After the start cycle is completed, the main engine drives the accessory gearbox. An added advantage of using an APU starting system is that it can be operated to provide power for ground checkout.

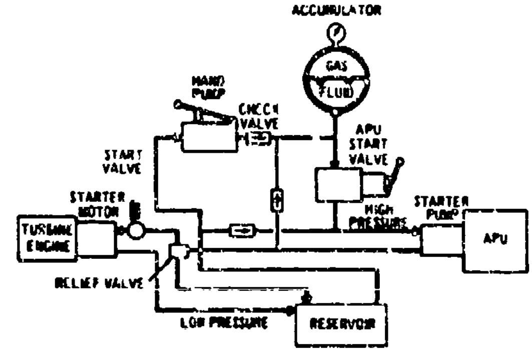

ACCUMULATOR
GAS
FLUID
HAND PUMP
CHECK VALVE
APU START VALVE
START VALVE
STARTER MOTOR
TURBINE ENGINE
HIGH PRESSURE
STARTER PUMP
APU
RELIEF VALVE
LOW PRESSURE
RESERVOIR

Figure 9-6. Hydraulic Starting; Power-limited System

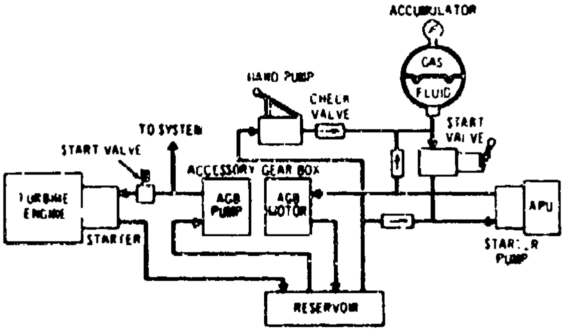

ACCUMULATOR
GAS
FLUID
HAND PUMP
CHECK VALVE
TO SYSTEM
START VALVE
START VALVE
ACCESSORY GEAR BOX
TURBINE ENGINE
AGB PUMP
AGB MOTOR
APU
STARTER
STARTER PUMP
RESERVOIR

Figure 9-7. APU Starting System

9-2.2.2 Cargo Door and Ramp System

Cargo and/or troop carrying helicopters normally will incorporate some type of cargo door and ramp system. The system shown in Fig. 9-8 is actuated by two direct-acting hydraulic cylinders. It is important to note that the actuators are self-locking in the retracted position. The manual control valve *shall* be located conveniently near or adjacent to the ramp. Actuation of the control valve directs pressure to release the actuator locks, and the ramp then is pulled down by the force of gravity. This is an important feature because the actuators may be unlocked via hand-pump pressure when utility system pressure is not available, allowing the ramp and door to free-fall open as an emergency measure.

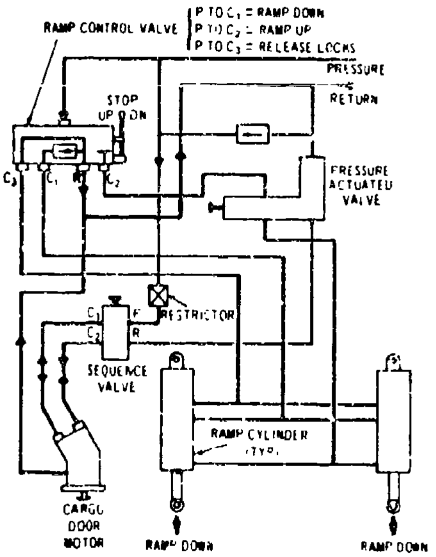

RAMP CONTROL VALVE
P TO C₁ = RAMP DOWN
P TO C₂ = RAMP UP
P TO C₃ = RELEASE LOCKS
PRESSURE
RETURN
STOP
UP DN
C₃
C₁
R
C₂
PRESSURE ACTUATED VALVE
RESTRICTOR
C₁
C₂
F
R
SEQUENCE VALVE
RAMP CYLINDER (TYP)
CARGO DOOR MOTOR
RAMP DOWN
RAMP DOWN

Figure 9-8. Cargo Door and Ramp System

In normal operation, initial movement of the ramp actuates a sequence valve that permits flow to a constant-displacement motor. The motor may be mounted inside the ramp, and connected to the cargo door by an endless chain, so that it can be retracted into the ramp structure. The sequence valve blocks downstroke flow from the ramp actuators until the door is completely retracted. When the door is positioned properly, a hydromechanical stop halts flow through the motor. Decay of motor back pressure opens the sequence valve, allowing the ramp actuator to bottom out fully or to travel to the point where the ramp touches the ground. Closing of the ramp is essentially the same but in the reverse order.

9-2.2.3 Cargo and Personnel Hoist

A utility hoist can be provided for loading and unloading cargo and for rescue operations. The hoist, as shown in Fig. 9-9, is powered by a hydraulic motor. The motor requirements are established so as to provide a particular hoist weight capacity and maximum reel-in speed. The speed can be made infinitely variable within the rated speed range by means of a hydraulic servo valve. The servo valve control signal is generated by a potentiometer incorporated into a control knob in the cockpit or the hoist operating station. For the hoisting operation, pressure is directed to the "in" port of the motor. For extending the hoist, a pressure reducer should be used to provide the relatively lower motor torque needed for cable extension. A flow regulator incorporated into the return line, downstream of the control valve, regulates flow in both directions. Limit switches can be provided to stop the hoist motor at the full reel-in and full reel-out positions. A hydraulically released, spring-loaded, "on" brake can be provided for fail-safe operation of the rescue hoist. Provisions should be incorporated to stop the motor as the hook passes through the cargo-rescue hatch in order to prevent whipping of the cable. In addition, a device can be incorporated to keep the cable under tension under no-load conditions.

9-2.2.4 Rotor Brake

The rotor brake *shall* be capable of stopping the rotor within a preselected time period for a specific range of rotor speeds following engine shutdown, and the brake *shall* be capable of holding the rotor stationary when full ground idle engine torque is applied. As shown in Fig. 9-10, the brake may be applied by a solenoid-energized valve that directs pressure to the rotor brake. A precharged accumulator provides a steady hydraulic pressure for braking when the main system is depressurized. A pressure switch should be incorporated in order to provide a cockpit warning light indication when the brake is on. To release the brake, the solenoid valve is de-energized to the off position, allowing the brake pressure to bleed off to the return system. Springs may be used in the brake assembly so as to overcome the return system back pressure in order to allow separation of the pucks from the rotor disks. The rotor brake *shall* be designed to be fail-safe. For small helicopters a simple rotor brake system may be satisfactory.

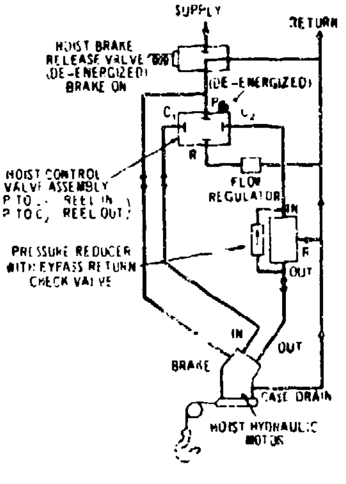

SUPPLY
RETURN
HOIST BRAKE RELEASE VALVE (DE-ENERGIZED) BRAKE ON
(DE-ENERGIZED)
P
C1
C2
R
HOIST CONTROL VALVE ASSEMBLY (P TO C1 REEL IN) (P TO C2 REEL OUT)
FLOW REGULATOR
IN
R
OUT
PRESSURE REDUCER WITH BYPASS RETURN CHECK VALVE
IN
OUT
BRAKE
CASE DRAIN
HOIST HYDRAULIC MOTOR

Figure 9-9. Cargo and Personnel Hoist (Constant Pressure) System

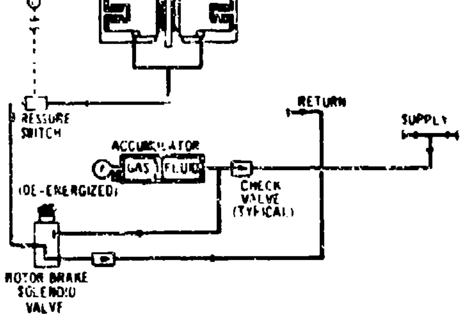

COCKPIT WARNING LIGHT
ROTOR BRAKE
DISK
PRESSURE SWITCH
RETURN
SUPPLY
ACCUMULATOR
GAS
FLUID
(DE-ENERGIZED)
CHECK VALVE (TYPICAL)
ROTOR BRAKE SOLENOID VALVE

Figure 9-10. Rotor Brake System

9-2.2.5 Wheel Brakes

Hydraulically powered wheel brakes *shall* be designed in accordance with MIL-P-8585. The circuit shown in Fig. 9-11 represents a braking system with differential control. Each brake pedal provides a direct input to a master brake cylinder. Hydraulic pressure may be reduced as necessary for brake operation. Parking brake capability also should be incorporated. A warning indicator *shall* be provided to signify when the parking brakes are "ON".

9-2.3 HYDRAULIC SYSTEM RELIABILITY

Good hydraulic system reliability can be obtained best by recognizing the probable weak links during the initial design. Because the system must be designed within the constraints of weight and cost, and must meet appropriate Military Specifications, reliability aspects of the design must be optimized.

9-2.3.1 Flight Control Redundancy

The most critical hydraulic failures are those that ultimately cause loss of the capability to operate the primary flight controls. MIL-H-5440 requires that, wherever hydraulic power is used for the primary flight controls, a completely separate system *shall* be provided for that purpose. Further, it is required that if direct mechanical control is not sufficient to allow controllability as defined in MIL-F-8785 in event of a hydraulic failure, an emergency power source *shall* be provided in order to supply the necessary controllability.

As a means of meeting the redundancy requirements, several design techniques may be considered:

1. A single primary flight control system with mechanical reversion. This method is simple and relatively lightweight, and can be used if operating loads are not above pilot and/or structural capabilities.

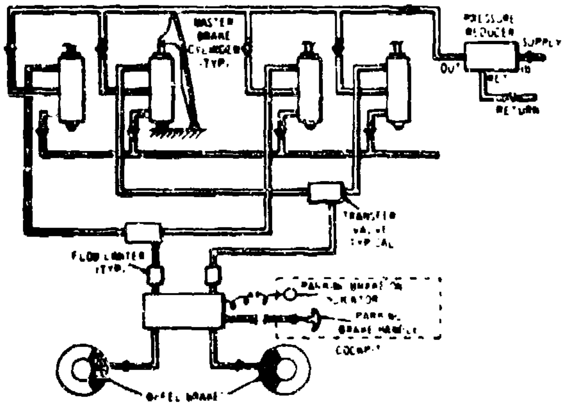


Figure 9-11. Wheel Brake System

2. A single primary flight control system with integrated, electrically powered hydraulic backup. The survivability characteristics of this design are good but may result in a heavier and costly installation. It also may present heat rejection problems.

3. Dual flight controls. This may be the best approach if high power must be delivered to the flight control actuators. Dual or tandem actuators can be used to enhance reliability further. However, this type of system involves more weight, more components, and, therefore, higher cost than other approaches.

4. A single flight control system with active utility system backup. This is an approach which should be considered if the utility system can supply power sufficient for the normal utility functions, as well as approximately two-thirds of the hinge moment required for the flight control functions. Use of priority valves in the utility system should be considered in order to insure that priority is given to the flight controls in the event of loss of the independent flight control hydraulic system.

9-2.3.2 Utility System Redundancy

In general, utility functions are not critical individually to the control of the helicopter; therefore, total system redundancy need not be considered. However, specific functions within the utility system may be critical during emergency conditions. If so, emergency or alternate modes of operation should be considered. Usually, the least reliable component within the system is the hydraulic pump, due to its relative complexity, high operating loads, and continuous operation. The use of two utility system pumps can provide additional reliability. Each pump can be sized to provide one-half of the maximum system power demand, thereby providing a good weight and cost trade-off in comparison with a single, large system pump. If one of the two pumps fails, system performance is reduced, but only the failure of both pumps can cause total system loss.

9-2.3.3 Miscellaneous Reliability Aspects

The key to high system reliability is reduction of the effects of single-point failures or elimination of their cause. Major types of recurring failures are:

1. External component leaks, causing loss of system fluid

2. Leakage of precharged accumulator gas into the hydraulic fluid, causing pump cavitation or dumping of fluid overboard.

The effects of component leakage can be alleviated significantly by use of a leakage isolation device in each flight control system branch circuit. The various types of such devices are discussed in par. 9-4.5.4, AMCP 706-201.

Loss of systems due to accumulator precharge gas leakage can be reduced to a great extent by eliminating the need for a main system accumulator. One method is to design a system relief valve having the response capability necessary to dump high-pressure fluid to the low-pressure return side of the system quickly enough to reduce the amplitude of pressure spikes.

If a variable-delivery, pressure-compensated pump is used, pressure surges can be reduced by designing the pump with a derivative compensator. The compensator anticipates pressure surges by sensing the rate of pressure rise, and thus acts to reduce the magnitude of that rise before it reaches a critical value.

The design approaches discussed in this paragraph represent the latest state-of-the-art developments and, therefore, may not prove to be the best methods. However, their considered usage could be worthwhile in increasing system reliability.

9-2.4 HYDRAULIC SYSTEM STRENGTH CONSIDERATIONS

Hydraulic system components and attaching linkages *shall* be designed so as to meet the most critical loads or combination of loads. Load factors or design factors *shall* be established for systems in order to insure adequate safety and life of the components. Where applicable, load factors must comply with Military Specification requirements. When no Military Specification requirement exists, these factors must be determined and assigned in accordance with good design practice. The following basic criteria *shall* be considered in establishing these factors:

1. The structure must not suffer fatigue when subjected to normal working loads. The unit stress under normal working loads must be limited so as not to exceed the fatigue strength of the material, under repeated loading, for the anticipated life of the structure, with stress concentration factors taken into consideration.

2. The structure must not yield when subjected to maximum expected loads. The unit stress at the limit load must not exceed the yield strength of the material. It should be recognized that test loads may be imposed upon the structure that may exceed the maximum limit load encountered after installation. No part of the structure should take any permanent set or experience any damage when subjected to applicable test loads.

3. The structure must not fail at the ultimate load. Where applicable, ultimate strength of material should be determined by bending or torsional modulus of rupture.

4. Deflection of components must not cause malfunction. Deflection, rather than strength, is often the major criterion for design of hydraulic system components and associated structural elements. When this is the situation, the design *shall* be based upon limit loads rather than ultimate loads.

5. Temperature variations must not cause malfunctioning or excessive stress. Consideration must be given to expected temperature variations so that no binding, sticking, or malfunctioning of components will result. Internal stresses, such as those resulting from the use of dissimilar materials in combination, should not exceed allowable stresses under the most adverse temperature conditions. Where components are expected to operate at extremely high temperatures, allowable unit stresses may be reduced.

Hydraulic system design pressures (operating, proof test, and burst) *shall* be determined in accordance with Table I of MIL-H-5440. The design should be based upon the most critical condition. In addition, MIL-H-5440 requires that all hydraulic systems and components that are subjected, during operation of the aircraft, to structural or other loads not of hydraulic origin *shall* withstand such loads when they are applied simultaneously with appropriate proof pressure as specified in Table I, without exceeding the yield point at the maximum operating temperature. MIL-H-5440 also requires that actuating cylinders and other components, and their attaching lines and fittings, if subject to accelerated loads, *shall* be designed and tested on the basis of a pressure equal to the maximum pressure that will be developed, without exceeding the yield point at the maximum operating temperature.

9-2.5 HYDRAULIC SYSTEM TEMPERATURE CONSIDERATIONS

Hydraulic fluid selection criteria include the expected range of operating temperatures, time at extreme temperatures compared to available means of temperature control, and fluid physical properties at expected temperature levels. Where ambient and structural temperatures are above the hydraulic fluid flash and/or fire points in a compartment, the potential fire hazard must be considered. Fluid stability is affected by thermal stress, which can result in changes in viscosity and formation of volatile components, insoluble materials, and corrosive deposits.

Hydraulic system efficiencies are reduced by high fluid viscosity at lower temperatures, which results in inlet problems with pumps, sluggish response of critical actuators, power loss in transmission, and weight penalties due to line size. At high temperatures, low fluid viscosity can cause internal leakage and slippage in pumps, actuators, and valves. Compressibility of a fluid increases with pressure and temperature, and the resultant loss of volume output of pumps is a significant design consideration. In control systems, compression of fluid provides a mass-spring condition that can limit system response. Successful operation of the hydraulic system throughout the design temperature range depends upon the interaction of the hydraulic fluid with all other components of the system.

The designer *shall* consider the temperature distribution within the helicopter in order to achieve judicious placement of hydraulic system plumbing and components in the cooler regions. However, it may be necessary to locate portions of the hydraulic system in high-temperature regions of the airframe. Radiation shielding and proper ducting for air cooling may be necessary for such equipment. The design of actuators located in extreme-temperature areas should be such as to provide for continuous exchange of fluid to assist in controlling temperature. This may be accomplished by controlled internal leakage, which likely will be inherent in flight control actuators, but special consideration also must be given to utility actuators, which may be subjected to extreme thermal environments for extended periods of time when in a static or nonoperating condition.

Although intermittent-actuating systems must be designed to operate with cold hydraulic fluid, continuous-power-transmission systems will reach their normal operating temperatures rapidly regardless of original ambient temperature. Heat exchanger and viscosity considerations favor use of the highest operating temperatures that are possible without encountering fluid breakdown and excessive wear of moving parts. For MIL-H-83282 fluid, the normal upper limit is 275°F maximum (Class II system). However, high-temperature design tends to impose a cost penalty due to the attendant requirement for special materials, along with a reduction in system life because of the reduced lubricating properties of the fluid. In many applications, normal heat losses from lines and components are adequate to maintain hydraulic system temperatures within design limits; thus, use of heat exchangers is unnecessary.

For analytical purposes, the assumption of uniform temperature throughout a hydraulic circuit usually is quite accurate. When a pressure drop occurs without external work resulting — i.e., losses through orifices and tubing — the hydraulic fluid temperature rises by 7°F per 1000 psi drop for each circuit. However, this rise normally is dissipated as the fluid passes through the system.

9-2.6 HYDRAULIC SYSTEM DESIGN

Hydraulic system design begins with an evaluation of the system as it evolved in the preliminary design phase. The production configuration requirements are compared with the preliminary design, and the necessary changes are incorporated. In addition, a final examination of the system *shall* be made in order to ascertain not only that it meets the procurement specification, but that it is optimum with regard to weight, cost, and performance. MIL-H-5440 and its associated specifications will be part of the procurement specification.

The subsequent discussion covers the major system design areas, including number and type of systems, operating pressure selection, fluid media selection, filtration, fittings, power levels and transmission system optimization, and heat rejection requirements. The starting system is discussed, as are system analysis, including failure mode and effect analysis (FMEA), and reports required for meeting the helicopter procurement specifications.

The component selection and design requirements also are discussed, as are system installation design and areas of good design practice with general application to components or installation.

The hydraulic system is influenced by the size and complexity of the helicopter and by the procurement specifications. Following is a discussion of the critical elements of hydraulic system design.

9-2.6.1 Survivability, Reliability, and Safety Tradeoffs

The problems of designing for combat survivability and for system reliability are similar. Redundancy of systems and components may be required in either case. For survivability, the requirement may be a tolerance of two hits anywhere in the flight control system without loss of the capability of returning safely to base. The elimination or reduction of the possibility of hydraulic system fires associated with incendiaries also may be a requirement. Reliability aspects involve losses due to equipment failures. The loss requirement may be stated in terms of "no more than *n*" losses per 100,000 noncombat missions. The component and/or subsystem reliability requirements may be stated as mean time between failures (MTBF), which must be demonstrated by analysis, test, and service.

Trade-off/optimization studies are required in order to establish the basic system configurations that will meet the previously mentioned goals. The alternatives that may be considered are:

1. Two or three independent, normally operative systems
2. Normally operative system plus emergency backup systems, the backup(s) normally being inoperative until loss of a primary system
3. Normally operative systems plus hydraulic circuit breakers (HCB)
4. Intersystem switching for redundancy with appropriate provisions against loss of fluid in the newly applied system if first system failure is due to loss of fluid
5. Combinations of independent system plus intersystem switching and hydraulic circuit breakers
6. Use of armor in conjunction with system redundancy
7. Reversion to manual control where applicable and possible.

For example, computer analyses indicate that two systems with HCB incorporated are nearly as survivable as three independent systems, and that significant weight and cost savings may result from this approach. A complete analysis of weight and cost impact versus survivability and reliability *shall* be conducted as early in the program as possible. Close coordination is required with the procuring activity during this phase in order to insure that contract requirements are being met and that the design is approved.

The combat safety aspects are concerned primarily with fire resulting from battle damage. Fire-resistant fluids are developed and *shall* be considered.

HCB concepts also should be considered as to their fire minimization impact since they can limit significantly the amount of fluid dumped into a fire.

Noncombat safety aspects are focused primarily on ground maintenance aspects of the system operation. This facet is covered in detail in par. 9-2.8. However, a basic flight control safety requirement dictates that if there is a complete power failure, at least one system *shall* be driven by the autorotating rotor.

9-2.6.1.1 Reservoir Level Sensing

Reservoir level sensing (RLS) is a technique which uses the reservoir to operate subsystem pressure shutoff valves mechanically. The return portion of the system is isolated by a check valve located in the return line. Both the mechanically operated pressure shutoff valve and the return check valve *shall* be located as close to the central power source as possible in order to provide the maximum central power source protection. RLS allows a reduction in pump replacement, filter changing, and flushing system maintenance. The RLS concept can sense and isolate a leak of any magnitude, thus reducing subsystem component and/or line leakage failures that normally cause complete system loss. RLS is detailed in par. 9-4.5.4, AMCP 706-201.

9-2.6.1.2 System Switching Concepts

When normally operated or passive backup systems are used as backups for flight control systems, intersystem switching is an important consideration. The switching function must be very reliable, and must provide for elimination of intersystem leakage as a steady-state or transient condition.

One method of accomplishing the switching function is through the use of a lapped spool and sleeve, with pressure and return of both systems on the same spool. To minimize intersystem leakage, only the returns are associated directly so that differential pressure between systems is minimized (Fig. 9-12).

A second alternative is use of check valves to separate the pressure side of the systems. A power (pressure and spring) shuttle is used to switch the return systems. The pressure and return switching are accomplished independently. This approach may not be acceptable under certain dynamic conditions of system operation (Fig. 9-13).

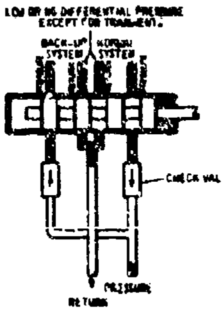


Figure 9-12. Combined Spool Switching Valve

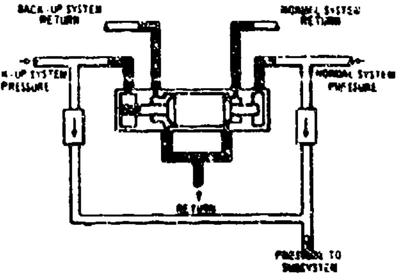


Figure 9-13. Pressure Check Valves Plus Power Return Switching

A third alternative is the use of check valves to separate the pressure side of the systems, in conjunction with an inline relief valve in a common return line. The relief valve represents an energy loss in the normal system, and acts to keep the backup system fully serviced as a part of the normal mode of operation (Fig. 9-14). Upon reversion to the emergency (backup) mode, the relief valve setting is high enough so that maximum generated return pressures will not operate it. Where this concept is used as a part of a dual system actuator, anticavitation valves are required in order to keep the relief valve setting at a reasonable level.

A variation of Alternative 3 is the use of a mechanical locked-out relief valve to eliminate energy loss during the normal mode of operation. This is accomplished by using the last portion of the backup system reservoir stroke on filling to hold the relief valve off its seat mechanically. The initial stroke of the reservoir allows the valve to reseat and preserve the system integrity upon reversion to the backup system (Fig. 9-15).

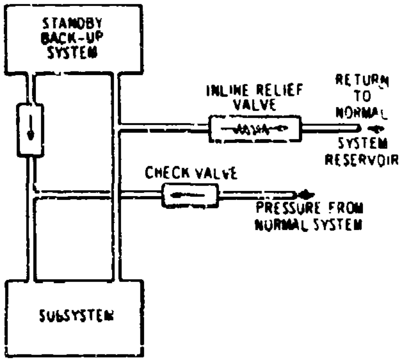

STANDBY
BACK-UP
SYSTEM
INLINE RELIEF VALVE
RETURN TO NORMAL SYSTEM RESERVOIR
CHECK VALVE
PRESSURE FROM NORMAL SYSTEM
SUBSYSTEM

Figure 9-14. Pressure Check Valves Plus Inline Return Relief Valve

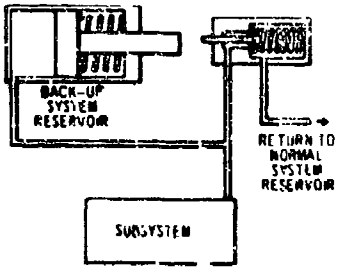

BACK-UP
SYSTEM
RESERVOIR
RETURN TO NORMAL SYSTEM RESERVOIR
SUBSYSTEM

Figure 9-15. Inline Mechanically Locked-out Relief Valve

A variation of Alternative 1 involves use of poppet valves and a mechanically operated cam in lieu of the lapped spool and sleeve (Fig. 9-16).

Pyrotechnically operated valves also may be used to provide switching. This approach generally is irreversible, and replacement of the valves is required after operation.

9-2.6.1.3 Return Pressure Sensing

Return pressure sensing (RPS) is most appropriate for use with full-trail solenoid selector valves. The cylinder port(s) is connected to the return when the valve is in the de-energized position. A spring-opposed, pressure-operated shutoff valve is located in the system return at the valve return port or downstream. Normal system return pressures, by design, overpower the RPS piston and allow normal valve operation. Combat damage, or component or line failure resulting in external leakage, causes reduced subsystem pressure because the return check valve prevents reverse flow. The spring then operates the RPS valve, thus inhibiting operation of the solenoids and, in effect, preventing use of the damaged subsystem and loss of the complete system.

9-2.6.1.4 Switching and Return Pressure Sensing

The pressure-sensing concept can be used in conjunction with switching functions as a means to prevent switching a good backup system into a subsystem that has lost its pressure vessel integrity. A time delay of several seconds is integrated into the switching function, i.e., the initial motion blocks both

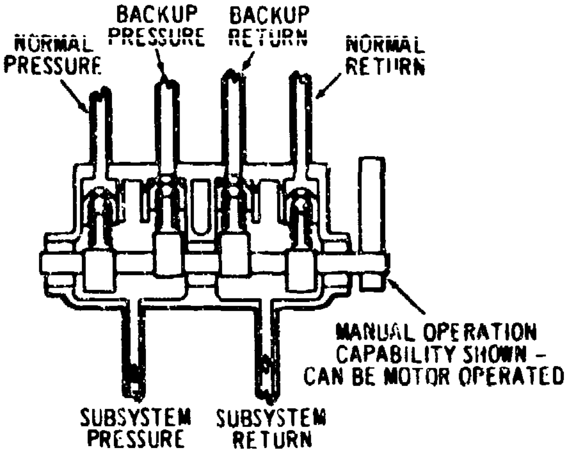

BACKUP PRESSURE
BACKUP RETURN
NORMAL PRESSURE
NORMAL RETURN
MANUAL OPERATION CAPABILITY SHOWN - CAN BE MOTOR OPERATED
SUBSYSTEM PRESSURE
SUBSYSTEM RETURN

Figure 9-16. Cam-operated Poppet Switching Valve

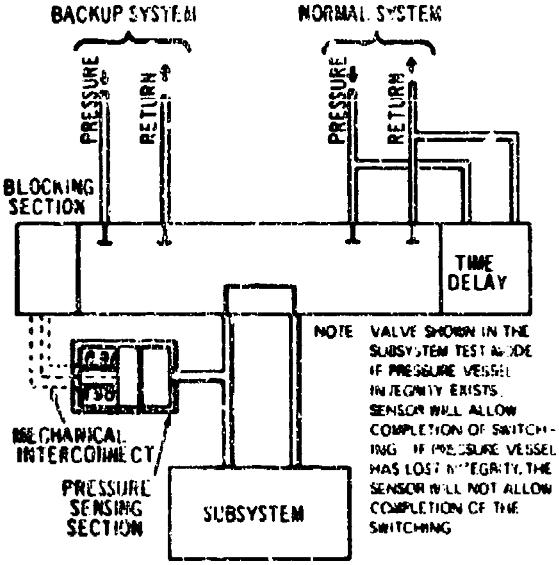
the normal and the backup systems. During the time delay, the pressure-sensing system tests the subsystem and mechanically inhibits the switching function if the subsystem does have an external leak. Fig. 9-17 contains a schematic of this approach.

9-2.6.2 Operating Pressure Considerations

Operating pressure is a function of helicopter size, complexity, and performance. Smaller helicopters with manual reversion capability may use low pressures, such as 1000-1500 psi, without significant weight penalties. Generally, dynamic seal life is better with reduced pressures. Larger helicopters will require 3000-psi systems in order to attain reasonable volumes and system weights.

The impact upon development, qualification, test and maintenance equipment requirements may be a strong motivation for maintaining 3000 psi or lower pressures. However, it is desirable to conduct a system pressure-weight-cost trade-off study as a means of determining optimum helicopter hydraulic system configuration. Par. 9-4.2, AMCP 706-201, contains a discussion of the pressure selection considerations as related to preliminary design functions.

The basic selection decision usually is made in the preliminary design phase, and a later re-evaluation may be necessary as helicopter requirements may change as the design advances into the hardware phase.



NOTE VALVE SHOWN IN THE SUBSYSTEM TEST MODE IF PRESSURE VESSEL INTEGRITY EXISTS. SENSOR WILL ALLOW COMPLETION OF SWITCHING. IF PRESSURE VESSEL HAS LOST INTEGRITY, THE SENSOR WILL NOT ALLOW COMPLETION OF THE SWITCHING

Figure 9-17. Switching Valve

9-2.6.3 Selection of Fluid Medium

MIL-H-5606 fluid is the most commonly used medium and has present widespread usage throughout the military world.

A synthetic hydrocarbon defined by MIL-H-83282 now is being considered for use in Army aircraft. Its primary attraction is that it is significantly less flammable than MIL-H-5606. Operational characteristics at very low temperatures have not as yet been fully established.

For a thorough discussion of hydraulic fluids, refer to AMCP 706-123.

9-2.6.4 Filtration (Contamination)

System fluid filtration and external contamination factors must be considered since they have a direct effect upon the reliability and serviceability of the hydraulic system.

9-2.6.4.1 Fluid Filtration

Several filtration methods are available. These include: central filtration, subsystem filtration, return-case drain concepts, and suction line filtration.

Central filtration involves a pressure filter, a return filter, and a pump case drain filter. This filtration method requires that all critical components, such as flight control actuators, *shall* have inlet screens. These screens *shall* be in the size range of 100-150 micron absolute particle, i.e., particles with two dimensions larger than the maximum allowable *shall* be blocked or prohibited from passing through.

Subsystem filtration involves a pressure filter that either is integrated into or is immediately upstream of each component or subsystem. The return filter generally will be of a common single configuration. The case drain from each pump *shall* have a separate filter.

The return case drain philosophy applies to both of the foregoing categories. The case drain and return system filter functions may be combined within a single filter assembly. The assembly may contain either a single- or two-stage element. If the element is two-stage, the full or transient flow stage *shall* be per MIL-F-8815 (5 microns nominal, 15 microns absolute) and the low flow or bypassing stage *shall* be 3 or 5 microns absolute. The capability of the single-stage element may be a compromise between the two, such as a 5-micron absolute capability.

Suction line filtration generally is not used because the suction pressure capabilities of pumps demand that the suction line pressure drop be kept to a minimum. This means a suction line element would be quite large and, therefore, bulky and heavy. The weight penalty usually is unacceptable.

Filtration is further discussed in pars. 9-4.2.2.3 and 9-4.4.5, AMCP 706-201.

9-2.6.4.2 Ground Operation Filtration

Ground carts generally incorporate their own filtration systems, which may be of 3-5 micron absolute capability. The helicopter system *shall* incorporate provisions for introduction of the ground cart fluid through the system pressure filter(s) as shown in Fig. 9-18. This provides for helicopter system protection if the ground cart filtration is defective.

The two available alternatives in filter elements are the noncleanable throwaways and the reusable, cleanable types.

9-2.6.4.3 Filtration Level

The required filtration level is defined in MIL-H-5440, which specifies that the filtration meet at least the requirements of MIL-F-8815.

The elements may be either noncleanable throwaways or cleanable, reusable elements. The cleanable units normally are fabricated from metal screens made with a Dutch twill weave, and are essentially two-dimensional; i.e., there is no relative depth in the fluid flow direction. The throwaway elements usually are three-dimensional in restricting particle size. While both may meet the same glass bead test criteria, the depth filter will do a better job of stopping fine particles that are smaller than the nominal rating of the element.

9-2.6.4.4 External Contamination

Experience has shown that sand and dust are primary contributors to poor hydraulic system component life. Damage to the hydraulic seal areas and to external rod finish by hard, cutting contaminants has caused many problems. The use of flexible protective boots is an obvious and desirable means of minimizing such effects. In addition, improvements in seals and rod scraper designs are required. Still another alternative is the use of rotary outputs in conjunction with crank arms for conversion to linear motion in lieu of linear actuators.

In any event, qualification and environmental test requirements that are much more stringent than those in present Military Specifications must be used in order to ascertain that new equipment will perform satisfactorily.

9-2.6.5 Fittings

To facilitate installation, system components and lines must have disconnect-connect points consistent with specific helicopter installation requirements. Separable, reusable connections are required for ease in removal and maintenance. However, in many cases, permanent fittings may be used in joining runs of tubing where access through the installation is not required. Par. 9-4.3, AMCP 706-201, contains a discussion of available fittings of both types.

There are several reusable fitting alternatives, including the MS flareless and AN flare standards, that are used for tube-to-tube and tube-to-component connections. These also include the Resistoflex "Dynatube" and several others, and are described in par. 9-4.3, AMCP 706-201.

MIL-F-18280 covers the reusable fitting requirements. The general test requirements are applicable to most fittings and aircraft. However, the specific usage area in the helicopter must be analyzed, and additional testing must be specified as considered necessary in order to insure that the fitting chosen will perform satisfactorily in service.

The Rosan fitting recently has become available as a component-line interconnect. This fitting is shown in cross section in Fig. 9-19. Its advantages over the standard AN boss and fitting combination include a superior seal design and a locking feature that eliminates the need for the use of a second wrench to hold the boss fitting when removing the line.

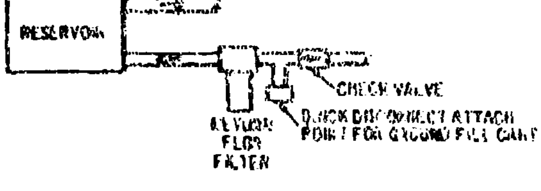


Figure 9-18. Hydraulic System Ground Fill Provisions

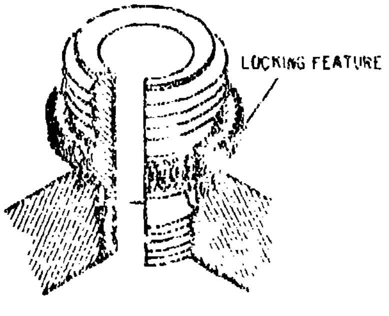


Figure 9-19. Rosan Boss Fitting

The use of titanium is being initiated in lines and fittings with significant weight savings. Techniques for eliminating galling between fittings have been perfected. A weight-cost trade-off covering the use of titanium may be performed in the final design configuration commitment phase.

9-2.6.6 Dynamic Fluid Connections

Swivels, hoses, and coiled or torsion tubing are alternatives to be considered where relative motion between an actuator and a structure must be allowed. Because these types of fittings may impose weight and cost penalties, other means of solving the relative motion problem must be considered seriously. The use of an articulating link between the structurally mounted and immobilized actuator and the moving control surface or other subsystem is an alternative that has been used frequently. Fig. 9-20 is an example of such an installation. Par. 9-4.3, AMCP 706-201, presents information covering dynamic fluid connections.

If hoses are the only alternative, close attention must be given to design of the installation. The minimum allowable bend radius of the hose must not be exceeded during motion of the actuator. The hose position relative to structure and areas on the actuator during motion must be analyzed in a layout drawing in order to determine that interference does not exist. In addition, a protective cover must be installed to protect the hose from abrasion damage due to vibration and g's, which can cause the hose to deflect outside its normal path during motion (Fig. 9-21).

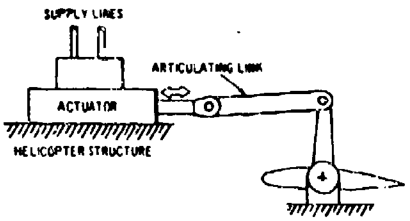


Figure 9-20. Use of an Articulating Link

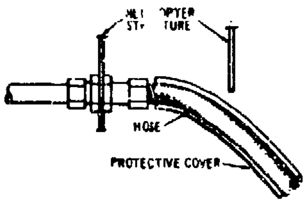


Figure 9-21. Use of Protective Cover on Hoses

Swivels may be used in conjunction with hoses, or as scissor assemblies, to resolve relative motion problems. The swivel assembly rigidity minimizes vibration and acceleration problems. However, the joints involve the use of dynamic seals, which can result in nuisance or catastrophic leaks.

The use of coiled or torsional tubing or articulating links is the preferred method.

9-2.6.7 Peak Power Levels

The determination of peak power levels, and, therefore, of hydraulic pump and line size, is detailed in par. 9-4.6, AMCP 706-201. It is desirable to develop a precise knowledge of the parallel-series operation of the subsystems so that adequate performance is attained at minimum weight and cost. Undue conservatism in hydraulic system design will penalize helicopter performance, insofar as weight and power extraction are concerned.

The same techniques of power system analysis that are discussed in par. 9-4.6, AMCP 706-201, *shall* be used in establishing system peak power requirements. In addition, a mission profile analysis *shall* be conducted to determine total energy requirements and the system heat load. Fig. 9-22 is a typical example of a mission profile requirement.

9-2.6.8 APU and Engine Starting

It is customary to use hydraulics for starting APU's and turbine engines, since the hydraulic starting system is self-contained and provides a capability for multiple starts. No external electrical or ground hydraulic carts are required, and a hand pump allows recharging of the accumulators for subsequent starts in the event that the first attempt is unsuccessful. The basic hydraulic starting system consists of air- or nitrogen-charged accumulators that are discharged through a hydraulic motor that is connected mechanically to the turbine engine or APU via

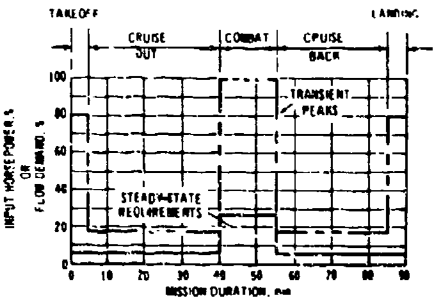


Figure 9-22. Typical Mission Requirement Profile

a manually operated selector valve. Self-displacing accumulators can be used to insure that the associated hydraulic system reservoir is kept to a minimum size and weight.

The cold start is a primary design point due to high line-loss characteristics with −65°F fluid temperatures. The motor requirements for equivalent warm oil output torques are not increased appreciably by −65°F fluids; however, flow rate will be greatly reduced until fluid temperature has increased.

Instantaneous or fast opening of the manual control valve can cause hydraulic motor shaft shearing or other damage due to the high-pressure shock wave. The control valve design should include features to provide for slow buildup of pressure (0.5 to 1.0 sec is reasonable). At high temperatures, the extra fluid energy available as a result of decreased line losses may require control if the APU or turbine engine is acceleration-limited due to a characteristic of a component or the basic gear train.

9-2.6.9 System Heat Rejection Characteristics

Low-power-level systems (low pressure of 1000 psi or less, and low flow of 5 gpm or less) generally do not require any special cooling equipment. Large, high-power-level systems operating in relatively warm ambient temperature regions may require heat exchangers in order to maintain fluid temperatures below the Type II upper limits of 275°F.

The performance specification for the heat exchanger includes the following requirements:

1. Media maximum inlet and outlet temperatures allowed
2. Media minimum and maximum mass-flow rates
3. Hydraulic fluid maximum inlet and outlet allowable temperatures
4. Hydraulic fluid minimum and maximum mass-flow rates
5. Allowable pressure drops in both the media and hydraulic sections.

The heat exchanger qualification test *shall* specify the following basic requirements in addition to a demonstration of performance:

1. Realistic impulse testing as a pressure vessel so as to insure adequate fatigue life
2. Environmental tests, including vibration, shock, and corrosion testing.

9-2.6.10 System Analysis

The system performance analysis necessitates definition of the requirements of each of the subsystems in terms of output requirements and resulting input needs. Flight control subsystems and utility subsystems are the basic divisions. Par. 9-4.6, AMCP 706-201, discusses the analysis or determination of power requirements for both types of subsystems. Where there are several subsystems, the maximum simultaneous need must be determined in order to define the peak output capability. As a part of the performance analysis, line-sizing criteria based upon maximum allowable transients must be used to insure that MIL-H-5440 and practical limits are met.

In order to insure that performance goals — including combat-damage tolerance and single- and dual-failure tolerances — are met, a failure mode and effect analysis (FMEA) must be conducted. This *shall* be conducted in as much detail as is practicable. This analysis *shall* be prepared in accordance with the system safety program plan of Chapter 3, AMCP 706-203.

MIL-H-5440 defines the hydraulic system data and reports that are required, including:

1. System requirement studies data
2. Design selection data
3. Developmental data
4. Production data
5. Schematic diagram
6. Hydraulic system design report
7. Hydraulic system nonstandard component cross-sectional assembly drawings.

The requirements of the helicopter detail specification, together with the MIL-H-5440 requirements, *shall* constitute the basic hydraulic system report requirements.

9-2.7 HYDRAULIC COMPONENT DESIGN AND SELECTION

The detail requirements for the various components needed to complete the hydraulic system are discussed in the paragraphs that follow.

9-2.7.1 Actuators

Actuators may be separated into two basic classes: flight control and utility. These two classes may be divided into rotary and linear output types. The flight control actuators may be dual- or single-system types with manual, mechanical, pilot control input signals. In addition, some units include an integrated control augmentation system (CAS), a stability augmentation system (SAS), and/or an autopilot system. The controls — manual and electronic — generally are combined with the electrical control mode, either in series or in parallel with the manual mode (Fig. 9-23).

The series mode provides limited authority and flight control surface motion without motion feedback to the pilot's control stick. The parallel mode of operation provides unlimited authority; in effect,

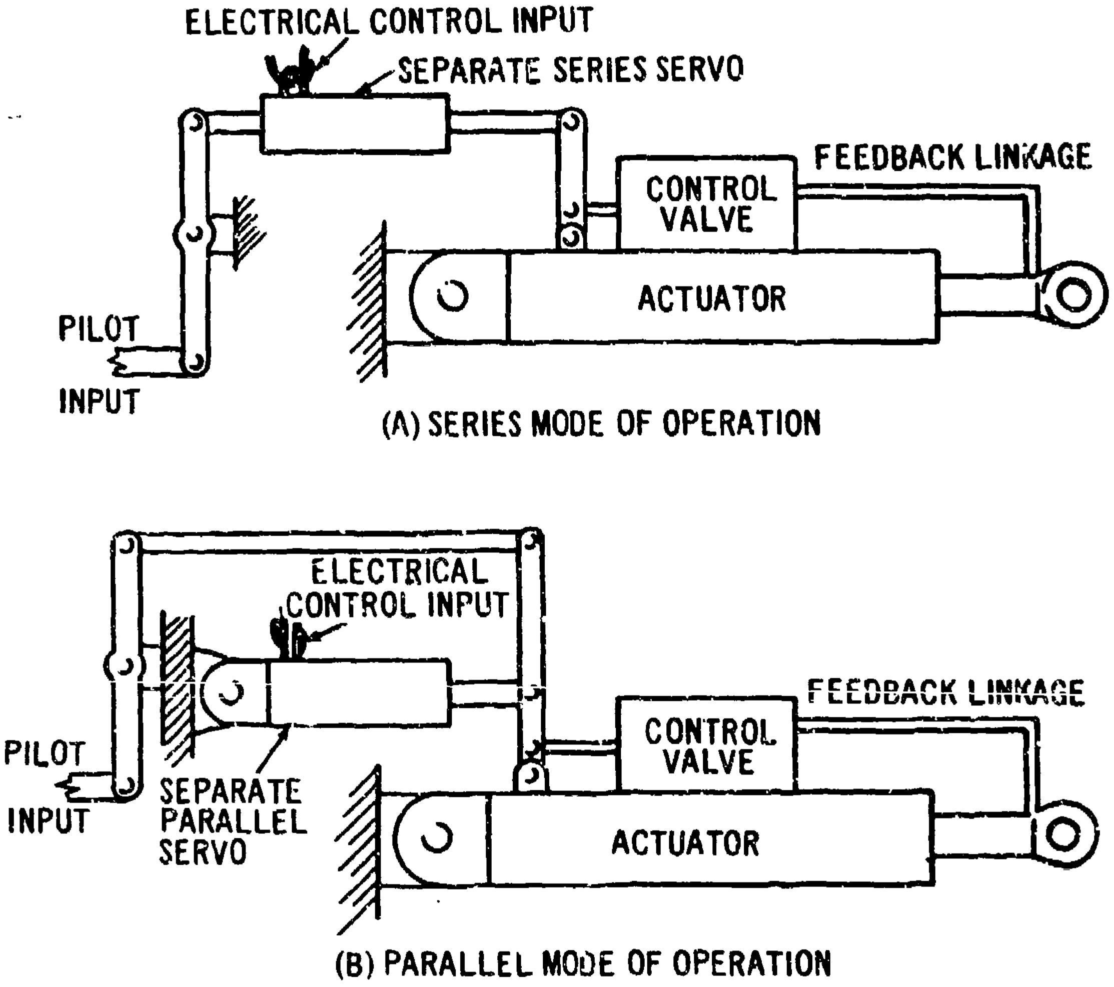

ELECTRICAL CONTROL INPUT
SEPARATE SERIES SERVO
FEEDBACK LINKAGE
CONTROL VALVE
ACTUATOR
PILOT INPUT
(A) SERIES MODE OF OPERATION
ELECTRICAL CONTROL INPUT
FEEDBACK LINKAGE
CONTROL VALVE
PILOT INPUT
SEPARATE PARALLEL SERVO
ACTUATOR
(B) PARALLEL MODE OF OPERATION

Figure 9-23. Examples of Parallel and Series Control Modes

dragging the stick along with the surface during motion. The augmentation and autopilot modes of operation use electrohydraulic valves, which receive electrical signals generally in proportion to the magnitude of the desired surface position change and then translate the signal into a hydraulic command to the actuator. The electrical feedback signal cancels out the electrical command signal when the control surface reaches the commanded position. Flapper and nozzle or jet pipe electrohydraulic control valves are used. Figs. 9-24 and 9-25 present schematics of the valves. The jet pipe is preferred because it has an inherently greater tolerance to contamination; however, this type valve has a higher leakage rate than the flapper and nozzle type which increases the hydraulic system heat load. This contribution to the heat load requires consideration in the decision on type of valve to be employed.

The master control valve that accepts the manual, electrical, or combined signal is a four-way, closed-center valve (Fig 9-26). For balanced-area cylinders in the neutral position, the cylinder port pressures are generally established at one-half the system pressure at the no-load position. In unbalanced actuators, the cylinder port neutral pressures are unbalanced in proportion to the area unbalance as required to obtain the necessary force balance. The spool of this servo is subject to jamming, particularly by thread-shaped particles which may have been scraped from surfaces of components by abrasives (predominately sand) which are introduced from the environment. Beside scrupulous care during maintenance and servicing to

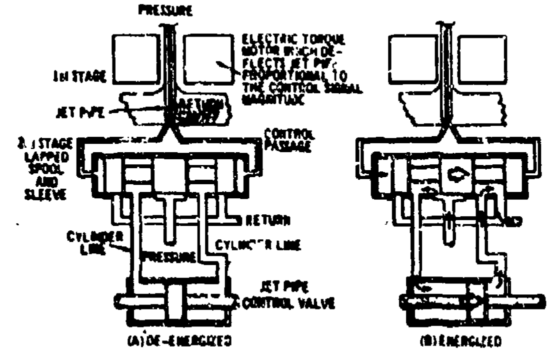

PRESSURE
ELECTRIC TORQUE MOTOR WHICH DEFLECTS JET PIPE PROPORTIONAL TO THE CONTROL SIGNAL MAGNITUDE
1st STAGE
JET PIPE
RETURN
2nd STAGE LAPPED SPOOL AND SLEEVE
CONTROL PASSAGE
RETURN
CYLINDER LINE
CYLINDER LINE
PRESSURE
JET PIPE CONTROL VALVE
(A) DE-ENERGIZED
(B) ENERGIZED

Figure 9-24. Schematic of Jet Pipe Electro-hydraulic Control Valve

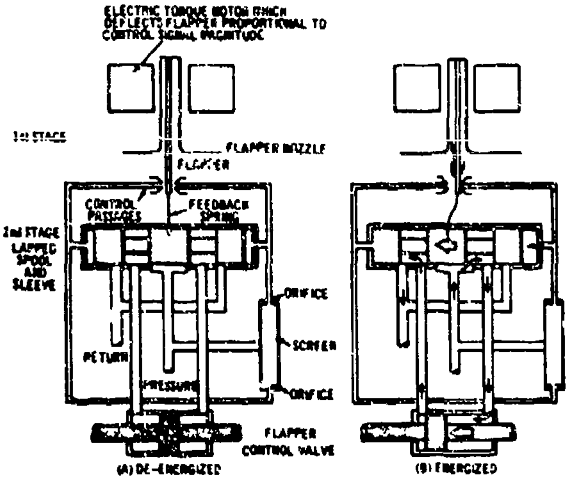

ELECTRIC TORQUE MOTOR WHICH DEFLECTS FLAPPER PROPORTIONAL TO CONTROL SIGNAL MAGNITUDE
1st STAGE
FLAPPER NOZZLE
FLAPPER
CONTROL PASSAGES
FEEDBACK SPRING
2nd STAGE LAPPED SPOOL AND SLEEVE
ORIFICE
SCREEN
RETURN
PRESSURE
ORIFICE
FLAPPER CONTROL VALVE
(A) DE-ENERGIZED
(B) ENERGIZED

Figure 9-25. Schematic of Flapper Electro-hydraulic Control Valve

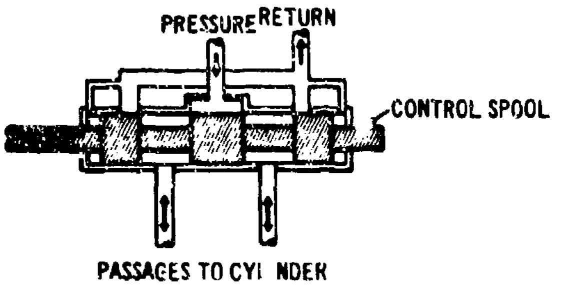

PRESSURE
RETURN
CONTROL SPOOL
PASSAGES TO CYLINDER

Figure 9-26. Typical Master Control Valve

preclude entry of contaminants, the obvious preventive measures are boots and effective scrapers and use of depth type filters which trap these thread-shaped particles far better than screen type filters can.

Where the electrical and manual signals are mixed, a linkage mechanism is used to combine the signals at the master control valve. The manual signal comes directly to the valve through an appropriate linkage. The electrohydraulic valve output flow is directed to a small auxiliary ram that, through an appropriate linkage, can move the master control valve as directed by the electrical signal.

The utility cylinders generally are relatively simple and do not include the control function of the subsystem.

9-2.7.1.1 Rip-stop Protection

When dual actuators are employed, or two independent systems are used simultaneously in a flight control actuator, experience indicates that a fatigue failure may originate in one system and propagate into the other, resulting in loss of both. If control cannot revert to manual mode in such a situation, the helicopter can be lost. This potential problem can be eliminated by use of the rip-stop technique, which requires the use of separate pressure vessels. The vessels can be joined by fasteners, brazing, or other techniques, and the discontinuity will preclude the possibility of a fatigue failure causing both systems to be lost. The present state of the art allows use of this technique without significant weight penalty. Therefore, it *shall* be used where possible.

9-2.7.1.2 Endurance Testing Requirements

Adequate endurance tests are essential if the actuator service lifetime is to be satisfactory. Two important design considerations are the dynamic seals and the integrity of the actuator as a pressure vessel. The endurance test program *shall* provide for a rigorous test of the basic and detail design concepts. Toward this end, the usage must be defined in depth; where test requirements in addition to those specified in the applicable Military Specification are desirable, they *shall* be specified. The additional test may include environmental, operational cycle, and/or pressure vessel impulse testing.

9-2.7.1.3 Seal Alternatives

Military Standard (MIL-G-5514, etc.) O rings, backup rings, and glands are used in dynamic seal applications. The backup ring is required for pressures above 1000-1500 psi, depending upon the specific application. Frequently, seal life requirements are such that special, nonstandard seals are used. The various types of nonstandard seals are discussed in par. 9-4.4.6, AMCP 706-201.

A useful technique involves the use of dual seals with the section between the seals vented to the return side of the system. The first stage is then the high-pressure seal, and the second stage must seal only the lower return pressures. Restrictors may be used in the vented section in order to inhibit large, short-circuit leakage flows.

Use of truly redundant seals is a new technique. This involves use of two seals in series without venting in between. The designer should, however, insure that fluid cannot be trapped between seals or that thermal effects will not cause failure.

Backups for standard O ring seals may be either standard scarfed or unscarfed (solid). The unscarfed backup *shall* be used whenever possible in lieu of the standard scarfed types.

Surface finish is important to seal life, and should be kept below 8 μin. A 16-μin. finish may be acceptable for short-life applications.

Static seals generally in use are the MS standard O rings with MS standard scarfed backups. As with the dynamic seals, unscarfed (solid) backups are preferred where the diameter is large enough so that they can be installed without damage. A 16- to 32-μin. finish is required for the O-ring glands.

The types of seals currently in use are discussed in par. 9-4.4.6, AMCP 706-20

9-2.7.1.4 Materials and Stress Considerations

Actuator stress considerations include control of stress raisers and use of appropriate materials with proven stress corrosion resistance and predictable fatigue life. Use of adequate internal and external radius is highly desirable. The finish must be kept in the range of 32 rms or below. A significantly rougher finish can offset generous radii by introducing many no-radius stress raisers.

Steels generally have better, more consistent fatigue-life characteristics than does aluminum, and, therefore, *shall* be used where long life is mandatory and flight safety important. MIL-C-5503 is applicable in this case, and contains detail requirements including environmental and life tests.

The lapped spool-sleeve control valve materials may be either SAE 52100 series steel, or corrosion-resistant steels such as Type 440C stainless. These materials have proven satisfactory in service usage, but 440C stainless must be cold stabilized to insure dimensional stability in order to avert subsequent jamming.

9-2.7.1.5 General Requirements

This discussion applies primarily to flight control actuators with integrated CAS, SAS, or autopilot modes of operation. For satisfactory service usage, it is desirable that the necessary control linkage be immersed in fluid inside the actuator. Jamming, corrosion, high friction, and unacceptable control system slop due to wear are problems that are eliminated and/or minimized by this method. It also allows the use of a rotary low-pressure dynamic seal as the access from the external manual control system to the actuator internal linkage. Such a seal is considered superior to a linear seal because more efficient sealing minimizes introduction of contaminants to the seal.

Mechanical override of a jammed servo requires an appropriate margin of yield/failure strength in the control linkage. Reliability of override success depends upon the severity of the jam, which, in turn, varies in some manner with size, shape, and material of the jamming particle. Dependence on mechanical override (as the sole back-up for a jammed servo) is not an acceptable technique except for small (light control force) helicopters. Hydraulic redundancy is the prescribed method for larger helicopters.

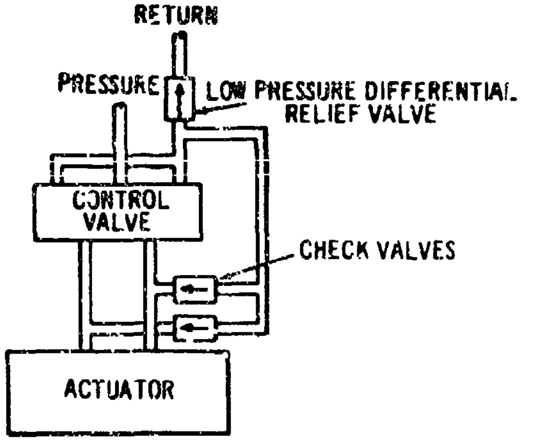
The linkage and the force input capability *shall* be designed so that chips that may get to the control valve can be sheared, thus avoiding catastrophic jamming of the actuator control. Generally, a 1000-lb force at the valve spool centerline is used to define the ultimate load-carrying capability of the control system linkage between the pilot input point and the valve spool. This is conservative, but usually deflection and other criteria are such that no weight penalty is involved.

For stable control system operation, both with power on and with power off, actuator stiffness can be important. The fluid spring rate capability of the actuator is a prime factor, because it is generally the softest portion of the total actuator spring rate capability. Adequate stiffness may be attained in several ways. The actuator area may be increased beyond that required for aerodynamic purposes so as to increase the fluid spring rate. Another alternative is to use hydraulic dampers in conjunction with actuator capability. Power-off damping may require the use of separate dampers in any event.

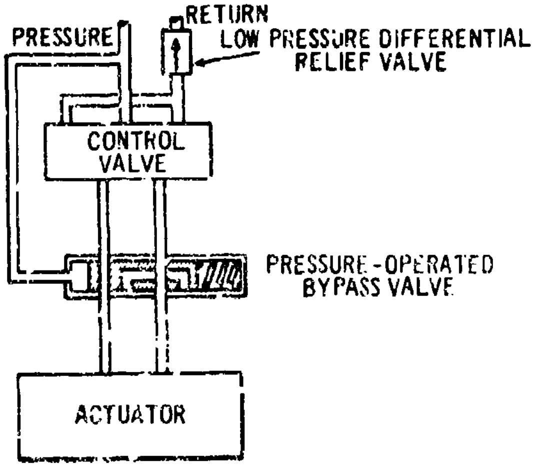
Dual actuator operation on a single system may require a hydraulic bypass or anticavitation feature. This prevents the pumping of fluid back to the system reservoir, and a resulting requirement for a larger, heavier reservoir. Another benefit is a significant reduction in the energy required from the remaining operative system in order to circulate fluid in the failed system. Anticavitation can be accomplished either by a pressure-operated bypass valve or by check valves that allow return flow to short circuit the control valve. Fig. 9-27 contains schematics for these two approaches.

In either case, an inline relief valve compensator is required to insure that fluid is retained within the actuator. The relief setting must be higher than the bypass flow pressures that can be developed during the maximum-flow condition.

The use of snubbing *shall* be considered wherever large assisting loads can exist. Snubbing reduces the impact load, but causes a local pressure intensification that must be taken into account during design and test. Therefore, a weight/design effectiveness trade-off is required to determine the optimum approach.


RETURN
PRESSURE
LOW PRESSURE DIFFERENTIAL RELIEF VALVE
CONTROL VALVE
CHECK VALVES
ACTUATOR

(A) USE OF CHECK VALVES


RETURN
PRESSURE
LOW PRESSURE DIFFERENTIAL RELIEF VALVE
CONTROL VALVE
PRESSURE-OPERATED BYPASS VALVE
ACTUATOR

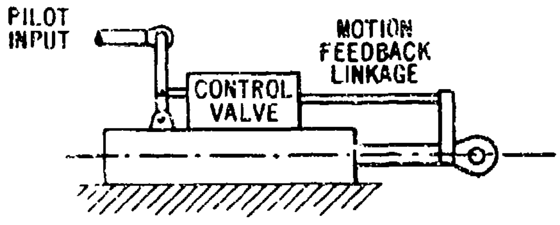
(B) USE OF PRESSURE-OPERATED BYPASS VALVE

Figure 9-27. Anticavitation Approaches

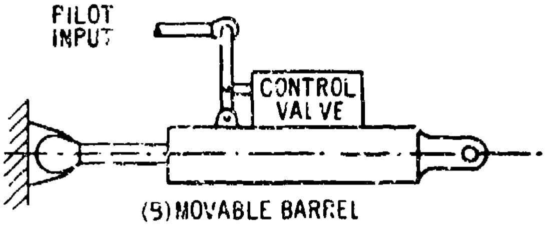
Feedback for position control may be attained in at least three ways: with a fixed barrel, with a movable barrel, or electrically. The fixed barrel involves a moving rod and an associated mechanical linkage interconnect between the moving rod and the pilot input linkage. This approach allows the use of direct tubing connections since there can be little or no relative motion. The moving barrel gives direct feedback, but requires hoses, swivels, or other means of handling the motion. The third alternative involves the use of transducers in conjunction with electrical control. Fig. 9-28 contains schematics of the alternative approaches.

Use of an inlet check valve is desirable, particularly when dual actuators are used. If the aerodynamic load during single-system operation can overpower the system, the use of a check valve will allow holding of the position involved when the first system failure occurs until the hinge moment or load can be reduced to a controllable level by reduction in helicopter speed or other means.

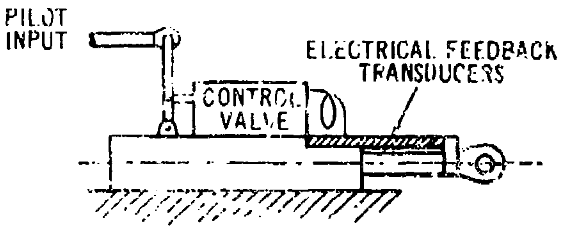
The vane or motor type of rotary actuator may be used to advantage in helicopters. An obvious area is in servo motor gun turret drive applications. Vanes may be used as combination actuator-dampers.


PILOT INPUT
MOTION FEEDBACK LINKAGE
CONTROL VALVE

(A) FIXED BARREL


PILOT INPUT
CONTROL VALVE

(B) MOVABLE BARREL


PILOT INPUT
ELECTRICAL FEEDBACK TRANSDUCERS
CONTROL VALVE

(C) ELECTRICAL FEEDBACK

Figure 9-28. Feedback Techniques

9-2.7.2 Hydraulic Pumps

The hydraulic pump is the heart of the hydraulic system, and, therefore, is of prime importance. It is, however, the primary source of system heat energy due to its inherent inefficiencies.

The basic alternatives for pumps are the fixed-displacement and the variable-displacement, constant-pressure types. The fixed-displacement unit can be used with "open center" utility systems, or with relief valves or combination unloader valve-accumulator systems, for pressure control. The fixed-displacement system is relatively heavy. In addition, it tends to have higher heat generation levels than the variable-displacement type. Consequently, the fixed-displacement pump is not in general use except in low-pressure-level systems.

The conventional variable-displacement pump has constant-flow capability through speeds up to the pressure control cut-in point, which is approximately 2850 psi for a 3000-psi system. As the demand decreases below the full displacement capability, the control de-strokes the pump until, at zero output flow, the pressure is maintained at 3000 psi nominal. The compensated pressure setting can vary from 2950 to 3050 psi between pumps. Fig. 9-29 shows these characteristics.

Modifications of the basic control are available. One variation used for horsepower limiting is the soft-cutoff, relatively constant horsepower approach. Fig. 9-30 shows the control reducing flow at 1500 psi and continuing on a linear basis until it reaches the standard sharp pump level. There, it reverts to sharp cutoff control. Superimposed on Fig. 9-30 is a constant-horsepower line that is tangent to the soft cutoff and that, when compared to the peak horsepower of a sharp cutoff pump, shows a significant reduction in the power required to drive the pump. This is a definite advantage where the driving horsepower is limited, and significant weight can be saved in, for example, an electric motor. However, some of the weight savings is offset by the larger lines required to maintain subsystem rates with 1500 psi available for load and line loss versus the conventional 3850-psi capability.

Another type of control offers a high response capability because of the large control valve used with pump hanger feedback as required for stabilization of the pump control system. The capability to respond to both on and off demands allows the deletion of the accumulator, resulting in savings in weight, maintenance, and servicing. In addition, the pump has better overshoot-undershoot characteristics than has the conventional pump-accumulator system. Therefore, pressure peaks are lower and system fatigue life is improved.

Pump case drain characteristics are important to satisfactory system and pump operation. The pump *shall* have a minimum case drain flow as a function of back pressure. This is required in order to eliminate pump overload and failure during operation in the compensated or near-compensated mode. An upper limit on case pressure buildup due to flow restriction is necessary in order to avoid loss of the system as a result of failure of the pump or case drain system as a pressure vessel. These minimum-maximum case drain characteristics are shown in Fig. 9-31.

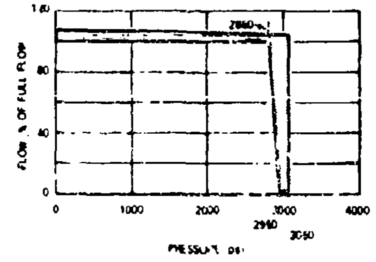


Figure 9-29. Hydraulic Pump Flow vs Pressure Characteristics

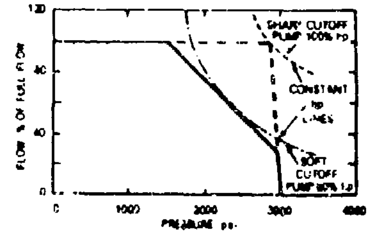


Figure 9-30. Hydraulic Pump Soft Cutoff Characteristics

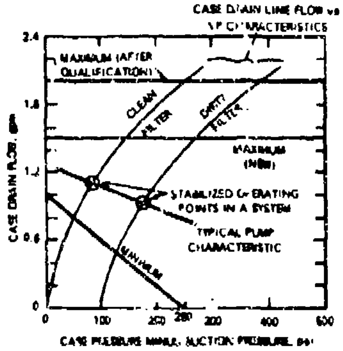


Figure 9-31. Hydraulic Pump Case Drain Flow Characteristics

Pump suction characteristics are important to pump response and pump life. The reservoir pressurization level in relation to the suction line loss and the minimum operating temperature characteristics *shall* be such that the pump suction does not cavitate when going to full stroke from compensated (no-flow) stroke. Fig. 9-32 shows characteristic system design requirements for reservoir pressures for typical operating temperatures and suction line lengths for a selected suction line diameter and required flow.

Pump pulsation characteristics and their compatibility with the system dynamic characteristics are extremely important for satisfactory system and helicopter performance. Pump system compatibility tests must be conducted as early as possible in the hardware development cycle. Appendix lines *shall not* be used close to the pump without extreme care, because the pump pulsation characteristic as an exciting frequency can drive the appendix line at its resonating frequency. The result is pulsations an order of magnitude higher (or more) than the pump pulsation level (2000 psi versus 200 psi peak-to-peak). These pulsations are converted into mechanical vibrations that induce stress levels high enough to cause fatigue failure in a few minutes.

If the general pulsation level is unacceptable, there are several alternative fixes. Line lengths may be modified in the resonating area so as to move the resonating speed point out of the using speed range of the pump since resonance can occur between the pump and an inline suppressor. There are several types of suppressors; the Helmholtz resonator is described in par. 9-4.5, AMCP 706-201, and a hydraulic pulsation suppressor is shown in Fig. 9-33. Accumulators also can be used for pulsation attenuation. Small volumes with a relatively high precharge (2000+ psi) and low impedance (large opening to system) are required in order to attain the necessary high-frequency response characteristics.

The basic applicable pump specification is MIL-P-19692, covering variable-delivery pump requirements. MIL-P-7858 covers fixed-displacement unit requirements.

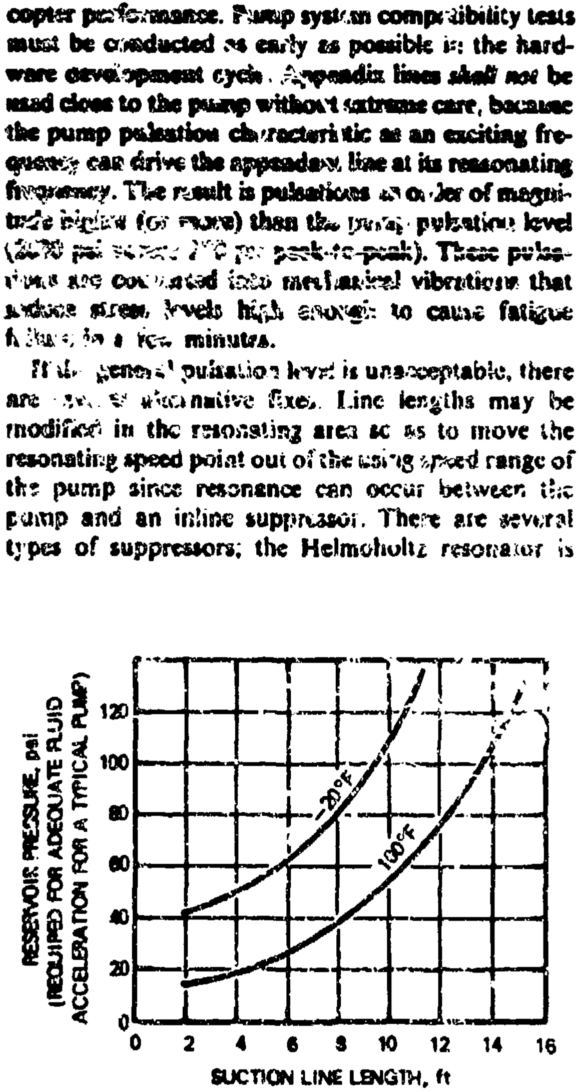


Figure 9-32. Suction Line Length — Reservoir Pressure Characteristics

9-2.7.3 Accumulators

Accumulators are used for energy storage and/or for pressure transient and pulsation attenuation. They may be of the piston, bladder, or diaphragm-in-gas-fluid-separation type. A variation of the piston type is the self-displacing variety, which avoids any adverse impact upon the reservoir as a result of size or transient high-velocity motion.

Seal, bladder, and diaphragm leakage, and its control, are the basic accumulator problems. The piston seal can be complemented with a piston face seal that provides redundancy when the system is not operating.

Another alternative for minimizing leakage of gas into the hydraulic system is use of redundant piston seals, with a vent between them giving access to the atmosphere; this may require a tail rod and an extra seal. The evolution of high-response pumps, and the resultant elimination of accumulators, is a desirable way of solving accumulator problems. The only basic area then requiring accumulators will be the engine-starting system.

Accumulator sizing and precharge are a function of the temperature range through which the accumulator must be used. The fluid volume available at the minimum pressure and the minimum operating temperature obviously must meet the subsystem energy requirements. Accumulators for tactical helicopters *shall* be designed to retain their integrity when exposed to gunfire.

MIL-A-5498 and MIL-A-8897 cover the detail accumulator requirements, and par. 9-4.4.3, AMCP 706-201, contains a discussion of accumulator techniques.

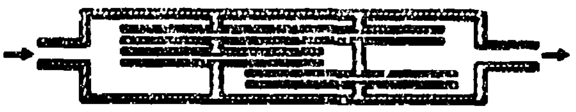


Figure 9-33. Hydraulic Pulsation Suppressor

9-2.7.4 Reservoirs

Reservoirs provide pump suction line pressurization, store fluid from subsystem differential volumes, provide for thermal expansion and contraction, and provide for acceptable external leakage at dynamic seals during system operation. There are several types of reservoirs, including bootstrap, separated-air pressurized, and nonseparated-air pressurized. These types are discussed in more detail in par. 9-4.4.2, AMCP 706-201. The various design requirements and features of the reservoirs are discussed in MIL-R-5520 and MIL-R-8931.

The reservoir pressure level essentially is defined by the pump suction requirements. These requirements involve both fluid acceleration and the pressure required to eliminate cavitation at the maximum speed, maximum flow, and minimum operating temperature points. The pump characteristics must be defined before the reservoir and pump suction line can be analyzed and designed. MIL-H-5440 defines general suction requirements; however, the specific pump requirements may override these.

The seals in the bladder and diaphragm reservoirs are not considered dynamic in the sense of a piston seal. The diaphragm and bladder do have relative motion between themselves and the reservoir. The bootstrap reservoir does incorporate piston seals. Where the weight penalty is acceptable, bellows are preferred to dynamic seals for the low-pressure side. Where dynamic seals are used, friction and life expectancy are primary considerations. The friction level *shall* be estimated conservatively and is to be used in designing the reservoir to meet the acceleration and anticavitation requirements.

The reservoir *shall* include, as an integrated or separate element, a relief valve for protection against overfill, in conjunction with accepting the maximum flow back to the reservoir as a result of system differential during operation.

Fill provisions *shall* be in accordance with MIL-R-5520. Filling is a function primarily of the type of reservoir. The open or air-pressurized reservoir may be filled by hand with fluid poured from a can. However, this method has obvious disadvantages in that contaminated fluid easily could be poured into the system. Separate air-fluid reservoirs and bootstrap reservoirs cannot be filled conveniently and easily by pouring. They require fill carts, or equipment such as hand pumps that can generate positive flow at low pressure This type of filling normally is accomplished through the system return filter. This is desirable because contaminated fluid will be filtered prior to reaching the pump inlet.

The use of HCE techniques such as RLS affects the reservoir requirements (refer to pars. 9-2.6.1 and 9-2.6.1.1). Mechanically, the RLS unit must be integrated into the design in the optimum manner: minimum external dynamic seals, lightweight, reliable operation, etc. Functionally, volume must be added, depending upon the specific subsystem characteristics, to allow for the search and fault isolation processes.

The suction outlet should be at the bottom of, or no higher than the middle of, the reservoir, relative to normal full levels. In the case of nonseparated reservoirs, provisions for negative g-transient operation *shall* be made.

Fluid level indication *shall* be in accordance with MIL-R-5520 and detail specifications.

For bootstrap reservoirs, the high-pressure side passages *shall* be large enough to maintain the transients due to reservoir high-volume change rate from exceeding 3750 psi.

Provisions *shall* be included for an air bleed at the highest point of the reservoir as it is installed in the helicopter. Manual-operation capability is required. Consideration *shall* be given to the possibility of automatic bleed on startup in addition to the manual bleed.

9-2.7.5 Pressure Relief

System relief valves are required to provide for possible pump control failure. The requirements are covered in MIL-V-8813, and a Military Standard (MS) series of qualified valves is available. However, MS valves are in separate housings; where integrated modules and service centers are chosen, the functional parts of the MS valve may be integrated into the package. Consideration should be given to routing the relief flow through the hydraulic system cooler. This could prevent system overheat under conditions where the relief valve on the pump compensator malfunctions. The standard system relief valve rarely operates, and therefore can be quite simple in design.

Where high response, frequent operation, and/or narrow reseat-full-flow requirements dictate a nonstandard design, consideration must be given to the dynamic performance characteristics. This will include damping provisions, such as a hydraulic dashpot.

9-2.7.6 Pressure Regulation

Pressure regulators or reducers generally are used to step the system pressure down for use in a specific subsystem. The detail requirements for these units are covered by MIL-V-8566. The reducers may include relief valves for protection of the subsystem in the event the reducer fails. The performance or regulation may be absolute (based on atmospheric pressure) or differential (maintenance of a specific pressure level above the return pressure at the reducer installation point).

9-2.7.7 Filters

Adequate filtration is essential to satisfactory operation of the hydraulic system. MIL-F-8815 covers the detail requirements for the housings and elements. Filter elements are discussed in par. 9-2.6.4.3.

The dirt-holding versus differential-pressure characteristics are shown in Fig. 9-34. They are quite nonlinear, i.e., a significant amount of contamination will be retained before there is a noticeable increase in differential pressure. Differential pressure indicators are required in order to signal that the element is loaded with contamination and requires replacement.

MIL-F-8815 requires that an automatic shutoff be provided so that a minimum amount of fluid is lost when the element is removed.

Thermal lockouts are used to keep the differential pressure indicators from operating below a specified fluid temperature level. This allows cold starts, in which the high-viscosity fluid would cause high-pressure drops across the element, thus causing the differential pressure indicator to operate.

A time delay is used to keep the differential pressure indicator from operating as a result of transient peak flow pressures. The contamination-holding capability then is referenced to steady-state flow pressures, and the element life thus is increased without penalty to the system.

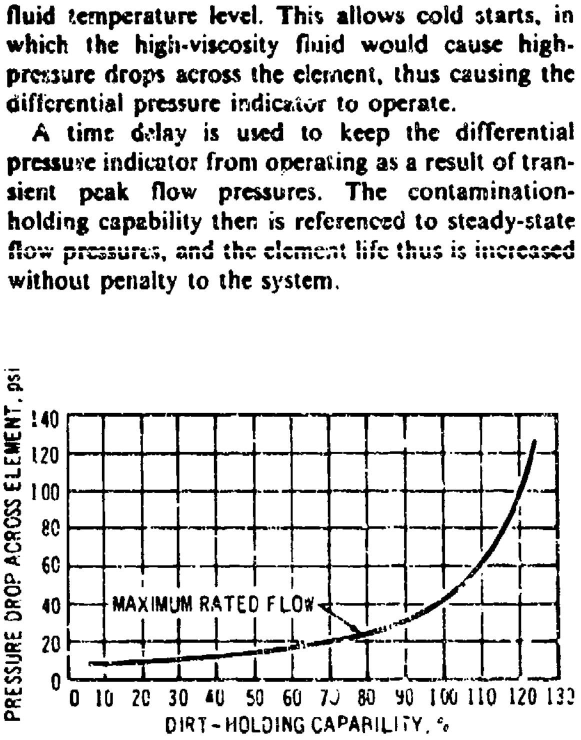


Figure 9-34. Filter Element Dirt-holding Characteristics

A bypass relief is used to limit the differential pressure across the filter element so that system performance degradation is controlled during transient low-temperature operation.

The interactions of the relief valve, thermal lockout, differential pressure indicator, and element clean-dirty differential pressure characteristics are important. The various requirements must complement each other if the filtration is to function effectively. Fig. 9-35 shows a typical composite performance curve. This curve must be used as a tool during design in order to determine the various performance requirements and to insure they are complementary.

The requirement for an element differential pressure indicator is covered in MIL-F-8815, and calls for 70 ± 10 psi differential capability for a 3000-psi system element.

Migration of the fluid filter media is not acceptable. This facet must be considered during selection of the media to be used. If necessary, additional testing *shall* be conducted so as to confirm that the media is adequate.

9-2.7.8 Check Valves

Check valve requirements are covered by MIL-V-25675, and AN check valves are available that meet Type I (160°F max) system requirements. Miniaturized check valves are available for Type II systems (275°F max), and provide a weight and cost savings. Nonstandard cartridge check valves that can be used in integrated packages also are available.

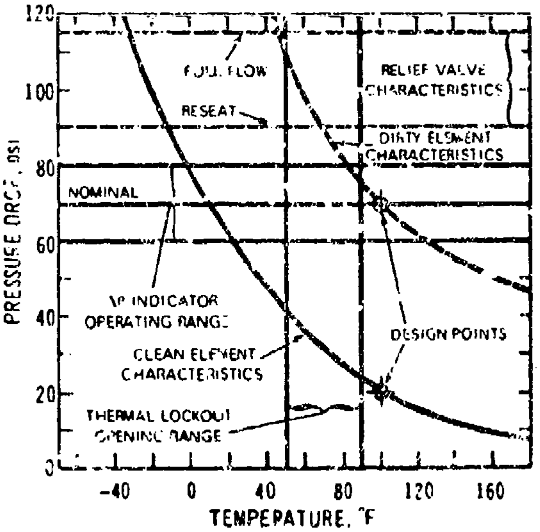


Figure 9-35. Filter Element Performance

The miniaturized check valve, with its inherent low-inertia, high-response characteristics, can be used quite effectively to keep transients of one subsystem from imposing unnecessary, potentially damaging stresses on the other subsystems. Par. 9-3.2.6.1 contains further discussion of check valves.

9-2.7.9 Pressure Switches

The pressure switch generally is used to operate a system failure warning light in the cockpit. The switches may be sensitive to either absolute or differential pressures. The absolute or atmospheric pressure-related type generally is used. A desirable feature of this type is its ability to retain the system fluid if the fluid seal fails, by using the case as a pressure vessel.

9-2.7.10 Pressure Transmitters

Pressure transmitters are required in order to convert pressure in an electrical signal for transmission to the cockpit in order to provide the pilot with a visual indication of the system pressures. Although MS standard units are available, they are large, heavy, and obsolete. Miniaturized, state-of-the-art units, which also provide a fail-safe feature, are available. The transmitter case of such a unit is designed to tolerate full system pressure, thus eliminating loss of the system when the sensing unit fails as a pressure vessel. The MS standard unit requires a pressure transient snubbing device for protection, plus a fuse in the event the sensing device fails; the improved, miniaturized units do not need snubbers and fuses.

9-2.7.11 Control Selector Valves

The selector valves for nonmodulating control of subsystems may be operated either manually or electrically. MIL-H-8775 covers their design requirements.

Valves may be classed as nontrail, full-trail, or half-trail. The "trail" indicates whether the cylinder or subsystem operating ports are connected to return when the valve is in the neutral-pressure blocked position. The valve is in the full-trail condition if the operating ports are connected to return. Fig. 9-36 is a schematic representation of the various trail conditions.

The valves may have two, three, or four commanded positions. Specialized or multiple valves may have more positions. Fig. 9-37 shows examples of the two-, three-, and four-position valves.

The valves may be two-, three-, four-, five-way, or more. The "way" refers to the number of ports on the valve. A configuration with several outlet bosses for one return section in a manifold would be counted as one-way.

The valves may be either pilot- or direct-operated. Direct operation usually is associated with low flows. Staging, using pilot valves, is required in order to handle large flows effectively and efficiently. Figs. 9-38 and 9-39 present schematics of both control modes.

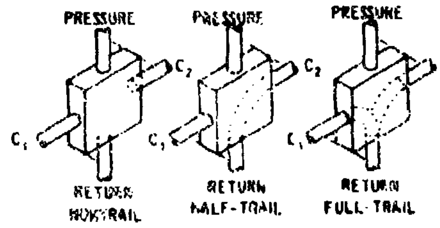

PRESSURE
PRESSURE
PRESSURE
C_2
C_2
C_2
C_1
C_1
C_1
RETURN
NONTRAIL
RETURN
HALF-TRAIL
RETURN
FULL-TRAIL

Figure 9-36. Hydraulic Valve "Trail" Configurations

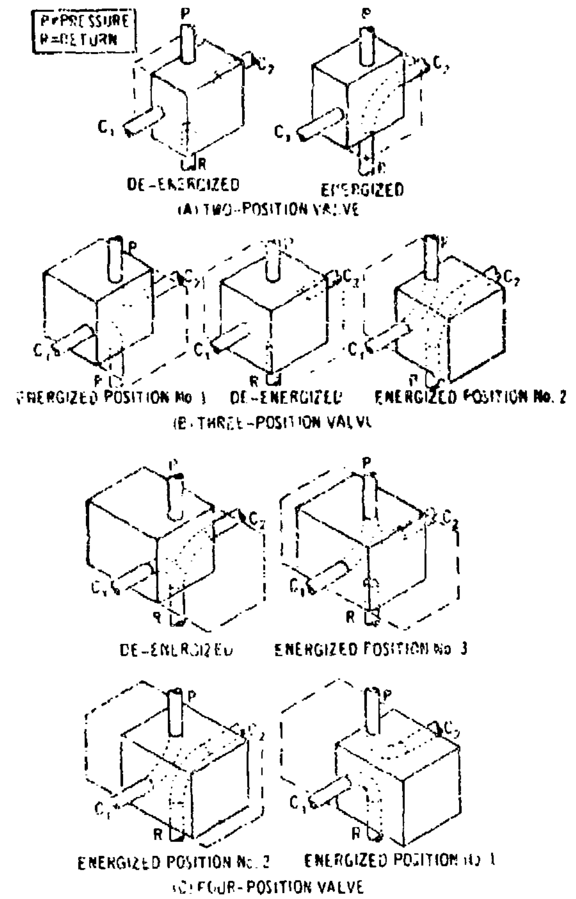

P=PRESSURE
R=RETURN
P
C_2
C_1
R
DE-ENERGIZED
ENERGIZED
(A) TWO-POSITION VALVE
ENERGIZED POSITION No. 1
DE-ENERGIZED
ENERGIZED POSITION No. 2
(B) THREE-POSITION VALVE
DE-ENERGIZED
ENERGIZED POSITION No. 3
ENERGIZED POSITION No. 2
ENERGIZED POSITION No. 1
(C) FOUR-POSITION VALVE

Figure 9-37. Hydraulic Valve Configurations

Minimum force levels are required for satisfactory service in cases where contamination of any significance is possible. For the smaller, lapped spool-sleeve and ball-poppet type valves, a 10-lb minimum operating force is required. For the larger, second-stage, lapped spool-sleeves, a 40-lb minimum operating force is required.

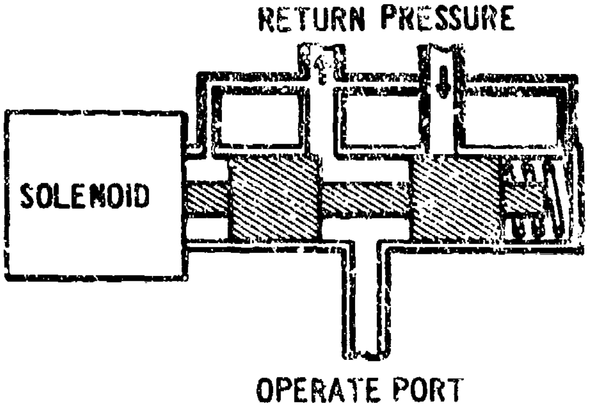
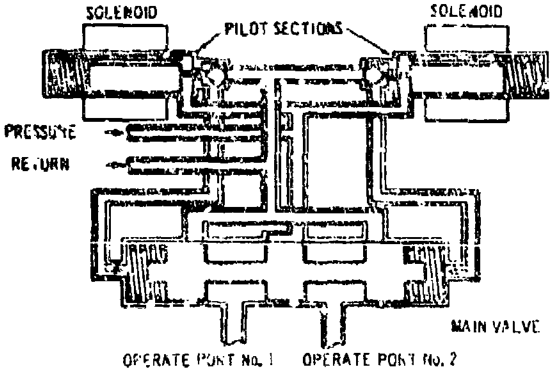
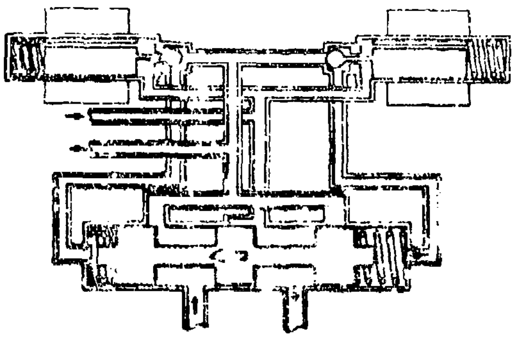

RETURN PRESSURE
SOLENOID
OPERATE PORT

Figure 9-38. Direct-operated Valve


SOLENOID
PILOT SECTIONS
SOLENOID
PRESSURE
RETURN
MAIN VALVE
OPERATE PORT No. 1
OPERATE PORT No. 2

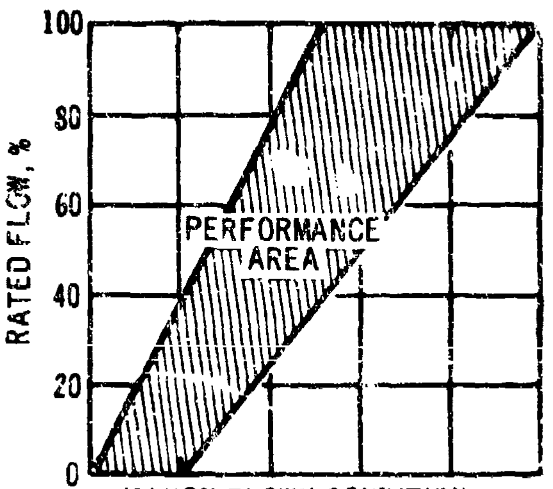
(A) DE-ENERGIZED



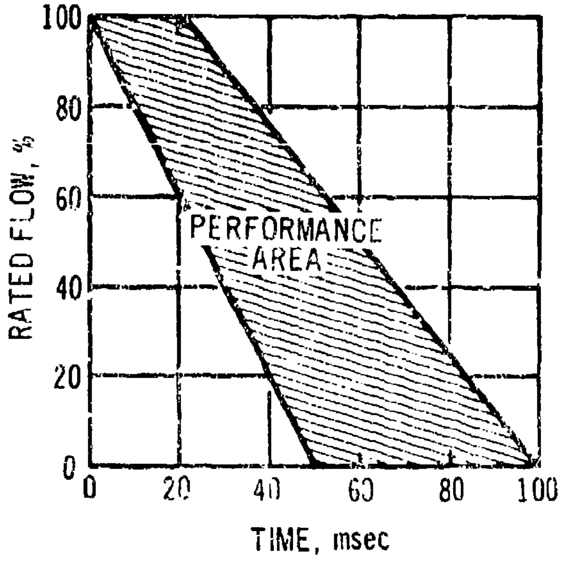
(B) ENERGIZED

Figure 9-39. Pilot-operated Valve

The valve operating time is important in controlling "water hammer" and return side transients due to release of 3000-psi levels into the return instantaneously. Slow operational time requirements are oriented primarily toward fast-operating solenoid valves. Fig. 9-40 shows characteristics that are satisfactory for most systems.



(A) "ON FLOW" CONDITION



(B) "OFF FLOW" CONDITION

Figure 9-40. Valve Operating Time

The normal pressure-to-return leakage characteristics of the valve will provide a fail-safe feature if the subsystem is affected by entrained air or inadvertent cavitation. The level of this leak protection is such that the reservoir will not be depleted on a normal-duration flight. Fig. 9-41 shows the features of this RPS technique (refer to par. 9-2.6.1.3). HCB concepts are discussed in AFAPL TR-70-50 and in par. 9-2.6.1.

The solenoid-operated pilot valves may be configured so that they are returned to neutral either by spring force or by the solenoid. The springs generally are configured in the 300- to 500-psi equivalent pressure level due to other constraints. Therefore, the six-to-ten-times-force capability of full-system pressure provides that much margin for handling contaminants and high friction due to galling or scoring, and is the preferred method of second-stage operation. Fig. 9-42 contains schematics of both approaches.

9-2.7.12 Restrictors

Restrictors are used as subsystem rate-control devices where precise control is not required. They

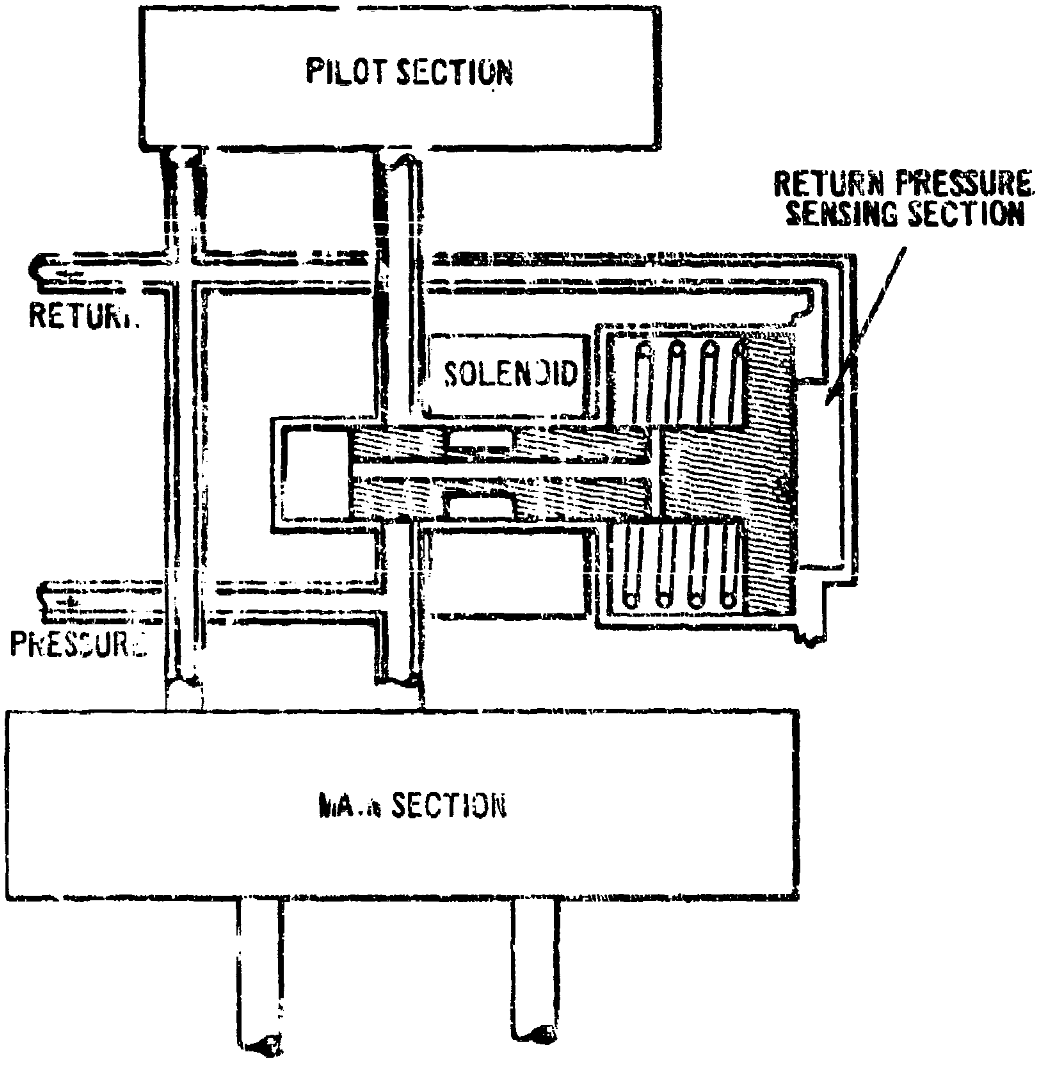
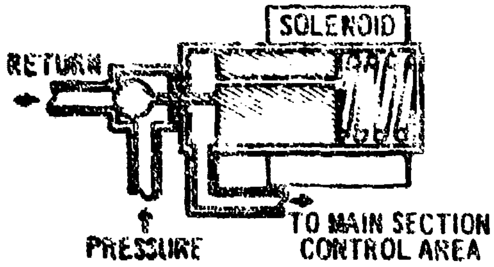
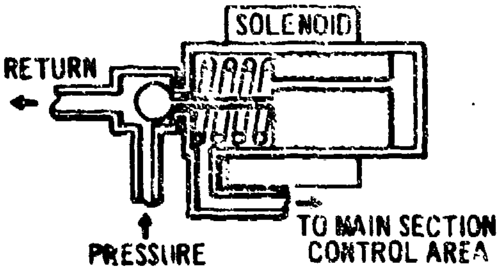


Figure 9-41. Solenoid Operated Valve Incorporating Return Pressure Sensing



(A) THE PILOT VALVE CONFIGURATION SHOWN IS FOR POWER RETURN OF MAIN SECTION TO THE DE-ENERGIZED POSITION



(B) THE PILOT VALVE CONFIGURATION SHOWN IS FOR SPRING RETURN OF MAIN SECTION TO THE DE-ENERGIZED POSITION

Figure 9-42. Power and Spring Main Section Valve Return to Neutral

may be either two-way or one-way. Screen contamination protection is required for orifices with a diameter smaller than 0.070 in., as specified in MIL-H-5440.

Generally, the ideal location for the restrictors is in the actuator device. They may be cartridge or press-in units such as the Lee jets. This arrangement localizes design requirements in situations where assisting loads can cause high pressures. In addition, external leak points are reduced. Flow from a restrictor can have destructive effects when exhausted directly into a hydraulic hose line or into a sharp tube bend. This should be avoided. A primary advantage of orifice restrictors is that they are relatively insensitive to temperature change; the pressure drop for a given flow varies with the density of the fluid.

The detail requirements for restrictors are covered in MIL-H-5440 and MIL-V-25517.

9-2.7.13 Separate Servos

Separate servos are required for SAS, CAS, and other such applications where requirements are not or cannot be integrated into the flight control actuators. Servos may be used for either single or dual systems. The control of the output ram is via an electrohydraulic valve that converts the electrical commands into the appropriate actions. The total package includes electrical signal feedback devices, and the preferred device is a linear variable-differential transformer (LVDT). The preferred electrohydraulic concept is the jet pipe valve, which is inherently tolerant of contamination. Fig. 9-43 shows a schematic of a typical single-system servo. Separate servos generally are series-type control devices, and incorporate a position lock that is activated when the system is not actuated or when system pressure is lost. MIL-V-27162 covers servo control valves.

9-2.7.14 Allowable External Leakage

Consideration should be given to leakage levels for dynamic seals in both static and operational situations. ARP 1084 defines realistic in-service requirements for static and dynamic seals.

9-2.8 HYDRAULIC SYSTEM INSTALLATION

The paragraphs that follow discuss the proper installation and support of hydraulic lines, hoses, and components. The requirements also are discussed in ARP 994.

9-2.8.1 Use of Hoses and Swivels

Hoses and swivels may be heavy, costly, and can be a maintenance problem. Therefore, use of these items *shall* be minimized. Desirable alternatives for handling relative motion include coiled tubing and fixed-body actuators, with articulating links between the actuator and the control surface or subsystem function where applicable.

9-2.8.2 Maintenance Access

In order for the weight benefits of permanent fittings to be realized, the hydraulic installation must be located behind other removable equipment and installations.

Filter elements *shall* be located so as to permit easy access via doors. Differential pressure indicators *shall* be either flush with the skin or visible through transparent skin sections or nonstructural single- or two-button doors. This is necessary because a check of the pressure level is required during preflight.

All separable connections *shall* be relatively accessible since they are employed to permit component removal and/or removal of lines for access to other equipment.

A visual check of the reservoir fluid level condition should be possible without removal of access panels. If fill and/or bleeding is required, access to these

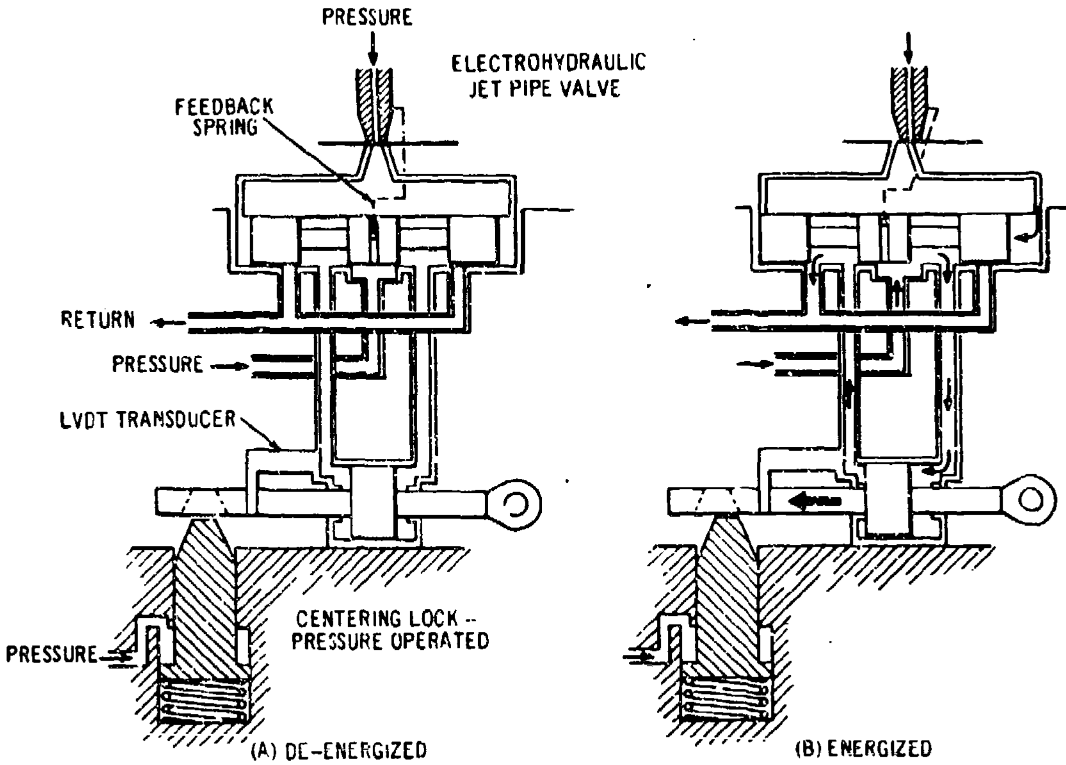


Figure 9-43. Typical Separate Servo Actuator

areas *shall* require no more than removal of quick-access doors.

RLS devices *shall* have ground-checkout capability (refer to par. 9-2.6.1.1). Manual operating buttons may be provided, along with operating indicators on each subsystem to indicate which subsystem is shut off. Visual access to the operating indicators *shall* be provided — preferably without need for door opening. Access to the manual operating buttons may be through conventional access doors since their checkout will be required only periodically.

Access requirements are discussed in MIL-H-5440.

9-2.8.3 Hard Versus Soft Installations

The tubing installations may be hard (rigidly held in place) or soft (compliant). Each approach may work satisfactorily; however, they cannot be mixed successfully. The use of a hard point as a fitting in a relatively noncompliant bulkhead — for instance, in conjunction with clamps some distance away — can be disastrous. At worst vibration condition, the clamps will allow enough motion of the line so that flexure at the bulkhead fitting will result in excessive bending stresses and fatigue failures. The preferred installation is the soft or compliant type, with clamps or blocks close enough to the component-attach points so that excessive stresses do not develop. The installation *shall* be designed with the known or estimated vibration environment in mind so that system resonance does not occur.

Clamps and blocks may be used to advantage. However, it is not desirable that they be mixed. ARP 994 discusses their use in detail.

9-2.8.4 Component Mounting Concepts

Component installation can be classified as follows:

1. Separate, independent components
2. Integrated packages with cartridge components in a common housing
3. A pressure-return manifold with "mount-on" components.

The installation trend is to integration or manifolding. Primary reasons are the resultant reduction in external leak points and the weight savings. An optimum approach may be an integrated pump-reservoir package, a separate service center that would include all other components except the control surface actuation or subsystem operating device. In any event, each helicopter design must be assessed individually in conducting the system installation trade-off.

9-2.8.5 Miscellaneous Installation Considerations

Components requiring frequent maintenance *shall* be accessible. Such units as fluid-driven air compressors, pumps, and filters must have unusually good access in order to allow adequate servicing.

Components that require frequent maintenance, such as filters and pneumatic chemical driers, *shall* be mounted rigidly in order to avoid damage during servicing.

Natural dirt-collecting areas, such as brake valves under the cockpit floor, *shall* be considered in planning system installation.

A typical example of improper installation is routing of lines through a wheel-well area, where leaking air can cause strumming of both normal and emergency lines. Attachment of lines to a panel that may vibrate will have a similar effect. Proximity to a common fire hazard also must be avoided. Separation of lines *shall* be accomplished after consideration of all environmental hazards.

The designer *shall* avoid installations in which an engine mount failure can drop the engine on critical lines or components and thereby increase the possibility of losing the helicopter.

9-2.9 MISCELLANEOUS DESIGN CRITERIA

Industry experience with previous hydraulic systems and components has revealed many miscellaneous design aspects that *shall* be considered. The paragraphs that follow present some of these considerations.

9-2-9.1 Actuators and Associated Equipment Design

For protection against dynamic shaft seal failures, the designer should consider use of dual seals, with a return vent incorporated between the seals (Fig. 9-44).

The design of the electrical system with regard to electrohydraulic servo valves *shall* avoid dither signals that cause valve oscillation and actuator dynamic seal wear. Protection against electromagnetic interference susceptibility *shall* be required.

Linkage pivots on flight control actuators, consisting of friction-held journal bushings with loose tolerances, require critical shimming in order to obtain the alignment necessary for free operation. Ball-bearing pivots, with provisions to prevent overtorquing and operating in system return fluid, are preferred in order to prevent binding of control-valve linkage.

The design of actuator valving should include provisions to prevent fluids being trapped at the ends of the main control-valve spool.

Actuator pistons *shall not* bottom against internal actuator stops during autopilot (stability augmentation) input, coupled with the extremes of manual input and rigging and manufacturing tolerances. Erratic stability augmentation inputs during taxiing can cause excessive loads due to hammering against stops. This condition *shall* be taken into account during testing, as well as in design.

Internal hydraulic stops in cylinders *shall* incorporate snubbing, or *shall* be strengthened sufficiently to prevent fatigue loading failures (Fig. 9-45). Rigging alone should not be considered in order to prevent fatigue loading. However, the designer should emphasize the rigging of controls to stick-stop limits to avoid actuator bottoming.

To prevent blow-by, Teflon piston seals should be avoided in an actuator that also must operate pneumatically. Piston rings *shall* be used on piston heads. Appropriate seals should be considered for long-stroke, large-bore actuator cylinder applications to alleviate chances of rolling and subsequent damage to seals.

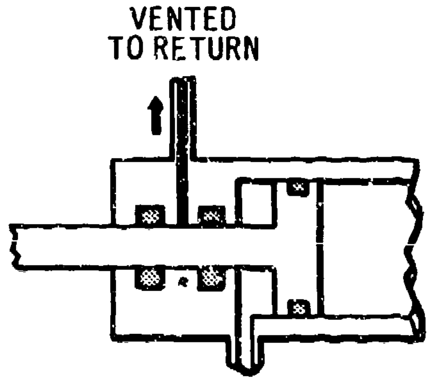


Figure 9-44. Dual Seals With Return Vent

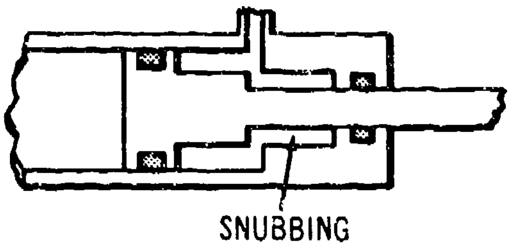


Figure 9-45. Hydraulic Snubbing for Internal Stops

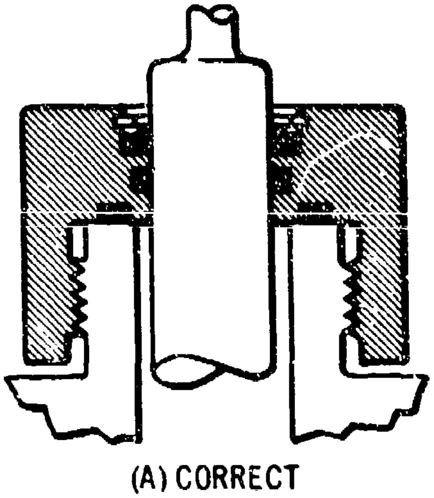
Internally threaded actuator endcaps utilizing AND 10050 boss sealing design, as shown in Fig. 9-46(B), should be avoided. Leakage can be caused by squareness of threads to boss surface and torquing problems. In addition, end caps that are threaded internally into a cylinder barrel and locked with a jam nut are subject to barrel stretch under pressure. This can result in leakage or loosening of the jam nut. Sufficient material thickness in the cap area is required.

Viscous dampers *shall* be self-servicing from system return fluid.

Use of Zerk grease fittings for servicing of hydraulic components *shall* be avoided.

In order to minimize binding and seal wear from actuator side loading, a minimum overlap of one piston diameter *shall* be provided for piston and rod bearing areas in the fully extended position (see Fig. 9-47).

Safety-wiring of piston head retaining nuts to a piston head that can rotate on the piston shaft is not acceptable as a locking method. Use of locking devices such as the NAS 559 keys as shown in Fig. 9-48 should be considered.



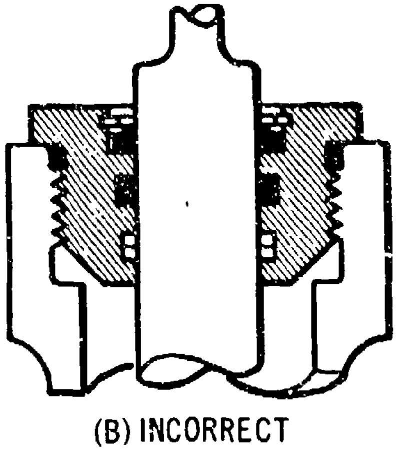


Figure 9-46. Avoid Internally Threaded End Caps

9-2.9.2 Brake Design

Excessive wheel brake torque can cause brakes to grab. Reduction of effective wheel brake piston area in order to increase the pressure required for a given brake torque is a suitable corrective action.

The brake control valve input shaft *shall* have adequate bearing surface area, and should not be subjected to side loading from hoses, etc., so as to insure smooth brake valve operation and full release.

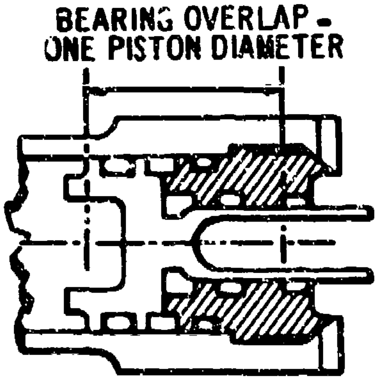


Figure 9-47. Bearing Overlap

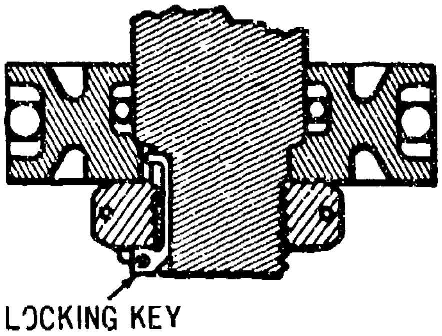


Figure 9-48. Piston Head Retaining Nut Locking Key

9-2.9.3 Control System Design

Flight control system assemblies *shall* have adequate clearance guards, or otherwise be protected, to afford maximum protection against jamming by foreign objects.

The designer *shall* avoid routing flight-control linkage through areas in which its removal is required in order to replace the engine.

Cable tension-retaining devices *shall* be considered as a means of preventing control cable tension changes.

Overtorquing of control system bolts *shall not* result in increased friction during operation.

The use of special bolts *shall* be avoided.

9-2.9.4 Electrical Design

Electrical connections to hydraulic components *shall* have a mechanical strength requirement consistent with maintenance handling requirements. Wires should be buried in the installation if possible.

To simplify troubleshooting and component replacement, hydraulic or pneumatic components incorporating an electrical function *shall* have integral electrical connectors for removal and replacement.

Potting compounds must not require a higher heat cure than can be withstood by electrical insulation. The compounds also must be compatible with subsequent processes applied to the assembly during manufacture, such as welding or baking for epoxy cure or strain relief.

Proper manufacturing of electrical connectors requires that only the wire should enter the soldering connection. The first layer of insulation of the wire should enter the potting compound so as to provide moisture leakage protection (Fig. 9-49). The braided insulation should be clamped adequately at the connector inlet, and should not enter the potting compound in such a manner as to provide a leakage path.

Higher quality electrical parts should be used in place of MS parts in critical applications where failure creates a high probability of catastrophic effect. Detailed failure analyses of electronic and electrical circuits are required in order to determine where use of such higher priced parts is justified.

A positive fix is required in order to prevent runaway trim actuators. One possibility is stepped motor operation.

Two electrical actuators in parallel, with braking when de-energized, must have independent electrical inputs so that the first actuator to complete its stroke can be de-energized and braked.

9-2.9.5 Filter Design

Filters *shall* be installed in the pump drain line prior to its entry into the oil coolers. (Installation downstream of the coolers will allow trapped pump particles in the cooler to recontaminate a replace filter element.) Proper flushing of a cooler is important.

For T-valve installations, central filtration should be used in order to avoid differential flow as the individual filter pressure differential changes (Fig. 9-50).

All restrictors with hole sizes of under 0.070 in. should incorporate filters.

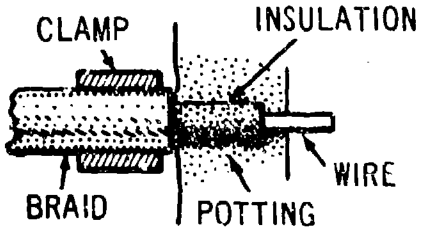


Figure 9-49. Properly Assembled Electrical Connector

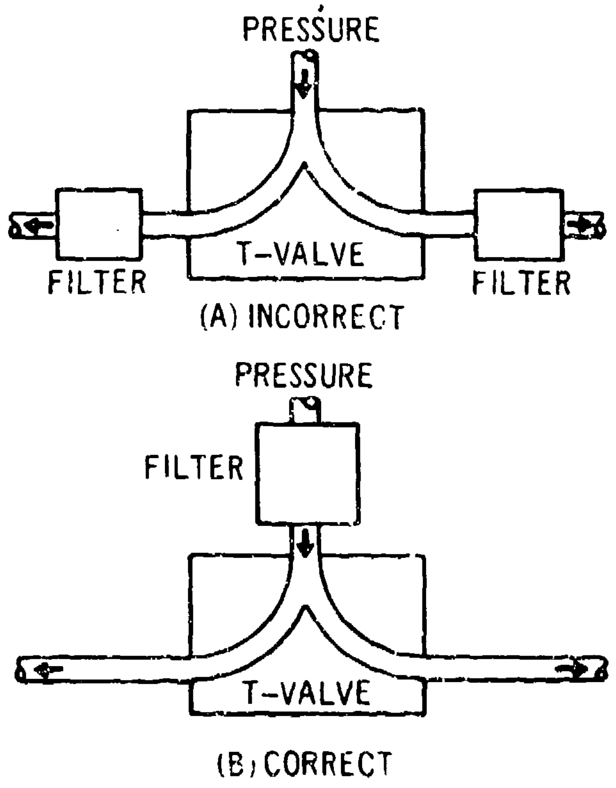


Figure 9-50. T-valve Central Filtration

Test filtration should not exceed that of the component in actual use.

9-2.9.6 Fittings Design

Hydraulic fittings, such as AN 833 universal bulkhead types, when installed in a valve port, can result in internal valve interference or restricted fluid flow. Component port design and fitting selection *shall* be such that interference cannot occur.

Pump fittings — and suppressors, if incorporated — *shall* be torqued to maximum allowable values in order to prevent loosening and subsequent loss of fluid. Use of an acceptable locking device is advisable for any large fitting in high-pressure application.

The use of pipe plugs for external sealing of drilled passages can lead to internal stresses in component housings, resulting in cracks. The Lee plug produces less stress concentration and should be considered.

9-2.9.7 Gage and Indicator Design

It generally is good design to include fuses as well as snubbers at the upstream ends of the lines leading to gages and indicators. An alternative is to install the pressure sensors into fittings in the system line instead of in an appendant line. Helicoil sensing elements are recommended instead of Bourdon tube types.

Gages with Bourdon tube or rack and pinion gearing *shall* be avoided for use as pressure indicators. The transmitter should be connected to the system with flexible hose or with a tube having sufficient bend to absorb vibrations.

Gage cases *shall* incorporate a vent hole to prevent buildup of case pressure if the sensing element leaks. The vent hole should be covered with tape to prevent corrosion.

9-2.9.8 Hose Design

Realistic design and testing of devices such as brake control valves must take hose side loads into account.

Hose routing or sizing *shall* prevent cross-connection at actuators. However, deviations from normal hydraulic practices, such as use of return hoses that are smaller than pressure hoses, *shall* be avoided because such deviations can have an adverse effect.

9-2.9.9 Pump Design

Insuring compatibility of the pump with the system requires determination of the effects of low inlet pressure, high case drain (or bypass) back pressure, and the interaction of the two on the internal balance of the pump. Back pressure also can cause reduced cooling flow, leading to shortened pump life. Compatibility determination includes analysis of the nature of the contamination generation properties of the pump; sufficient filters must be used to keep back pressure low within a reasonable cleaning schedule while maintaining a clean fluid supply for the pump. Two-pump system design *shall* consider large, nonbypass filters in the drain line of each pump. Should bypass-type filters be used to insure low pump case pressures, the flow *shall* be routed through a second, larger return filter.

Pulsations resulting from pump ripple, which may be intensified by system resonance, can be determined by oscilliscope scanning of the pressure through the range of operation. Peak pulsations *shall* be kept below ±150 psi (300 psi total). Pressure pickups must be in the line (not on appendages), and *shall* be located at the pump and, at least, at the first downstream component. The optimum design furnishes some elasticity to the system at the pump outlet port. Short, dead-ended lines near the pump require particularly close scrutiny, and should be avoided.

Pump cavitation will result if reservoir pressurization is not sufficient to accelerate the fluid in the suction line to a flow rate compatible with pump displacement. This condition is likely to be a more critical design condition than is the steady-state flow requirement. Qualification testing of pumps, particularly those for use in power-control systems, should include suction and demand requirements based upon the condition of the application.

Connecting two or more pump systems from a common routing can result in priming problems, with a momentary interruption of inlet flow resulting if reservoir pressurization is low.

If the normal system pump is pumping air and cavitating a bootstrap reservoir, the auxiliary pump can be affected adversely by the low bootstrap supply pressure, and may not prime. A check-off accumulator may be required in order to maintain bootstrap pressure with loss of normal system pressure.

Centrifugal pump operation with an outlet flow blockage can result in overheat, thus causing seal or case (structural) failures. Therefore, bypass flow for cooling *shall* be provided.

Pump testing *shall* include realistic case drain system characteristics. The internal leakage of a hydraulic pump is necessary for lubrication and cooling of the pump mechanisms. However, the subsystems into which the flow is discharged vary among helicopter designs. The designer *shall* specify the case drain system characteristics to insure adequate housing strength, shaft seal capability, and pressure conditions during the pump qualification testing.

Water hammer limiting is discussed in par. 9-4.2, AMCP 706-201.

9-2.9.10 Reservoir Design

Bootstrap reservoir design *shall* incorporate sufficient piston force, in a static, no-pressure condition, to facilitate reservoir servicing and bleeding.

Reservoirs *shall* be designed with the air bleed vent high and the suction outlet low. The overboard relief flow capability *shall* be sufficient to prevent reservoir damage during improper or emergency operations, such as system operation with an overfilled reservoir or overfilling during reservoir servicing.

The designer should avoid the connection of two or more drain or vent lines together, to a common overboard vent, where back pressure can cause back flow through the second vent system.

With hydraulic power present in one system only, high rates of motion in large, tandem actuators can pump the fluid from the unpowered section back to the return system without recovering equal fluid from the pressure side of the unpressurized system. Unless provisions exist to dump the returned fluid at a low pressure, damage to the reservoir and other low-pressure components can occur. An alternative to dumping is to equip ground test carts with multiple connections so that both systems may be pressurized simultaneously during checkout.

Test reservoirs *shall* be representative of the actual system reservoir. This will allow viscosity, fluid temperature, fluid settling, and fluid aeration test conditions to be realistic.

9-2.9.11 Valve Design

Check valves *shall* be installed in subsystem return lines so as to prevent back pressures from high return flows from acting upon cylinder locks, differential cylinder areas, and return cavities in components that may fail under repeated return transient pressures. The miniature check valve should be used for this application since it has a faster response time than the standard AN type. In addition, balanced areas for lock devices are recommended.

Check valves *shall* be installed in the pressure lines of subsystems where airloads can cause a flow reversal when system pressures are reduced because of an operational demand upon the system. A relief check valve *shall* be considered if overloading can occur at high speeds.

Regulated pressure can be affected by transient back pressures at the valve return port. This can be corrected by installing a fast-acting miniature check valve in the return line.

Soft seals or poppets that depend upon assembly compression to prevent secondary leak paths around the material may leak due to distortion or compression under operating pressures. This can be prevented by incorporating static O rings to protect secondary leak paths (Fig. 9-51).

The designer *shall* avoid use of self-locking nuts to hold spring-loaded adjustments.

Part concentricity *shall not* be dependent upon thread concentricity.

Flow paths within valves *shall* be considered. Indexing radial holes in spools or placement of springs can affect flow paths. Poppets with flutes can rotate with flow, and thus may be desirable in some designs.

Inadequately designed spring guides may allow spring loads to cock spools. Center point loading at both spring ends is recommended. Reduction of load upon springs that are heat-soaked while loaded must be a design consideration. The designer should avoid tension and plated springs. For springs immersed in fluid, etc., 17-4 PH spring material should be used (refer to MIL-HDBK-5).

Low-operating force valves, such as solenoid-valve pilot sections, should be designed with poppets since spool valves are subject to sticking from contamination (silting, etc.).

Vent holes between two seals of differential areas, such as are incorporated in return-line dampers and inline balanced relief valve designs, *shall* be multiple or indexed to insure that a leakage trap does not occur. Unit malfunction may result if the vent chamber becomes filled with fluid that is not readily dischargeable through a small vent hole.

Direct-operated solenoid valves *shall* have adequate return spring force to overcome silting action (chip shearing).

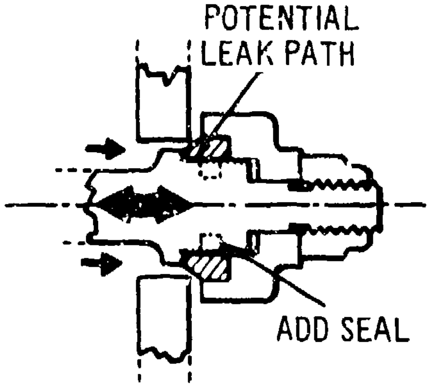


Figure 9-51. Secondary Leak Path Seal

Split-coil (holding) solenoids should be avoided in valve design. Unreliable operation, caused by starting-coil switch settings and malfunctions, results in overheated solenoids. This problem usually overrides such desirable features as the lower weight of split-coil designs.

Servo-valve-cover deflections or shocks can cause valve malfunctions if the cover is mounted upon the servo valve motor instead of the main valve body. Adequate clearance between cover and motor should be provided in order to minimize effects of minor dents. Covers should be of rugged design.

Hydraulic pressure surges due to valve spool shift may be prevented by the use of slow valving (adequate dead band) or electric time delay.

In the case of electrically operated valves, consideration *shall* be given to valve positioning, and to the effect upon the system should electrical power be applied inadvertently to two electrical inputs due to shorting, etc. Relays can insure predictable operation in this abnormal situation.

Hydraulic emergency overboard dump valves *shall* have sufficient flow capacity to avoid back pressure buildup that, in some system designs, can divert fluid to the reservoir through selector valves and check valves. Directing fluid to the reservoir may result in reservoir overpressurization and failures under certain conditions.

Differential area vent seal wear and leakage can occur from plunger motion during normal system pressure fluctuations. Long-life seals, or valve designs incorporating little or no plunger motion during normal system pressure fluctuation, are desirable.

In the case of half-trail valves, internal leakage when in the neutral half-trail position should be evaluated for its effects upon subsystem operation. When leakage from pressure to blocked cylinder port is greater than leakage from block cylinder port to return, pressure buildup in the blocked circuit can occur, resulting in unwanted motion or loads in the blocked circuit. Excessive leakage from the blocked cylinder port to return can result in unwanted motion from external loads.

High-pressure tests with return ports capped can result in overpressurization of components. The return proof pressure requirement should be compatible with system operating pressure.

9-2.9.12 Lubrication

Experience indicates that graphite-loaded grease tends to dry up in high-temperature antifriction bearing applications, leaving a residue of hard graphite that interferes with proper bearing function.

All critical joints *shall* be lubricated and protected. The lubricant must be compatible with oiling.

Lubrication of mechanisms that are located in a high-temperature area can result in jamming as a result of burned oil carbon.

Left- and right-hand component lubrication fittings *shall* be multiple, or *shall* be located so as to be accessible.

Long lubrication paths result in frozen grease and blocked fittings.

Unclamped monoball bearings must have two grease fittings in order to insure proper lubrication on both ID and OD.

9-3 PNEUMATIC SYSTEMS

9-3.1 PNEUMATIC SYSTEM DESIGN

The decision to use a compressed gas rather than a pressurized liquid as a working medium in a fluid power control system is made during the preliminary design phase. The various trade-offs to be considered in making the choice are presented in Chapter 9, AMCP 706-201. The paragraphs that follow describe the detail design considerations for a pneumatic system.

9-3.1.1 System Analysis

The design and analysis of a pneumatic system become considerably involved when nonlinearities are considered. The derivation of a mathematical model describing the physical phenomena of compressible fluid through a system — where the fluid passes through restrictions, expansions, changes in direction, etc. — proves difficult and results in cumbersome, complicated equations.

For example, the flow within the system, or the pressure drop, will vary between the extremes of adiabatic flow (no heat transfer) and isothermal flow. The basic formulas for adiabatic and isothermal flow are given in par. 9-5.5, AMCP 706-201. The formula for calculating the maximum mass flow of air within a system is presented in the same paragraph.

The calculation of flow through nozzles, orifices, piping, valves, and fittings may be simplified by use of charts and graphs for expansion factors, orifice coefficients, critical pressure ratios for nozzles, and relative roughness and friction factors for piping and tubing. Familiarity with the simplified equations of Refs. 2 and 4, and use of the tables, graphs, and charts contained therein, will allow a good analysis of an entire system or component to be made.

The verification of the design through actual operational test of the system or component performance is the designer's ultimate goal.

To aid in the selection of particular components, a list of commonly used components, and an operational description of each, is provided in par. 9-3.2. Refs. 1 through 4 are additional sources of information regarding design and analysis of pneumatic components and systems.

9-3.1.2 System Redundancy

All pneumatically operated services that are essential to safety in flight or landing *shall* be provided with emergency devices per MIL-P-5518. The emergency systems must be completely independent of the main system up to, but not necessarily including, the actuating cylinder or motor. These emergency systems should be designed to be actuated only by compressed air, direct mechanical connection, electromechanical units, gravity, or combinations of these.

Where dual pneumatic lines are used to provide emergency operation of a mechanism, the normal and emergency lines *shall* be separated by as great a distance as is practicable, so that the possibility of both lines being ruptured by a single projectile is remote. Where shuttle valves are necessary in order to connect the normal and emergency systems to an actuating cylinder, they *shall* be built into the cylinder. The emergency line from the shuttle valve should be vented to the atmosphere when not in use.

When an air bottle is used as an emergency backup energy source, a standard pressure gage *shall* be installed to allow maintenance personnel to check the pressure. The air bottle should be located so as to produce a minimum length of line between it and the shuttle valve.

An APU may be used for providing emergency pneumatic power. The APU should be designed to use stored compressed air for starting and then to provide a limited amount of power for the essential subsystems of the helicopter.

9-3.2 COMPONENT DESIGN

9-3.2.1 Air Compressors

The air compressor maintains the pneumatic system pressurization during flight. It can be driven by direct drive from the helicopter engine gearbox, by an electric motor, or more commonly, by a hydraulic motor powered by the utility hydraulic system. Compressor operation usually is controlled by a manifold pressure sensing switch, with the compressor cutting in when system pressure drops to a preset minimum and cutting out at a preset maximum.

Compressors can be classified into two basic groups: positive displacement, and dynamic, or nonpositive, displacement.

9-3.2.1.1 Positive Displacement

In this type, pressure is increased by confining a gas in a progressively diminishing space. There are a number of different arrangements, among them the axial piston and the rotary.

In the axial system a piston moving within a cylinder (Fig. 9-52) alternately traps and compresses the gas. This is the most widely used type, and sizes range from less than 1 hp to 5000 hp. Good part-load efficiency makes this type most acceptable where wide variations in capacity are required.

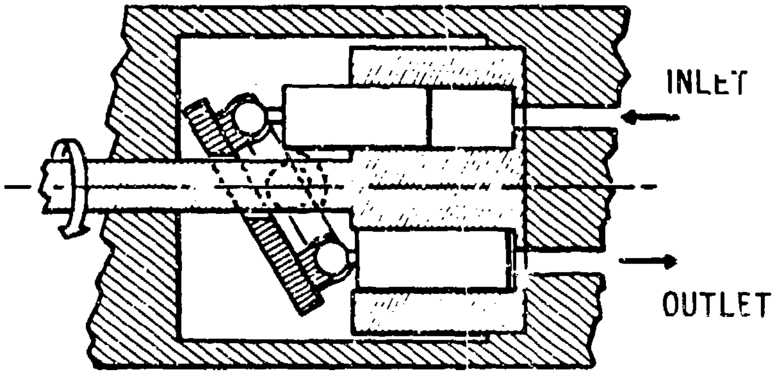


Figure 9-52. Axial Piston Compressor

In the rotary system, the rotating motion of single or mating elements compresses the gas. Major types of rotary compressors are sliding vane (Fig. 9-53), lobed-rotor, liquid piston, and helical.

9-3.2.1.2 Dynamic Displacement

In this type compressor, a high-speed, rotating element imparts velocity to the gas. This velocity is converted into a pressure rise in the compressor volute or other diverging passageways. There are two main arrangements, centrifugal and axial flow.

Centrifugal compressors have an impeller similar to a centrifugal pump. Impellers can be arranged singly, or in multiple units for higher discharge pressures. At a constant speed, a centrifugal compressor delivers nearly constant discharge pressure over a considerable range of inlet capacities.

Axial-flow compressors move air parallel to the rotor axis. They are made in single- or multiple-stage versions. In the latter, matching stator blades redirect the flow of air to the proper entrance angle for succeeding rotating blades. Generally, axial compressors are used for ultra-high capacity. However, there are many special applications for smaller units.

9-3.2.2 Compressed Air Supply System Selection and Operation

There is overlap in the performance of different compressor types, and sometimes several can be used for any given service. Narrowing the choices is a process of considering such factors as space and weight limitations, power ranges, and capacity ranges of the different types.

In a typical helicopter compressor of the reciprocating type, the air entering the compressor through the air inlet cap on the first-stage cylinder head passes through a filter to allow removal of any particles that might damage the internal components of the assembly. The filtered air is drawn through the intake valve into the first-stage cylinder by means of the suction created on the downward, or intake, stroke of the first-stage piston. On the upward (compression) stroke, the intake valve is forced shut by the increasing pressure, and the spring-loaded discharge valve is forced open when the pressure reaches a predetermined value. The compressed air is directed through the discharge valve and into the first intercooler connected to the first-stage head, where the heat created during compression is dissipated through forced convection by the airflow from the fan directed over the intercoolers. The flow of cooled, compressed air next passes through the first-stage relief valve — connected between the first intercooler and through the inlet port in the second-stage cylinder head — and then into the second-stage cylinder during the downward (intake) stroke of the second-stage piston. The operation inside the three subsequent stages is identical to that of the first stage.

9-3.2.3 Moisture Separators

Moisture separators (Fig. 9-54) are used in conjunction with a chemical drier as the dehydration equipment of a high-pressure pneumatic system. These units, working together, deliver dry air having

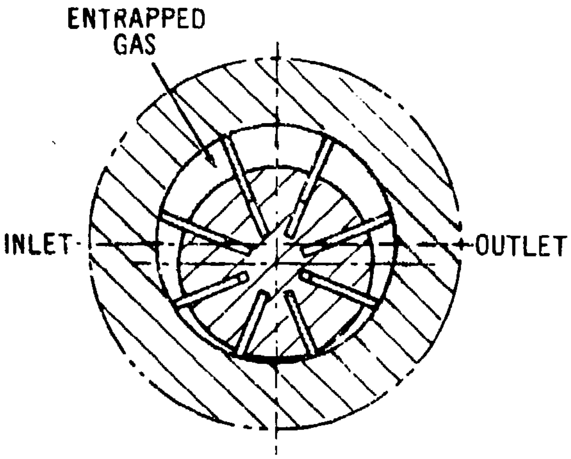


Figure 9-53. Sliding Vane Compressor

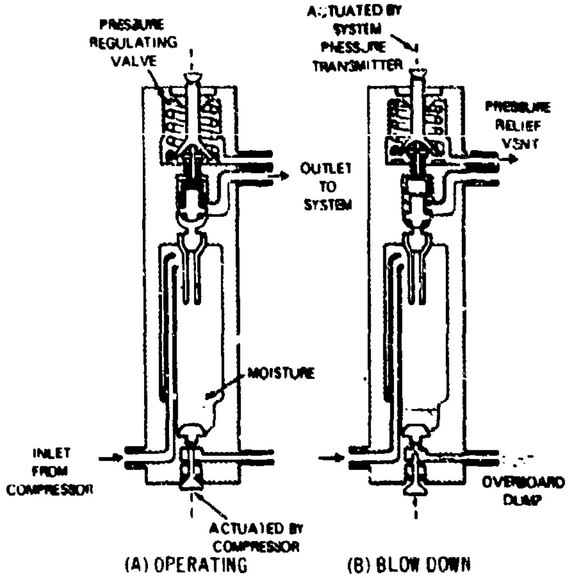


Figure 9-54. Moisture Separator Incorporating System Pressure Regulating Valve

a free air dewpoint of −65°F by trapping and collecting droplets of moisture that literally have been squeezed out of the air during compression. In a system pressure of 200 atmospheres, the separator reduces the free air dewpoint to −15°F. Collected moisture is drained from the separator either by a dump mechanism that operates automatically when system pressure drops upstream of the separator, or by a mechanism that discharges the moisture at frequent intervals while the compressor is supplying air to the system.

The air compressor incorporates a bleed valve that allows the separator to blow down automatically when the compressor stops running. During this operation, the compressor and the interconnecting lines to the separator also are blown down. The moisture separator includes a heating unit that prevents the accumulated water from freezing. Also included is a safety disk that protects the separator from the effects of overpressurization. A back-pressure valve is used directly after the moisture separator in order to build up pressure in the moisture-separation chamber before it can build up in downstream components. This valve insures immediate moisture separation when the compressor starts running. The valve either is integral to the separator or is installed separately. Valves are available with various back-pressure settings.

9-3.2.4 Dehydrators

The basic types of vapor-removal equipment are mechanical and chemical.

Mechanical dehydration usually includes a refrigeration cooling process that lowers the air temperature below the required dewpoint. The condensed water then is collected and eliminated. The limiting factor is the temperature to which the air can be lowered. This type usually is not found on airborne systems due to the weight penalty of the refrigeration equipment.

Chemical dehydration normally is used in conjunction with a moisture separator in order to provide maximum efficiency. The chemical drier is placed immediately after the moisture separator (refer to par. 9-3.2.3). The pneumatic system beyond this dehydration equipment thus operates with dry air, reducing the possibility of freezing in lines or components. Although they are called chemical driers, these units reduce moisture content by the process of absorption, and no chemical change takes place. Each unit consists of two parts: (1) a metal housing that acts as the pressure container, and (2) a replaceable cartridge containing the drying agent. The life of a cartridge will depend not only upon the rate of airflow, but also upon the ambient and inlet air temperatures and the moisture content.

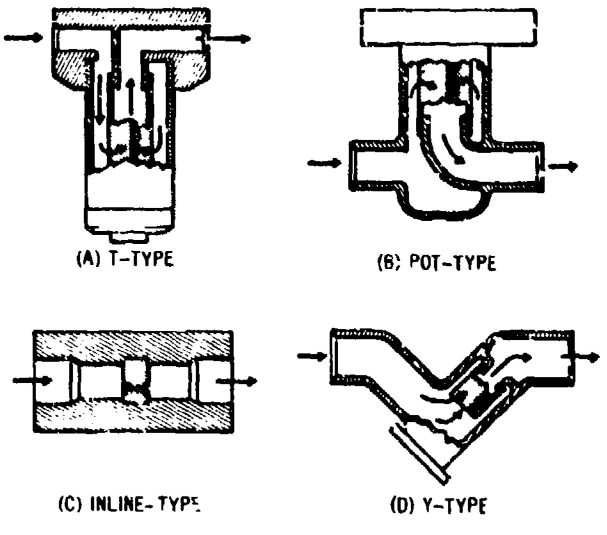
If the chemical drier is to be serviced manually, a filter *shall* be included at the outlet port so as to prevent downstream migration of particles of the replacement compound. Some means of indicating when the compound is no longer removing water vapor effectively *shall* be included. In the manually reactivated types, ease of compound removal for replacement or reactivation should be considered. The compound should be in cartridge or capsule form to prevent spilling.

9-3.2.5 Filters

For long life and trouble-free performance of system components, the air should be kept as clean as possible. Fig. 9-55 shows some filter configurations. There are many sources of contamination: the air itself may be contaminated by dirt from the atmosphere, or from system hose connections and other transfer devices. Particles from worn system components are significant sources of contamination.

All filter media act to varying degrees as both depth and surface filters. However, they normally are classified on the basis of the predominant type of filtration provided.

Depth media depend upon long, tortuous flow paths to remove contaminants. Examples of these media include paper, cellulose, felt, glass fiber, wood pulp, and sintered powder. In fiber filters, variations in thickness, density, and fiber diameter are combined to produce nominal filtration of 0.5 to 100 microns. Absolute ratings vary from 2 to 50 times the


(A) T-TYPE

(B) POT-TYPE

(C) INLINE-TYPE

(D) Y-TYPE

Figure 9-55. Filter Housing Designs

nominal rating. Sintered powders of metal, ceramic, or plastic provide depth filtration with nominal ratings of 2 to 65 microns and absolute ratings of 13 to 100 microns. Media migration is the primary disadvantage of sintered powder, but this can be avoided by proper manufacturing techniques.

Surface media remove contaminants by means of a surface that contains fairly uniform orifices. Thus, the contaminants are retained on the media surface.

In wire mesh types, small, uniform-diameter wires, woven into a Dutch-twill or square pattern, provide nominal filtration ratings of 2 to 100 microns and absolute ratings from 12 to 200 microns. Wire mesh has good strength and is free from media migration. Dirt capacity per unit area is low, but the thinness of the mesh permits use of multiple layers.

9-3.2.6 Valves

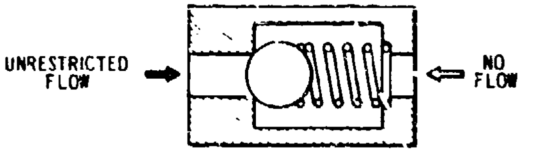
Valves include any device that stops, starts, or otherwise regulates the flow of a fluid by means of a movable element that opens or obstructs a flow passage. The most commonly used valves in any airborne pneumatic system are described in the paragraphs that follow.

9-3.2.6.1 Check Valves

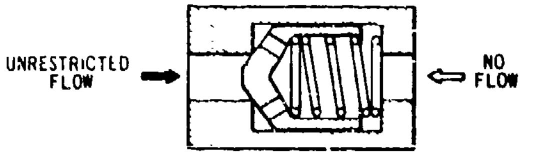
The primary function of a check valve is to prevent flow reversal. Check valves pass air freely in one direction and, if pressure reverses, close quickly to stop flow in the other direction. Flow reversal in fluid systems may be programmed as a normal occurrence, or may be caused by accidents or failures. Accidental flow reversal must be halted promptly and effectively. If this is not done, accumulators may be overpressurized, rotating equipment may overspeed, or other types of equipment damage may occur. Check valves are automatic in their operation, with their valving elements being activated by the forces of the following media. Four types of check valves are shown in Fig. 9-56.

In the ball check valve, a hardened ball serves as the closure element and is spring-loaded against a circular, conical, or spherical seat. Flow forces lift the ball off the seat and against the loading spring. Because the flow must proceed around the ball, this type of valve shows more of a tendency toward turbulence and pressure drop than do other types. During normal operation, the ball rotates slightly on the retaining spring, thus allowing even wear on the ball and the valve seat and minimizing the effects of contamination. Because of inherent simplicity and low cost, ball check valves are used frequently in applications involving small line diameters, where pressure drop is not of particular concern. Practically no damping can be incorporated into the mechanism of the ball check valve, and the chattering tendency cannot be eliminated. Therefore, ball check valves are not recommended for applications where chattering is unacceptable.

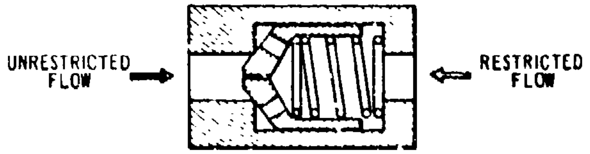
In cone check valves, the ball is replaced by a sliding element with a conical seating surface at one end. This surface seats against a circular sharp edge or another conical surface. Cone check valves generally have less pressure drop for a given size than do ball check valves, and have less tendency to chatter because of the guided movement and resultant damping of the valving element. Cone check valves are susceptible to dirt in the seating area and between the piston and body; this can cause cocking, with a resultant leakage between the piston and the body seating area. Cone check valves generally are used in the same types of applications as are ball check valves. However, cone types can be used to produce a reduced pressure drop in a valve of given size, and also can be used in applications where the tendency to chatter cannot be tolerated.



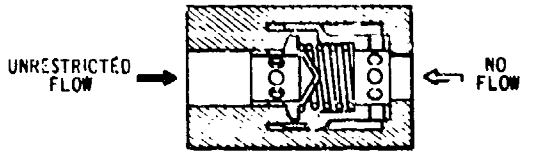
(A) BALL CHECK VALVE



(B) CONE CHECK VALVE



(C) RESTRICTION CHECK VALVE



(D) POPPET CHECK VALVE

Figure 9-56. Check Valves

A variation of the cone check valve is the restriction check valve. In this type, full flow is allowed in the forward direction, while a restricted flow is obtained in the reverse direction by means of a small orifice in the conical seating element. This type of action cannot be achieved in a ball check valve.

Poppet check valves consist of a mushroom-shaped poppet, with the stem closely guided in the valve body and the head sealing against a flat or tapered circular seat. In this valve, flow forces in the forward direction lift the head of the poppet off the seat, and flow proceeds through the stem of the poppet, around the head, and through the body of the valve. In general, poppet-type check valves have less pressure drop for a given flow rate than do either cone or ball types. Poppet check valves can be designed so as to eliminate any tendency toward chatter or hammering by the incorporation of damping chambers in the valve. Because of the close clearances between the poppet stem and the valve body, contamination can cause sticking and leakage. Poppet check valves require more parts than ball or cone types, and, therefore, usually are more costly. They are used most commonly in applications where it is desirable to improve flow characteristics.

9-3.2.6.2 Relief Valves

The primary use of relief valves is to control fluid pressure in a tank or system by discharging excess flow to an area of lower pressure. A relief valve is a pressure-relieving device that opens automatically when a predetermined pressure is reached. Relief valves may have a full opening "pop" action, or may open in proportion to overpressure. Valves that open rapidly to full flow generally are referred to as safety valves or pop valves, and are considered a special form of relief valve.

A relief valve consists of a valve body, a reference load, and a closure that serves as a control element and seat (Fig. 9-57). The reference load is linked to the closure, and opposes the pressure buildup in the tank or system. The magnitude of the load determines the relief pressure setting. As the internal system pressure increases to nearly the relief pressure necessary to balance the reference load, leakage usually begins. When the internal pressure reaches the relief pressure level, the valve opens and discharges the upstream air. As internal pressure decreases below the set pressure, the reference load opposing the pressure force closes the valve.

A relief valve is considered to have good operating characteristics when the pressure for rated flow and reseat closely approaches the cracking pressure. The cracking pressure is the relief pressure setting of the valve, defined as the pressure where leakage flow reaches some specified value. The cracking pressure always is set below the allowable working pressure of the tank or system, and commonly is not more than 110% of normal operating pressure. The rated capacity usually is established for flows at pressures 10% greater than the pressure setting of the relief valve. The reseat pressure is some value below the cracking pressure, depending upon the closure configuration; a reseat pressure of 95% of cracking pressure is common.

Relief valves may be either direct-acting or piloted. Direct-acting valves can be either of the conventional type, where the control element moves relative to the seat, or of the inverted type, where the seat moves relative to the control element.

For airborne applications, the relief valve body is designed for minimum weight, consistent with pressure rating, and for passage of high flows with minimum pressure loss. Lightweight construction materials, such as aluminum, are used extensively. To achieve minimum pressure drop, some manufacturers use a venturi design in the discharge side, while others enlarge the outlet port even to the extent of using larger connections.

The reference load is the force opposing any pressure buildup until relief pressure is reached. The most common element used to establish the load is a compression spring. Weights could accomplish the same purpose, but seldom are used.

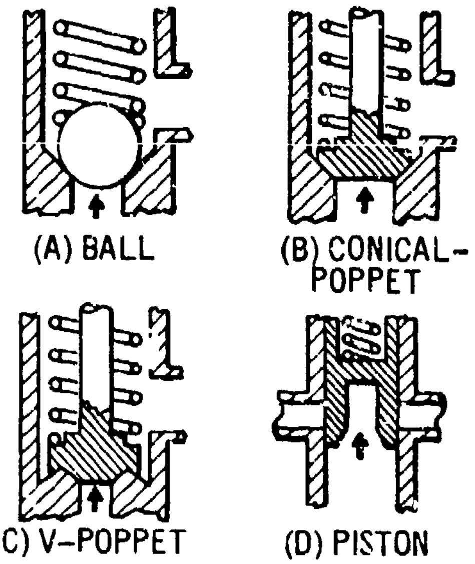


Figure 9-57. Relief Valves

The valving unit is composed of a seat and a control element. The seat may be flat, spherical, or conical in shape, with its configuration determining the sealing and opening characteristics of the relief valve to a large degree. There are four commonly used control element designs: ball, conical-poppet, V-poppet, and piston.

The ball control element is used extensively in both quick-opening and proportional relief valves because of its simplicity, low manufacturing cost, and inherent self-aligning capability as it reseats. When used in a quick-opening relief valve (safety valve), the ball tends to chatter when discharging fluid. It is limited to small valve sizes, and has a short life cycle.

Like the ball control, the conical-poppet may be used in both quick-opening and proportional relief valves. Conical-poppets lend themselves to larger port sizes, but require closer tolerances on the sealing surfaces of the poppet and seat. The control element stem must be guided in order to obtain alignment between the poppet and seat. When the valve is open, however, the inside surface of the guide is exposed to the fluid. If gas is discharged, cooling during expansion may result in an ice buildup on the guide that will prevent the valve from closing. This type is quieter in operation than the ball, due to frictional damping induced by the guide.

The V-poppet element is used only in safety valves. As soon as the valve starts to open, the fluid — by changing momentum due to the V-design — exerts a greater force against the poppet, causing it to pop open for full flow. The poppet uses only its inner cone for a sealing surface. Like the conical plug, the poppet stem is guided. Precision machining is required in order to obtain accurate poppet-seat concentricity and alignment.

A piston sometimes is used in relief valves for closed air systems. The piston offers no positive seating surface to prevent leakage, depending primarily upon close tolerances. Valve opening is proportional to the overpressure. Pistons are used commonly as the second stage in pilot-operated relief valves, rather than in single-stage valves.

9-3.2.6.3 Pressure-reducing Valves

The most practical components for maintaining secondary lower pressures in a pneumatic system are pressure-reducing valves. These are normally open, two-way valves that sense downstream pressure in order to close. There are two types, direct-acting and pilot-operated.

Direct-acting valves are usually of sliding spool design. Air flows from the high-pressure inlet to the low-pressure outlet. An adjustable spring holds the poppet or spool open, and reduced pressure acts to close the valve. When the valve is closed, a small quantity of air bleeds from the low-pressure side of the valve through the spring chamber to the atmosphere, usually through a fixed or adjustable orifice in the spool or body. The bleedoff prevents downstream pressure from increasing above the valve setting because of spool leakage when the valve closes. The spring chamber always is drained to the atmosphere in order to prevent fluid pressure from building up and holding the valve open. Direct-acting valves require a large envelope to provide space for the spring and adjustment. Also, spring ranges usually are narrow. As in relief valves, a small pilot section may be added to control the main valve.

In a pilot-operated, pressure-reducing valve, the spool or poppet is balanced pneumatically by downstream pressure at both ends. A light spring holds the valve open. A small pilot relief valve, usually built into the main valve body, bleeds air to the atmosphere when reduced pressure reaches the pilot valve spring setting causing a pressure drop across the spool or poppet. Pressure differential then moves it toward its closed position against the force of the light spring. The pilot valve relieves only enough fluid to position the main valve spool or poppet so that flow through the main valve equals the flow requirements of the reduced-pressure circuit. If no flow is required in the low-pressure circuit during a portion of the cycle, the main valve closes. High-pressure air leaking into the reduced-pressure section of the valve then returns to the atmosphere through the pilot relief valve. Pilot-operated, pressure-reducing valves generally have a wider range of spring adjustment than do direct-acting valves, and provide more repetitive accuracy. However, contamination can block flow to the pilot valve, causing the main valve to fail to open properly.

9-3.2.6.4 Pressure Regulators

The pressure compensator bypass flow regulator and moisture separator usually is a part of the air compressor, and controls flow by diverting excess compressor output overboard. In a typical example, flow-pressure drop across a metering orifice is used to shift a balanced spool against a control spring. This spool movement is used to maintain a constant pressure drop across the orifice, diverting or bypassing excess supply flow. The pressure drop, which is determined by spool area and spring force, is relatively low.

9-3.2.6.5 Directional Control Valves

This term describes all multiple-passage valves, because their primary function is to control the direction of flow from one fluid line to another. Common types include three-way, four-way, diverter, sequence, and shuttle valves. Actuation may be manual, mechanical, pneumatic, or electrical. These valves are identified by method of actuation, number of parts, number of positions to which the valve can be actuated, type of valving element (spool, slide, poppet, ball, etc.), and type of sealing.

A three-way valve (Fig. 9-58) is one with three external port connections and is either two- or three-position. The usual three-way valve has one common port that can be connected to either one or two alternate ports while closing the nonconnected port. Normally, these ports are identified as pressure, cylinder, and return (vent). When used to control a single-action cylinder, the cylinder port is the common port, and is connected alternately to the pressure port and to the return port.

A four-way valve (Fig. 9-59) has four external port connections, which usually are arranged so that there are two simultaneous flow paths through the valve. Four-way valves commonly are used to actuate double-acting cylinders. In such applications, the valve is connected so that when pressure is applied to one cylinder port, the other cylinder port is vented, and vice versa. Four-way valves normally are two- or three-position. In a three-position, four-way valve, there is a center position in which all ports are vented.

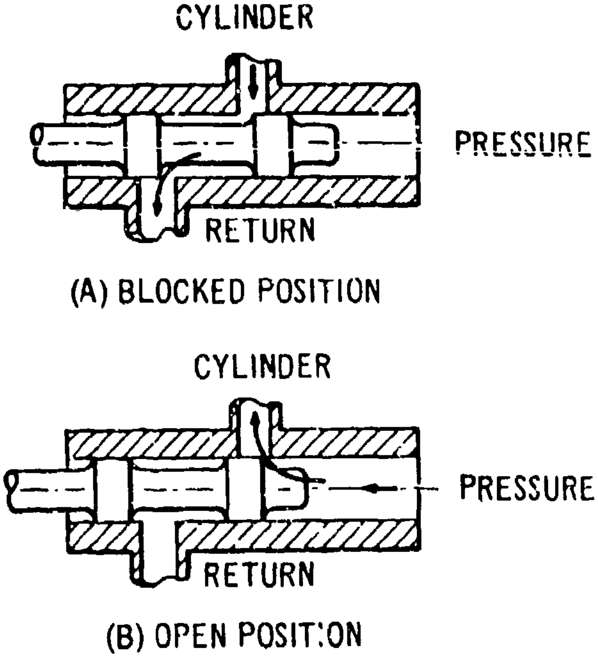


Figure 9-58. Directional Control Valve—Three-way

A diverter valve basically is a three-way valve, with the common port being the pressure port. Flow can be diverted from the pressure port to either of two alternate flow paths. Diverter valves also are called diversion valves.

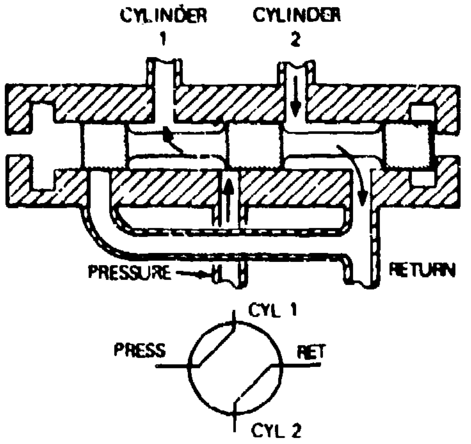
A selector valve functions similarly to a diverter valve, except that the common pressure port can be connected to an unlimited number of alternate flow paths.

A sequence valve is one whose primary function is to direct flow in a predetermined sequence between two or more ports. A shuttle valve is a type of sequence valve that is pressure-actuated in such a manner that when a preset system pressure has been reached, the valve automatically actuates, connecting two or more flow paths.

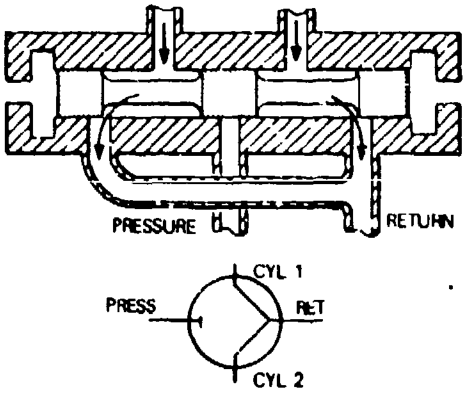
A spool valve controls fluid flow by covering and uncovering annular ports with lands on a sliding spool. The number of lands and ports on the spool and valve body determines the porting arrangements that can be achieved, and the geometrical relationship between the lands and ports determines the timing of the valve function. With a sharp-edged land and port, operation of a spool valve is abrupt. In applications where this arrangement would cause undesirable pressure surges, the land edges can be notched, tapered, or chamfered to modify the flow characteristics. Spool valves are classified as packed or unpacked, depending upon the sealing characteristics. Packed spool valves use O-rings or some other type of seal between the spool and the valve body in order to achieve tight shutoff. Unpacked spool valves possess internal leakage, depending upon the clearance between the spool lands and the valve body. Annular grooves usually are machined on the spool lands to improve lubrication of the valve and to equalize pressure all around the spool in order to prevent binding on one side of the bore. In addition to eliminating binding, the annular groove centers the valve, and in this position the leakage clearance is minimized. A unique feature of the spool-type, multiple-passage valve is that the end of the spool can be used as the actuator piston to position the valve.

Poppet-type, multiple-passage valves use two or more flat, conical, or spherical seats on a translating poppet. These valves lend themselves to three- or four-way operation with a variety of seating and sealing arrangements. For a solenoid-actuated, three-way, poppet-type valve in the de-energized position, pressure is applied to the cylinder.

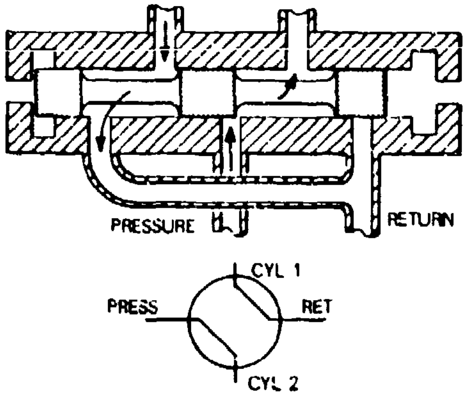
The sliding plate valve consists of three main elements, a slide and two plate enclosures. The slide element is sandwiched between the two plates, and contains cored holes and passages that mate with



(A) POSITION 1



(B) BLOCKED POSITION



(C) POSITION 2

Figure 9-59. Directional Control Valve—Four-way

ports in the plates. Many porting arrangements are available, and three- and four-way mulitple-passage valves can be achieved easily. One advantage of a sliding plate valve is that it can be reworked and lapped to compensate for wear.

A ball valve can be adapted readily to operate as a multiple-passage valve by the addition of outlets on the body and additional porting in the ball valving element. With three outlet connections on the body, the ball valve can be made into a variety of three-way valves, depending upon the porting utilized in the ball.

A rotary slide valve is used more commonly as a multiple-passage valve than as a two-way shutoff valve. Rotary slide, multiple-passage valves consist, essentially, of two parts, i.e., the body and a rotating plate (Fig. 9-60). The body contains either three or four outlets to provide a three- or four-way valve configuration, and the rotary plate contains various porting arrangements so as to achieve a multiplicity of three- or four-way valve types.

A valve actuator is a power unit that provides a mechanical operating force for positioning a valving element. The actuator may be either direct-acting or piloted. The direct-acting valve actuators include piston cylinders, solenoids, electrical motors, and servo torque motors, each used independently. Piloting, involving the use of a small power input to control a larger power source, is common with such piston-cylinder actuators as the electro-pneumatic, pneumatic-hydraulic, or pneumatic-pneumatic combinations. Most aerospace valves are powered by remote-control, or automatic, actuators. Linear actuators used in aerospace valves include solenoids, piston-cylinders, bellows, and diaphragms. Rotary actuators also can be used to impart linear motion, for example, through a rack and pinion. Linear actuators also can be used to impart rotary motion by driving an internally threaded valve stem attachment with a threaded rotating shaft. The actuator may be either an integral part of the valve, or a separate device linked mechanically to the valve. Solenoids, diaphragms, bellows, piston-cylinders, and servo torque motors are usually integral parts of the valve. Electric motors commonly are linked to the valve through a gear train, screw drive, etc. Valve actuator positioning requirements can be divided into two groups:

1. Two-position, or nonmodulating, actuators for "on-off" control valves and shutoff valves
2. Modulating actuators for positioning control valves; pneumatic and hydraulic piston-cylinder actuators commonly are used with valve positioners, and provide a feedback loop between the control

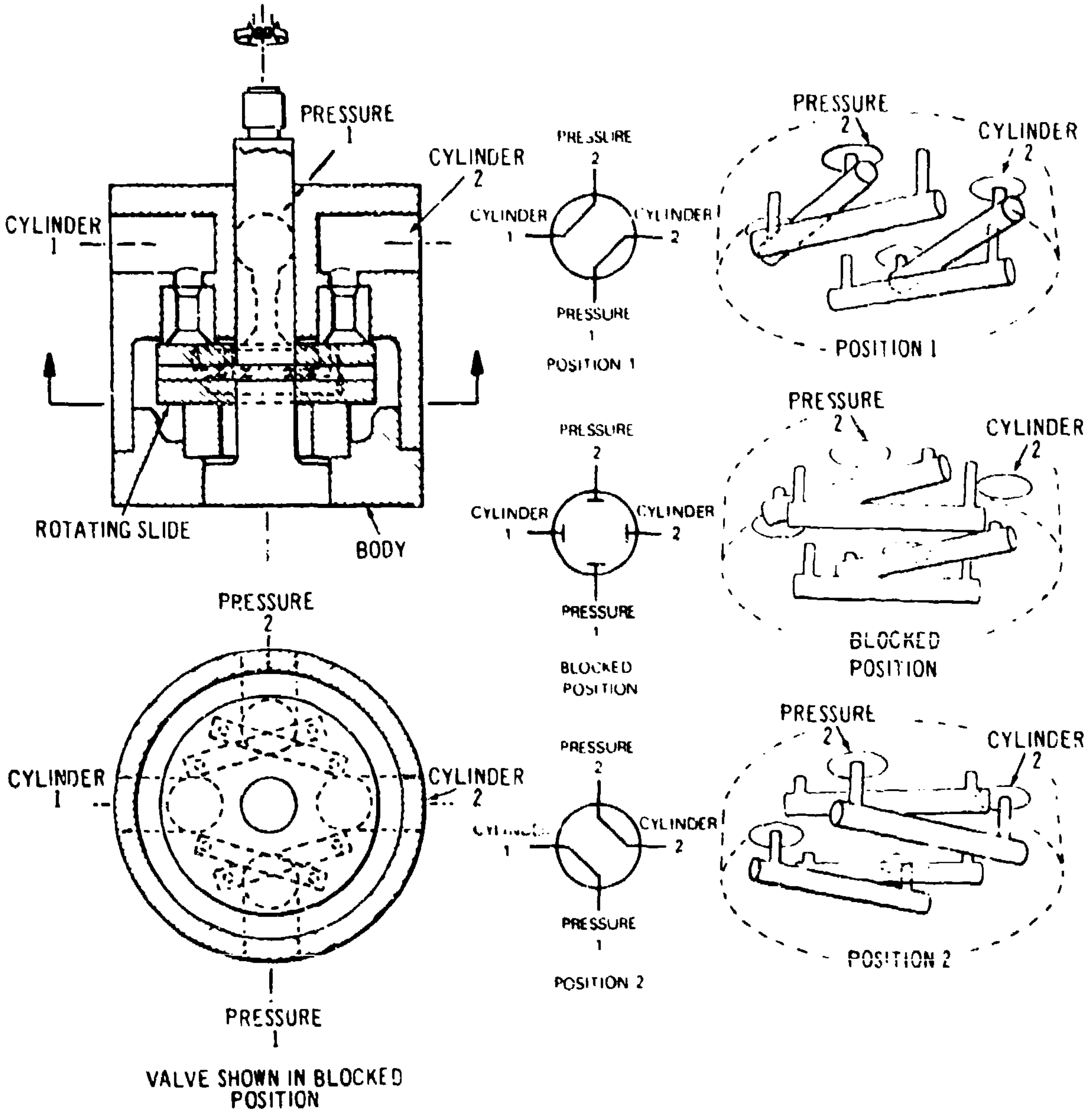

PRESSURE 1
CYLINDER 2
CYLINDER 1
ROTATING SLIDE
BODY
PRESSURE 2
CYLINDER 1
CYLINDER 2
PRESSURE 1
VALVE SHOWN IN BLOCKED POSITION
PRESSURE 2
CYLINDER 1
CYLINDER 2
PRESSURE 1
POSITION 1
PRESSURE 2
CYLINDER 1
CYLINDER 2
PRESSURE 1
BLOCKED POSITION
PRESSURE 2
CYLINDER 1
CYLINDER 2
PRESSURE 1
POSITION 2
PRESSURE 2
CYLINDER 2
POSITION 1
PRESSURE 2
CYLINDER 2
BLOCKED POSITION
PRESSURE 2
CYLINDER 2
POSITION 2

Figure 9-60. Directional Control Valve—Rotary—Four-way

signal and the stroke of the actuator.

The design or selection of a valve actuator is, at best, a trade-off among several interrelated factors Typical mechanical factors that must be considered include length of stroke, locking requirements, speed, limitation of envelope size and weight, and magnitude of required force. If a long stroke is needed, use of solenoid is eliminated automatically. If fast response times are desired, actuation by electric motor, solenoid, or explosive charge should be considered. If high forces must be overcome, either hydraulic or pneumatic pressure must be used.

9-3.2.7 Pressure Gages

Pressure gages are used in fluid-power equipment to provide:

1. An indication of operating pressure, especially where this pressure must be selected by the operator

2. An indication (alarm) of abnormal pressure within the system.

Pressure gages also are used to provide data in development of fluid-power equipment.

Pressure, or pressure change, within a system must be correct if pneumatically powered or controlled equipment is to operate properly. The proper gage indicates this pressure and helps to prevent malfunctions. Gages also can be calibrated in values proportional to pressure, such as total force exerted by a pneumatic cylinder.

Bourdon-tube indicating dial gages (Fig. 9-61) are used to measure pressure from 0.5 psi vacuum to 150,000 psi. Primary advantages are accuracy, ruggedness, reliability, simplicity, and low cost. Other methods of measuring pressure include electronic devices based upon strain gage readings, or, in the case of pulsating pressure, piezoelectric crystals. Such units, relatively, are costly and complex. At the low end of the pressure-measurement spectrum are bellows and diaphragm-type devices, which are used to measure relatively low pressures.

Components of all Bourdon-tube gages are similar. However, many styles and materials should be considered in selecting a gage. The following factors, listed in the normal order of consideration, influence this selection:

1. Measured medium, including pressure range and fluctuation
2. Environmental conditions, such as temperature and vibration
3. Wear conditions caused by pulsation and vibration
4. Connection of gage to measured medium
5. Mounting method
6. Size and weight
7. Accuracy.

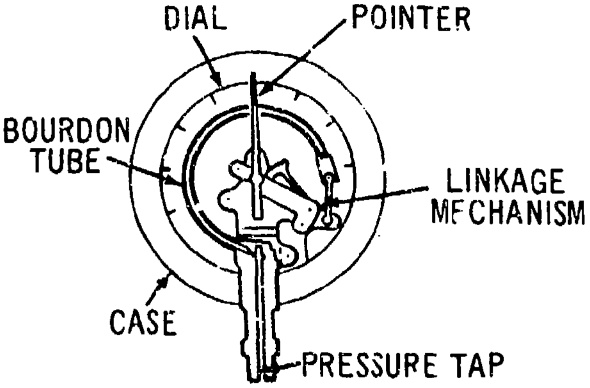


Figure 9-61. Pressure Gage—Bourdon-tube Type

9-3.2.8 Air Storage Bottles

Air storage bottles or vessels are often of conventional shapes, such as cylinders or spheres. On the other hand, limited space may require a conical, oblate spheroid, toroidal, or pear shape. Drawn cylinders up to 9 in. in diameter and up to 50 in. long are available. Welded containers can be made much larger. Capacities may range from 3.0 to 3500 in.' Cylinder bottoms may be concave, bump-shaped, spherical, or elliptical. Ports can be located wherever they are required. Either external or internal threading can be supplied.

Pressure vessels with pressure ratings varying from a few hundred psi to 25,000 psi are available. Pressure vessels can withstand high ambient temperatures in the range of 275°-600°F. In addition, they can meet vibration, shock, and other extreme requirements of modern helicopter environments. A varied choice of constructions is possible. Among the metals that have been used successfully are aluminum, low-alloy steels, high-strength steels, and a number of exotic metals. For highly predictable, multicycle performance, one of the chromium steel alloys is recommended.

Seamless cylinders are deep drawn, and wall thickness tolerances can be held precisely. An excellent surface is obtained. The dome and neck are hot spun on the open end. Heat treatment produces the metallurgical quality needed for best performance. Seamless drawn cylinders are relatively inexpensive to produce.

Welding often is used to fabricate the larger sizes of cylinders. Certain smaller cylinders for aerospace applications, where minimum weight is a more important consideration than is price, also are welded. Containers with a wide range of wall thicknesses, diameters, and alloys can be welded, as can vessels of exotic high-strength metals.

Fiberglas vessels usually are lighter than their all-metal counterparts. Advanced technology has made them highly reliable. Silane-finished glass, bonded with epoxy resin and protected against moisture penetration by an external coating, can be used to fabricate pressure vessels. Internal rubber linings effectively retain the air in instances where some permeability can be tolerated.

Composite cylinders offer extremely low weight without sacrificing reliability. In this type of construction, a cylindrical metal shell having hemispherical ends is wrapped with circumferential weldings — usually of bonded Fiberglas. Hoop loads are shared between metal shell and windings while longitudinal loads are carried by the metal alone. Wire winding may be applied in accordance with

Military Specifications in order to keep vessels from shattering under gunfire. Wire winding contributes added strength, but usually is not regarded as a lightweight composite construction.

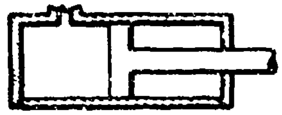
Pneumatic-system vessels, which must be charged and discharged several times daily, must withstand many thousands of cycles without failure. To insure that high-cycle-life vessels will meet their requirements in service, factors such as stress level, choice of material, shape of the vessel, surface conditions, joint design, heat treatment, and environmental conditions during use must be considered. The ratio of test pressure to service pressure is usually 1.67:1. However, the ratio is governed by individual specifications, and, in some cases, may range from 1.5:1 to 2.0:1. The ratio of minimum burst pressure to service pressure depends upon the design stress levels, but is commonly 2.22:1. Ratios of 2.0:1 to 4.0:1 often are indicated by specific cycle-life requirements and other service considerations.

9-3.2.9 Subsystem Components

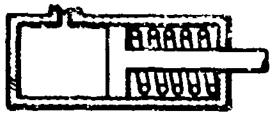
Subsystem components are discussed in pars. 9-3.2.9.1 through 9-3.2.9.4.

9-3.2.9.1 Actuators

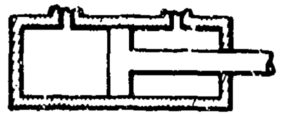
An actuator, as used in helicopter applications, is a power unit that produces a force or torque for positioning loads. Normally, pneumatic actuators are of the linear-motion type, and are designed to individual specifications. Among the types manufactured are specialized actuators of the piston type (with built-in dampers), used for the retraction of landing gear; landing gear up-lock actuators; high-temperature piston units for both high and low pressures; cargo and passenger door actuators; store-ejection actuators; screw-jack actuators for high-temperature applications, and air motor and screw-jack actuator assemblies that form a part of such systems as nosewheel steering. Advantages of pneumatic actuators include speed of operation, simplified power requirements, and ability to withstand ambient temperatures to 500°F. The inherent limitations of pneumatic actuators result primarily from the elastic properties of the compressed air working fluid. Fig. 9-62 illustrates typical linear pneumatic actuators.



(A) SINGLE ACTING



(B) SPRING RETURN



(C) DOUBLE ACTING

Figure 9-62. Typical Linear Actuator Types

In this single-acting actuator, the power stroke is in one direction only, and can be either the out-stroke or in-stroke. The return stroke is accomplished by some external means; a double-acting cylinder can be used for this purpose by connecting the actuating fluid line to only one port through a three-way valve, leaving the other port open. Special single-acting cylinders are designed with piston-sealing devices that seal in one direction only. These cylinders have a port hole in one head and a bleeder hole in the opposite head.

The spring-return actuator is a single-acting cylinder, with the return stroke effected by a spring. The length of the cylinder in the retracted position is at least twice the actual stroke length because of the spring length. The initial spring force, as well as the increase in spring force due to spring rate during compression, depends upon the amount of force required by the spring-actuated stroke.

In the double-acting actuator, the cylinder has a power stroke in both directions. The actuating fluid line is connected to both heads of the cylinder, usually through a four-way valve. Most standard catalog cylinders are double-acting. The sealing devices also operate in both directions.

Rotary actuators rotate an output shaft through a fixed arc to produce oscillating power, converting fluid-energy input to mechanical output. They are compact, simple in construction, and efficient, producing high instantaneous torque in either direction, and requiring only limited space and simple mountings. Rotary actuators consist of a chamber or chambers for containing the working fluid, and a movable surface against which the fluid acts. The movable surface is connected to an output shaft to produce the output motion.

The basic types of rotary reciprocating actuators are vane and piston. Basically, the vane actuator consists of a cylindrical chamber, a stationary barrier, a central shaft with a fixed vane, and end caps through which the shaft projects and which support the shaft-and-vane assembly. Fluid energy on one side of the vane produces an unbalanced force on the shaft. The shaft extension can protrude from either or both ends, and has key-ways, splines, or squared ends for mechanical connection of the load. Vane actuators usually have one or two vanes, but may have three or more. The arc of rotation for single-vane units is about 280 deg; for double-vane units, about 100 deg. Maximum arc varies with the size and construction of the unit. Besides the torque, there is a radial force on the vane shaft, and this side load tends to deflect the shaft. Thus, the output shaft must be large enough to withstand maximum torque and side load without excessive deflection, and bearing areas must be large enough to support these loads with minimum wear and friction. Efficiency of single-vane units varies from 70 to 95%, depending upon such items as bearings and bearing length-to-diameter ratio.

Double-vane units contain opposing vanes and opposing stationary barriers. Fluid enters one compartment from an external port and flows through internal passages to the opposite compartment. A force is exerted on each vane, and the force that tends to displace the shaft is balanced transversely. Because pure torque is the only load on the shaft, efficiency of the double-vane unit is high; torque output is double that of a single-vane unit of comparable dimensions. Vane-type actuators are available in a variety of standard sizes and mountings, with torque outputs ranging from 3 lb-in. at 50 psi to more than 700,000 lb-in. at 3000 psi.

Piston actuators are available in several types. In a helical spline actuator, fluid is applied to one side of the piston which is kept from rotating by guide rods. The actuator can be stopped at any point in its stroke. The helix angle on the shaft and piston is self-locking, preventing rotation of the actuator under external torque loads. Sealing is by ring seals around the piston, the fluid rods, and the helical screw. Standard units are available for a wide range of torque outputs and pressures. The arc of rotation can be larger than 360 deg. Adjustable cushions are available to reduce shock at each end of the stroke.

A variation of this type of actuator uses two pistons in the same cylinder. The shaft has a right-hand helix on half of its length and a left-hand helix on the other half. Fluid is introduced into the area between the pistons and causes linear movement of the pistons, thereby imparting rotary motion to the center shaft. The piston-rack actuator may have a cylinder with two pistons, each integral with a rack, or two cylinders with four pistons. In each case, the racks engage a pinion in the center of the cylinder and, as the pistons move the racks, the pinion is rotated. Equal torque thus is produced in each direction of rotation. Pistons usually are sealed by standard O-ring seals. Units are available with high torque ratings and for pressures to 3000 psi.

9-3.2.9.2 Brake Valves

Pneumatic brake valves provide a means of supplying operating pressure to helicopter wheel brakes. The pressure applied to the pedals (or hand lever, depending upon valve design) delivers a regulated, proportionate pressure to the valve outlet port, thus allowing air to flow to the brakes. Releasing of the brake pedal (or handle) vents the pressurized air on the brakes. Pedal or handle travel is proportional to brake valve outlet pressure, i.e., to the pressure applied to the wheel brake. The design of brake valves should feature minimum hysteresis characteristics. For increased release response, rapid-exhaust valves are located between the brake valves and the brakes.

9-3.2.9.3 Pneumatic Fuses

A fuse is used to protect a pneumatic system while permitting full, efficient operation of components in the remainder of the system. Essentially, the fuse operates on a rate of flow that is sensed by a pressure drop through the fuse itself. Once closed, the fuse resets automatically as soon as the pressure differential is removed by venting the upstream compressed air.

9-3.2.9.4 Quick-disconnects

Quick-disconnects provide easy, instant coupling and uncoupling of pneumatic systems and system components without loss of supply pressure. Their use greatly facilitates aircraft overhauls and service replacements. In order to meet varying requirements, two types of quick-disconnects are available: a lever type and a rotating type.

9-3.3 PNEUMATIC SYSTEM INSTALLATION AND QUALIFICATION

Pneumatic systems are classified into types and classes as follows:

1. Types:

a. Type A. Airborne compressor-charged system, in which system air pressure is maintained by a compressor mounted in the helicopter

b. Type B. Ground-charged system, in which system air pressure is obtained from ground-servicing equipment

2. Classes:

a. Class 1. Supply system is charged to a pressure of 1500 psi

b. Class 2. Supply system is charged to a pressure of 3000 psi

c. Class 3. Supply system is charged to a pressure of 5000 psi.

The qualification testing required for Type A and B components is similar. The tests include those for examination of product, proof and burst pressure, leakage, flow and pressure drop, extreme temperature, life cycle, vibration, humidity, fungus, sand and dust, salt-fog, and dielectric strength. General requirements for pneumatic-system component testing are given in MIL-P-8564. The conditions specified should include the test media, temperatures, and filtration. System installation testing requirements are listed in MIL-T-5522 and AMCP 706-203.

9-3.4 PITOT-STATIC SUBSYSTEM DESIGN

Airspeed, pressure altitude, and rate of climb are basic parameters in the performance of all helicopters. Instruments used to measure these quantities are the airspeed indicator, altimeter, and rate-of-climb indicator. The pressure inputs to the airspeed indicator are obtained from the pitot tube, which measures total pressure, and from the static pressure source. The latter also provides pressure for the altimeter and rate of climb indicator.

The pressure sources on the outside of the fuselage are connected directly to the instruments in the cabin area by means of leak-tight tubing. Design and construction of the tubing installation is governed by MIL-P-5518 and MIL-P-8564. No valves or severe restrictions are permitted. Drain fittings *shall* be provided, as necessary, at low points in the system in order to permit removal of condensed moisture.

Military Specifications governing pitot and static systems on all aircraft and missiles are MIL-P-26292 and MIL-I-6115. These specifications are written primarily for conventional aircraft, but are applicable to compound helicopters having alternate means of producing horizontal thrust and, therefore, increased forward airspeed capability.

9-3.4.1 Altimeters

Most altitude measurements are made with a sensitive absolute-pressure gage, called an altimeter, scaled so that a pressure decrease indicates an altitude increase in accordance with the US Standard Atmosphere (Ref. 5). If standard atmosphere conditions exist and the altimeter setting is 29.92 in. Hg, the altimeter will read the correct field elevation when the helicopter is on the ground. The altimeter in its simplest form is shown in Fig. 9-63, and consists of an evacuated diaphragm or capsule mounted in an airtight case or static-pressure chamber. The diaphragm responds to changes in pressure by expanding and contracting, and the movement of the diaphragm is transmitted to a main pinion assembly. The dial is calibrated to read pressure altitude. The static pressure measured P_m at the static source of the altimeter may differ slightly from the true atmospheric pressure p. For any P_m, the altimeter, when corrected for instrument error, will indicate the measured pressure altitude corrected for instrument error H_m. The instrument error is an error built into the altimeter, consisting of such things as scale error. The quantity $(P_m - p)$ is called the static pressure error or position error and is determined through flight tests. The value that is added to H_m to determine true pressure altitude H is termed the altimeter position error correction ΔH_c.

$$\Delta H_c = H - H_m, \text{ ft} \tag{9-1}$$

where

H = true pressure altitude, ft

H_m = measured pressure altitude, ft

ΔH_c = altimeter position error correction, ft

Minimum performance standards for a pressure-actuated, sensitive altimeter are given in FAA TSO-C10b. There also are a number of Military Specifications available that cover specific altimeters currently used by the military.

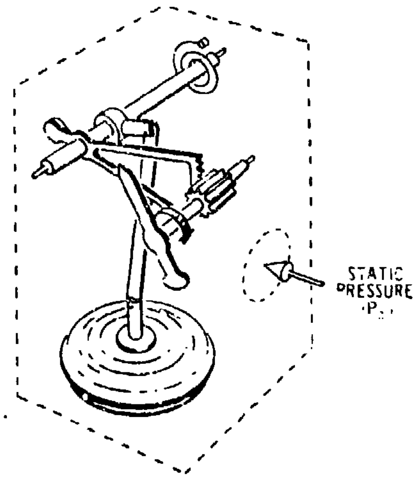


Figure 9-63. Altimeter Schematic

The design and installation of the altimeter system *shall* be such that the error in indicated pressure altitude at sea level in standard atmosphere, excluding instrument calibration error, does not result in a reading more than 30 ft high nor more than 30 ft low in the level flight speed range from 0 mph to 0.9 times the maximum speed obtainable in level flight with rated rpm and power.

9-3.4.2 Rate-of-climb Indicator

The rate-of-climb indicator uses the same static pressure source as the altimeter. A calibrated flow restriction is placed in the unit to restrict the passage of air into the instrument case when the measured static pressure changes. The time lag associated with this pressure change is used to obtain a pressure differential on two sides of a diaphragm. This pressure differential is displayed mechanically as change in pressure altitude with time dH/dt at standard sea level conditions. The relationship is in accordance with the hydrostatic equation for small differences.

$$\Delta P = \rho_{sl} g(\Delta H), \quad \text{lb/ft}^2 \qquad (9\text{-}2)$$

where

g = acceleration due to gravity, ft/sec²
ΔH = height difference, ft
ΔP = pressure difference, lb/ft²
ρ_{sl} = density at standard sea level conditions, slug/ft³

The slow response time of those basic mechanisms is now generally corrected by the incorporation of accelerometers which provide an artificial boost of air for an instantaneous needle movement.

9-3.4.3 Airspeed Indicators

True airspeed V is the velocity of the helicopter with respect to the air through which it is flying. It is difficult to measure true airspeed directly. Instead, calibrated airspeed V_c is measured. Calibrated airspeed is determined from the difference between total pressure and static pressure using Bernoulli's compressible equation for frictionless adiabatic (isentropic) flow. Calibrated airspeed is the adopted standard reading of an airspeed indicator, and is the same as true airspeed under standard sea level conditions. The difference between total pressure P_t and static pressure P is called true impact pressure q_c.

$$q_c = P_t - P, \quad \text{psi} \qquad (9\text{-}3)$$

where

P = static pressure, psi
P_t = total pressure, psi
q_c = true impact pressure, psi

A tabulation of q_c as a function of V_c is given in Ref. 6 together with a complete derivation of the equations used in establishing the table. Airspeed indicators are calibrated according to these relationships.

In operation, the airspeed indicator is similar to the altimeter. However, instead of being evacuated, the inside of the capsule is connected to a total-pressure source and the case to the static-pressure source. The instrument then senses the differences between measured pitot (total) pressure P_{t_m} within the capsule and measured static pressure P_m outside as shown in Fig. 9-64. The pressure differential q_{c_m} (Eq. 9-4), as measured by the airspeed indicator when corrected for instrument error, will correspond to measured calibrated airspeed V_{c_m}.

$$q_{c_m} = P_{t_m} - P_m, \quad \text{psi} \qquad (9\text{-}4)$$

where

P_m = measured static pressure, psi
P_{t_m} = measured pitot pressure, psi
q_{c_m} = measured impact pressure, psi

In general, this will vary from the correct calibrated airspeed because of pitot and static pressure errors, and an airspeed position error correction ΔV_c must be added to V_{c_m} to obtain V_c.

$$\Delta V_c = V_c - V_{c_m}, \quad \text{kt} \qquad (9\text{-}5)$$

where

V_c = calibrated airspeed, kt
V_{c_m} = measured calibrated airspeed, kt
ΔV_c = airspeed position error correction, kt

To determine its significance, this correction is evaluated by flight testing of the entire airspeed system.

Specifications have been written to establish the acceptable instrument errors and general construction of airspeed indicators. Table 9-2, AMCP 706-203, specifies requirements for airspeed indicators.

Because of the complex flow patterns around the helicopter in flight, the airspeed system must be flight

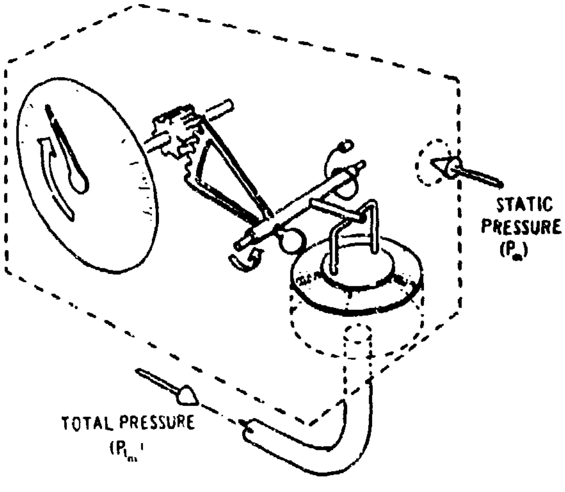


Figure 9-64. Airspeed Indicator Schematic

tested in order to evaluate performance. The flight test for airspeed calibration is described in par. 9-5.2, AMCP 706-203. This flight testing must be accomplished over the full range of flight capabilities of the helicopter, including climbout, cruise over the entire speed range, autorotation, and sideslip. Flight test techniques, established primarily for conventional aircraft, are described in Refs. 7, 8, and 9. Some specific flight test instrumentation for V/STOL aircraft is described in Ref. 10.

9-3.4.4 Total-pressure Sources

Pitot or pitot-static tubes are used to determine total or pitot pressure, which is defined as the pressure of the air when it is brought to rest isentropically. True total pressure is measured by a sharp-lipped pitot opening that is faced directly into the airflow. Pitot pressure errors can develop when the airflow impinges the tube at an angle of attack or incidence. A wind tunnel investigation of a number of total-pressure tubes at high angles of attack is described in Ref. 11. Results indicate that a sharp-lipped opening with shallow or no internal and external tapers has the least sensitivity to angle of attack.

The pitot opening is very susceptible to water ingestion. Therefore, it usually is desirable to place a water-collection chamber and drain hole in the pitot tube. The small amount of airflow that passes through the pitot tube and out the drain hole also causes a pitot pressure error, and should be investigated for each design.

Helicopters can and do operate in atmospheric conditions that are conducive to producing ice, e.g., from supercooled water droplets in the air or from freezing rain. The tip of the pitot opening, because it is in a stagnant-flow region, is susceptible to blocking by ice, and can become completely plugged and inoperative even before an appreciable accumulation of ice has developed on the rotors and other parts of the vehicle. It, therefore, is important to use electricially heated pitot tubes that are capable of deicing and anti-icing under the most severe atmospheric conditions that are likely to occur in flight.

Pitot tubes normally are placed in the forward area of the helicopter, outside of the airflow boundary layer. Locations high on the fuselage are desirable in order to avoid ground damage. The pitot tube *shall* be pointed into the nominal flow direction, and should not be placed behind any protrusion that could cause flow separation ahead of the tube. It also should not be located aft of windows or openings that could exhaust airflow into the pitot opening. It *shall* be located where the downwash from the rotors does not cause large-flow angles of attack and excessive flow pulsations. Locations on the forward top of the canopy or above the rotor on a stationary rotor hub have been found satisfactory.

Rotor downwash can cause pressure pulsation in a pitot tube. Ref. 12 evaluates the effect of these pulsations on total pressure. Proper design of the pitot tube and connecting lines for pressure-lag response, together with a favorable mounting location out of severe downwash, can eliminate the pulsation problem. Flight test investigation of pressure-lag problems is documented in Ref. 7.

Although one pitot tube is sufficient for obtaining total pressure, it is recommended that, if there are two sets of instruments (i.e., two airspeed indicators), separate pitot sources be used for each indicator. This redundancy permits detection of a faulty pitot pressure reading caused by a plugged or iced-over pitot opening or a pressure leak in the system.

9-3.4.5 Static Pressure Sources

A number of helicopters are designed with flush static pressure vents located on the fuselage. These static vents or ports *shall* be located such that vehicle speed, the opening or closing of windows, airflow variation, and moisture or other foreign matter will not affect their accuracy seriously. Each altimeter, airspeed indicator, and rate-of-climb indicator *shall* be connected into the system in an airtight manner, except for the static vents.

Two static vents normally comprise the static pressure system. They are located symmetrically on the right and left sides of the fuselage, and are interconnected to the flight instruments by a single tube containing a T-fitting located halfway between the two vents. This right- and left-hand installation is used in order to reduce sideslip errors in the system, and also provides a partial redundancy if one of the vents becomes plugged or damaged, or if a leak develops. For helicopters with advanced speed and altitude capabilities, a leak to a pressurized cabin area could cause a serious static pressure error. In this case, two completely separate static systems with separate right and left vents should be used if there are two sets of flight instruments.

Common locations for static vents are on the sides of the fuselage aft of the cabin or on the tail boom. A location *shall* be selected that is not sensitive to rotor downwash or forward speed, and that is away from windows, doors, or air vents that could produce a variable airflow geometry in the vicinity of the vents. A flight test program is necessary in order to determine the pressure influence of all variables, in addition to the calibration under conditions of climbout, cruise over the speed range, and autorotation.

Static vents *shall* be located so that no moisture can enter the openings under any service conditions. The static vent plate should be heated if there is a probability that ice could seal over the static vent. Other pertinent design information for flush static vents can be obtained from MIL-I-6115 and MIL-P-26292.

9-3.4.6 Pitot-static Tubes

On recent helicopters, the pitot and static pressure sources have been combined into a pitot-static tube. This tube can be straight for boom mounting, or L-shaped for mounting directly to the fuselage. Acceptable mounting locations are on a short boom ahead of the nose of the helicopter on a stationary hub on the top of the rotor, and on the top of the cabin — toward the front, yet near the rotor hub. Locations on the forward sides of the fuselage also are acceptable. However, low locations on the bottom of the fuselage usually are susceptible to ground damage. If the tube is located at the rear of the fuselage, right- and left-mounted units should be used in order to reduce the influence of sideslip on the static pressure measurement. For redundancy, two pitot-static tubes are recommended for all helicopters having two sets of flight instruments. Pitot-static tubes also can be designed with two sets of static ports if additional static sources are required.

Minimum requirements for pitot-static tubes are specified in MIL-P-33136. It is recommended that all pitot-static tubes be capable of completely deicing and anti-icing under the most severe environmental conditions likely to be encountered in flight. Placement of the static ports on a pitot-static tube also assures adequate deicing of the static vents.

Pitot-static tubes offer the possibility of aerodynamic compensation for static-pressure errors. This is accomplished by selectively designing the shape of the tube and the location of the static pressure ports. A general description of aerodynamic compensation is given in Ref. 13. The concept is used extensively for conventional aircraft, both commercial and military. MIL-P-83207 applies for straight-boom-mounted, aerodynamically compensated pitot-static tubes; and MIL-P-83206, covers L-shaped, compensated tubes.

Placement of the static ports on the pitot-static tube and away from the fuselage skin also can reduce errors caused by local skin irregularities in the vicinity of flush static vents. An extensive investigation of surface irregularities and nonreproducibility has been performed for conventional aircraft (Ref. 9).

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CHAPTER 10
INSTRUMENTATION SUBSYSTEM DESIGN

10-1 INTRODUCTION

This chapter discusses the instrumentation necessary in the helicopter cockpit to permit assigned missions to be performed. Because missions must be conducted at night, or under IFR conditions, the requirements for lighting of the instrument and control panels are included.

The instruments required in order to provide the pilot with the information necessary to conduct assigned missions normally should be grouped by functional categories. Included are flight, navigation, helicopter subsystem, and weapon system instruments. Installation requirements for instrumentation are discussed separately. Because of the high vibration environment in which helicopter instruments operate, they *shall* be designed and qualified to survive curve M (5G's) of method 514 of MIL-STD-810B.

The definition of the total helicopter instrumentation package *shall* include the arrangement of all displays and controls in the cockpit. Aircrew station instruments and displays *shall* be located in accordance with MIL-STD-250, unless unique mission requirements and supporting human factors engineering (HFE) analysis dictate otherwise. Displays for other crew stations *shall* be arranged to provide good control/display compatibility, along with minimum workload and minimum opportunity for human error. The most satisfactory method of assuring that these goals are achieved is by a systematic HFE analysis which should be accomplished as early in the design phase as possible. It is discussed in more detail in par. 13-3, AMCP 706-201. Formal mock-up review and evaluation are discussed in Chapter 5, AMCP 706-203. The formal function of the aircraft mock-up is not limited to the evaluation of the instrument subsystem.

10-2 INSTRUMENTATION LIGHTING REQUIREMENTS

10-2.1 GENERAL

The lighting concepts in use today vary from red to white in the color spectrum, and from direct lighting to diffused indirect lighting techniques. A number of solid-state, light-emitting instruments recently have been developed, and these may be appropriate for use in helicopter cockpits. Such applications have used electroluminescent (EL) lighting, liquid crystals, and light-emitting diodes (LED).

The lighting concept *shall* consider the total cockpit instrumentation requirements rather than individual needs. The necessity to consider all requirements cannot be overemphasized. Also, because the airframe developer usually procures cockpit instrumentation from many different sources, he must recognize his role as the cockpit lighting integrator early in the detail design process or the resulting cockpit lighting will suffer from problems of varying colors, imbalance, and unevenness even though all vendors are designing to the same requirements. Control of cockpit glare and reflections also must be addressed by the airframe developer.

Except for the primary instrument lighting, the cockpit lighting for all helicopters *shall* be designed in accordance with the applicable provisions of MIL-L-6503. Integrally lighted instruments having a white lighting system *shall* be designed according to MIL-L-27160. MIL-L-25467 *shall* be used if operator requirements dictate the use of red lighting. All switches, radio controls, auxiliary controls, and circuit breaker panels *shall* be illuminated by plastic-plate, edge-lighted panels, as specified in MIL-P-7788 and MIL-L-81774 for red lighting or MIL-P-83335 for white lighting. The secondary instrument panel lighting system *shall* provide a minimum of 10 ft-candles of either red or white illumination on the surface of the main instrument panel.

10-2.2 LIGHTING INTENSITY CONTROL

Unless otherwise specified by the procuring activity, the instruments *shall* be grouped on continuously variable lighting intensity controls as follows:

1. Side-by-side:
    a. Pilot's basic flight and navigation displays on main instrument panel
    b. Co-pilot's basic flight and navigation displays on main instrument panel
    c. Propulsion and other subsystem displays on main instrument panel
    d. Center console
    e. Overhead console
    f. Secondary panel lighting
2. Tandem:
    a. Pilot's basic flight and navigation displays on main instrument panel
    b. Pilot's propulsion and other subsystem displays on main instrument panel

c. Pilot's side consoles
d. Pilot's secondary panel lighting
e. Co-pilot/gunner's main instrument panel
f. Co-pilot gunner's side consoles
g. Co-pilot/gunner's secondary panel lighting.

Lighting rheostats *shall* be capable of continuous adjustment from FULL "ON" to FULL "OFF" to provide the low settings required for use with light amplification devices such as night vision goggles. A single switch to control all cockpit lighting *shall* be provided if night vision goggles will be used extensively.

10-2.3 LOW INTENSITY READABILITY

Particular attention must be directed toward optimizing the primary instrument lighting and edge-lit panels for readability at low intensity settings. All pointer, scale markings, and nomenclature *shall* be readable at 30% rated voltage when viewed from a distance of 32 in. under fully dark adapted conditions. This can best be accomplished by maintaining uniformity in scale design, and letter size and font. Two instruments may differ significantly in the total areas of the dial face markings and the instrument with the greater total marking area will appear brighter. Careful attention to balancing the area between instruments will reduce the differences. In some cases a resistor may be added in the lighting circuit of the instrument with greater apparent brightness if matching the area of the dial face markings is not practical. Additional design guidance in instrument lighting design may be found in Refs. 4 and 5.

10-2.4 WARNING, CAUTION, AND ADVISORY SIGNALS

All warning, caution, and advisory signals *shall* be designed, displayed, and operated in accordance with MIL-STD-411. A warning signal is a signal assembly indicating the existence of a hazardous condition requiring immediate corrective action. A caution signal is a signal assembly indicating the existance of an impending dangerous condition requiring attention but not necessarily immediate action. An advisory signal is a signal assembly indicating safe or normal configuration, condition of performance, operation of essential equipment, or to attract attention and impart information for routine action purposes. Special consideration *shall* be given to minimizing erroneous signals, and to combining several input parameters through logic networks in order to provide a more credible signal for such complex and critical situations as an engine-out condition. The use of a voice warning system (VWS) also should be considered. VWS is advantageous particularly during mission phases when the crew's task loading is high and their attention is directed outside the cockpit. Under such conditions, a light signal frequently may go undetected for long periods of time. In addition when a problem is detected via a master caution or warning light, the specific caution or warning light then must be located and read, and corrective action initiated or deferred, depending upon the criticality of the problem. With VWS, the crew is made aware immediately of the exact nature of the problem and can decide whether to initiate or defer corrective action without diverting attention from primary tasks. However, as with a visual overhead, noise and auditory load may not provide an environment conducive to the detection and recognition of aural caution or warning signals. The final mix of visual and auditory caution/warning signals *shall* be based on a human factors analysis of (1) criticality, i.e., time available to respond to each caution warning signal; and (2) the visual and audio workload of each mission segment for which a caution/warning signal is critical. The lists of warning, caution, and advisory signals that follows are provided as a suggested baseline. The final configuration must be determined from the subsystem failure modes and effects analysis, and the previously mentioned HFE analysis.

10-2.4.1 Warning Signals

Warning signals should include, but not be limited to, the following information:

1. Engine out (identify engine if multiengine)
2. Engine fire (identify engine if multiengine)
3. Landing gear up (if retractable gear installed)
4. APU fire (if applicable)
5. Other fire zones (as appropriate)
6. Low/high rotor/engine RPM.

10-2.4.2 Caution Signals

Caution signals should include, but not be limited to, the following information:

1. Low transmission oil
2. Low engine oil pressure (identify engine if multiengine)
3. Low hydraulic fluid pressure (identify system)
4. High engine oil temperature (identify engine if multiengine)
5. Low engine fuel pressure (identify engine if multiengine)
6. Engine fuel pump inoperative (identify engine if multiengine)

7. Low fuel quantity (20-min warning)
8. Fuel filter bypass operating (for each filter)
9. Oil filter bypass operating (for each filter)
10. Chip detector (engine)
11. Chip detector (accessory section)
12. Chip detector (transmission)
13. Chip detector (tail rotor gearbox)
14. Engine inlet icing (if applicable)
15. Other icing detectors (where appropriate)
16. Electrical system failure (both AC and DC)
17. Essential AC bus OFF
18. Main transmission oil pressure
19. Main transmission oil temperature
20. APU low oil pressure (if applicable)
21. APU high oil temperature (if applicable)
22. APU rotor speed (low/high) (if applicable)
23. SAS failure
24. Oil cooler bypass operating
25. Low oil level for each independent oil subsytem.
26. Drive system overtorque (if engine rating is significantly higher than drive system).

10-2.4.3 Advisory Lights

Advisory lights should include, but not be limited to, the following information:

1. AFCS disengage
2. Pilot heat ON
3. Parking brake ON
4. Anti-ice ON (if applicable)
5. External power ON
6. Starter ON
7. Rotor brake (if applicable)
8. APU ON (if applicable).

10-3 FLIGHT INSTRUMENTS

10-3.1 GENERAL

Helicopter flight instruments are basically similar to those of fixed-wing aircraft although frequently they are optimized or provided with additional features to make them more suitable for helicopters.

This paragraph discusses preferred arrangements and other characteristics for the selection of helicopter flight instruments. The detail specifications will define when multiple installations are required.

10-3.2 AIR SPEED INDICATORS

In view of the ability of the helicopter to fly at very low airspeeds, including hover, airspeed indicators suitable for use in fixed-wing aircraft are not acceptable for installation in helicopters. Instruments having increased accuracy in the low-airspeed range have been qualified, and *shall* be specified for helicopter installation. Airspeed systems that are capable of measuring and displaying both the magnitude and direction of the relative wind should be considered when accurate relative wind data are required to improve weapon or navigation system accuracy, or when relative wind limitations during hover are critical.

10-3.3 ALTIMETERS

Barometric altimeters installed in Army helicopters *shall* be of the counter-drum-pointer configuration. A maximum altitude reading of 30,000 ft is adequate for use in helicopters. One of the altimeters in each helicopter *shall* provide encoded altitude information to the transponder compatible with the CONUS air traffic control system.

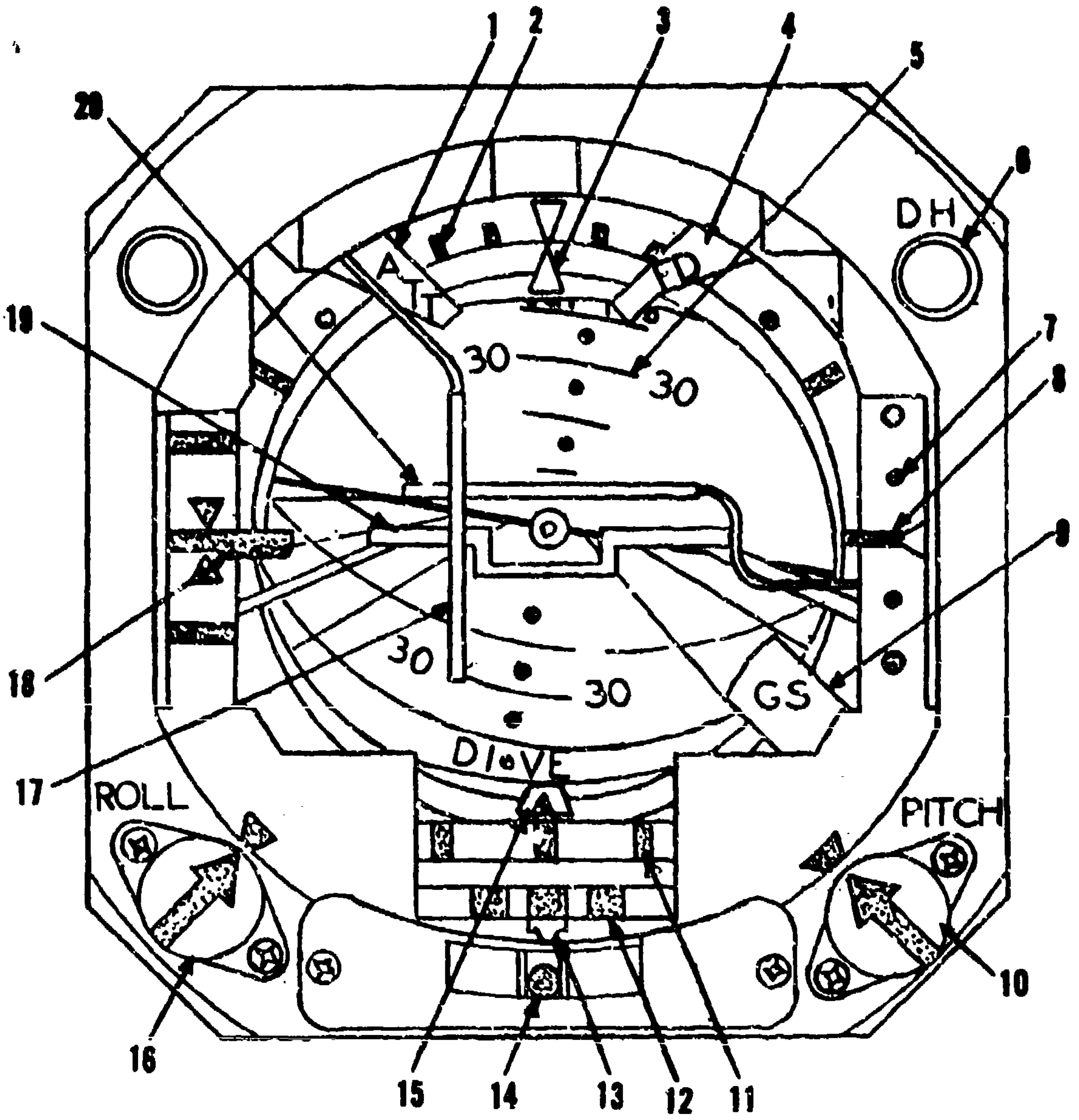
Some mission applications may require an altitude display that provides an accurate and direct indication of relative altitude or height above the terrain. In these cases, radio or radar altimeters will be specified in addition to barometric altimeters.

10-3.4 TURN-AND-BANK INDICATORS

The turn-and-bank indicator provides rate of turn, and/or needle and ball sideslip information. In some cases, the turn-and-bank indicator should be combined with the attitude indicator. However, the turn rate gyro and the attitude gyro *shall* be provided with independent power sources.

10-3.5 ATTITUDE INDICATOR

The attitude indicator provides the pilot with a substitute for the real horizon as a reference for maintaining desired aircraft pitch and roll attitude under all flight conditions. A typical attitude indicator is the IND-A5-UH1. This device is an electrically driven (400 Hz) gyro that is housed in a standard 5-in. instrument case, a size that is preferred for ease and accuracy of reading and interpretation. Because of the magnitude of instrumentation required in most Army helicopters, the basic attitude indicator generally is replaced with a more highly integrated display such as a vertical situation indicator (VSI). The VSI, if specified, *shall* provide the following as a minimum: attitude (pitch and roll), rate of turn, inclinometer information (slip and skid), FM homing and station passage, glide slope, pitch and roll trim, and weak signal flag alarm. When a flight director system is provided, command bars are added to the above resulting in an instrument referred to as an attitude director indicator (ADI). An example of an ADI which has been optimized for helicopter application, including a collective pitch command, is shown in Fig. 10-1.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1. GYRO FLAG
2. ROLL ATT. SCALE
3. ROLL ATT. INDEX POINTER
4. FLIGHT DIRECTOR CMD. FLAG
5. PITCH ATT. SCALE
6. DECISION HEIGHT LAMP
7. VERTICAL DEVIATION SCALE
8. VERTICAL DEVIATION POINTER
9. VERTICAL DEVIATION FLAG
10. PITCH TRIM KNOB
11. LATERAL DEVIATION SCALE
12. RATE OF TURN SCALE
13. RATE OF TURN POINTER
14. INCLINOMETER
15. PAD SYMBOL
16. ROLL TRIM KNOB
17. ROLL CMD. POINTER
18. COLLECTIVE CMD. POINTER
19. HELICOPTER SYMBOL
20. PITCH CMD. POINTER

Figure 10-1. Typical Helicopter Attitude Director Indicator

10-3.6 RATE-OF-CLIMB INDICATORS

The mechanization of rate-of-climb indicators is described in par. 9-3.4.2. Rate-of-climb indicators installed in Army helicopters *shall* be the rapid response accelerometer-aided type with a scale range of ± 6000 fpm.

10-4 NAVIGATIONAL INSTRUMENTATION

10-4.1 GENERAL

The types of navigational systems used in Army helicopters are dependent upon the mission assigned. The types of equipment to be installed will be defined by the detail specification for each model of helicopter. The detail specification also will indicate Government-furnished and contractor-furnished equipment. This paragraph discusses the types of navigation instrumentation most commonly employed in Army helicopters. The navigational systems that these displays are based upon and the functions performed by them are described in par. 8-3.

10-4.2 TYPES OF INSTRUMENTS

The instrumentation required for IFR flight is defined in AR 95-1. The minimum required navigation instrumentation is:

1. Magnetic compass with current calibration card
2. Clock with sweep second hand
3. Gyro-stabilized heading reference
4. Automatic direction finder (ADF)
5. VOR receiver (if VOR facilites are to be used).

Typically, these minimum requirements are met with three instruments: clock, standby magnetic compass, and radio magnetic indicator (RMI) which displays the last three of the given functions on an instrument similar to that shown in Fig. 10-2.

If mission requirements include extensive CONUS IFR flight, a simplified course indicator similar to that shown in Fig. 10-3 may be added to provide VOR/ILS course deviation; or the functions of the RMI, and course indicator, combined on a horizontal situation indicator (HSI) similar to that shown in Fig. 10-4. The HSI provides the additional capability of displaying the distance to the selected navigation aid or way-point. The capability to display both range (in kilometers) and bearing to selected waypoints *shall* be provided if mission requirements include nap-of-the-earth (NOE) navigation.

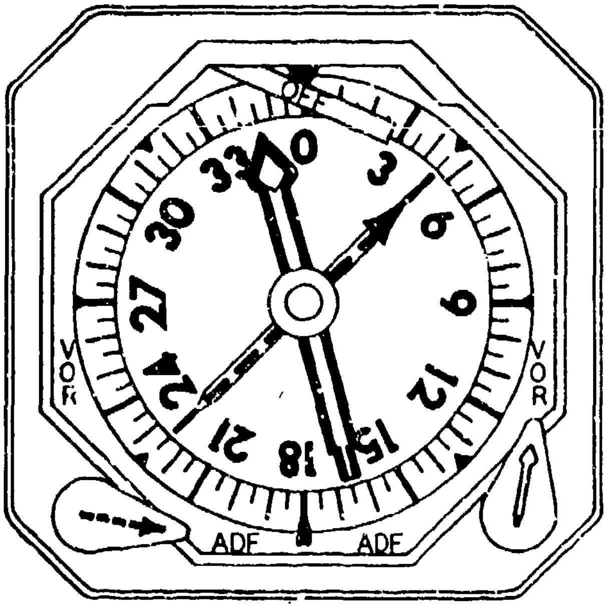
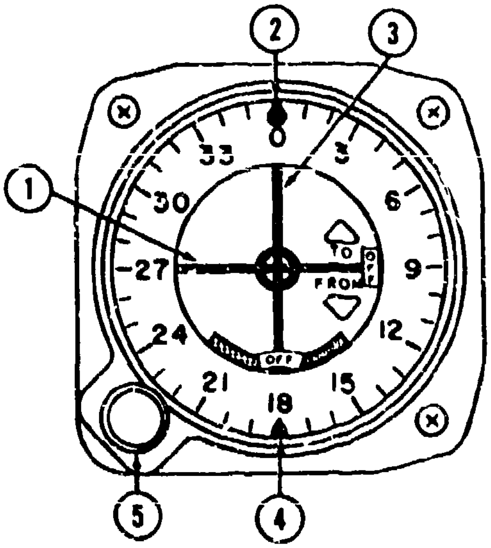
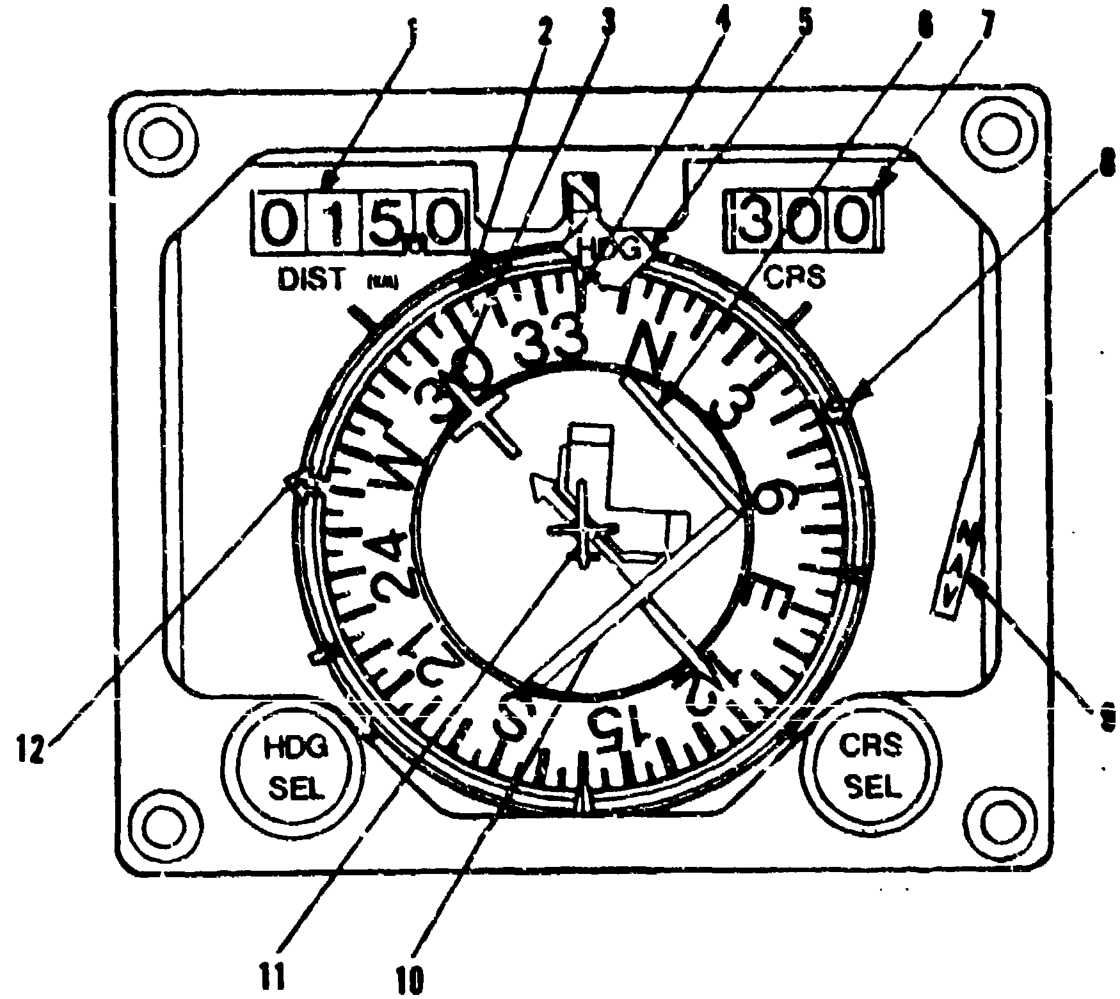


Figure 10-2. Radio Magnetic Indicator



1. HORIZONTAL POINTER
2. RECIPROCAL POINTER
3. VERTICAL POINTER
4. COURSE POINTER
5. COURSE SELECTOR KNOB

Figure 10-3. Course Indicator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NAV
HDG SEL
CRS SEL

1. DISTANCE READOUT
2. COURSE ARROW
3. HEADING BUG
4. LUBBER LINE
5. HEADING FLAG
6. LATERAL DEVIATION BAR
7. DIGITAL COURSE
8. BEARING POINTER NO. 1
9. NAV FLAG
10. LONGITUDINAL DEVIATION BAR
11. HELICOPTER SYMBOL
12. BEARING PONTER NO. 2

Figure 10-4. Horizontal Situation Indicator (HSI)

10-4.3 Map Displays

Because controlling the helicopter and maintaining geographical orientation are both very demanding tasks in NOE flight, a map display may be considered to reduce crew workload. The capability to display present position continuously is particularly valuble when terrain obstacles require frequent heading changes. Map displays are generally either of the projected film or paper roller type. The primary advantages of each of these types are:

1. Projected Map Display Advantages:
   a. Simpler map preparation for a specific mission
   b. Larger map storage capability
   c. Display can be oriented either track-up or north up
   d. Simple in flight scale change
2. Roller Map Display Advantages.
   a. Lower initial cost
   b. Easier preflight and inflight annotation

10-5 HELICOPTER SUBSYSTEM INSTRUMENTATION

10-5.1 GENERAL

Subsystem instrumentation provides cockpit references that describe the condition of engines, secondary power systems, and ancillary equipment. When pilot and copilot are seated side-by-side, the most common instrument panel arrangement groups the subsystem instruments in the center of the panel. When the pilots are seated in tandem, it is necessary to duplicate some of the subsystem instruments and to place them on each pilot's panel. Preferred locations are given in MIL-STD-250.

10-5.2 INSTRUMENTATION REQUIRED

The number and complexity of the instruments are limited to the minimum required for safe and efficient operation of the individual subsystems.

The amount of instrumentation required depends upon the size of the helicopter and the complexity of its subsystems. A single-engine, light helicopter obviously requires less instrumentation than a multiengine, transport helicopter. Cost, weight, and panel space savings frequently can be realized by having several similar parameters share the same display. Since many subsystem parameters are of concern only in event of a malfunction, the appropriate parameter can be selected manually or automatically for display when a caution light illuminates. Rapid advances in electronics may result in the cost-effective replacement of many individual instruments with a CRT display, symbol generator, and digital processor. This approach becomes particularly attractive in the event sufficient onboard computer capacity exists. Unless otherwise specified in the detail specification, the following subsystem parameters *shall* be displayed:

1. Gas generator rotor speed, calibrated in percent, for each engine, and for the APU (if applicable)
2. Turbine gas temperature, calibrated in °C, for each engine, and for the APU (if applicable)
3. Output shaft speed, calibrated in percent, for each engine
4. Output shaft torque, calibrated in percent, for each engine
5. Total torque, calibrated in percent for all engines, if multiengine
6. Rotor speed, calibrated in percent, for the main rotor
7. Oil temperature, calibrated in °C, for each engine
8. Oil pressure calibrated in pounds per square inch (gage), for each engine
9. Fuel quantity, calibrated in pounds, for each fuel tank
10. Total fuel quantity, calibrated in pounds, for all fuel tanks
11. Oil pressure, calibrated in pounds per square inch (gage), for each pressure lubricated gearbox
12. Oil temperature, calibrated in °C, for each drive subsystem gearbox.

Electrical, hydraulic, and pneumatic subsystem instrumentation should be based on subsystem capacity, redundancy, and failure modes. When redundant hydraulic systems or redundant generators — either of which is capable of carrying the entire electrical load — are provided, caution lights indicating generator failure or loss of hydraulic pressure may prove sufficient.

10-6 WEAPON SYSTEM INSTRUMENTATION

10-6.1 GENERAL

This paragraph describes the required design standards for controls and instruments for the helicopter armament subsystem. Contrary to flight and navigation displays, which frequently can be selected off-the-shelf with little or no modification, weapon system instrumentation is generally unique to the weapons mix on a specific airframe. Careful attention to mission requirements and established principles of human engineering are required to achieve an optimized man/machine weapon system. The armament controls and instruments should provide the operator with rapid armament subsystem status indication, and with rapid control of the particular systems and selection of various available options. Shape coded controls should be considered to allow the operator to select the various sight modes, and types and quantities of ordnance with a minimum diversion from his search or tracking tasks. The design *shall* provide the operator with up-to-the-minute store inventory information in order to insure the intelligent choice of ordnance for firing. Design consideration also shall be given to preclude inadvertent activation of the weapon systems.

10-6.2 DESIGN REQUIREMENTS

For all weapon control systems, the following design requirements *shall* be applied:

1. Multiple, and preferably sequential, actions are required from the initial control operation to the normal firing or release of the ordnance. All weapon controls and circuits *shall* be fail-safe so as to prevent firing or release of ordnance in the event of improper control operations or sequences. Where practicable, the design should make it either mechanically or electrically impossible to actuate control circuits in an improper sequence. The operator *shall* be provided with feedback to indicate improper sequencing. If deemed necessary, it is permissible to have controls serving dual functions; however, in no case should safety be compromised. Controls used for prearm or release *shall* be made unique to that operation.

2. Where practicable, all armament controls and indicators with the exception of the firing switch *shall* be grouped together. The group should be outlined by a 3/16-in red border (Color 3116, FED-STD-595). Orange-yellow (Color 23538, FED-STD-595) and black (Color 27038, FED-STD-595) striped borders *shall* be used to outline armament groups when red compartment lighting is used. Placard abbreviations *shall* be made for each indicator and control in accordance with MIL-STD-783 and ANA BUL 261. Preferred locations are shown in MIL-STD-250.

3. A complete failure mode and effect analysis (FMEA) during the design stage is desirable in order to preclude the inadvertent design of unsafe failure modes into the system.

10-6.2.1 Arming, Fuzing, and Suspension and Release Control Design

Requirements for the helicopter arming, fuzing, and suspension and release control systems include:

1. The helicopter commander *shall* be provided with the capability to permit and/or to prohibit prearming and arming of the weapons.

2. Two activations are required, preferably by two separate controls which should be separated so that they cannot be actuated by one movement. For example, most turrets require that both an "Action" or "Deadman" switch and a trigger be depressed before the weapon will fire. Lever-lock and/or hooded configurations *shall* be used for all master arming switches. These controls *shall* be designed so that inadvertent activation is prevented. A will-to-use control capability should be provided. Both the design and the crew responsibility for this control should be developed based upon human factor studies of the particular system.

3. Reversible, inflight capability of arming and returning to an unarmed, or safe condition *shall* be provided for weapons and/or suspension and release mechanisms. The ARMED state *shall* be designed to return automatically to SAFE, and the SAFE state should remain unchanged in the event of an aircraft power failure.

4. The wiring *shall* be designed to preclude any interaction between power and critical armament circuits. Control power *shall* not be applied to the weapon unless it is turned on intentionally by the operator of the system.

5. Jettisoning of ordnance may be effected individually or in multiples, provided that the warheads are in an unarmed state. Depending upon the armament system design, missiles may be jettisoned either by free fall or by being fired from their launchers.

10-6.2.2 Human Factors Considerations

In all close-proximity control groups, the individual controls *shall* be arranged so that those operated in sequence are in line and in their normal order of operation progressing from left-to-right or from top-to-bottom. When two or more switches must be activated simultaneously or in a rapid sequence they should be located so that they can be reached simultaneously from a fixed position. Each control should be placed so that it does not hinder the operation of another control in the sequence, with adequate clearance for a 95th percentile gloved hand.

10-6.2.3 Indicator Design

The following are weapon system indicator design objectives:

1. Indicators *shall* have high reliability.

2. A minimum number of indicators *shall* be provided in the crew compartment to show the condition (armed or safe) of critical weapon components.

3. Indicator systems *shall* be current-limited so as

to preclude indicator current from activating any weapon or suspension and release component. If [illegible]-based continuity monitoring is required, the monitoring currents *shall* be limited to a value below that which will activate the most sensitive component. Indicator circuits that are integral to control circuits cannot meet this requirement, and thus, should be avoided.

4. Indicator systems *shall* be designed so that indicator power is not available to any part of the weapon system unless it is turned on intentionally by the operator.

5. Indicator tests *shall* be possible in flight, independent of the indicator-related components.

6. The operator *shall* be provided with visual indication of a "hot" trigger condition. This indication should be in the operator's direct line of sight. The most common indication is the use of an amber light, which alerts the operator to use caution when the weapons are armed.

7. The operator *shall* be provided with a visual or aural missile condition indication (missile launched or being launched) signal.

8. Immediate visual indication of hangfires or misfires *shall* be provided.

9. Identical visual indications are to be employed whether live or training missiles are aboard the helicopter. This will insure that the crew acts at all times as though live missiles were aboard.

10. Arming and fuzing indicators *shall* be fully automatic. Except for the power-on function, and the press-to-test feature of the monitor testing function, no manual operation should be required. For multiple carriage, each weapon should be monitored individually. This may be accomplished either selectively or continuously.

11. Weapon malfunction, if it occurs during the prearm cycle, *shall* be indicated.

10-6.3 WEAPON SELECTION CONTROLLER/PROGRAMMER

A weapon selection controller and programmer *shall* be required for helicopters carrying a variety of ordnance. This unit also is required for configurations incorporating selective or automatic interval sequence launching of missiles and rockets from alternate sides of the helicopter.

The controller provides the operator with a choice of type and quantity of ordnance. If necessary, the controller also can provide controls for missile guidance operations. The programmer portion of this unit stores basic information about store availability and ammunition depletion status, in addition to performing weapon selection sequencing at the command of the controller.

The controller/programmer unit(s) must be mounted in a readily accessible area, but not necessarily in the control panel area. However in the case of a guidance control system for missiles, location in the panel area may be required. The controller/programmer function may be divided between any reasonable number of subunits. The subunits may be mounted in functionally convenient locations, e.g., one per store location; or the entire function may be handled at one or two units mounted in a central area. In all cases, the programmer portion(s) must be easily accessible during ordnance loading. The programmer portion(s) of the unit(s) *shall* provide some means of programming into the unit the type of ordnance loaded into the various stores locations. This input may be provided by switches, keypunch, patchboard, or any other suitable means. Based upon the information provided by the programmer unit(s), the controller portion(s) *shall* provide the function or ordnance selection, firing, and guidance (when required), and the rate of ordnance delivery as selected by the operator.

In order to simplify wiring requirements, portions of the controller which need not be accessible during normal usage of the system may be located wherever convenient on the aircraft. The controller *shall* provide an effective ground of all electrically fired ordnance. This ground *shall* be lifted only during firing. Parts of the controller/programmer that must be accessible on a regular basis — e.g., rocket on stub wing of the helicopter — may be mounted inside the leading edge of the stub wing. This would require box sizes limited to approximately 3 × 4 × 10 in. at each location.

10-7 TYPES OF INSTRUMENTS

The type of instrument to be used depends upon the condition that the instrument is recording and the required ease of interpretation on the part of the crewmember monitoring it. The designer *shall* consider: the best type of display for the information to be provided (qualitative, quantitative rate, trend, etc.); proper scale design to cover the required range, yet provide adequate discrimination in critical ranges; proper alphanumeric design to assure readability under low-level illumination and in a typical helicopter vibration environment and the reliability/maintainability features of potential designs. In addition to MIL-STD-1472 and AFSC #DH 1-3, Refs. 1, 2, and 3 should be reviewed for additional information.

The two types of instrument actuation are direct and remote indicating.

10-7.1 INSTALLATION

MIL-I-5997 covers the general requirements for the installation of aircraft instruments and instrument panels. However the vibration mounting specifications are inappropriate for most helicopter installations and generally are waived for the rigid mounting described in par. 10-7.2.

10-7.2 VIBRATION

The installation of instruments requires that special attention be given to the vibration of the complete instrument panel. Because each instrument has its own set of vibration requirements as set forth in its specification, the designer must review the applicable procurement specification for each instrument. He must determine the maximum frequency and double amplitude permissible, and must select a set of instruments that is compatible in this regard.

Ideally, the method of establishing the vibration and test criteria for a new helicopter should be based upon a thorough vibration analysis that defines the anticipated vibration conditions. Reference to MIL-STD-810 will assist in this analysis.

In general, and specifically for new designs, the normal procedure is to make the instrument panel as rigid as possible in order to avoid any resonances that may be excited by rotor fundamental frequencies. (This conflicts with the vibration requirements of MIL-I-5997, which usually is waived.) Should the vibration characteristics of an individual instrument be incompatible with the environment provided by the rigid panel, vibration isolators may be used in the mounting of the critical instruments.

10-7.3 ACCESSIBILITY AND MAINTENANCE

All panel-mounted instruments ***shall*** be mounted with the case lugs or mounting ring against the front of the panel so that the instrument may be installed and removed from the front. Each instrument ***shall*** be installed with enough electrical wiring or conducting tubing to permit the instrument to be pulled out of the panel to expose its connections. Where necessary, suitable means should be provided to prevent fouling or objectional interference of slack wiring or tubing when the instrument is installed in the panel. Each connection should be well identified so as to preclude its being hooked up to another instrument inadvertently.

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3. Semple, Heapy, Conway, and Burnette, *Analysis of Human Factors Data for Electronic Flight Display Systems*, Technical Report AFFDL-TR740-174, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, OH 45433, April 1971.
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CHAPTER 11
AIRFRAME STRUCTURAL DESIGN

11-0 LIST OF SYMBOLS

a_L = limit cargo acceleration, number of g's, dimensionless

g = acceleration due to gravity, 32.2 ft/sec^2

H = peak acceleration, number of g's, dimensionless

K = acceleration ratio, a_L/H, dimensionless

n_x = longitudinal acceleration, number of g's, dimensionless

$n_{z_{lim}}$ = limit flight load factor, dimensionless

t_L = time for acceleration to reach limit cargo acceleration, sec

t_0 = time to reach peak acceleration, sec

V_L = velocity at time t_L, ft/sec

V_0 = initial velocity, ft/sec

w = width of the cargo floor, ft

X = controlled deflection, in.

11-1 INTRODUCTION

Structural considerations for helicopter major component design were discussed in Chapter 1. This chapter is concerned with the detail structural design of the airframe only; i.e., fuselage, tail boom, stabilizers, fins, and auxiliary lifting wings. Secondary structure, such as doors, cowlings, and fairings, is included. Detail design of transparent areas also is discussed.

The basic helicopter configuration is chosen and the external loads are developed during the preliminary design of the helicopter. Fundamental airframe decisions, such as whether to use monocoque or semimonocoque construction, also are made during preliminary design. The task in detail design is to confirm that the airframe structure designed in preliminary design meets the mission performance and survivability requirements, and that it can be developed and produced within the budget cost established for it.

The bases for detail design are the helicopter detail specifications, MIL-S-8698 and MIL-A-8860 through -8871, and the design criteria defined by or developed from these documents.

The detail design involves selection between alternative types of local structure by application of certain trade-off criteria. Material properties, method of fabrication, weight, and strength limitations are all important considerations in the design confirmation. The determination of how a particular portion of the airframe is to be fabricated — i.e., by welding, forging, steel metal buildup (riveted or bonded), or from composite materials — is the principal decision to be made in detail design. The trade-off criteria to be used during the investigation of fabrication techniques include weight, surface finish, stiffness and ruggedness, fatigue sensitivity, cost, and properties of materials.

11-2 DESIGN CONSIDERATIONS

11-2.1 WEIGHT

At the beginning of the detail design effort, the weight group provides the weight budget to the design group. The weight budget is based on statistical analyses and estimates of the preliminary design. A state-of-the-art design, therefore, normally would meet the weight allowance, while advanced design techniques and new materials should produce a structure weighing less than the allotment. Incorporation of new design techniques in order to save weight must be considered in conjunction with the other requirements since, for instance, advanced designs may increase the airframe cost. In any case, a strict accounting of weight with respect to the budget must be maintained throughout detail design.

11-2.2 SURFACE SMOOTHNESS

Surface smoothness is another structural quality that affects aircraft performance. Criteria that limit the use of protruding fasteners, stipulate faired areas, and dictate external contours are provided by the aerodynamic group. The extent to which these criteria are applied depends upon the impact upon manufacturing cost. It may be possible to trade off the additional costs of meeting these criteria through simplification (therefore, lower costs) of the structure itself.

A curved surface often will be more costly, but it will be stiffer, more rugged, lighter, and less fatigue-sensitive than the simpler alternative. Limiting flush fasteners to the forward 25% of the airframe aerodynamic length may be an acceptable compromise that will yield the required drag reduction.

11-2.3 STIFFNESS AND RUGGEDNESS

Airframe components must have adequate stiffness to meet stated vibration criteria. There must be adequate separation between the natural frequencies of prime modes and the exciting frequencies. This frequency separation will reduce the internal stresses caused by amplification (a phenomenon experienced when operating near resonance). Vibration levels at the pilot and crew positions often are excessive (uncomfortable and distracting) due to poor airframe design; rotor vibration has been amplified by structural elements whose natural frequency is too close to the operating frequencies or to multiples thereof.

Vibration of structural components can cause audible noise or pilot fatigue. In addition to crew discomfort, vibration can cause structural fatigue and possible catastrophic failure.

Ruggedness is a quality that prevents denting or puncturing of structure by rotor-induced debris, ground handling, erosion, or brush. It is difficult to prescribe physical characteristics that would prevent these types of damage. Flight testing of a prototype should include simulated operational conditions so that ruggedness can be checked. These tests will show those areas that must be strengthened, resulting in the minimum weight increment to obtain the required capability.

11-2.4 FATIGUE SENSITIVITY

Statically strong, lightweight designs may be fatigue-sensitive. Attention must be paid to the details of the airframe structure to prevent fatigue sensitivity. For example, designers would like to eliminate the clips attaching the stringers to frames in the fuselage, thus saving both weight and cost. However, experience has shown that skin and bulkhead flange cracks occur when the clips are omitted.

A highly loaded airframe fitting may have sufficient static strength because a high heat treatment is provided, but, as a result, the material may be notch-sensitive and prone to fatigue failure.

Grain orientation in highly loaded fittings is important. The most efficient structural design is obtained by orienting the longitudinal grain in the direction of the primary load. A fitting may be fatigue-sensitive if the transverse grain is oriented in the primary load direction.

11-2.5 COST

In the design of the airframe, three cost areas should be considered: cost to develop, cost to manufacture, and cost to operate.

The cost of development includes that of design, development of new methods, and testing required to prove the design. Development testing is a very important design tool. A number of design alternatives can be tested under identical conditions to find the best design. The costs involved may be high, particularly where fatigue testing is involved.

Manufacturing costs must be estimated during the design stage. Materials, tooling, labor, quality control, and facility costs are involved. Make-or-buy decisions may influence detail design of components. For example, a fairing might be manufactured in house if molded from Fiberglas; but if pressed from aluminum, it might have to be developed outside. In such an instance, weight, stiffness, etc., also must be considered when the method of manufacture is being selected. Table 11-1 indicates the cost impact of various detail design alternatives.

Each manufacturer can prepare such a table during preliminary design and update it during detail design using specific cost data. The table is a particularly useful design tool if it is stated on a cost-per-airframe-pound basis.

Operating costs for airframe structures consist of repair, maintenance, and replacement parts costs. The design objective should be a maintenance-free life equal to the anticipated service life of the helicopter.

11-2.6 MATERIALS

In preliminary design, material selection has been completed. A most important detail design consideration is material verification. A wide variety of ferrous metals, nonferrous metals, and nonmetallic materials is available. Characteristics of these metals and materials are described in Chapter 2. Additionally, Table 11-2 summarizes materials, characteristics, and their uses in helicopter airframe construction.

11-2.7 SURVIVABILITY

The survivability characteristics of an Army helicopter design include:

1. Detectability
2. Vulnerability to enemy ballistic threats
3. Crashworthiness.

Detection methods to be considered are radar, infrared (IR) radiation, acoustics, and visual. In the design of the airframe structure, the greatest contribution to reducing the detectability can be made by reducing the radar cross-section of the helicopter fuselage. The techniques available include the use of radar-transparent materials and/or nonreflective coatings. Reflectivity characteristics of the key aspects of the helicopter also may be reduced, at least for selected radar frequencies, by careful attention to the shape of the target presented to the transmitted beam.

The specific ballistic threats to which the helicopter will be exposed and the desired levels of protection will be stated by the helicopter system specification. The use of armor materials to defeat ballistic threats

TABLE 11-1. COST IMPACT, AIRFRAME DETAIL DESIGN

COST AREA	WELDED	RIVETED	CAST	FORGED	BONDED
DEVELOPMENT					
DESIGN	LOW	MEDIUM	HIGH	HIGH	MEDIUM
METHODS	NONE	NONE	LOW	MEDIUM	MEDIUM
TESTING	NONE	LOW	HIGH	MEDIUM	MEDIUM
MANUFACTURING					
MATERIAL	LOW	LOW	LOW	LOW	MEDIUM*
TOOLING	MEDIUM	MEDIUM	MEDIUM	HIGH	HIGH
LABOR	HIGH	MEDIUM	**	**	MEDIUM
QUALITY CONTROL	HIGH	LOW	HIGH	MEDIUM	HIGH
FACILITIES	LOW	LOW	MEDIUM	HIGH	HIGH
PROOF OF ADEQUACY					
ANALYSIS	LOW	MEDIUM	LOW	LOW	MEDIUM
GROUND TEST	NONE	HIGH	HIGH	LOW	MEDIUM

*IF BORON FILAMENTS OR THE LIKE ARE USED, THIS COULD BE HIGH.

**DEPENDS ON QUANTITY--HIGH PRODUCTION RATE RESULTS IN LOWER COST.

TABLE 11-2. MATERIAL SELECTION — AIRFRAME DESIGN

	WEIGHT, lb/in³	STIFFNESS psi	TENSILE STRENGTH psi	COST, $/lb	AIRFRAME USE
FERROUS METALS:					
CARBON STEELS				LOW	
ALLOY STEELS	APPROXIMATELY 0.3	30 x 10^6	180,000	MEDIUM	FITTINGS
STAINLESS STEEL				MEDIUM	FIREWALLS
PH STEELS			300,000	HIGH	
MARAGING STEELS				HIGH	
NON-FERROUS METALS:					FITTINGS SHEET ELEMENTS
ALUMINUM	0.1	10 x 10^6	75,000	LOW	SANDWICH FACINGS
MAGNESIUM	0.06	6 x 10^6	30,000	MEDIUM	CASTINGS
TITANIUM	0.15	15 x 10^6		HIGH	FORGINGS
NON-METALLIC:					
THERMOPLASTICS		5000		LOW	TRIM
THERMOSETTING	0.03			LOW	FAIRINGS
ELASTOMERIC		LOW		LOW	WEATHER STRIP
GLASS		5000		HIGH*	GLAZING
LAMINATES:					WORK PLATFORMS
FIBERGLAS	0.06				DOORS
GRAPHITE				60-400	FAIRINGS
BORON					COMPLETE STRUCTURES

*WHERE OPTICAL QUALITY IS HIGH

is discussed in Chapter 14. In the design of airframe structure, care must be taken to minimize the possibility that a single hit by the stated threat — including explosive and/or incendiary projectiles when so specified — will cause a crash, crash fire, in-flight fire, or comparably catastrophic result. In the design of structure surrounding fuel tanks, it is essential that redundant loads paths be provided. The principal load-carrying members may be damaged by hydraulic ram effects following projectile impact, particularly if the hit is by a high explosive or armor-piercing incendiary (HEI or API) projectile. Ref. 1 provides additional design guidance for the reduction of vulnerability to ballistic threats.

Regardless of the intensity of combat or the severity of the ballistic threat(s) to which a helicopter will be exposed, the design must be crashworthy. Crashworthiness design criteria for Army aircraft are given by MIL-STD-1290. Additional guidance and specific design techniques for meeting the stated criteria are provided by Ref. 2. Some of the crashworthiness considerations applicable, particularly to the airframe structure, include:

1. The incorporation of crushable structure outside the occupied zones to assist in the absorption of impact energy while maintaining a protective shell
2. The incorporation of turnover structure adequate to maintain the integrity of the protective shell following impact with the ground in either a rolled (90 deg) or inverted (180 deg) attitude
3. The provision of support for the main transmission and rotor mast so that the transmission is not displaced into the the protective shell following either specified crash conditions or the strike of a rigid object by the main rotor
4. The incorporation of seats and litters, when appropriate, with restraints adequate to retain the crew and other occupants within the protective shell following specified crash conditions
5. The provision of emergency exits of sufficient number and size to permit the evacuation of all occupants in the minimal time available regardless of the postcrash position of the helicopter. Crashworthiness considerations pertinent to the cargo compartment are discussed in par. 11-4.2.

11-3 DESIGN AND CONSTRUCTION

During detail design, overall layouts of the airframe are prepared from the preliminary design data available. Final locations of the major subsystems to be supported by the airframe are shown, together with directly associated assemblies such as fins, stabilizers, and stub wings. Major subsystems are the power plant, transmissions and drives, rotors and propellers, flight controls, electrical system, avionics, hydraulics and pneumatics, instrumentation, landing gear, crew stations (furnishings and equipment), armament, armor, and protective devices. Weight and balance — as well as operational and functional factors — are to be considered while locating the subsystems. The mounting points of each subsytem are arranged so that the overall load distribution system of the structure can accept the local load feed-in most efficiently. Shelves, beams, frames, and bulkheads are types of local structures used to take the mounting hard-point or fitting loads and to distribute these loads into the basic structure.

Methods of locating and attaching fittings, supports, frames, bulkheads, skin and longeron systems, corrosion protection, and electrical bonding are major design requirements to be considered while accommodating the subsystems in the airframe and applying the trade-off criteria discussed in par. 11-1. Fig. 11-1 illustrates some of these structural components.

During airframe detail design, all areas subject to repeated high loads should be analyzed for the incorporation of fail-safe features. This design philosophy — to prevent catastrophic failure — requires provision of redundant load paths so that, if one path fails due to fatigue, the load is carried by the remaining structure until the next inspection uncovers the failure.

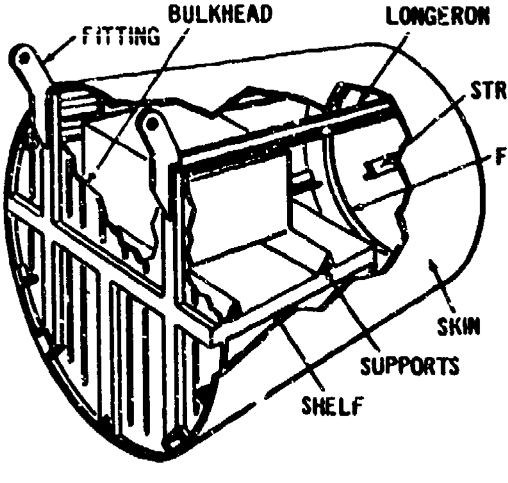


Figure 11-1. Airframe Components

11-3.1 FITTINGS

Fittings provide the structural transition between two different types of structure, and also serve as a convenient point for disassembly. They may be designed as welded, cast, machined, or forged substructures. Appropriate safety factors must be used when calculating fitting strength. For example, castings require analytical safety factors ranging from 1.33 to 2.0, depending upon the quality control standards to be applied. Safety factors are governed by the casting classifications in MIL-C-6021. The decision as to type of fitting depends upon weight, production quantity (or cost), and the nature of the loads to be transmitted. Castings generally have low elongation characteristics, and, therefore, have poor fatigue characteristics.

Meticulous care should be given to keeping the local eccentricities in the fittings to a minimum. All changes in cross section should be made as gradual as possible by using generous fillet radii. Abrupt changes in cross section cause stress concentrations, and, therefore, must be avoided.

Lug analysis requires special consideration. Refs. 3 and 4 contain discussions of this subject.

The use of a variety of fasteners, such as both rivets and bolts, for a single fitting attachment should be avoided. Rivets fill the holes and, therefore, pick up load before bolts do. It is important to preload bolts so that clamping of the facing surfaces is accomplished and bending of the components is minimized.

11-3.2 SUPPORTS

Shelves, beams, and brackets that support equipment and subsystem parts generally are constructed by assembly of sheet-metal components. The support must be strong enough to take the design load -- the weight of the item supported times the design load factor — plus the applicable mechanical reaction forces and the vibratory loads. The loads being distributed into the primary structure from the support should not induce secondary stresses, which can cause the primary structure to fail. Secondary loads are caused by the deflection of the support.

Beams having an open section should be investigated for shear center location. When a beam is loaded off its shear center, twisting will occur, as indicated in Fig. 11-2(A). Secondary loading and lower subsystem natural frequency generally result.

Brackets should be designed so that the load feed-in or feed-out is parallel to a sheet-metal face. If the load is perpendicular, local bending will occur and subsequent cracking will result. Radii should not be counted upon to carry load. Softness in the support will be evident, and, again, cracking will occur after a period of service. Figs. 11-2(B) and (C) show correct and incorrect design concepts for bracket attachments.

11-3.3 FRAMES

Frames are used to reduce skin panel size and stringer column length, to maintain aircraft contour, to transfer various local loads to the outer skin as shear loads. Fig. 11-3 shows an example of a frame reacting shear flow from a tank liner. The weight of the tank and the fuel result in a bending load on the liner, and the resultant shear flow in the liner is opposite in direction to that shown acting on the frame.

The frame flanges that support the skin must be sufficiently thick to prevent tension field wrinkles from propagating when the panels are not shear-resistant. The moments of inertia of the cross sections of the frames must be adequate to restrain the stringers against column failure. Junctions between frames, skin, and stringers must be clipped.

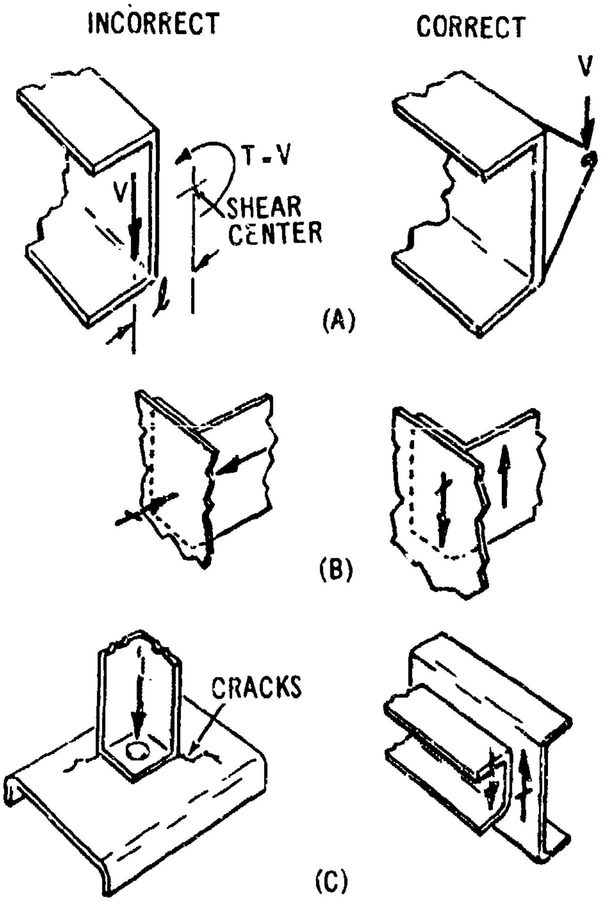


Figure 11-2. Correct and Incorrect Load Distribution

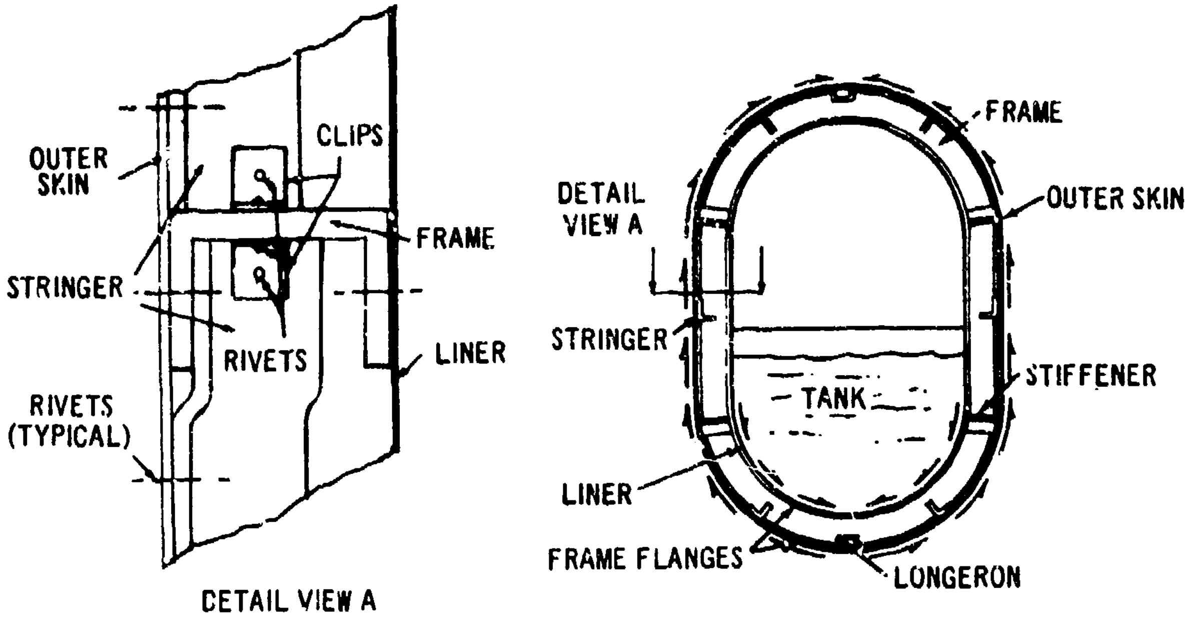


Figure 11-3. Frame Used for Tank Support

11-3.4 BULKHEADS

Bulkheads function much the same as frames, but have added capability and utility. Generally, major loads — such as landing loads — are introduced at bulkheads and are redistributed within their own plane. Out-of-plane restraints, such as fore or aft fuel tank loads, also may be provided.

Stiffeners in the plane of the bulkhead distribute fitting loads into the web. The stiffeners must have sufficient inertia to stiffen the web against catastrophic buckling. The webs of all bulkheads should be designed to prevent oil-canning. Bulkhead flanges must be stiff enough to resist column buckling, if so loaded, as well as to provide skin restraint. Loads from fittings should be sheared into the bulkhead with as little eccentricity as possible.

Edge distances of fasteners should not be less than two times the fastener diameter wherever possible. Low-strength blind fasteners should be avoided. Cutouts should be reinforced, while attachments to an unsupported web should not be made because vibration may cause cracking.

The strength required for major bulkheads usually is attained by employing forgings. Final dimensions are obtained by machining.

11-3.5 SKIN SUBSYSTEMS

Skin subsystems are defined as structural coverings composed of longerons, stringers, and skin. If sandwich construction is used in a monocoque configuration, some of these elements are not used.

Longerons are the primary axial-load-carrying members, and are situated at optimal distances from the bending neutral axes of the airframe. They are designed to withstand column loads and the secondary effects of skin tension field. It usually is necessary to locate intermediate stringers between longerons to control skin panel size.

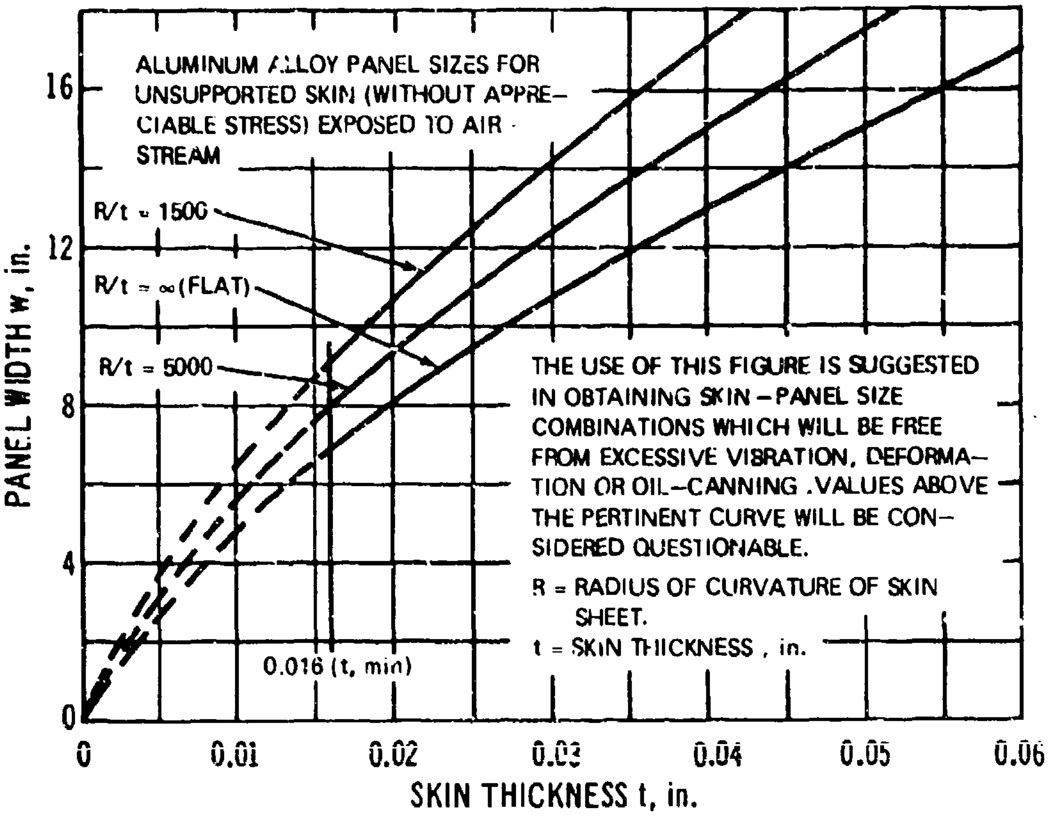
Bending material (stringers) should be distributed among a large number of elements in order to limit the reduction of static strength following damage to one member. Again, oil-canning should be prevented by proper panel sizing. Fig. 11-4 presents suggested panel dimensions.

Stringer and longeron splices should be scarfed and multiple attachments provided. The fasteners should be loaded in shear. Fastener types should not be mixed in a splice.

Doublers should be provided around excess holes through the skin. Because repeated removal enlarges the fastener holes, access hole covers should not be designed to be load-carrying. Minimizing the number of cover fasteners makes servicing faster and, hence, more economical.

11-3.6 CORROSION PROTECTION

Protection of parts against corrosion can be accomplished in two ways. One method is to apply coatings to prevent corrosive atmospheres from coming in contact with airframe parts. The other is to in-



THE USE OF THIS FIGURE IS SUGGESTED IN OBTAINING SKIN – PANEL SIZE COMBINATIONS WHICH WILL BE FREE FROM EXCESSIVE VIBRATION, DEFORMATION OR OIL–CANNING. VALUES ABOVE THE PERTINENT CURVE WILL BE CONSIDERED QUESTIONABLE.

R = RADIUS OF CURVATURE OF SKIN SHEET.

t = SKIN THICKNESS , in.

WHEN PANEL LENGTH EXCEEDS 3.0 x WIDTH, DIVIDE LENGTH BY 3 TO OBTAIN WIDTH FOR USE IN DETERMINING THICKNESS FROM CHART.

Figure 11-4. Aluminum Alloy Panel Sizes

duce an oxide coating that prevents further oxidation.

Many coatings can be applied, ranging from zinc chromate primer for internal surfaces to epoxy paints. Par. 2-6 provides detailed information about paints and finishes, as well as about special processes, such as anodizing for inducing oxidation on aluminum parts.

Treatments such as anodizing on aluminum, and cadmium- and nickel-plating on steel, have deleterious effects upon the fatigue strength of the metal. Parts that have been so treated and are subject to alternating loadings should be fatigue-tested.

11-3.7 ELECTRICAL BONDING

In order to provide a continuous ground throughout the airframe structure so that remote electrical components may be grounded adequately, nonconducting structures must have a conductance bond built in.

Par. 7-6 includes detailed bonding requirements necessary to produce electrical and electronic system installations having acceptably low levels of electromagnetic interference (EMI).

11-4 CARGO COMPARTMENT

Cargo compartments pose special design requirements in that cargo floor and tiedown fittings must be designed to accept static and crash loads.

11-4.1 STATIC LOADS

Static limit flight loads *shall* be used as the basis for design of all structure and fittings within the cargo compartment. The primary areas of concern are the cargo floor and the tiedown fittings.

For helicopters having cargo transport as a primary mission, the basic cargo floor should be designed for a limit-flight-load floor pressure of $300n_{z_{lim}}$ psf , where $n_{z_{lim}}$ is the maximum flight limit load factor. For light utility helicopters, which are not primarily cargo transports, the limit-flight-load floor pressure can be reduced to $100n_z$ psf over that portion of the floor that can be used for cargo. Baggage compartment floors should be designed for a limit-load pressure of $100n_z$ psf.

Cargo transport helicopters having the capacity for vehicular loads must be designed for the concentrated loads applied by the vehicle wheels. These wheel loads should be assumed to be acting at any point within the treadway areas shown in Fig. 11-5. The dimensions shown on the drawing describe the suggested minimum treadway area.

The portions of the treadway that are used only for ground operations (loading and unloading) should be designed using a limit-load factor of 1.0 applied to the maximum wheel load. The areas of the treadway upon which the vehicle rests in flight must be designed to carry the maximum wheel load multiplied by the maximum flight limit load factor n_z.

Maximum wheel load should be equal to 1/3 of the maximum anticipated vehicle weight. This assumes a laterally symmetrical vehicle with 2/3 of the weight carried by one axle. The maximum wheel load, multiplied by the appropriate limit-load factors, will be applied to the treadway areas through pneumatic tires inflated to a maximum pressure of 100 psi. This local pressure of 100 psi can be assumed to remain constant with the application of helicopter flight limit-load factors, because the action of the pneumatic tire under moderately increased load is simply to enlarge the wheel contact area in proportion to the load under constant pressure. Solid rubber tires or metal wheels do not offer this flexibility; accordingly, concentrated pressure loads applied to the floor by these wheels must be assumed to increase in proportion to the flight load factor. Performance of flooring under solid-wheel loading should be determined experimentally, with pertinent operational limitations specified. In most contemporary helicopters, the use of shoring planks is recommended to protect the floor from solid-wheel damage.

MIL-A-8865 specifies wear test requirements for newly designed flooring. Nontreadway areas also should have the capability of carrying limited concentrated loads in excess of $300n_{z_{lim}}$ psf. A load strength of $1000n_{z_{lim}}$ psf typically is used in the design of cargo aircraft. The extent of the $1000n_{z_{lim}}$-psf area of application is limited by weight and balance considerations and running-load limits, which interact.

The running-load limit should be established by the greater of the following: twice the maximum wheel load (i.e., the axle load), or $300n_z w$ lb/linear ft, where w is the width of the cargo floor in feet. The running-load limit is the maximum load that can be applied to any single foot of floor length.

Floor tiedown fittings should be available in sufficient quantity and capacity to restrain the maximum design cargo weight under the ultimate crash loads.

The cargo tiedown restraint factors must be de-

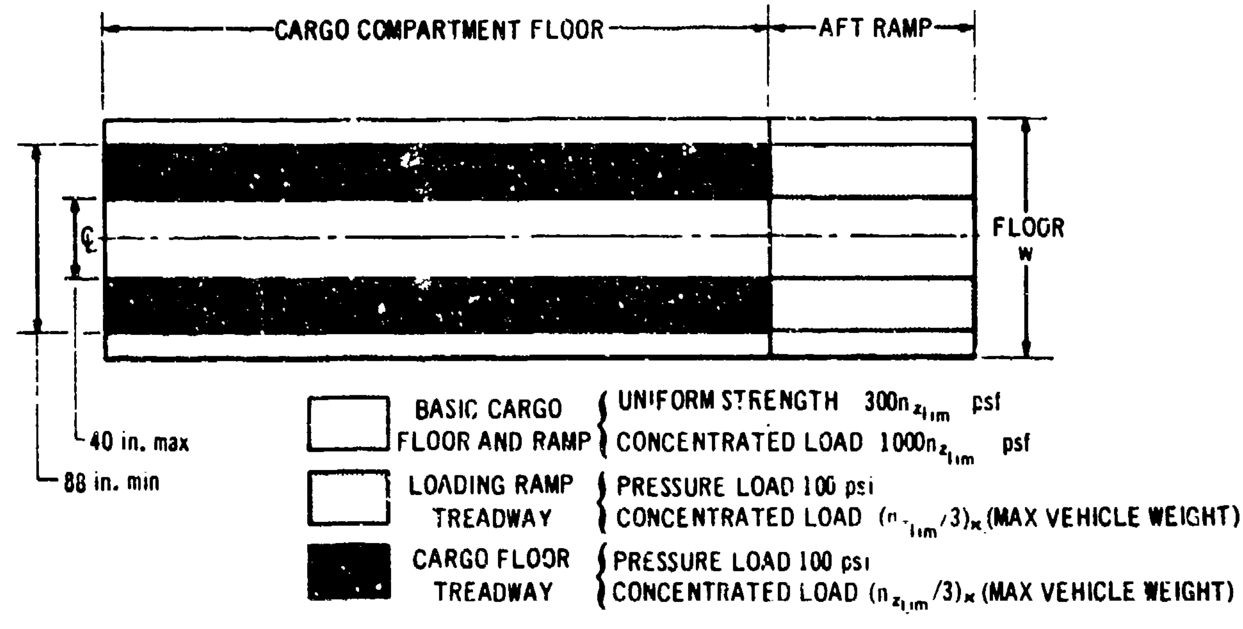

CARGO COMPARTMENT FLOOR
AFT RAMP
℄
FLOOR w
40 in. max
88 in. min
BASIC CARGO FLOOR AND RAMP
UNIFORM STRENGTH $300n_{z_{lim}}$ psf
CONCENTRATED LOAD $1000n_{z_{lim}}$ psf
LOADING RAMP TREADWAY
PRESSURE LOAD 100 psi
CONCENTRATED LOAD $(n_{z_{lim}}/3) \times$ (MAX VEHICLE WEIGHT)
CARGO FLOOR TREADWAY
PRESSURE LOAD 100 psi
CONCENTRATED LOAD $(n_{z_{lim}}/3) \times$ (MAX VEHICLE WEIGHT)

Figure 11-5. Minimum Floor Strength Requirements

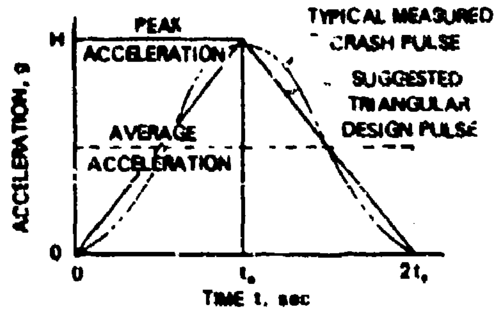
termined and specified in the helicopter cargo loading technical manual. Cargo tiedown restraint factors are discussed in MIL-A-8865.

The task of securing each specific piece of cargo to withstand the specified load factors is the responsibility of the user. The designer's responsibility is to provide tiedown fittings that will carry their rated load without failure. The installed fitting should swivel and rotate through a hemisphere bounded by the floor. The rated load should be considered as the ultimate load for the purpose of designing the fitting and its supporting structure. For large helicopters, it is recommended that the standard floor tiedown fitting be rated at 5000 lb ultimate strength. The row of fittings nearest the edge of the floor should be rated at 10,000 lb each. This facilitates the restraint of vehicular loads with a minimum of tiedowns.

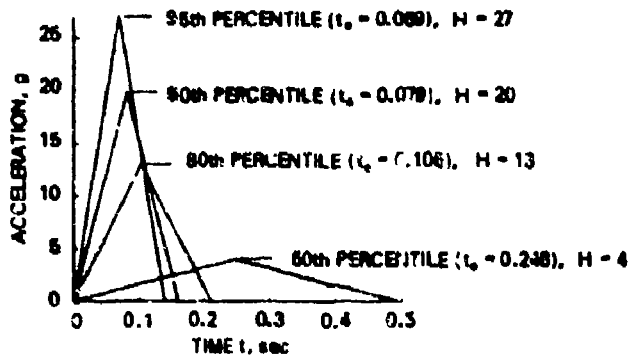
11-4.2 CRASH LOADS

It has been found that cargo compartment longitudinal accelerations can exceed 27 g during a survivable crash landing. Fig. 11-6(A) shows a typical acceleration-time history measured at floor level during the test crash of an instrumented helicopter (Ref. 2).

Considerable experimental work in this area has shown that the typical major impact accelerations can be represented for engineering purposes by a simple triangular pulse. Such curves have been developed for the 50th-, 80th-, 90th-, and 95th-percentile survivable helicopter accident, and are shown in Fig. 11-6(B). It can be seen that, in order to restrain cargo rigidly to the helicopter floor throughout a 95th-percentile helicopter crash, it would be necessary to employ seven times the number of tiedowns required for the mean (50th percentile) crash. This becomes prohibitive in terms of weight and complexity.



(A) TYPICAL MAJOR IMPACT ACCELERATION PULSE

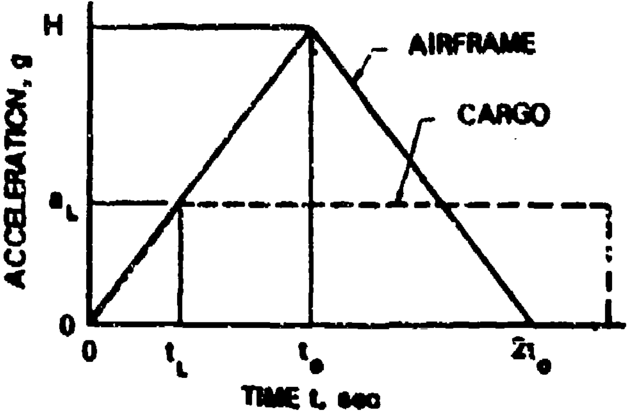


(B) DESIGN PULSES, CARGO COMPARTMENT FLOOR

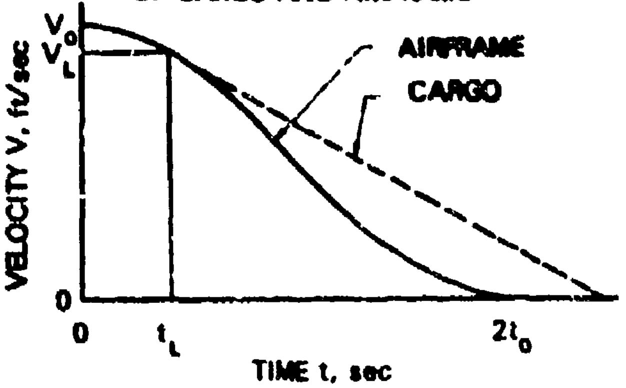
Figure 11-6. Major Impact Acceleration Pulse

Recent efforts in the development of load-limiting cargo tiedowns have shown these devices to be an effective solution to the crash-load restraint problem (Ref. 5). Load-limiting tiedowns are energy-absorbing devices that link cargo to the existing helicopter floor tiedowns. They are designed to resist an increasing tension load rigidly until, at a predetermined force, they yield and deflect over some distance while maintaining a constant resistance. When connected in series with the cargo tiedown chains, such devices act to attenuate the accelerations transmitted from the helicopter floor to the cargo, as shown in Fig. 11-7(A).

Fig. 11-7(B) shows the velocity-time curves of the cargo and the airframe as they act separately during



(A) ACCELERATION—TIME HISTORY OF CARGO AND AIRFRAME



(B) VELOCITY—TIME HISTORY OF CARGO AND AIRFRAME

Figure 11-7. Effect of Load-limiter

the crash event. These curves are derived by integration of the respective acceleration curves in Fig. 11-7(A). Because it is under a lesser acceleration, the cargo must move slightly farther than the fuselage during the period of the acceleration pulse. This additional distance is the distance through which the load-limiter must deflect, and is determined by integrating the two velocity-time curves (cargo and airframe) to find the distance each travels during the event, and then finding the difference between the distances. Eq. 11-1 can be used by the designer to determine load-limiter stroke required, assuming a triangular acceleration input pulse:

$$\text{stroke} = gHt_0^2 \left(-\frac{K^2}{24} + \frac{K}{2} + \frac{1}{2K} - 1 \right) \text{, ft} \quad (11\text{-}1)$$

where

H = pulse peak acceleration, number of g's, dimensionless

g = acceleration due to gravity, 32.2 ft/sec^2

K = acceleration ratio a_L/H, dimensionless

a_L = limit cargo acceleration, number of g's, dimensionless

t_0 = time to reach peak duration, sec

This equation is derived from Ref. 5.

The load-limiting cargo restraint concept has been developed to the prototype stage. Cost and weight permitting, it is recommended that consideration be given to incorporating such devices into the basic design of new helicopters. Such load-limiting restraint fittings would provide a maximum degree of crashworthiness. Details of load-limiter design and application are available in Ref. 6.

When specifying load-limiters, the designer must make a trade-off between rated strength and load-limiter stroke. Fig. 11-8 shows the various ways of achieving adequate restraint of Class A cargo in a 90th-percentile crash, using load-limiters having different energy absorption characteristics.

It can be seen by referring to the sample curves that an additional restriction on the performance of the load-limiter is that the load-deflection curve must avoid the shaded area below the base curve. This base curve indicates the load below which the load-limiter must act as a rigid link in the tiedown chain. This load should define the rated yield strength of the limiter. An optimum load-limiter, from the standpoint of weight and number required, would be one

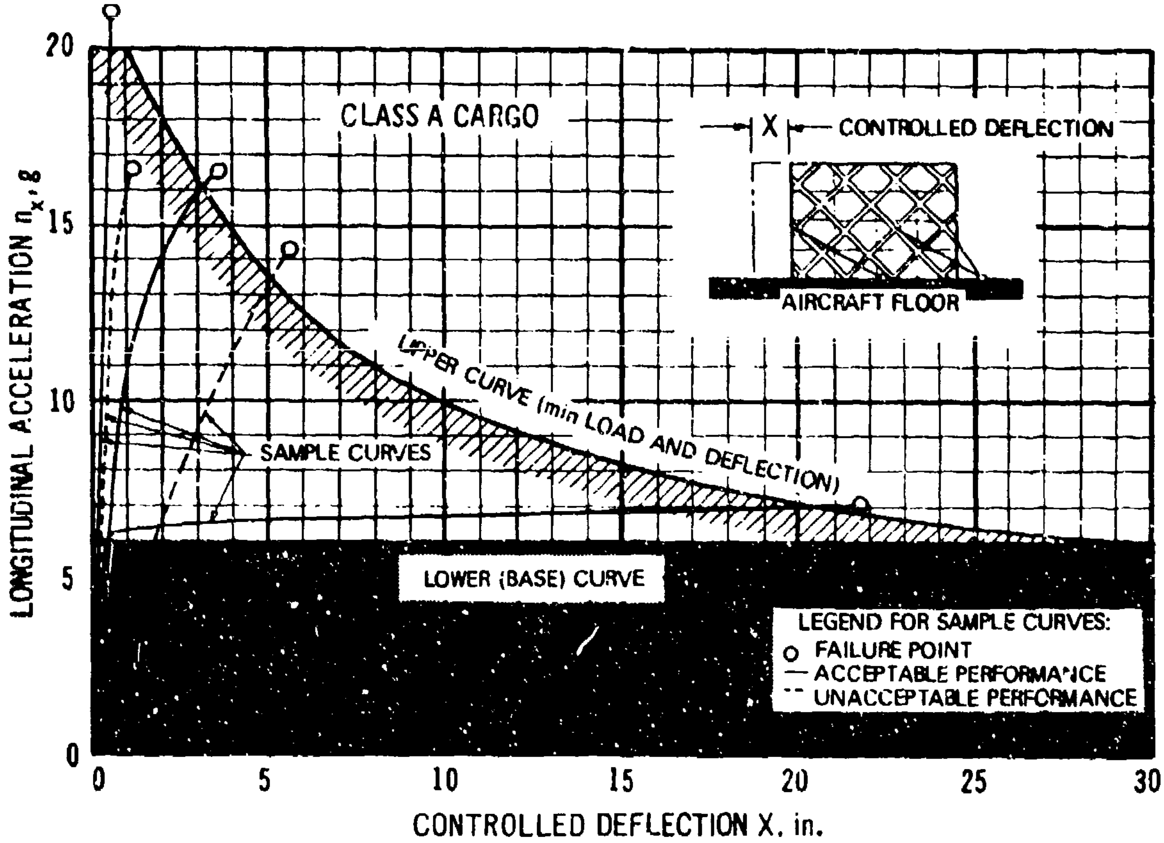


Figure 11-8. Class A Cargo Forward Load-deflection Envelope

with a load-deflection curve conforming as closely as possible to the base curve. The long stroke of such a limiter, however, could allow the cargo to move an excessive distance within the compartment.

It is recommended in Ref. 5 that Class B cargo (vehicles, etc.) be restrained to withstand the 80th-percentile crash pulse of 13 g, rather than the 90th-percentile pulse that defines the restraint criteria for Class A cargo. The rationale is that personnel rarely are carried in the cargo compartment with Class B cargo, and, therefore, that only the relatively remote flight crew need be protected from shifting Class B cargo. Fig. 11-9 shows the load-deflection requirements pertaining to load-limiters restraining Class B cargo.

11-5 TRANSPARENT AREAS

The design requirements *shall* be as established by the helicopter system specification. Two categories of transparent areas are apparent. One category is represented by a windshield that must withstand direct airstream loads, possibly including impact by birds. The other category is represented by an observation window located on the side, roof, or floor of the crew or passenger compartment. Such a window is subject to indirect pressure loading, positive or negative.

The characteristics of available glazing materials are reviewed in par. 2-3.5. Additional information on the properties of the materials are available in Part II, MIL-HDBK-17.

The optical quality of the windshield must be maintained under all conditions of loading. To keep optical distortion to a minimum, flat panels should be used to the maximum extent practicable. In any case, consideration must be given to the deflection of the windshield under load and to the effect of these deflections upon the optical characteristics. The loadings that result from thermal gradients across the windshield must be considered separately as well as in combination with pressure loadings in order to determine the critical loading condition. The effect of an antireflective coating used to reduce the glint from the windshield and canopy must be included in determining the thermal loading.

Installation of glazing materials is described in par. 2-3.5. Part II, MIL-HDBK-17 also contains a thorough discussion of the design of the "edge attachments", or means of fastening a glazing material to an airframe. With regard to the structure, it is recommended that openings for transparent areas be self-supporting, i.e., not dependent on any structural support from the window. This is particularly important for window openings within doors. The de-

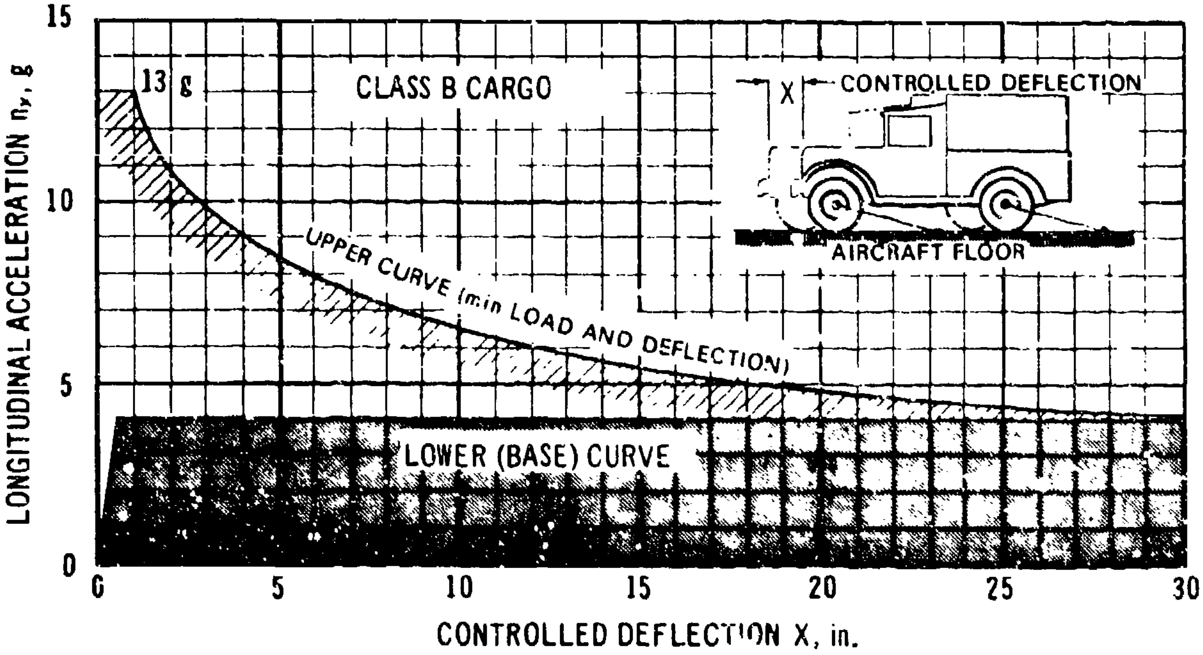


Figure 11-9. Class B Cargo Forward Load-deflection Envelope

flections of the opening should not be large enough to cause failure of the glaze-supporting system.

Thickness of the glaze — determined by analysis or test — should be adequate to support the pressure and impact loadings. In addition, the glaze material must satisfy any ballistic-resistance requirements contained in the system specification.

11-6 DEVELOPMENT

The iterative process of design, build, test, and redesign is called development. Several different versions of the same airframe component may be designed, built, and tested to determine trade-off parameters such as weight, strength, and cost. With these data available, a decision can be made as to which structure best fits the requirements.

Development, including testing of alternative manufacturing methods, is a costly process. It should be used only for redundant structures, where analysis is either more costly or is impossible to accomplish within the alloted schedule, or for new construction techniques in which experimental manufacture is required to prove feasibility and to determine costs.

For purposes of this discussion, development does not include the testing of parts to determine points of local failure and the changes necessary to obtain adequate strength. This testing, rework, and retesting is considered a part of the demonstration of structural adequacy and is discussed in par. 11-8.

11-7 MANUFACTURE

The designer has the responsibility to provide an airframe requiring as low a manufacturing cost as is possible. Generally, the most easily manufactured airframe also is the least costly one. A reasonable number of subassemblies should be planned to allow ease of manufacture.

Accessibility to all areas of the structure usually results in lower cost, because more workers can be applied during a given period of construction.

The structure should be designed to permit use of automated machines (e.g., riveters) and numerically controlled machine tools during manufacture in order to minimize production costs.

In Chapter 4, AMCP 706-201, the consideration for practical production tolerances is discussed. The preliminary design will have considered the tolerances that can be achieved during manufacture and will have been the basis for design selection. The example used in Chapter 4, AMCP 706-201, is the preliminary design of the doors and hatches.

During detail design the exact production tolerances required must be established. The tolerances chosen will be the result of many considerations. These include:

1. Number of units. The total number of helicopters to be produced influences the type and extent of the tooling that can be used. A large production run will permit intricate tooling with costs that can be spread over many units.

2. Fabrication method. The way the helicopter will be fabricated on the production line influences the tolerances. If the helicopter is fabricated from many components that have been subcontracted to many different vendors, there is a need for closer tolerances than if the helicopter is entirely fabricated on one production line.

3. Assembly method. There are various methods of assembly that have different tolerance requirements. Close tolerances will permit complete interchangeability of parts without additional work. Selective assembly will permit less strict tolerances. A large production run will make parts available for selective matching with adjacent parts. If assembly methods and production time is available, the saw to suit, file to fit, approach may be the best.

4. Performance. The need for flushness and gap tolerances to meet performance requirements will be an influence on production tolerances.

5. Interchangeability and replaceability. The extent that parts and components should be interchangeable and how they should be replaceable have a direct influence on production tolerances. Replacment requirements at the lower maintenance echelons will require tolerances that will permit replacement with a minimum amount of match fitting.

6. Cost. Cost production tolerances where they are not required result in added expense. Loose tolerances that present assembly problems also may result in added expense.

Careful consideration must be given to production tolerances. The final tolerances used will be the result of coordination between the procuring activity, engineering, manufacturing, and purchasing.

See Chapter 11, AMCP 706-201, for preliminary design treatment of accessibility/interchangeability/replaceability.

11-8 SUBSTANTIATION

In addition to design, development, and manufacture of an airframe, it is necessary to demonstrate that the structural subsystem meets the design requirements. Two methods are used to demonstrate the structural adequacy of the airframe: analysis and testing. Generally, it is acceptable to use one or the other method to prove that strength and deformation, utility, dynamics, and weight and CG are within the required limits. In special cases — e.g., structures that are redundant or subject to fatigue — it may be necessary to employ both methods.

11-8.1 ANALYSIS

As the development progresses, substantiating data are prepared in accordance with the contract data requirements list (CDRL). Specific data requirements are coordinated with the Airworthiness Qualification Specification (AQS) prepared and approved for an individual helicopter program. The possible scope of these requirements is discussed in Chapter 4 of AMCP 706-203. Chapter 9, AMCP 706-203, presents a comprehensive discussion of the final qualification of the airframe. The analysis and testing performed during the design phase are discussed in this paragraph.

At the start of the detail design effort, design parameters established during the preliminary design are confirmed or amended. The loads applicable to all airframe components under design flight and ground loading conditions must be established and the critical design conditions determined. In Chapter 4, AMCP 706-201, the design loading conditions are described and procedures for the determination of internal load distributions are discussed. Use of a comprehensive computer program such as NASTRAN (Ref. 7) is recommended.

Following determination of the loadings that are critical for individual parts, in many cases structural adequacy can be substantiated by stress analysis alone. Examples are parts that involve either a single load path or simple redundancy, for which therefore the stress and analysis is both simple and accurate. Other parts for which analysis alone is adequate are simple fittings for which dynamic (fatigue) loadings are not significant, and secondary structure and components that are classified as nonstructural. However, because the airframe structure generally includes multiple load paths for which even the most sophisticated stress analysis methods often provide unreliable results, substantiation is based largely upon the results of structural tests.

11-8.2 TESTING

Portions of the structure will be tested after brief preliminary analysis. These include fatigue-loaded primary structure, redundant structures, components manufactured using new processes or materials, and castings with low safety factors.

Other portions of the structure will require detailed analysis and backup testing. Where expensive processes and/or large, costly structures are involved, extensive analysis is necessary in order to produce an optimum design that will be proven adequate by test.

When both analysis and testing are used, significant savings in weight can be obtained with designs for which the analysis indicates a small negative or zero margin. During testing, the areas in which failures occur can be strengthened. Some of these areas will have sufficient strength as is. In this way, the minimum weight can be achieved.

Some portions of the airframe are designed with multiple load paths. In the event that one path fails, the others will continue to carry expected loads. It will be necessary to test these areas with selected load paths failed, and with the applicable fatigue spectrum applied, to demonstrate that the structure will continue to support all loads until the next mandatory inspection. It also is necessary that inspection of these areas be possible, and particularly that portions of structure that are designed as fail-safe not be obscured. Chapters 7, 8, and 9, AMCP 706-203, describe component tests, surveys, and demonstrations that are applicable.

Fatigue testing may be generalized in that only basic data (such as S-N) are collected. Statistical analyses that consider operating load frequency will insure adequate structural reliability. The number of tests conducted will determine the confidence level for the design. Fatigue life determination is discussed in detail in Chapter 4, AMCP 706-201, and the required tests are outlined in Chapters 7 and 8, AMCP 706-201.

After the helicopter prototype is manufactured, flight test loads will be gathered as basic data to be used for structural qualification analysis and testing. It may be necessary to redesign locally before qualification if the flight loads are in excess of those predicted by analysis.

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CHAPTER 12
LANDING GEAR SUBSYSTEM

12-0 LIST OF SYMBOLS

A_c = tire contact area, in.^{2}
D = outside diameter, in.
W = tire width, in.

12-1 GEAR TYPES

Conventional helicopter landing gear configurations include wheel, skid, and float. Normally, the desired configuration will have been selected during preliminary design. The detail designer's objective is to verify a type, or combination of types, that provide the best performance for the least weight, cost, and maintenance. The system specification and mission requirements will influence the selection by specifying the following:

1. The environmental and operational landing conditions, which will indicate the roughness of operating terrain and the requirements for snow or water landing capability
2. Descent velocities, allowable load factors, and ground clearance, which together with weight and CG location, will dictate gear location, size, and axle vertical travel
3. Performance, which may require low-drag or retractable landing gear, thus requiring a compact gear configuration
4. Overload conditions and growth factors, which would require a gear with increased capacity.

As a rule, the skid gear is lighter in weight, is less expensive to manufacture and replace, and requires less maintenance than other types. One disadvantage of the skid gear is the necessity for special ground-handling wheels or dollies for moving the inoperative helicopter. In addition, a method of raising the helicopter to permit installation of these wheels must be provided. Some vehicles carry this equipment along and thereby suffer the additional weight and drag penalties.

The wheel gear provides the capability and directional control necessary to maneuver the helicopter on the ground, as well as during landing and takeoff. In addition, this type of gear allows a running takeoff, which provides additional lift and thus permits an increase in allowable payload in the event of an overload condition. The tires can absorb part of the impact energy during landing, and also can act as a cushion over obstacles during taxi operations.

Where the primary mission of the helicopter involves flight over waterways or areas with a number of lakes, the use of some type of water-landing gear is required. Depending upon the mission requirements, one of the following options could be selected:

1. A type with full seaplane capability (either floats or a boat hull) and having little or no land capability
2. An amphibious type, with either floats or boat hull plus land gear
3. A type with primary water capability, using fabric (inflatable) floats that also are capable of being used for land operations
4. A type with secondary water capability, using packaged floats that can be inflated for water landings if desired. With this option, the helicopter still maintains its land gear. Depending upon the sea state, it may or may not be possible to take off from the water.
5. A ditching capability, which keeps the helicopter upright (although it may be partially submerged) long enough for personnel to evacuate. Common practice here is to make provisions on the helicopter to attach the flotation gear and then to provide the major portion as a kit; thus, if the helicopter mission is not over water, only a small weight penalty is incurred.

Skis are required for operation from snow- or ice-covered areas. Current practice is to maintain the normal landing gear and to adapt the ski to it. This gives the helicopter a greater versatility and also improves ski life by preventing the skis from scraping when encountering areas bare of snow. Bear paws are smaller than skis and are used in marsh or bog areas, where normal wheels or skid gear would not be capable of supporting the helicopter weight.

12-1.1 WHEEL GEAR

12-1.1.1 General

There are four types of wheel gear in present use:

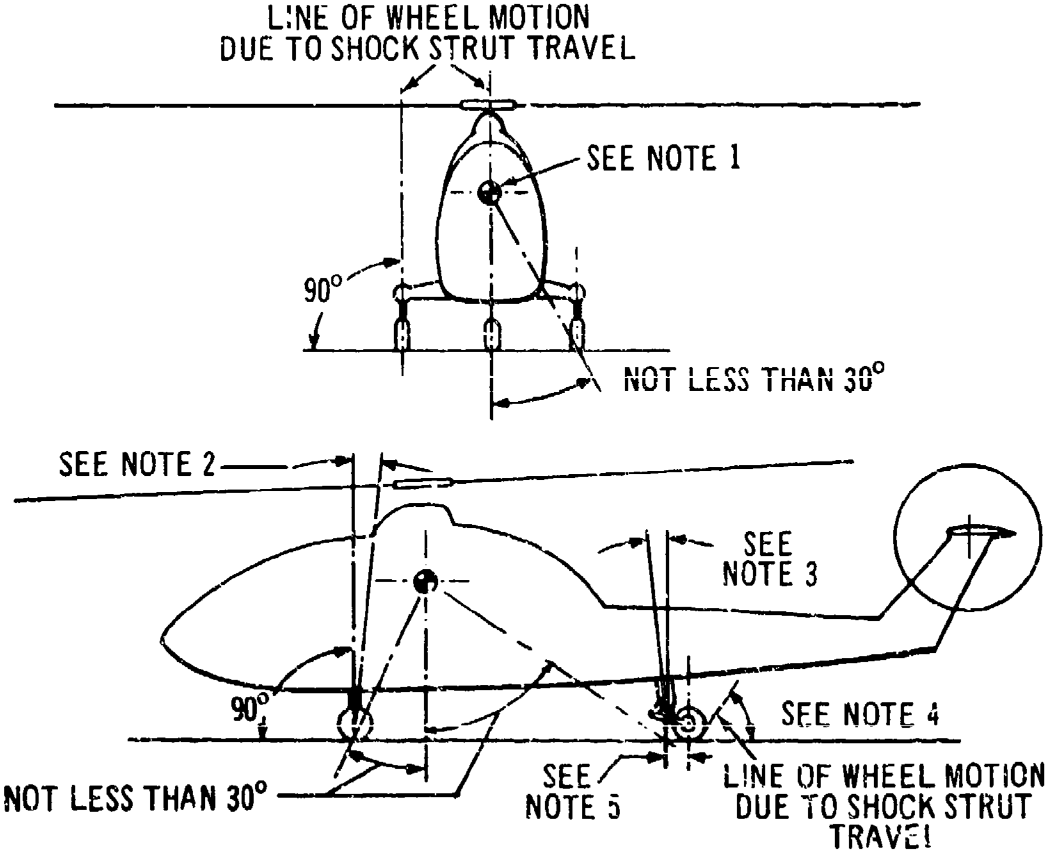
1. The conventional or tail wheel type
2. The tricycle or nose gear type, with and without tail bumper
3. The quadricycle arrangement
4. The bicycle type gear with outrigger wheels.

The recommended location and rollover angles for the tail wheel amd nose gear types are shown in Figs. 12-1 and 12-2, respectively. Because of the requirement for Army helicopters to operate on or from surfaces with as much as 15 deg slope, the turnover angle in any direction should be at least 30 deg (par. 13-

1.1.8, AMCP 706-201). Other than turnover angles, there are no specific requirements for or limitations on the location of the individual gear fore and aft of the CG. The quadricycle and bicycle arrangements may fall into either nose or tail category; thus, the same guidelines shown in Figs. 12-1 and 12-2 apply. The tail gear arrangement automatically, with no increase in weight, provides protection to the tail rotor during tail-down landings, whereas the nose gear arrangement usually requires a tail bumper, or, in essence, a fourth gear. However, no one arrangement inherently is superior or preferable. Many design criteria and operational requirements — such as safety, ground handling, and transportability — must be considered during selection of the landing gear arrangement for a new helicopter.

Because the pertinent trade-off studies will have been made and the optimum type of gear selected in the preliminary design phase, the following items should be required at the inception of the detail design phase:

1. Gear and location, rollover angles, and other



NOTES

1. CG LOCATION TO BE EITHER AT WEIGHT EMPTY OR AT BASIC STRUCTURAL DESIGN GROSS WEIGHT, WHICHEVER IS MORE CRITICAL. STATIC TIRE AND OLEO DEFLECTIONS AT BASIC STRUCTURAL DESIGN GROSS WEIGHT ARE TO BE USED FOR ALL CONDITIONS.
2. THE LINE OF MAIN WHEEL MOTION DUE TO SHOCK STRUT TRAVEL MUST BE AT AN ANGLE OF FROM 0 TO 5 deg AFT OF THE VERTICAL
3. THE TAIL WHEEL SPINDLE AXIS MUST BE INCLINED FORWARD AT AN ANGLE OF 5 deg.
4. THE LINE OF WHEEL MOTION DUE TO SHOCK STRUT TRAVEL CAN RANGE FROM 45 to 90 deg. ABOVE THE HORIZONTAL THE OPTIMUM VALUE IS 60 deg.
5. TAIL WHEEL TRAIL DISTANCE MUST BE GREATER THAN 10% OF THE TAIL WHEEL DIAMETER.

Figure 12-1. Wheel Locations and Motions for Tail Wheel Helicopters

constraints shown in Figs. 12-1 and 12-2 must be met.

2. Ground clearance with struts and oleos in a static position must be adequate for the operational environment anticipated for the specific helicopter mission class. A minimum clearance of 6 in. with all oleos and tires bottomed out, or with any one oleo and tire bottomed out and all others in a static position, is recommended.

3. Tire size must be adequate to satisfy ground flotation requirements at the most adverse CG position.

4. A shimmy analysis should be performed to insure that the best combination of trail angle and damping is being used.

5. A ground resonance analysis must be performed to assure that spring rates and damping coefficients of the oleo and the tire have the proper values to keep the helicopter out of the resonant frequency range (See par. 12-3).

6. Braking capability adequate for both stopping and parking the helicopter on a required slope must be established.

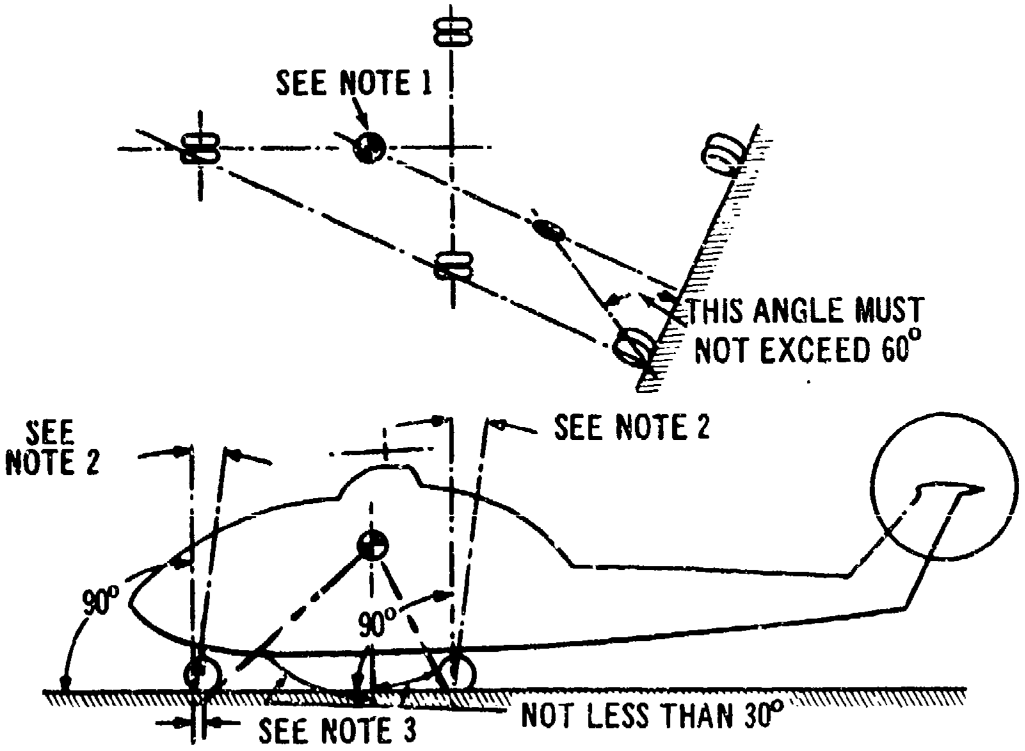
7. Final ground loads must be calculated, and the most critical loads on the gear determined.

12-1.1.2 Component Design and Selection

12-1.1.2.1 Tires

Ground flotation requirements, operating terrain and environment, and ground-resonance criteria are the main factors that influence tire selection. Design and construction features and qualification tests as listed in MIL-T-5041 apply to aircraft tires. When these tires are used on helicopters, standard aircraft ratings may be adjusted as shown in Table 12-1.

Because of the low landing speeds and short taxiing distances required in helicopter operations, vertical tire deflection greater than for standard aircraft



NOTES

1 CG LOCATION TO BE EITHER AT WEIGHT EMPTY OR AT BASIC STRUCTURAL DESIGN GROSS WEIGHT, WHICHEVER IS MORE CRITICAL. STATIC TYRE AND OLEO DEFLECTIONS AT BASIC STRUCTURAL DESIGN GROSS WEIGHT ARE TO BE USED FOR ALL CONDITIONS.

2 THE LINE OF WHEEL MOTION DUE TO SHOCK STRUT TRAVEL MUST BE AT AN ANGLE OF FROM 0 TO 7.5 deg AFT OF THE VERTICAL

3 REFER TO PAR. 12–2.2.3. FOR TRAIL DISTANCE REQUIREMENTS.

Figure 12-2. Wheel Locations and Motions for Nose Wheel Helicopters

TABLE 12-1. LOAD FACTORS FOR HELICOPTER TIRES

TIRE OUTSIDE DIAMETER	*LOAD FACTOR	*NORMAL INFLATION FACTOR
26 in. AND UNDER	1.67	1.50
OVER 26 in.	1.50	1.50

*TO BE APPLIED TO BOTH STATIC AND DYNAMIC RATINGS

is permissible at static load. Type III tires are classified as low-speed, low-pressure tires having a large cross section, and are used where good flotation capabilities are desired. Type VII tires are classified as high-pressure, high-speed tires with a smaller cross section than Type III types. Type VII tires have a high lateral spring rate, which may be required to control ground-resonance tendencies. In addition, their smaller size allows their use in high-speed compound aircraft, where limited space within the airframe for gear retraction requires a smaller, more compact tire. For Type III tires, a tire deflection not to exceed 40% is allowed; for Type VII tires a deflection of 37% is permitted. Tire efficiency for calculation of gear energy absorption is taken as 45%. To determine tire stroke under load, a tire deflection versus load curve should be used. This curve normally is available from the tire manufacturer for any standard tire size.

Ground flotation is the measure of the ability of a tire to remain upon the surface without causing breakdown or failure of the soil under the tire. The factors affecting ground flotation are:

1. Load per tire
2. Tire pressure
3. Wheel spacing
4. Strength of surface and subsoil.

Flotation requirements for a helicopter usually are dependent upon its primary mission. This usually is specified as a California Bearing Ratio (CBR) value coupled with the number of passes the helicopter can make before failure of the surface occurs. The method of analysis of the flotation capabilities of a helicopter on unprepared surfaces is discussed in Ref. 1; additionally it is important to note that the deceleration rate due to braking should be 6 ft/sec^2 for rotary-wing aircraft.

A more precise method for determing tire contact area A_c is

$$A_c = 2.0 \quad WD, \text{in.}^2 \qquad (12\text{-}1)$$

where

W = tire width, in.
D = outside diameter, in.
2.0 = empirical constant, in.

This equation has been derived by analysis of a curve faired through test data from current tires.

Sufficient clearance must be provided in the landing gear design to prevent tire chafing against the airframe or gear structure under all conditions of loading and operation. MIL-STD-878 establishes the procedure for determining the clearances required due to growth of tires and increase in diameter due to centrifugal force. In addition to these values, a 4% increase in section width and height *shall* be allowed to compensate for the overinflation allowance for tires used on helicopters. An allowance for growth in gross weight (25% minimum) should be made when wheel and tire sizes are selected and clearances are established. To provide for such weight growth, the addition of plies to increase the load rating of a tire otherwise suitable for the design and/or dynamic loads is acceptable. A change of wheel and/or tire sizes during the service life of the helicopter can be expected to require changes to other landing gear and airframe structural components and therefore should be avoided.

12-1.1.2.2 Wheels

Factors that affect wheel design include notch sensitivity, fatigue, corrosion, heat damage due to sustained braking, and wheel disintegration due to tire failure. Current practice is to specify the use of forged aluminum alloy materials (usually 2014-T6) for new designs. Cast magnesium is not desirable because of its susceptibility to corrosion and its tendency to fragment or shatter upon failure.

Helicopter wheels *shall* be designed and tested to the requirements of MIL-W-5013. The rated load for the wheel *shall* be equal to or greater than the maximum static load to which the wheel will be subjected at maximum towing or alternate gross weight, which ever is greater. Roll tests *shall* be for 250 mi compared to 1500 mi for fixed-wing aircraft. Bearing sizes can be established by the methods outlined in Ref. 2 or a comparable roller bearing handbook. A seal or other means of preventing water from entering the wheel bearings must be provided to prevent bearing corrosion.

12-1.1.2.3 Shock Struts

The most commonly-used devices for absorbing energy during a landing, and for supporting the helicopter during taxiing and ground operations, are mechanical springs, liquid springs, and air-oil struts. Other methods of absorbing energy (usually referred to as secondary energy or one-time application usage) are:

1. Crushable structures, e.g., honeycomb
2. Friction devices, i.e., two materials in contact with each other
3. Extruding devices, e.g., metal drawn through an undersized hole
4. Cutting devices, e.g., a sharp edge slicing through metal as it moves
5. Plastic deformation, i.e., material stretched beyond the elastic limit.

These devices are mentioned as possible solutions when an energy absorption beyond the normal landing gear capability is desired.

The most commonly-used mechanical spring is the cantilever type, with either a flat or a tubular cross section. These can be considered to have a linear load deflection curve if they are not stressed beyond the elastic limit. Other types of mechanical springs — such as Belleville washers, ring springs, and rubber — have nonlinear load deflection diagrams. Damping is small (usually due only to loss of energy at the support, or frictional forces between elements). Because the springs return far more energy than they dissipate, the use of spring landing gear usually is limited to light aircraft. The lack of damping also increases the likelihood of ground resonance.

Both the liquid spring and the air-oil strut absorb energy by the dashpot principle, forcing fluid under pressure through an orifice. In either application the area of the orifice may be fixed or may vary with displacement, as a metering pin of varying diameter moves through a fixed orifice.

In the liquid spring the compressibility of the fluid is used to store energy. Special consideration must be given in the gland designs to the high pressures at which these units operate. The sensitivity to changes in temperature also must be considered. For example, to accommodate low temperatures, the unit probably will be pressurized in the extended, or unloaded, condition to 2000 psi and special features such as a "decuperator" unit must be provided to limit the pressure rise due to high temperature. A liquid spring is particularly well suited for short stroke applications with levered gear arrangements, but any length stroke can be achieved by proper combination of piston rod diameter and total volume of oil. In fact, with tail rod liquid springs, exceptionally long strokes can be achieved. For servicing a liquid spring without jacking the helicopter a charging pressure of 20,000 psi is required, while the maximum charging pressure required probably is 2000 psi when the spring unit is unloaded and fully extended by jacking the helicopter.

In the air-oil strut a chamber of compressed air is used to restore the strut to a static position and to provide a cushion while taxiing and maneuvering the helicopter on the ground (Fig. 12-3). The efficiency of this type of shock absorber is shown by drop tests to be between 80 and 93%. A conservatively low value for the shock absorber efficiency should be used, together with the specified reserve energy descent velocity and the design limit landing load factor, to establish the maximum strut stroke required to absorb the total reserve energy. This design calculation is discussed in par. 4-10.3, AMCP 706-201.

The lower values of efficiency are applicable to struts containing fixed orifices. The higher values are achieved by the introduction of a metering pin. This configuration provides a variable orifice, which can

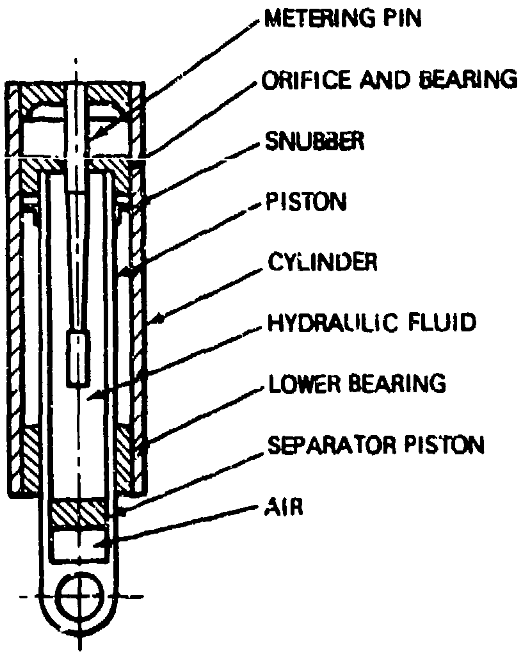


Figure 12-3. Typical Air-Oil Strut in Static Position

be larger at the point at which spin-up, or high drag, loads are applied. The orifice area then decreases further on in the stroke and more energy is absorbed at lower load levels. However, this type of strut is more complex and the implications of increased weight and cost relative to a strut with fixed orifice must be weighed.

The design of air-oil struts *shall* be in accordance with MIL-L-8552, with two exceptions:

1. A seal other than the standard O-ring to avoid spiral failure
2. A special scraper ring that is more effective in keeping out dirt than the specified MS 33675 scraper rings. The latter have no means of preventing dirt from entering the strut past the outside of the scraper, which fits loosely.

The first step in the design of the oleo strut is to determine the size of the piston. The static load and static pressure define the area, and consequently the diameter, of the piston rod. The static pressure is determined from an isothermal air compression-stroke curve developed for the strut. A pressure of 3000 psi can be assumed at the compressed position, with provision in the strut design for a maximum of 4000 psi to allow for growth (increased gross weight) of the helicopter. A compression ratio of 2.5-3.0:1.0 from the static to the compressed position commonly is applied during preliminary design, with the value selected being dependent upon the landing load factor. Thus, the static air pressure used in the determination of piston diameter will be in the range of 1000 to 1250 psi.

To avoid full extension of the gear when the helicopter is lightly loaded, the pressure in the extended position is set at approximately 25% of the static pressure. The static position is set between 66 and 85% of the strut stroke, with 83% providing smoother taxiing. As an example, for a strut with a total stroke of 12 in., the static position would be 10 in. (83% of 12 in.) from the fully extended position. A typical shock strut compression curve is shown in Fig. 12-4. When the static positions of the individual struts are being selected, the overall landing gear configuration also must be examined to assure that all applicable ground clearance requirements, including flat tire and flat strut conditions, are met.

As previously noted, the piston area is determined by the static load and the static pressure. The wall thickness of the piston rod is determined by the critical combination of bending moment and compression load. Of the ground loads specified (pars. 4-5 and 4-6, AMCP 706-201) the braking conditions or obstruction load condition probably will produce the maximum value of bending moment in the piston rod, which occurs at the point that the piston enters the lower bearing. Materials with ultimate tensile strength of 260,000 psi or greater should be considered for the piston rod, subject to the approval of the procuring activity (MIL-L-8552). The ratio of piston rod diameter to wall thickness also must be examined to assure that the design is neither unnecessarily heavy nor impractically difficult (hence expensive) to manufacture. Should the wall thickness, as determined by the critical loading condition, produce an unacceptably heavy strut, the conditions governing the static position should be re-evaluated.

There are two ways in which a shock normally is mounted, cantilever and universal (Fig. 12-5). The

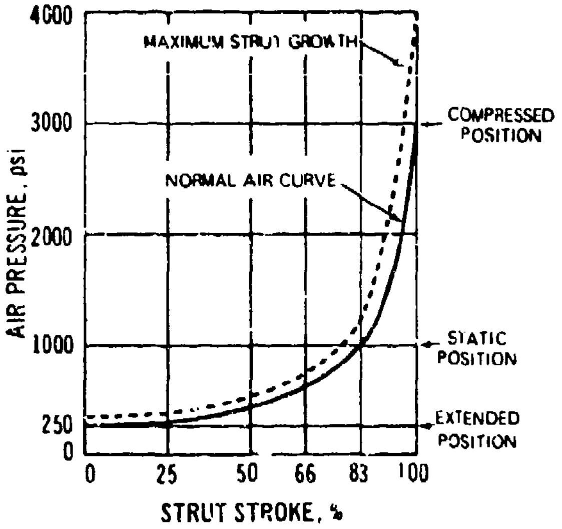
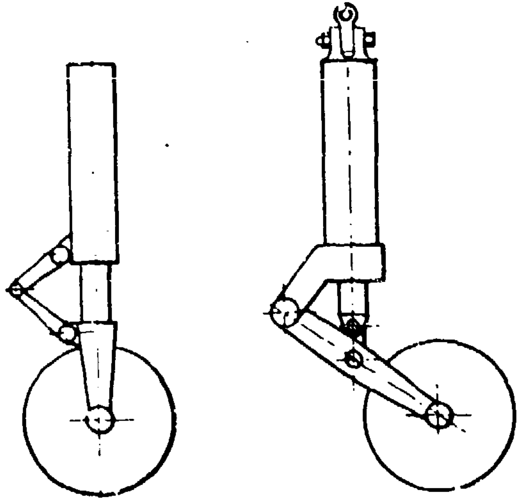


Figure 12-4. Typical Isothermal Shock Strut Air Compression Curve



CANTILEVER GEAR

ARTICULATING GEAR WITH UNIVERSAL-MOUNTED STRUT

Figure 12-5. Cantilever and Universal Struts

cantilever design is used more commonly because it is lighter in weight, even though large bending moments may be induced under loading conditions that include side loads, or components other than along the axis of the strut. These bending moments are detrimental not only from the standpoint of the resultant stress, but also because they increase the possibility of leakage through the strut bearing and increase strut friction. This friction can cause erratic strut operation during landing and taxiing operations. However, use of proper bearing materials, proper location of seals in the bearing, and low bearing pressures are suitable preventatives.

For cantilever-mounted struts, the bearings should be spaced so that bearing stress does not exceed 6000 psi under the design side load. The distance between the outermost ends of the upper and lower bearings with the strut fully extended should be at least 2.75 times the piston diameter. For universal (pin ended) struts, which have little tendency to bind, this distance is only 1.25 times the piston diameter. With these parameters known, the length of the strut can be determined. The length of a cantilever strut in the extended position is equal to (Fig. 12-6):

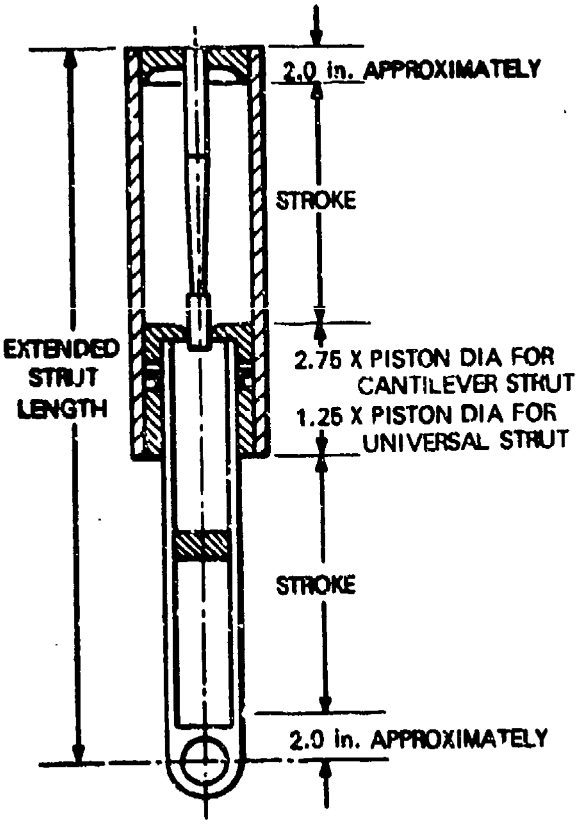

2.0 in. APPROXIMATELY
STROKE
EXTENDED STRUT LENGTH
2.75 X PISTON DIA FOR CANTILEVER STRUT
1.25 X PISTON DIA FOR UNIVERSAL STRUT
STROKE
2.0 in. APPROXIMATELY

Figure 12-6. Determination of Strut Stroke

2.75 x (piston diameter) + 2x (stroke)
+ (allowance for upper cap and lower axle clearance)

This allowance, shown in Fig. 12-6 as approximately 4.0 in., will vary with strut size. For universal-mounted struts, the length is calculated in the same manner as for the cantilever strut, except that the lower value of minimum bearing spacing is used. The elimination of bending loads by a levered suspension thus results in a shorter strut. The use of this configuration is recommended for tail gear where long trail arms are desirable to reduce shimmy tendencies.

A snubber or rebound ring is provided to reduce the impact forces upon the gear during sudden extensions, such as those encountered on takeoff and landing rebound. The strut in the fully extended position and not touching the ground must be capable of reacting to a rebound load factor of 20 times the unsprung weight of the landing gear, or three times the load induced by air pressure within the gear, whichever is greater.

Mixing of air and oil in the oleo can cause loss of strut efficiency as the hydraulic fluid becomes aerated. Also, foaming of the fluid takes place during servicing, increasing the possibility of having the incorrect amount of fluid in the oleo and thus reducing some of the energy-absorption capability of the strut. Current practice is to install a separator piston, which creates two separate chambers; thus, no mixing of air and oil is possible. The housing or outer cylinder is usually of forged aluminum construction and consists of bosses or lugs to provide the method of attachment to the airframe. A set of torque arms is necessary to react the torque loads on the gear due to pivoting, and to maintain wheel alignment by joining the sliding piston and fixed housing so as to permit only an axial degree of freedom. Main gear torque arms usually are constructed of forged aluminum, with the angle between the arms limited to 135 deg. Nose gear torque arms, being susceptible to shimmy, may be of steel construction with a lesser angle, and may use a knuckle design (multiple lugs) at the apex end.

The design of nose and tail wheels must include centering springs to insure proper alignment of the wheels during landing and sufficient damping of angular motion to prevent shimmy during run-on landing and taxiing. Shimmy is a self-excited, rapid oscillation of the landing gear that occurs at or above critical landing speeds. Basically, shimmy is the result of a lateral misalignment between the helicopter CG and the center of contact of the tire with the ground. The gear is deflected to one side, but a restoring force due to the elasticity of the gear and its supporting structure causes the wheel to move back. It then overshoots the center, with a subsequent lateral misalignment on the other side.

An analysis of shimmy is quite complicated if an attempt is made to incorporate all of the variables; however, a simplified approach that calculates the speed at which shimmy starts to occur has proven acceptable for landing speeds below 100 kt. This method uses Moreland's stability criteria (Ref. 3), which express the equations of shimmy in a nondimensional form containing all of the major aircraft parameters. A digital computer program has been developed to indicate the stable and unstable regions for any given aircraft configuration.

To reduce the possibility of shimmy, several design practices based upon experimental work should be observed for helicopters equipped with nose gears:

1. The trail distance (Fig. 12-2) should be less than 8% or greater than 50% of the tire diameter.
2. Dampers mounted at the wheels or at the strut are acceptable. For dual wheels, a damper connecting the two wheels is preferred because the amount of play in the system thereby is minimized. Dual wheels are preferred for dynamic loads above 20,000 lb.
3. A short trail distance usually requires more damping than a long trail arm. Therefore, an articulating or semiarticulating gear with a long trail arm is probably lighter than a gear with a short trail arm.
4. Hydraulic viscous damping is preferred. Friction damping is not desirable due to the large variations encountered.
5. Torque arms should be as stiff as possible. The apex should be a knuckle design to avoid any offset in the line of action of the two members. The use of steel instead of aluminum should be considered.

In the case of single-rotor configurations, the tail rotor provides excellent control of the helicopter during landing, takeoff, and taxiing; therefore, steering of the nose or tail wheel is not necessary. A device should be provided, controllable by the pilot, to lock the tail wheel in a trail position during landing and takeoff to assist in directional control. Nose gear helicopters are inherently stable and, therefore, usually require no lock. Both nose and tail gears should contain cams or other centering devices to maintain the gear in a trail position prior to landing.

As a rule, antiskid devices are not required for vehicles with landing speeds below 100 kt.

12-1.1.2.4 Brakes

Braking system requirements are governed by the system specification, and by MIL-B-8584 and MIL-W-5013. Current cargo and crane helicopters use Type IV systems because the energy-absorption requirements dictate a power-operated system. This type of system must operate from either set of pedals, and must permit parking of a 10-deg slope without application of external power.

Main gear wheel braking usually is sufficient to achieve the specified deceleration; however, each helicopter must be analyzed individually to determine brake adequacy.

A parking brake handle accessible to both pilot and copilot must actuate a parking brake with pressure sufficient to hold the helicopter on the specified slope with the power off.

Sizing of brakes is dictated by prior constraints in addition to the applicable Military Specification. The size of the wheel and tire, previously selected, defines the volume available for the brake assembly, and hence the area of the friction pads and the number of actuating cylinders.

Master brake cylinder sizing and detail design are dependent upon the brake pressure and actuation volume requirement, the speed of response, the linkage ratio to the pedals, and the available pressure supply.

12-1.2 SKID GEAR

12-1.2.1 General

The major advantages of skid gear are lightweight, low cost, and simplicity. The initial cost of skid gear is less than that of a conventional oleo gear. The elimination of wheels, tires, brakes, and braking system also results in reduced maintenance. The disadvantages are the need for support wheels or dollies to handle the helicopter on the ground, a limited running-landing capability, the inability to perform running takeoffs and to thereby increase payload with the resultant increased lift, and the high rate of wear of the bottom of the skids.

Skid gear is used on many lightweight helicopters where the normal landing energy is stored in tubular or rectangular spring members. For harder landings, the landing energy is absorbed by permanent deformation of the spring members.

The static deflections of the skid gear usually are less than those of oleo gear. The efficiency of skid gear is approximately 50% until the load in the spring member exceeds the elastic limit. When the load is above the yield strength of the member, the efficiency of the skid gear is comparable to that of the conventional gear.

Ground clearances of structure, control surfaces, or external items for skid gear *shall not* be less than 6 in. with skids flat on the ground in a static position.

12-1.2.2 Ground-handling Wheels

Location and number of wheels are dependent upon helicopter weight and CG location. Units should have the capability of being removed easily and quickly.

Wheels should be capable of rolling over a 4-in. obstacle. A braking device is not required.

12-1.2.3 Scuff Plates

Factors that affect skid wear are speed of landing, bearing pressure, resistance of skid material to abrasion, and type of landing surface. Removable wear plates should be located at critical wear points along the bottom of the skid to prevent permanent damage to the skid and supporting structure when a landing is made on a hard surface at a speed of 35 kt. These wear plates must be designed to prevent excessive digging into the pavement surface as a result of the scuffing action resulting from gear motion. Steel wear strips seem to give the best results.

12-1.3 RETRACTABLE GEAR

12-1.3.1 General

Extension and retraction requirements for the landing actuation system are given by the helicopter system, or detail specification.

The gear actuation system must include a mechanical lock at either extreme of travel; must provide an indication of gear status — i.e., up, down, or in transit — and must provide a method for emergency actuation. Retraction can be accomplished by folding foward, aft, or laterally, or by telescoping the gear along a fixed oleo axis. Forward retraction is favored for the main gear in order to permit the airstream to assist in emergency extension. The telescopic retraction method may be employed on the nose gear, either because structural considerations preclude a large, open wheel well in the nose, or because the nose gear strut length must be controllable in order to tilt the fuselage for cargo loading.

A typical retraction system, wherein the gear pivots up into a wheel well, contains the following:

1. An actuating cylinder on each gear, which either acts as a drag strut or drives the drag strut linkage, which in turn actuates the gear
2. Mechanical up and down locks
3. Mechanical lock limit switches
4. Limit switches on each gear scissors, or on the oleo itself, which are deactivated when the helicopter weight is on the gear.

The down lock limit switch, the landing gear control handle, the scissors switches, and the hydraulic control valve up coil are wired in series so that raising can commence only with gear unload. Similarily, the up lock limit switch and the control valve down coil are wired in series through the control handle.

The design of the actuating systems *shall* be in accordance with MIL-C-5503 and MIL-H-8775. However, the specified seals (MS 28775) and scraper rings (MS 28776) should be replaced where possible. O-ring shaft seals are prone to spiral failure because the seal works on an unlubricated shaft. Improved service life has been achieved by using a seal consisting of a T-shaped elastomer supported by two Teflon backup rings. MS scrapers allow entrance of sand and dirt because the sealing surface is discontinuous. A filled Teflon scraper, preloaded with an O-ring, provides longer seal life.

12-1.3.2 Actuation

Indication usually is provided for three modes on each gear. Engagement of the mechanical up and down locks actuates the gear up and down indications, respectively, in the cockpit.

Disengagement of both up and down locks indicates an in-transit condition by illuminating the gear control handle from within and/or by uncovering a striped "barber pole" indicator.

12-1.3.3 Emergency Extension

Emergency extension of the gear should be manual. An air bottle may provide the energy necessary to assist in lowering the gear and to overcome air loads on the gear door. Provisions *shall* be made for emergency extension in the event of loss of hydraulic pressure or of failure of the landing gear directional control valve.

12-1.4 SKIS AND BEAR PAWS

12-1.4.1 General

Skis and bear paws are similar. Bear paws are used primarily on snow or soft terrain for nearly vertical descents. Skis are used primarily for landings with some forward speed in snow-covered areas. Skis are

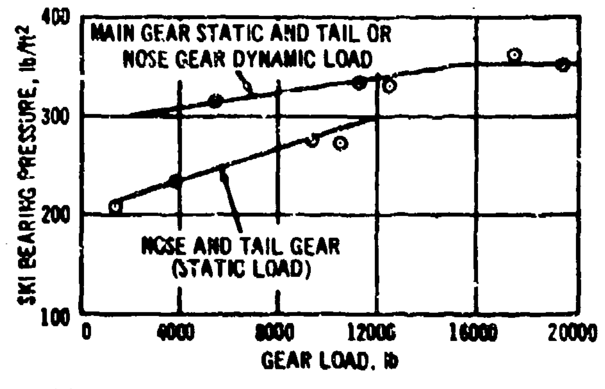


Figure 12-7. Landing Gear Static Load vs Ski Bearing Pressure

larger than bear paws, and have a lower bearing pressure, a longer nose section, and a greater length-to-width ratio. Landing gear static load versus acceptable ski-bearing pressure data are shown in Fig. 12-7.

Note that the main gear has a higher allowable bearing pressure than do the nose or tail gears. The tendency of pilots is to feel for the ground, or land tail down, when operating tail wheel helicopters. This increases the effective tail ski load. As a result, relative to the main gear static load, a lower bearing pressure is required to stay above the snow surface. The nose gear must react the dynamic loads caused by drag and friction forces at the main gear, and thus a lower bearing pressure is necessary to keep the nose ski from submerging in the snow. A comparison of typical missions indicates that lighter vehicles require a greater mobility than does a heavy cargo helicopter. This is evidenced by the lower bearing pressures found to be acceptable for helicopters with lower gear static loads.

Limited available experience indicates that above a bearing pressure of approximately 250 psf, ski operation becomes marginal, while a reasonable bearing pressure for bear paws for soft-soil operation should not exceed 1500 psf (approximately 10 psi).

The length-to-width ratio of skis is not of great importance at the low forward speeds encountered in helicopter operations. A ratio of 2.5:1 currently is in use for helicopters. On snow-covered terrain, a running landing speed of at least 15 kt is required to enable the pilot to maintain a clear field of vision by keeping the helicopter in front of the blizzard created by the rotor downwash.

Ski friction is due to compacting of the snow (which can vary greatly with the moisture content or density of the snow) and sliding friction. A coefficient of friction of 0.25 *shall* apply for landing conditions and a coefficient of 0.40 *shall* apply for ground-handling conditions (MIL-A-8862).

The relation between snow depth and ski track depth is presented in Fig. 12-8.

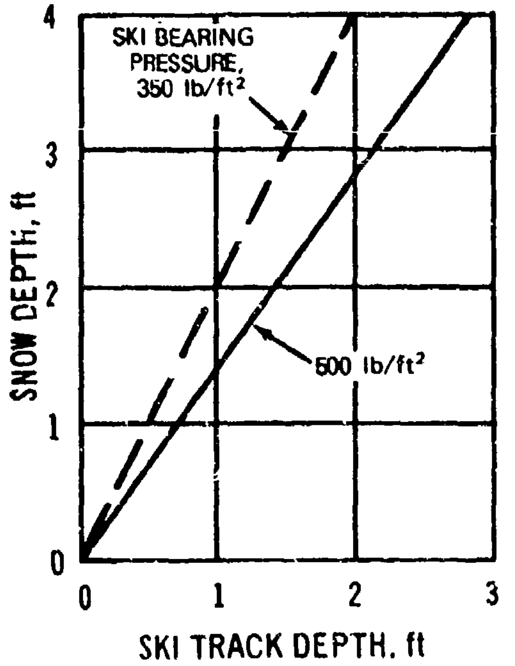


Figure 12-8. Snow Depth vs Ski Track Depth

12-1.4.2 Installation

Current practice is to adapt skis to the standard wheel arrangement. The tire is allowed to protrude through an opening in the ski in order to permit landing on hard-packed snow and ice. This permits the tire to absorb some of the landing energy that otherwise would be transmitted through the ski. This design also permits ground maneuvering of the helicopter to work areas that normally are cleared of soft snow.

The skis are usually of metal or Fiberglas construction with a honeycomb or balsa wood core. A long planing nose (Fig. 12-9) keeps the ski from digging into the snow during landings involving foward speed. The aft portion also is raised upward, although to a lesser degree, to permit rearward movement over snow or other obstacles without digging in or snagging. Replaceable chafing strips may be attached to the bottom of the ski to protect it from being scraped and damaged if it contacts the ground while landing or taxiing on snow-cleared surfaces.

The ski is attached to the extended wheel axle by pedestal fittings that position the ski in relation to the wheel and permit pivoting of the ski above or below its normal horizontal position, thus allowing it to follow the terrain without imposing high loads on the ski or gear structure. Fore and aft cables with adjusters position the ski in flight to minimize drag at cruising speed. Cables are attached above the shock strut and become slack as the oleo compresses upon

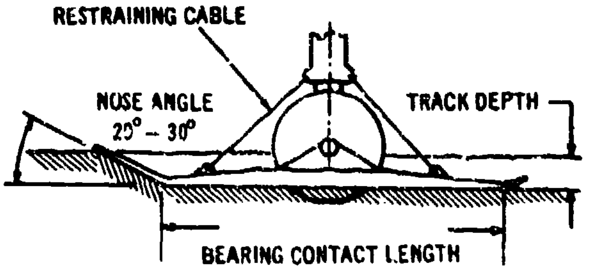


Figure 12-9. Ski Configuration

landing, allowing the ski freedom to seek its own position on the snow. A spring or bungee cord sometimes is used to keep the ski clear of the ground on snow-free runways, although some designs favor a caster wheel at the aft end.

12-2 LANDING LOAD ANALYSIS

The operational environment to which Army helicopters are exposed is sufficiently severe that it has been found necessary to increase the sinking speed requirements specified in MIL-S-8698. The applicable design criteria are given in Chapter 4, AMCP 706-201. The requirements for both symmetrical and asymmetrical landing conditions are discussed therein, inasmuch as a comprehensive design analysis of the landing gear subsystem is required during the preliminary design of a new model of helicopter.

Landing loads *shall* be determined by a rational analytical procedure that has been approved by the procuring activity. One such procedure is outlined in MIL-A-8862. The adequacy of the landing gear subsequently will be demonstrated by drop test. The qualification requirements, including the drop test, are described in Chapter 9, AMCP 706-203.

In addition to the landing loads, the loads created by taxiing over obstructions, turning, braking, towing and backing also must be determined using the design criteria of Chapter 4, AMCP 706-201. The landing and ground-handling loads then are distributed as shear loads, and bending and torsion moments at selected points on the helicopter. The shear loads and moments resulting from flight loads are distributed similarly, and the critical loads at each point are determined.

Normally, the energy-absorbing capability of the landing gear is designed so that the landing and ground-handling loads are critical only for the landing gear attachment and support points. Should the specified landing or ground-handling loads exceed the flight loads, it usually is appropriate to revise the landing gear energy-absorbing system to reduce the load factor at the CG, and/or the local loads. The airframe is overweight and structurally inefficient if the landing loads are critical. As with other characteristics, it is necessary to examine the trade-off between energy-absorbing-system weight and airframe structural weight.

12-3 AVOIDANCE OF GROUND RESONANCE

Ground resonance occurs due to coupling between the main rotor blades oscillating about their lag hinges and the airframe excited at its rigid-body natural frequency in the roll, lateral, and/or pitching modes. Instability can result in oscillations which buildup sufficiently to destroy the helicopter in a matter of seconds. Because the theory is understood (Refs. 5 and 6), the problem can be approached analytically during detail design.

The factors governing stability are a combination of rotor and landing gear parameters. Several of the parameters are fixed by basic constraints beyond the control of the landing gear designer. The critical parameters are:

1. Blade mass, offset, and moment of inertia
2. Number of blades
3. Blade damping about the lag hinge
4. Rotor rpm
5. Fuselage mass
6. Fuselage polar mass moment of inertia about the lateral and longitudinal axes
7. Landing gear geometry
8. Tire vertical and lateral spring rate
9. Oleo strut damping
10. Structural spring rate of gear.

Items 3, 7, 8, 9, and 10 offer the most flexibility in assessing and resolving ground resonance.

The problem is further complicated by the following factors:

1. Nonlinear variation of the tire and oleo spring rates caused by changes in helicopter gross weight or CG
2. The percentage of gross weight supported by the rotor
3. The effect of improper servicing on the spring rates of the struts and tires
4. The effect of a flat or soft tire on the spring rate.

A detail design approach for assessing stability and determining the parameter changes required to resolve the problem includes the following procedures:

1. Determine the lateral, roll, and longitudinal rigid-body natural frequencies of the airframe, considering Items 5 through 10 plus the aforementioned nonlinearities. Operational considerations will determine the range of available parameters in some cases. A range of oleo spring rates is established by the deceleration loads imposed upon the airframe and by the ratio among static, extended, and compressed lengths. Tire selection is dictated by terrain, load, and CBR, leaving some latitude in tire vertical and lateral spring rates. The range of oleo-strut damping is established by the necessity for an efficient energy-absorption curve. The possible rigid-body natural frequencies, and the extremes of the ratio of rotor rpm to rigid-body natural frequency, should be tabulated.
2. A ground resonance stability plot should be

generated (see Ref. 4 and Chapter 5, AMCP 706-201). This curve describes a center of instability by defining, as a function of blade properties, the ratio of rotor rpm to undamped airframe rigid-body natural frequency that leads to instability. A band of instability on either side of the center of instability is superimposed using Coleman's technique (Ref. 4). The analytical technique establishes instability in the absence of oleo- or rotor-blade damping. A comparison with the results of Step 1 determines what conditions of rotor rpm and strut and tire stiffnesses lead to instability.

3. Stabilize the system by reducing the width of the instability band to zero through the introduction of both rotor hub damping and oleo damping. Deutsch's criteria (Ref. 5) specify the product of blade and oleo damping necessary to reduce the unstable range to zero. Required strut damping is obtained from Eq. 6, Ref. 5. Strut damping will be nonlinear with respect to stroke and load if a tapered metering pin is used, resulting in a variable orifice area. Here, the damping should be defined as the tangent to the force/velocity curve at each discrete combination of strut loading. The conditions for which ground resonance must be avoided (see par. 5-3.5, AMCP 706-201) include one blade damper inoperative and the combination, on a single strut, of flat tire(s) and flat strut (shock absorber pressure at zero). These cases are discussed in Ref. 5. The methods of analysis also are discussed in Chapter 5, AMCP 706-201, while substantiation requirements are given in AMCP 706-203.

12-4 WATER-LANDING CAPABILITY

12-4.1 GENERAL

The design criteria applicable to water landings are a sink speed of 8 fps in combination with 2/3 rotor lift, and appropriate head moment and drag at the basic structural design gross weight. Specific landing conditions to which these criteria apply are:

1. Zero forward speed
2. Forward speed of 30 kt
3. Asymmetrical drop, with the hull rolled 10 deg and no forward speed
4. A forward speed of 30 kt and a yaw angle of 15 deg
5. A forward speed of 30 kt and nose-up pitch angles of 3, 6, and 9 deg

When determining the lateral stability of the helicopter, the following lateral imbalances must be considered:

1. Lateral displacement of the helicopter CG from the centerline as inherent in the construction of the helicopter
2. Effect of partially empty fuel tanks
3. Placement of passengers, crew, and cargo
4. Rotor blade lead-lag effect.

For helicopters with a water-takeoff capability, a minimum clearance of 6 in. between the rotor blades and the water must be provided at the required sea state with rotor shut down. This precludes possible damage to the blades while the helicopter is rolling and drifting during engine shutdown.

Sea state is a condition that comprises height of waves, wind velocity, and wave length. Current practice is to use significant wave-height values, which are considered conservative, for design. A graph of wave height versus wind velocity for different sea states is shown in Fig. 12-10.

When calculating the buoyancy of a float, it is customary to consider the weight of fresh water as 62.4 lb/ft^3.

Inverted V-type hull shapes are preferred over flat or round shapes. Inverted V hulls have low water resistance and reasonable aerodynamic characteristics, whereas floats with circular bottom sections tend to stick and to exhibit undesirable spray characteristics. Methods to prevent spray from obscuring the pilot's vision or damaging the blades during landing, takeoff, or taxiing should be incorporated in the initial design stage. Spray deflectors are a possible solution, but they result in increased drag, i.e., poorer performance. Buoyancy of float hulls is dictated primarily by interior cabin size, with the result that hull helicopters generally have substantial amounts of excess buoyancy.

12-4.2 PRIME CAPABILITY

For water-based helicopters, the basic flotation design will consist of metal floats or hull-shaped fuselage with some form of outriggers for stability. The hull and auxiliary floats must have enough watertight compartments so that, with any single compartment of the hull or float flooded, the buoyancy of the helicopter still will provide sufficient stability to prevent capsizing in the sea state in which it is to operate.

The high CG inherent in helicopter design, together with the large droop of the blades, makes helicopter operation in the open ocean sea state difficult without the imposition of large performance and weight penalties to obtain the required stabiltiy and blade clearance (Fig. 12-10). The use of a sea anchor to maintain a heading into the wind and waves is one approach to improving the roll stability for operation in this sea state.

12-4.3 ADDITIONAL CAPABILITY

Many helicopters are primarily land-based, with some phase of their operation performed on sheltered or inland waterways. For this type of operation, hull and auxiliary floats, if used, must be divided into compartments in such a manner that, if any one compartment is flooded, the buoyancy of the helicopter still will provide sufficient stability to prevent capsizing. If floats are used, the buoyancy necessary to support the maximum weight of the helicopter in fresh water must be exceeded by 50% for single floats and 60% for dual floats.

The most straightforward approach to an amphibious helicopter is to incorporate a watertight hull with more than adequate buoyancy and with little performance or weight penalty. Auxiliary floats must provide lateral stability for the sea state condition specified. The landing gear can be retracted into the auxiliary floats to reduce drag.

Another way to obtain a water capability for land-based helicopters is to add separate fabric (bag) or metal floats. The wheels and shock absorbers are attached to the floats in order to keep weight to a minimum. A single main float is not used on a helicopter due to the absence of a wing or other structure to support the auxiliary floats that are necessary for lateral stability. The permanently inflated bag floats are attached to the airframe by tubular members that support two fore-and-aft tubes to which the floats are attached. These latter tubes provide both stiffness and stability for the floats. The underside of the float may have an additional tubular member to accommodate wear. This also can serve as an attaching point for a damping unit if such a device is required to damp out any tendency of the helicopter to shake or bounce.

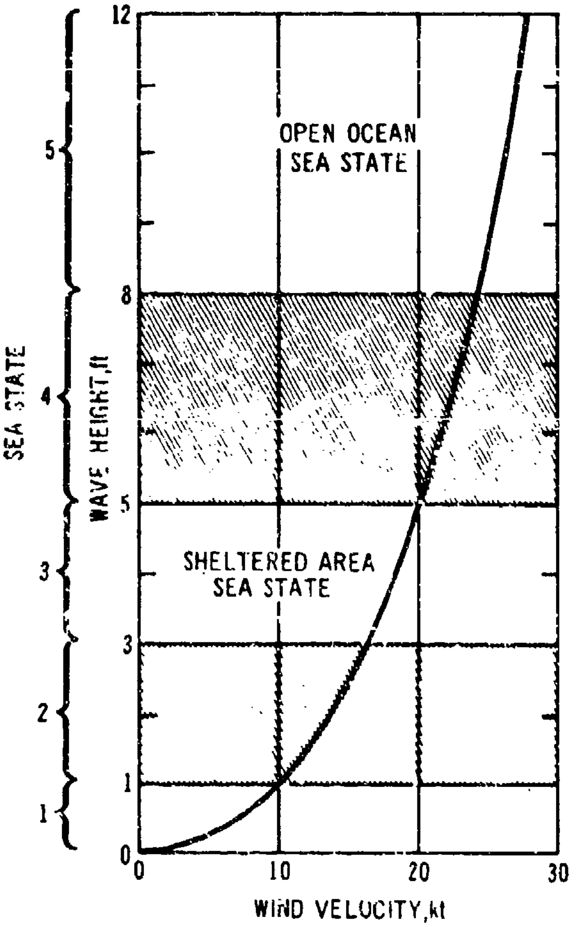


Figure 12-16. Definition of Sea State Conditions

Larger amphibious helicopters have both floats and wheels. The wheels keep the float clear of the ground during land operations in order to eliminate chafing and wear on the underside of the float and to allow taxiing to loading areas without the need for dollies or other special ground-handling equipment.

Floats, either primary or auxiliary, must be able to withstand the maximum pressure differential that might be developed at the maximum flight altitude without exceeding limit pressure.

12-4.4 EMERGENCY FLOTATION CAPABILITY

The purpose of emergency floats is to enable the helicopter to remain afloat long enough for the occupants to leave safely. The helicopter should be capable of remaining afloat in a condition of sea-state three if the normal helicopter mission is flown over sheltered areas or inland waterways; or in a condition of five if missions are performed over open waters.

Two classes of floats have emergency flotation capability: inflatable floats and ditching floats. The inflatable floats, which can be stowed or folded at the landing gear, use a storage charge of air or other gas for inflation. Inflatable floats are a lightweight and compact system, with a drag increase that is only a fraction of that of permanent floats. Inflatable floats can maintain the fuselage clear of the water so that, in the event of a forced landing in relatively calm waters, repairs of a minor nature may be made to allow the helicopter to take off again and complete its mission. Because the floats are inflated only a few hundred feet above the water (inflation time ranges from 3 to 7 sec), pressure change with altitude is not a design factor. Floats normally are pressurized at from 0.75 to 1.25 psi. For use with skid gear, these floats are cylindrical shaped and are stowed on top of each skid. For wheel gear, the float is doughnut-shaped and stored in a metal cage around each wheel.

Ditching floats usually permit the water level to cover the floor of the crew or passenger compartment. This has the effects of:

1. Lowering the CG, thereby improving lateral stability
2. Reducing the weight of the submerged portion of the helicopter by the buoyant force of the displaced water, thereby requiring less buoyance from the flotation bags
3. Keeping the helicopter intact so that the crew can be rescued and a salvage operation can be undertaken.

Normally, two main ditching floats are forward of the CG and are powered by a cool-gas generator or other method to give rapid inflation, usually in under 3 sec. Inflation may be activated by the pilot or by submersible valves that automatically trigger the generating unit when it is immersed in water. A third float, permanently inflated, is mounted in the tail cone to provide fore-and-aft stability.

12-4.5 MODEL TESTS

To substantiate the analytical data for any type of float system, a hydrodynamically complete scale model is tested in a model basin capable of generating different wave forms. The model is complete with rotors and ballast weights to permit varying helicopter weight, CG, and moments of inertia. Stability, roll response, and landing impact load data then are recorded and compared with analytical data. On models tested to date, excellent correlation between analytical and model test data has been noted. With the model in the tank, wave forms can be generated and model tendencies such as heading, rolling, and pitching can be observed. Methods of eliminating tail rotor submergence or other undesirable characteristics, if they exist, also can be evaluated.

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CHAPTER 13

CREW STATIONS AND CARGO PROVISIONS

13-0 LIST OF SYMBOLS

K = spring rate of suspension, lb/ft
K = total spring rate of two-spring system, lb/ft
K_1 = spring rate of first spring, lb/ft
K_2 = spring rate of second spring, lb/ft
L = length of suspension, ft
M_H = mass of helicopter, slug
M_L = mass of external load, slug
N = yaw restoring moment, ft·lb/deg
W = weight of external load, lb
x = distance between suspension attach points, ft
x = longitudinal distance in case of four-point suspension, ft
y = lateral distance between suspension attach points, ft
y = four-point suspension, ft
ω = natural frequency of suspension system, Hz

13-1 INTRODUCTION

This chapter addresses the requirements for personnel accommodations and cargo provisions to be incorporated in the detail design of helicopters. The discussion of the cockpit includes the pertinent requirements for crew station geometry, passenger compartment arrangements, seats and restraint systems, control/display arrangements, map and data cases, etc. Also included is a discussion of the interface criteria pertinent to the installation and employment of cargo handling and survival equipment. Design requirements for environmental control systems and lighting guidelines are included.

The basic design of personnel and cargo accommodations is determined during preliminary design. The detail designer should adopt an integrated systems approach to optimizing provisions for personnel and cargo in such a way as to obtain maximum mission effectiveness. To accomplish this goal, the detail designer will consider parameters such as the exact location of displays, controls, mission equipment, and emergency equipment and must optimize on the basis of anthropometric data and a human factors engineering analysis of crew tasks. He must recognize existing design boundaries as specified by the preliminary design and the appropriate Military Specifications. Pertinent Military Specifications are listed within the chapter where applicable.

13-2 PERSONNEL ACCOMMODATIONS

13-2.1 COCKPIT

There is a growing body of literature, methods, and practices which provide guidance in the application of system engineering to the development of cockpits. However, there are few permanent, specific, absolute requirements. The mission profile, the vehicle constraints, function analysis, time-line analysis, link analysis, and the development of new materials and components must be considered in determining cockpit design.

In addition to design handbooks and manuals — e.g., Refs. 1 and 2 — reference should be made to MIL-STD-250, MIL-STD-850, MIL-STD-1333, MIL-STD-1472, Ref. 3, AFSC DH 2-2, and pars. 13-3 and 4-5 of AMCP 706-201.

The general cockpit dimensions, seating arrangements and external vision envelope will have been determined during preliminary design. The trade-offs available to the detail designer include standardization of dimensions and controls versus opportunities for improvement resulting from a particular aerodynamic shape. The goal of maximum vision in all directions may compete with instrument panel location, airframe weight, comfort, armor protection, and aerodynamic shape.

The detail designer must consider the entire anthropometric range of user population for which the crew station is designed. The reach and body clearance envelopes must consider the heaviest clothing, survival, and protective equipment likely to be worn or used. Guidance for the application of anthropometric factors is contained in Chapter 13, AMCP 706-201.

Crew station arrangement and geometry *shall* be in accordance with MIL-STD-250 and MIL-STD-1333 unless otherwise specified by the procuring activity. Deviation from these specifications, or the location of controls or equipment not covered by these specifications, *shall* be based on a human factors engineering analysis of crew tasks. Controls, switches, and levers that require frequent actuation in flight should be conveniently accessible to the pilot's left hand to minimize removal of his right hand from the cyclic pitch stick during normal flight.

13-2.1.1 General Vision Requirements

MIL-STD-850 defines the requirements for aircrew external vision. Figs. 13-1 and 13-2 illustrate the Aitoff's equal area projection of the sphere showing single-pilot/tandem-pilot and side-by-side helicopter vision plots.

The problems associated with designing for optimum vision include reflections, glare, distortion, light transmission, and angle of vision. Antireflective coatings should be applied as necessary to minimize reflections and glare from instrument faces, windscreen, and windows. Analyses of crew stations under all parts of expected mission profiles should be conducted to determine that transparent areas are adequate for all mission requirements. Light transmission through transparent areas *shall* be maximized, consistent with mission requirements. Tinted windows or other design features which reduce light transmission are not acceptable because of their adverse effect upon external night vision.

Maximum obstruction due to transparency frame members is specified in MIL-STD-850. In helicopters where there are broad expanses of transparent areas that are used by more than one crew member, vertical obstructions should be no more than 2.5 in. wide projected on a plane normal to the line of sight. This will permit binocular vision to, in effect, see around the obstruction. Such obstructions should be located to avoid critical vision areas and to provide maximum distance between the crew member's eye and the nearest obstructions. Structural design and material selection guidance for transparent areas is discussed in Chapter 11. When necessary, a rear-vision mirror (see MIL-M-5755) should be positioned inside the windshield to give maximum visibility to the rear without obstructing forward vision.

13-2.1.2 Controls

Requirements for size, shape, location, range and direction of motion, and force-versus-displacement characteristics for helicopter controls are defined in the following documents: MIL-STD-250, MIL-STD-1333, MIL-STD-1472, MIL-G-58087, MS 87017, and AFSC DH 1-3.

The pilot-operated lighting and emergency controls should be shaped and located so that an aircrew member reasonably familiar with their arrangement is able to operate them without visual reference — the so-called "blind position reaction".

All controls of like function should be grouped together, and normal operating and emergency controls should have priority of position.

Conformity to established custom or standards must be carefully reviewed in cases where unique mission requirements or helicopter design possibilities present an opportunity for a more nearly optimum cockpit arrangement through deviation from standards. Standards, in general, represent minimum requirements and the result of experience, but are not intended to obstruct progress nor to stifle initiative. However, the possibility of improvement through deviation from standard should be subjected to thorough analysis before approval is requested, and the designer should make full use of human factors engineering techniques in order to determine the value of any deviation from standard practice. Human factors engineering principles also *shall* be used to determine the locations of any controls not regulated by current standards.

Control panels, control knobs, handles, or levers other than emergency controls and integrally illuminated controls should be black to minimize reflections. Control panels *shall* be designed in accordance with MIL-C-81774. All emergency switches, buttons, handles, knobs, and levers which require immediate corrective action by the operator in the event of an emergency should be identified with alternate orange-yellow and black stripes in accordance with MIL-M-18012.

The edges of adjacent circular nondetent knobs should be at least 1 in. apart. Knob diameters usually are determined by torque and setting requirements. Knobs of less than 0.5-in. diameter should be used for low torque application only. Knobs with diameters of more than 0.5 in. may somewhat reduce chances of error, but increase the demand for control panel area.

13-2.1.2.1 Pitch Controls

Separate collective and cyclic pitch controls should be provided for pilot and copilot. The characteristics of force vs displacement for these controls *shall* be in accordance with par. 6-3.6, AMCP 706-201. An adjustable friction device or irreversible mechanism should be incorporated in the pilot's collective pitch control.

Subsystem functions controlled from the cyclic stick *shall* be in accordance with MS 87017 and MIL-G-58087. Among the functions to be controlled from this location are lateral and longitudinal trim, microphone (radio or ICS), weapon firing, cargo or rescue winch, and cargo hook release. Subsystem function controls installed on the collective control head should be located in accordance with Ref. 4. Examples of cyclic and collective control grips incorporating subsystem controls are shown in Figs. 13-3 and 13-4.

13-2.1.2.2 Directional Control Pedals

The locations and ranges of motion of directional control pedals are specified in MIL-STD-1333. Pedals *shall* be 6 in. minimum width. If mechanical adjustment of the pedals is provided, it should be located under the instrument panel and along the centerline of the crew station. If a rotary pedal adjustment is used, the motion of control to extend the pedals should be clockwise The force displacement characteristics of the pedals *shall* be in accordance with par. 6-3.6, AMCP 706-201.

13-2.1.3 Seats, Belts, and Harnesses

13-2.1.3.1 Crew Seats

All crew member seats *shall* be positioned for ease of access to the seat, the helicopter exit, items such as chart boards, and necessary equipment controls. The pilot and copilot seats *shall* be easily adjustable both horizontally and vertically. The body contact areas of the seat *shall* be of open mesh construction or fabricated from permeable material. For helicopters of the utility class and larger, the arrangement of the seats *shall* be such as to facilitate inflight removal of an injured crewmember. Seats for pilots, navigators, engineers, radio and radar operators, and electronic

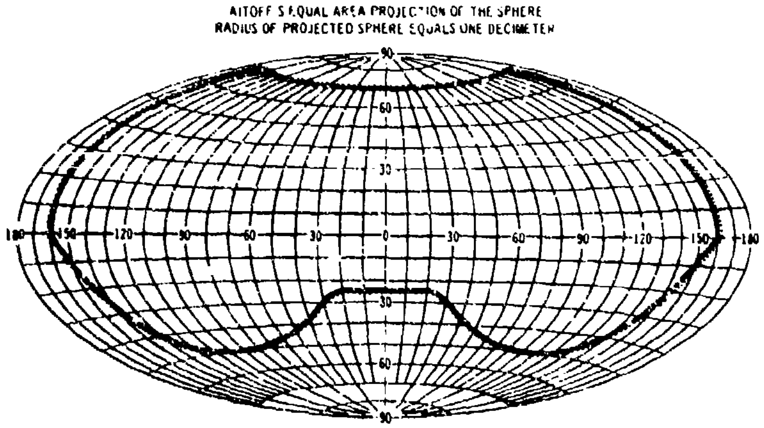


Figure 13-1. Single Pilot/Tandem-pilot Helicopter Vision Plot

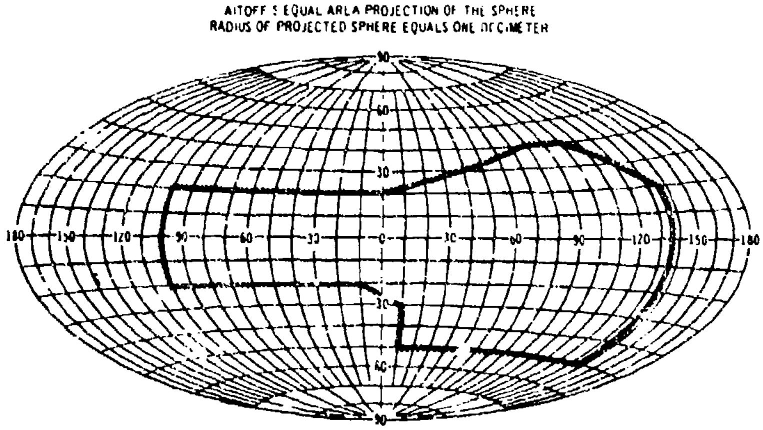


Figure 13-2. Side-by-side Helicopter Vision Plot

countermeasure crew members should conform to MIL-S-58095. The crew should be located to minimize danger from rotors, propellers, and turbine blades.

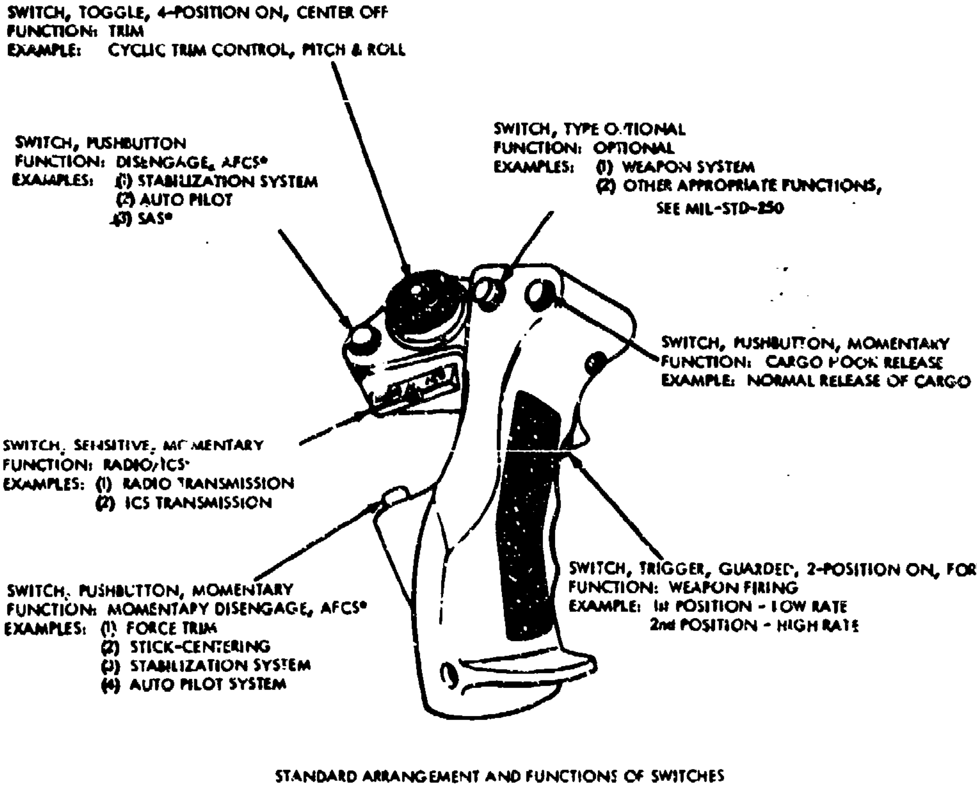
Critical dimensions of the seat *shall* conform to MIL-STD-1333 and MS 33575. Requirements for structural strength and controlled deformation are given in MIL-S-58095 and MIL-STD-1290. Crash-force attenuation *shall* be accomplished by plastic deformation of the seat structure, by load-limiting devices, or by a combination of the two methods.

Seat armor may be required. Design considerations for seat armor are given in par. 14-3.3.

13-2.1.3.2 Belts and Harnesses

Safety belts and harnesses should be installed for aircrew, troops, and passengers. The crew restraint harness *shall* include a lap belt, side strap, lap belt tie-down strap, two shoulder straps, and a single point of attachment-release with a single-action-release buckle in accordance with MIL-S-58095. Restraint devices must provide sufficient freedom of movement to permit crewmen to manipulate aircraft controls. The release mechanism should prevent accidental or unintentional release, but should also facilitate emergency release of injured crewmen, i.e., it should be possible to release the harness with one finger while tension equal to the occupant's weight is supported by the harness. It *shall* be possible for the seat occupant to make strap adjustments easily, with either hand. Ref. 5 describes the development and test of a restraint system to meet the design criteria given by Ref. 3.

For those crew members whose duties require them to stand in open doors or windows during flight, a retaining harness should be installed.


SWITCH, TOGGLE, 4-POSITION ON, CENTER OFF
FUNCTION: TRIM
EXAMPLE: CYCLIC TRIM CONTROL, PITCH & ROLL
SWITCH, PUSHBUTTON
FUNCTION: DISENGAGE, AFCS*
EXAMPLES: (1) STABILIZATION SYSTEM
(2) AUTO PILOT
(3) SAS*
SWITCH, TYPE OPTIONAL
FUNCTION: OPTIONAL
EXAMPLES: (1) WEAPON SYSTEM
(2) OTHER APPROPRIATE FUNCTIONS, SEE MIL-STD-250
SWITCH, PUSHBUTTON, MOMENTARY
FUNCTION: CARGO HOOK RELEASE
EXAMPLE: NORMAL RELEASE OF CARGO
SWITCH, SENSITIVE, MOMENTARY
FUNCTION: RADIO/ICS*
EXAMPLES: (1) RADIO TRANSMISSION
(2) ICS TRANSMISSION
SWITCH, TRIGGER, GUARDED, 2-POSITION ON, FORWARD OFF
FUNCTION: WEAPON FIRING
EXAMPLE: 1st POSITION - LOW RATE
2nd POSITION - HIGH RATE
SWITCH, PUSHBUTTON, MOMENTARY
FUNCTION: MOMENTARY DISENGAGE, AFCS*
EXAMPLES: (1) FORCE TRIM
(2) STICK-CENTERING
(3) STABILIZATION SYSTEM
(4) AUTO PILOT SYSTEM
STANDARD ARRANGEMENT AND FUNCTIONS OF SWITCHES

*AFCS - Automatic Flight Control System
*ICS - Intercommunication System
*SAS - Stability Augmentation System

Figure 13-3. Example of Cyclic Control Grip

13-2.1.4 Map and Data Cases

Map and data cases *shall* be located in accordance with MIL-STD-250 so that the crew member can reach and locate the cases without having to divert his attention from his displays and controls.

13-2.2 PASSENGER COMPARTMENT

The requirements of the mission influence seat design requirements. Historically, troops have been transported from point to point with little consideration given to their comfort or protection. Less than maximum combat-effectiveness often resulted, as well as lower survival probabilities for the passengers in case of a crash or rough landing. Therefore, improved techniques which deliver troops or cargo in condition to perform their specified mission must be a constant goal. When combat equipped troops are to be transported, compartment sizing and arrangement must be adequate to accommodate all gear and equipment.

13-2.2.1 Troop and Passenger Seats

The basic design criteria for troop and passenger seats are safety, comfort, and ease of ingress/egress. Trade-offs involving these criteria and considering the helicopter mission and configuration are necessary to determine seat orientation, i.e., forward or rearward facing seats. Ease of ingress and egress under all situations is assisted by simple, reliable restraint systems and flush stowage of the restraint system to prevent entanglement. The design must consider the requirement for ease of fastening and releasing the restraint system.

Suggested configurations for troop seats are provided by Ref. 6. Design requirements for crashworthiness and for restraint system design is given by MIL-STD-1290. Troop seats should fold and be capable of being stowed in a minimum volume, so as not to interfere with the transport of cargo.

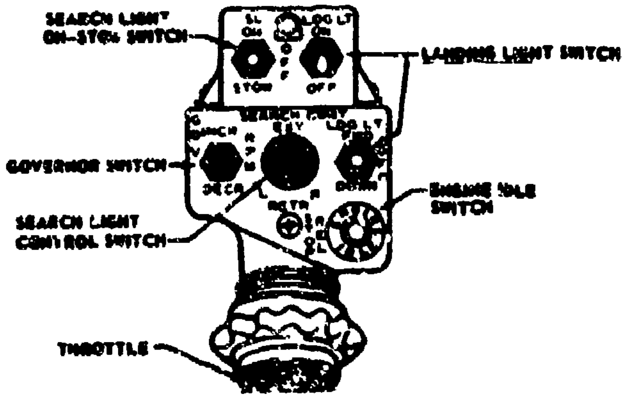


Figure 13-4. Example of Collective Control Head

Troop seats are to be equipped with lap belts and consideration should be given also to providing shoulder harnesses for the occupants of all forward or sideward facing seats. The lap belts should retract automatically to improve the ease of ingress and egress. The belts should adjust automatically to torso size without need for additional fittings, and locking and unlocking of the lap belt should be possible using only one hand.

13-2.2.2 Color

Interior color schemes *shall* be in accordance with TB 746-93-2 .

13-2.2.3 Upholstering and Carpeting

The choice of upholstering and carpeting is, to some extent, mission-dependent. Consideration should be given to availability, weight, resistance to combustion, wear, sunlight, oils and greases, and other degrading influences and stresses. Coated and simulated leather fabrics are suitable materials. Foam rubber per MIL-R-5001 and polyurethane foam per MIL-P-26514 are acceptable for US Army seating applications.

Seat cushions should minimize occupant submarining and dynamic overshoot. Net-type seat cushions may be used if they prevent contact between the occupant and the seat pan under design vertical loads, and if their rebound characteristics limit occupant return movement from the point of maximum deformation to 1.5 in. or less. Cushions should be so contoured so as to avoid constrictions or localized pressures that reduce comfort or inhibit body circulatory functions. Body contact portions of the seat back and seat cushion should be permeable.

13-2.2.4 Smoking Provisions

In an area in which smoking is permitted, removable self-contained ash trays are necessary. The flight compartment ash trays must be flush-mounted and preferably accessible to the crewman's left hand. Compartments where smoking is not permitted *shall* be designated by an appropriate placard.

13-2.2.5 Signal Lights and Alarm Bells

Crew station signals are governed by MIL-STD-411. Caution and warning signals are discussed in par. 10-2.2.

13-2.2.6 Aeromedical Evacuation

When required, the installation of the aluminum pole type of litters should be provided. A minimum vertical separation between litters of 18 in. is required. The entire installation should conform to the strength requirements of MIL-S-8698. MIL-STD-1290 provides design guidance for providing adequate crashworthiness in the litter installation. Litters may be installed either longitudinally or laterally. Care must be taken to provide adequate crashworthiness and unless the litter configuration is capable of providing restraint, the litter straps must be capable of restraining the patient against the survivable crash loads. A lateral orientation of the litters is indicated as preferable. Ease of litter loading and in-flight accessibility by medical attendants are important considerations. A litter lift device is generally preferable to adjustable litter support strap assemblies if combat evacuation of litter patients is required. Fixed litter support strap assemblies should be used only when required by the procuring activity.

The maximum dimensional requirements for aluminum pole litters are shown in Fig. 13-5. All other pole litters used by the services have the same length of pole, distance between poles, and length of canvas.

Oxygen lines or storage for portable oxygen units — for the use of all litter patients, medical personnel, and other personnel — should be provided.

Electrical outlets should be provided in order to operate electrical medical equipment and provide sockets for traveling loads.

As specified by the procuring activity, foldaway desk, medical chest and equipment storage space, intercom equipment for two-way communication between pilot and medic, and a loudspeaker system for giving emergency instructions to evacuees also should be provided.

13-2.3 SURVIVAL EQUIPMENT

13-2.3.1 Inflight Escape and Survival Equipment

Provisions for inflight escape and survival depend on the performance capability of the helicopter, the nature of the mission, configuration of the helicopter, crew training, and equipment provided. For most helicopters, the autorotative characteristics of the helicopter and the low altitude at which most missions are flown make inflight escape undesirable if not impossible. Therefore parachute provisions are

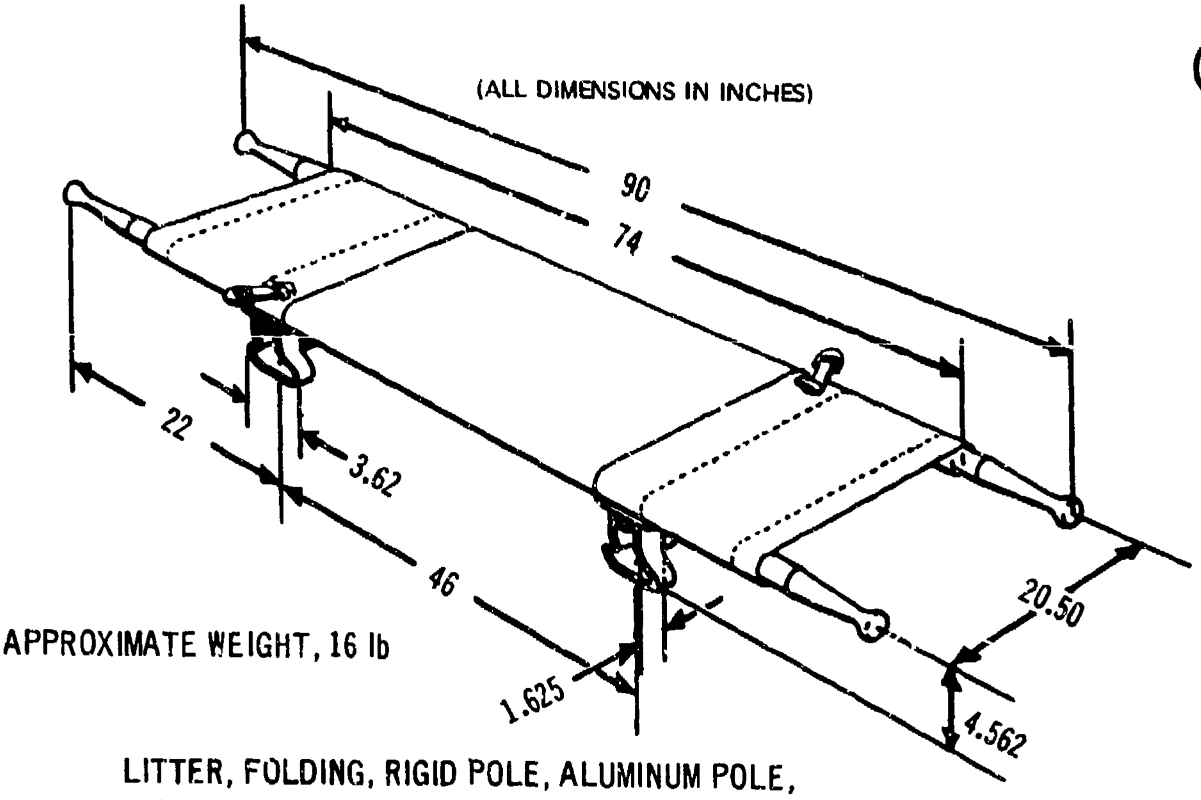

(ALL DIMENSIONS IN INCHES)
90
74
22
3.62
46
20.50
1.625
4.562
APPROXIMATE WEIGHT, 16 lb
LITTER, FOLDING, RIGID POLE, ALUMINUM POLE,
FEDERAL SUPPLY CATALOG NO. 6530-783-7905

Figure 13-5. Rigid Pole Litter

not required unless specified by the procuring activity. If feasible, the role of inflight emergency escape systems is to deliver the crewmen and passengers — with necessary survival equipment — to the earth's surface in such physical condition that they can perform the actions required to survive, evade capture, and take action necessary (such as the establishment of signals) to aid in rescue operations. Although inflight escape systems are not now operational, research and experimentation have been undertaken toward the development of such systems. An example of such an effort is the HEPS program of the US Navy. In this program a pyrotechnic device permits separation of the canopy and rotor system from the helicopter and extraction of the crew if autorotation is impossible.

13-2.3.2 Ground Escape and Ditching Provisions

Because of the variety of emergency situations that can occur, the design of emergency egress facilities and ditching capabilities is a difficult problem. Among the primary design parameters are the number, size, strength, and location of emergency doors; the use of slide ropes, slide poles, and ground evacuation slides; the possibility of explosively created exits; and performance features that maximize chances for successful ditching.

The scope of appropriate ground and ditching provisions will have been defined during preliminary design (see par. 13-3.2.1, AMCP 706-201). Mission requirements largely dictate the size and shape of the helicopter, as well as its basic performance characteristics. These in turn help to establish the nature and likelihood of the possible emergency environments. An analysis of the characteristics of the helicopter and the emergency environments can be used to create a weighted checklist of required ditching and emergency-egress provisions. Detail design requires that these provisions be continuously reviewed and re-optimized as adjustments or improvements in subsystems are made. Detail considerations include design of doors or escape hatches to minimize possibilities of jamming due to crash deformation of crew or passenger compartments. Design guidance is provided by Ref. 3. Included are recommendations regarding operation of emergency exit closures and required markings for emergency exits.

13-2.3.3. Emergency Lighting Provisions

Emergency lighting provisions should be installed independent of the craft electrical system in the passenger or cargo compartments. Preference should be given to an emergency lighting system that is self-contained, explosionproof, waterproof, and operable by a combination manual/inertia type switch. One emergency lighting unit should be provided at or near each emergency exit. Additional criteria and design guidance are provided by Ref. 3.

13-2.3.4 Life Rafts

The type and number of life rafts, if required, will be specified by the procuring activity. Design guidance for life raft installation directions is given in MIL-R-9131 and MIL-L-5567.

13-2.3.5 Survival Kits

Army aviators usually wear survival vests, which are personal equipment and for which the designer has to insure weight allowance, restraint capability, and absence of control or reach interferences. In addition to this, space provisions for survival kits *shall* be provided.

13-2.3.6 First Aid

The type of first aid kit will be specified by the procuring activity. Selection of the location of the first aid kit should be based upon the requirements of inflight access as well as easy access in case of emergency egress.

13-2.3.7 Fire Extinguishing Systems and Axe

The differences in size and configuration of helicopters preclude specifying a standard location for portable fire extinguishers. The portable fire extinguishers should be located in such a way that they are readily accessible to the crew members at their normal duty stations. Hand fire extinguishers may be mounted either vertically or horizontally. The extinguishers must not be mounted over or behind the heads of crew members when they are positioned in their normal or emergency stations. Each cargo or litter compartment should be provided with a minimum of two extinguishers at each end of the compartment and near entrance doors if it is practicable. Fire extinguishers for use in occupied areas *shall* use nontoxic agents. Hand emergency fire axes *shall* be provided as specified by the procuring agency.

13-2.4 ENVIRONMENTAL CONTROL

13-2.4.1 Ventilation, Heating, and Cooling

Ventilation, heating, and cooling requirements are defined in par. 13-3.2.3.2, AMCP 706-201. During detail design the specific environmental control unit must be selected, air duct routing defined, and the type and location of the air discharge ports defined. The location and controllability of the air outlets *shall* be adequate to meet the 10-deg maximum temperature spread in both the heating and cooling modes. At least one outlet for each crewman *shall* be located to provide variable airflow over the head and chest. This feature significantly increases comfort in the ventilating and cooling modes. Detail design guidance on environmental control and air distribution systems can be found in Refs. 19 and 20. An example analysis of heating and ventilation requirements is presented in Appendix A.

There should be a comprehensive investigation of the possible toxic elements from all of the materials that go into the construction of the aircraft. Consideration must be given to the removal and the detection of any toxic elements which may enter into or be generated within the crew compartments or the cockpit. As new materials are developed for use in the construction of the aircraft, they must be tested to determine possible toxicity. Since it is impossible to present a comprehensive, up-to-date list of toxic agents which may find their way into the cockpit, it is necessary that each design program consider toxic hazards.

13-2.4.2 Windshield Defogging and Deicing Equipment

The windshield defogging and deicing equipment should be provided to meet the requirements of the altitude, thermal, and weather conditions required by the mission profile of the helicopter. Detail requirements are given by MIL-T-5842. Automatic and manual override controls should be provided as appropriate. In the case of anti-icing through chemicals such as alcohol or other toxic substances, the design should be such that the fumes or liquid will not enter the cockpit or passenger compartments and thus adversely affect the crew or passengers.

13-2.4.3 Acoustical Environment

The noise within occupied compartments *shall not* be in excess of the maximum allowable levels prescribed in MIL-STD-1474, MIL-STD-740, or MIL-A-8806, as applicable. MIL-S-6144 provides the general specifications for the soundproofing of aircraft.

Special attention must be paid to techniques for reducing noise level at its source, and to the use of special materials and techniques for insulating against acoustical noise. Reference should be made to par. 13-3.2.2, AMCP 706-201, for information on maintaining noise levels within acceptable levels.

Auxiliary systems that normally operate for longer than 5 min should not produce an increase in noise levels in occupied compartments above that specified in MIL-A-8806.

Special missions which may require noise levels lower than those required by the general Military Specifications should have the requirements so stated in the detailed procurement specification and special care should be taken to insure that these levels are not exceeded.

13-2.5 SIGHTS AND SIGHTING STATIONS

Provisions must be made for safe, efficient positioning of sights and sighting stations required for direction of helicopter weapon systems. Design guidance for both direct and indirect sights follows.

13-2.5.1 Direct-viewing Sights

Direct-viewing sights may be fixed or slewable. Fixed sights generally are provided to permit pilot operation of fixed weapons, flexible weapons in the stow position, and rockets.

Fixed sights should not restrict pilot movement or field of view during normal flight conditions. Furthermore, the fixed sight must not interfere with emergency exit crew movements. The sight may be folded out of the operator's field of view until required. Sufficient clearance must be provided to permit the pilot (with gloves) to place the sight into its operating position and to perform any required hand-operated reticle adjustments.

A direct-viewing flexible pantograph sight may also be provided at the gunner's station. It generally is supported from the aircraft structure and is hand directed by the gunner. Alternately, the sight may be fixed to, and rotate with, the gunner's seat during target tracking and firing operations. The helicopter designer must insure that adequate clearance is provided for the gunner's head, hands, and other extremities to permit unrestricted movement of the sighting station installation.

Airframe and cockpit surfaces within the operator's f [illegible] iew should have dull (antiglare) finishes to prevent reflection and eye strain. The canopy enclosure (windshield) within the line-of-sight envelope should contain as few areas of curvature and thickness variations as possible to minimize optical refraction and distortion. That portion of the windshield within the sight-aiming envelope should have adequate defogging and deicing provisions.

Direct viewing sights have been a major source of serious or fatal head traumas in crashes in US military aircraft during World War II, Korea, and Vietnam. It is essential that all sighting devices be designed to minimize the potential for injury. Safety factors to be considered are:

1. Capability for instant removal, jettisoning, or storage during emergency periods should be provided.

2. Stowed sight positions should not create ad-

ditional hazards to other crew personnel in the event of emergency.

3. Adequate stowed tiedown strength should be provided to prevent the sight from rebounding during impact.

4. Jettisoned or removed sights should not represent lethal missile hazard in the event of a crash.

5. The guidance contained in Ref. 3 (Chapter 6) should be followed.

13-2.5.2 Helmet Mounted Sight

The helmet mounted sight is another type of direct-viewing optical sight currently used for helicopter applications. The same clearance considerations apply to this installation as to the aircraft-mounted flexible sight. Adequate head clearance must be provided between helmet projections and the aircraft structure for normal flight operation and for weapon control within the required azimuth and elevation flexibility ranges. Quick-disconnect provision must be made to allow dislodging of the helmet from its electrical or mechanical connection with the aircraft; provisions also must be made enabling such disconnection to be a "single-hand" operation.

One type of helmet sight incorporates a mechanical linkage to detect head motions and sum the aiming information. Other types of helmet sights use light sensors, sonic sensors, and electromagnetic fields to perform the same function.

Another type of direct sight is the periscopic sight which views the target through a mirror arrangement and permits the sight head to be located externally and in locations that allow unobstructed line of sight within a large envelope of azimuth and depression angle limits. The optical equipment may be supported from the aircraft structure, or may constitute an integral part of rotating gunner's station. Periscope installations of either type are sizable and require adequate airframe support structure. Power requirements for a flexible gunner's sight station may be significant and may require early consideration of sighting station cable or hydraulic line sizing and routing.

13-2.5.3 Indirect Sights

Indirect sights receive the target image from electronic sensors and project it to the observer by means of a panel-mounted or helmet-mounted display. The ability of the observer to detect and identify targets with an indirect sight is dependent on display resolution, contrast, number of shades of grey, display size, and eye-to-panel distance. Detail design guidance for indirect view sights and other CRT displays is available in Ref. 21.

13-2.5.4 Missile Sighting Stations

Sighting stations for missile installation generally will employ the sight installation used for guns and rockets. Provisions may be required for installation of retractable optical filters (in the gunner's line of sight) for use when the missile exhaust light intensity is sufficient to cause temporary blindness or a blurred view of either the target or the guidance references. Sight alignment with the aircraft datum plane may be obtained electrically or mechanically, and generally requires provisions for adjustment.

13-3 LIGHTING SYSTEMS

13-3.1 EXTERIOR LIGHTING SYSTEM

The exterior lighting system design may include an anticollision light system, formation lights, landing/taxi lights, searchlights, floodlights, and position lights. These are discussed subsequently. MIL-L-6503 provides specific design requirements for exterior lighting.

13-3.1.1 Anticollision Light System

Unless otherwise specified, the Army developed day/night anticollision light system *shall* be provided. This system is identified as Light Set, Navigation (AABSHIL) and provides a white daytime strobe of 3500 effective candle-power and a red night time strobe of 150 effective candle-power. This system is available in the following two types:

1. Type I (28 V) NSN 6620-00-361-0646
2. Type II (115 V) NSN 6620-00-361-0614

Field of coverage *shall* be as specified in MIL-L-6503.

13-3.1.2 Formation Lights

For helicopters, formation flying lights usually fall into two categories: fuselage formation flying lights and rotor tip lights.

Fuselage formation flying lights *shall* be so arranged that adjacent aircraft can fly in either stepped-up trail formation or vee formation by alignment of lights. Durable electroluminescent panels easily can be adapted for fuselage lighting. Refer to the specific aircraft system specification for detail requirements. A test installation is required to verify optimum system design.

Rotor tip formation flying lights should be considered on helicopters because the rotor blade usually extends beyond the fuselage, and movement of the rotor disc provides an indication of an impending maneuver. Due to the high centrifugal loads of rotor tips, multiple white lamps should be used minimizing filament and vibratory angle. Some type of slip ring normally will be required, as well as wiring in the rotor blade itself. Five different light intensity levels have been found satisfactory. Formation lights must not be visible from the ground, and should be further shielded when practical to be visible only from behind and at the same altitude and slightly above the lead aircraft.

13-3.1.3 Landing/Taxi Light

A 600-W or 1000-W retractable landing light (1000-W preferred) *shall* be installed. For small helicopters with limited power systems, lights of less wattage may be acceptable. MIL-L-6503 requires that this light be slewable from 20 deg above to 60 deg below the normal level flight position of the aircraft. For those helicopters having limited lower nose space available, the 20-deg light above normal level flight position may be difficult to meet, and a deviation should be requested.

13-3.1.4 Searchlight

A 450-W controllable searchlight, in accordance with MIL-L-6503, *shall* be provided unless the detail specification requires a larger searchlight, i.e., 600-W or 1000-W.

13-3.1.5 Floodlight System

Some detail specifications for rescue helicopters may require a floodlight system in addition to the landing lights and searchlights. The ground area to be illuminated and the helicopter altitude when using the floodlight system should be determined prior to the design or during a lighting mock-up.

13-3.1.6 Position Lights

All helicopters *shall* be equipped with fuselage side position and tail lights as defined in MIL-L-6503.

13-3.2 INTERIOR LIGHTING SYSTEM

The interior lighting system design may include cabin and compartment lighting, cockpit lighting, panel lighting, interior emergency lighting, portable inspection lights, troop jump signal lights, worktable light, warning, caution, and advisory lights, and instrument panel lighting. The applicable Military Specifications for the interior lighting system are MIL-L-6503, MIL-P-7788, MIL-L-5667, MIL-L-27160, and MIL-L-25467.

13-3.2.1 Cabin and Compartment Lighting

Cabins and compartments *shall* be provided with suitable lighting for passengers and crew. These lights *shall* be installed so that their direct rays are shielded from the pilot's eyes, and so no objectionable reflections are visible to the pilots. In aircraft where dark adaption is required, these lights *shall* be capable of providing both red and white illumination with separate dim controls in the cabin area. The required levels of illumination are tabulated in MIL-L-6503.

13-3.2.2 Cockpit Lighting

Cockpit lighting *shall* provide illumination sufficient to enable crew members to ascertain readily indicators and switch positions. A cockpit dome light, with controls accessible to both pilot and copilot, will normally meet this requirement. The dome light *shall* be dimmable and provide either red or white lighting.

13-3.2.2.1 Utility Lights

MIL-L-6503 gives applicable design requirements for cockpit utility lights. For most helicopters, one light is installed for each pilot.

13-3.2.2.2 Secondary Lighting

Secondary lights *shall* be installed in the instrument glare shield to provide dimmable red and white illumination for supplementary and thunderstorm lighting. These lights, connected to the essential bus, *shall* be in accordance with MIL-L-18276. Utility lights may suffice as a secondary light source in certain cockpits if they can be located to illuminate essential instruments while remaining readily accessible to be used as utility lights.

13-3.2.3 Panel Lighting

Control panels *shall* be sufficiently lighted to permit easy and accurate reading of the information contained thereon. Integrally illuminated panels *shall* be provided in accordance with MIL-P-7788 when a red lighted cockpit is specified, or in accordance with MIL-P-83335 when Air Force blue-white light is specified.

13-3.2.4 Interior Emergency Lights

An interior emergency lighting system when required *shall* be in accordance with MIL-L-6503. Design innovation radioactive luminous lighting panels may fulfill some of the requirements of MIL-L-6503 for emergency lighting.

13-3.2.5 Portable Inspection Lights

MIL-L-6503 requires that each helicopter be equipped with a hand-held scanning light. The light covered by MIL-L-7569 is approved for this application. Outlets *shall* be provided in the cockpit and crew compartment to permit the required inspections during hours of darkness.

13-3.2.6 Troop Jump Signal Light

As defined in MIL-L-6503, when required by the procuring activity, a troop jump signal light *shall* be provided.

13-3.2.7 Warning, Caution, and Advisory Lights

The warning, caution, and advisory lighting system is discussed in par. 10-2.2.

13-3.2.8 Instrument Panel Lighting

Instrument panel lighting is discussed in par. 10-2.

13-3.2.9 Cargo Compartment Lighting

Cargo compartment lighting should consist of, at the minimum, two rows of flush-mounted ceiling floodlights located along the edges of the ceiling. It also is advisable to provide external lighting in the general area of the ramp and other doorways, to facilitate night loading and enhance safety. A row of lights along the base of the side walls provides the illumination required for the rigging of tiedowns on vehicle frames and undercarriage.

13-4 CARGO PROVISIONS

13-4.1 INTERNAL CARGO

The provisions of this paragraph are applicable to helicopters having an all-cargo or combined cargo/passenger-carrying mission, with the cargo carried within a fuselage compartment or within a pod separate from the basic helicopter airframe. Baggage compartments, incidental cargo provisions, and equipment stowage bins should conform to the cargo compartment design criteria wherever possible.

Army cargo can be grouped into two classes. Class A cargo includes unpacked items and loose boxes smaller than a 3-ft cube which can be restrained by a net or similar device. Class B cargo includes larger, single-unit loads such as vehicles, artillery, and fuel barrels, which are secured individually within the cargo compartment. Incidental cargo, which falls into neither class, includes items such as spare parts, flight bags, or mission-oriented equipment carried onboard by passengers — e.g., tool kits, ammunition boxes, and weapons.

13-4.1.1 Cargo Compartment Layout

The basic envelope dimensions of the cargo compartment are established early in the preliminary design process. The detail designer must keep the various helicopter structural and mechanical components from intruding into this envelope as the design develops. Compartment walls and ceilings must remain unobstructed; even a minor protrusion can decrease the usable cargo volume considerably. Any structures or components which project unavoidably into the compartment must be marked conspicuously. In all cases, the protruding component must be suitably protected against impact from cargo and vehicles.

With the overall dimensions of the cargo floor determined by preliminary design, the cargo tiedown points can be located. These points must be arranged in a basic 20-in. grid pattern. Such a grid pattern has been standardized internationally (AFSC DH 2-1) and is shown in Fig. 13-6.

The requirement that cargo tiedown fittings be located on 20-in. centers must be considered early in the structural design process, since this is a major factor in the location of fuselage frames (see Chapter 11).

Helicopters which have the capability of hauling vehicles should have strengthened treadway areas on the floor, located to coincide with the wheel locations of all Army vehicles which might be transported by the helicopter. This subject is discussed in Chapter 11.

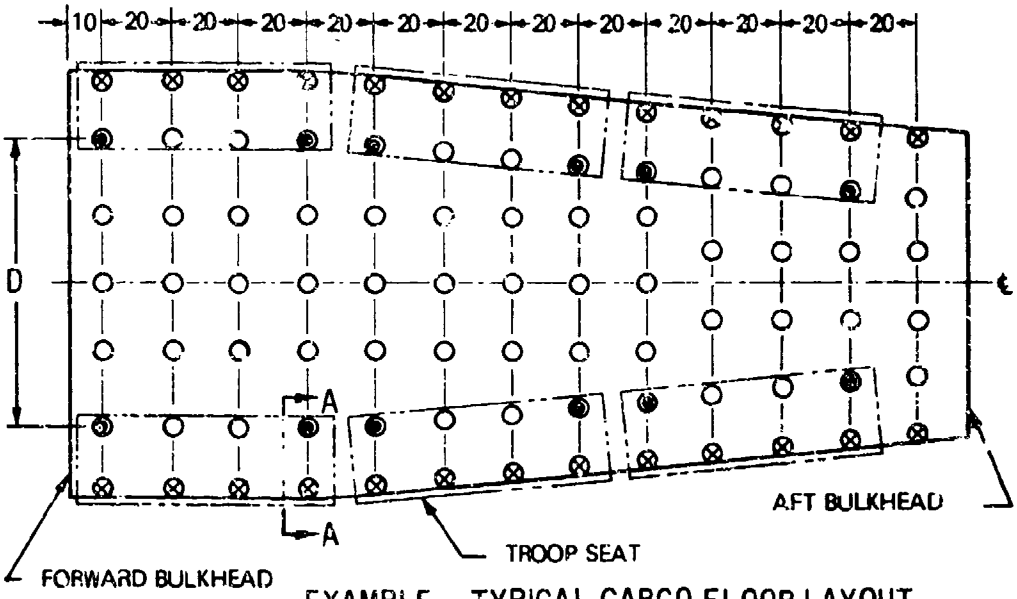
The main cargo door should have an aperture equal to the size of the cargo compartment, with the door sill on the same plane as the floor. Forward or aft doors should open clear of the extended wall and ceiling planes, to permit straight-in loading of the highest load which will fit into the compartment. In addition to the main door, at least one smaller, secondary door should be located at the opposite end of the cargo compartment, to allow access when the compartment is filled with cargo. If the secondary door is on the side of the fuselage, it should be located on the right-hand side of the helicopter to facilitate the pilot's surveillance of loading operations.

If required for a cargo hoist (Chapter 11), a hatch at least 30 in. square should be located on the centerline of the floor, at the approximate location of the helicopter CG.

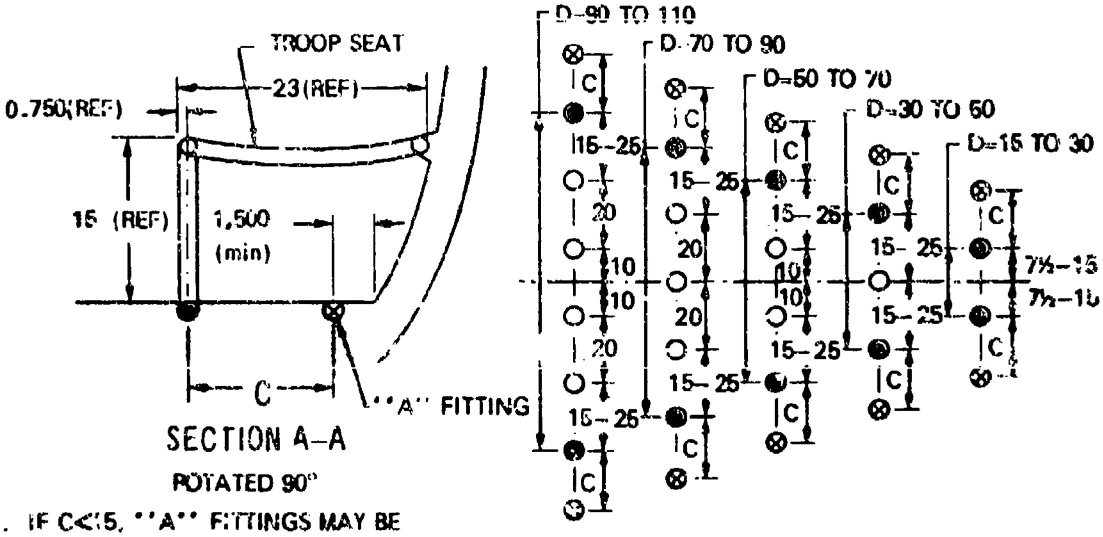
13-4.1.2 Detail Design

The cargo floor should be sufficiently flat and smooth to insure that boxes and fabric containers can be slid across the floor without being snagged or damaged. Tiedown fittings, floor panel fasteners, and seat/litter anchor studs must be flush with or recessed below floor level. Tiedown rings which cannot be installed flush with the floor must be made removable when not in use. Since they present a storage problem and are a general inconvenience, removable tiedown rings should be considered only as a last resort.

(ALL DIMENSIONS IN INCHES)


10
20
20
20
20
20
20
20
20
20
20
20
20
D
℄
A
A
FORWARD BULKHEAD
TROOP SEAT
AFT BULKHEAD

EXAMPLE – TYPICAL CARGO FLOOR LAYOUT


TROOP SEAT
0.750(REF)
23(REF)
15 (REF)
1.500 (min)
C
"A" FITTING
D=90 TO 110
D=70 TO 90
D=50 TO 70
D=30 TO 50
D=15 TO 30
C
15–25
20
10
7½–15

SECTION A–A

ROTATED 90°

1. IF C<15, "A" FITTINGS MAY BE ELIMINATED.
2. IF 15≤C<20, "A" FITTINGS SHALL BE LOCATED 1.5 INBOARD FROM INTERSECTION OF FLOOR AND FRAME.
3. IF C≥20, "A" FITTINGS SHALL BE LOCATED 20 OUTBOARD OF SEAT LEG

DETAIL B

LATERAL GRID PATTERN DEVELOPMENT BETWEEN LONGITUDINAL LINE OF SEAT LEG FITTINGS

● 5000 POUND FITTING LOCATED BY PLACEMENT OF TROOP SEAT LEGS

○ 5000 POUND FITTING LOCATED BY LATERAL PATTERN DEVELOPMENT PER DETAIL B

⊗ 10,000 POUND FITTING LOCATED FROM VIEW A–A

Figure 13-6. Standard Cargo Floor Tiedown Grid

Provision must be made for the inevitable damage which will occur occasionally to the floor during loading operations. The floor should be divided into panels that can be replaced readily during organizational maintenance. If it is not possible to make the load-bearing floor structure removable, then it should be protected from minor damage by replaceable covers. The floor should be made as stiff and puncture-resistant as possible, consistent with weight limitations. Material selection should be based on considerations of weight, stiffness, and resistance to corrosion, fire, moisture, and abrasion. Because of the relatively frequent replacement of floor panels, cost also is significant.

Because of the abrasive wear and severe environment to which the floor is subjected, the designer should pay particular attention to the potential effects of corrosion. Dissimilar metal joints in the vicinity of the tiedown fittings should be designed carefully, especially when the faying surfaces are subject to relative movement. The use of magnesium is not recommended for cargo floor components of tactical aircraft. If used, magnesium parts must be suitably protected, and shielded from any abrasion which could destroy the protective coating. If composite panels are used for the flooring, the core material must be highly resistant to water migration and fungus attack.

The floor should be covered with a nonslip material, such as the Type III matting described in MIL-W-5044, to provide good footing for personnel and good traction for vehicles. The cargo compartment walls and ceiling should be lined with protective paneling. This material should be as light as possible and should serve to protect the fuselage structure from minor damage and wear, furnish some thermal and sound insulation, and provide a smooth, snag-free wall. A commercially available Fiberglas-reinforced laminated sheet material is used widely as a cargo compartment liner. The thin material is constructed of parallel glass fibers bonded together in a cross-ply construction with epoxy resin. For additional insulation, Fiberglas batting material can be installed between the inner compartment liner and the helicopter skin.

All materials used for constructing and lining the cargo compartment must be fire-resistant. Any cargo compartment which is not accessible in flight must have a fire detector and remotely controlled fire extinguishing equipment. In any case, a means should be provided to seal off the cockpit from smoke and fumes originating in the cargo compartment.

Flight controls, critical mechanical components, and critical wiring and plumbing must not enter the cargo compartment. Any such components located immediately adjacent to the wall of the cargo compartment must be shielded properly to guard against the possibility that shifting cargo could deform the wall locally and jam the controls or damage the components.

13-4.1.3 Loading Aids

The detail design of the cargo compartment must include provisions for the handling of cargo during loading and stowage. The utility value of the helicopter will be affected greatly by the attention this area receives from the detail designer.

If cargo transport is one of the prime missions of the helicopter, an integral loading ramp is a necessity. This ramp should be at least equal in width to the cargo compartment floor, and should provide a continuous, smooth surface over this width, although it may be segmented into left and right halves for better conformance to rough terrain. It should have the capability of being adjusted to and locked at any height within the widest possible limits. Measured from its position when deployed on level ground, a minimum range of travel of −10 in. to +50 in. is recommended. The slope of the ramp when deployed on level ground should not exceed 13 deg. Allowance should be made for changes in helicopter floor height during loading. Jacks supporting the helicopter weight will eliminate such settling, but at the cost of considerable weight. Alternatively, a load limiter (see Chapter 11) should be incorporated into the ramp actuating mechanism, to allow the ramp to move upward as required when the helicopter settles under load. Without such a load limiter installed, the ramp actuator and its supporting structure must be designed to support the weight of the loaded helicopter.

Cargo helicopters must have power-operated ramps. Power for operating the ramp and associated doors must be of a type available from ground power units. Light observation and utility helicopters may use manually positioned ramps that can be remotely stored within the cargo compartment. Rear entrance ramps must be deployable in flight. Any doors operating in conjunction with these ramps must be capable of being opened in flight; alternatively, such doors may be removable on the ground, with their absence having no effect on structural integrity or flight characteristics.

The ramps should be strong enough to permit loading of the heaviest anticipated vehicular load, with only one corner of the ramp contacting the ground. The ramp may be strengthened locally to provide treadway areas at least equal in width and strength to the treadways within the cargo compartment. It is desirable, however, that the entire ramp have a uniform bearing strength, equal to that of the compartment treadways. Ramp edges and door sills must be designed to withstand the severe localized impact loadings encountered in cargo operations. In the absence of more specific design criteria, the ramp edges, door sills, and vertical door frames should be designed to sustain without damage a single randomly located load equal to the weight of the heaviest single item to be loaded, the load being applied to the structure through a 1-in. radius sphere.

Ramp extensions, if necessary, should conform to the strength requirements of the integral ramp and compartment treadway. Extensions should have a continuous width equal to that of the ramp. If this is not possible, the more narrow individual ramp extensions should be made reversible, with a smooth surface on one side for cargo loading and side flanges on the opposite side for guidance of wheeled vehicles.

A winch should be provided for the purpose of loading and unloading cargo from cargo helicopters. Although the winch should be located at the forward end of the cargo compartment, it may be located elsewhere provided that a suitable combination of blocks and pulleys can be arranged to guide the winch cable. As a minimum requirement, the winch should be capable of both pulling cargo on board and extracting cargo from the compartment to the ramp. Reversal of direction of pull can be accomplished by rerigging the winch cable over snatch blocks located on the ramp. Snatch blocks that can be attached to cargo tiedown fittings will greatly improve the flexibility and utility of the winch in shifting cargo within the compartment. All necessary blocks and pulleys must be provided as part of the basic equipment of the helicopter, and provisions for their mounting and storage must be included.

A desirable, but nonessential, secondary mode of operation for the winch is as a hoist within the cargo compartment, with the cable rigged over a ceiling-mounted pulley. If a floor hatch is available, the cargo winch can be used to a limited degree as an external load hoist. However, the designer should consider the conflicting requirements which this application imposes upon the cargo winch. A winch used only for cargo handling need not have high-speed capability. Thus, although force levels are quite high, the low cable speed tends to minimize power requirements, size, and weight. On the other hand, an external load hoist, to be of any value, requires a high reel-in rate under load, as well as a rapid deployment rate and such features as a high-capacity brake and pyrotechnic cable cutters.

The minimum usable cable length must be that required to retrieve a load which is 20 ft beyond the aft-most part of the helicopter (the tail rotor disk, or, in the case of the tandem-rotor helicopter, the aft main rotor tip). The winch system should have sufficient capability to haul a flat-bottomed package of a weight equal to the maximum payload up to the loading ramp and into the cargo compartment. On the assumption that the package is on metal skids, and the ramp is deployed on level ground, the use of snatch blocks and multiple purchase cable arrangements is recommended to achieve this maximum load capability, since a winch with a straight-pull capacity of such magnitude would be prohibitively large. The coefficients of friction shown in Table 13-1 should be used when determining winch capacity.

The selection of a power source for the cargo winch should take into consideration the fact that loading operations usually are conducted with the helicopter main power plant shut down, and power is supplied by either the onboard auxiliary power unit or a ground electrical power cart. Control of the winch should be by a remote pendant control with a cable long enough to permit the winch operator to move throughout the cargo compartment and ramp.

Table 13-2 lists the various tiedown devices which may be used to secure cargo within Army helicopters. Floor tiedown fittings must be compatible with all these devices. Provisions must be made for the storage of an appropriate number of these devices within the cargo compartment.

13-4.2 EXTERNAL CARGO

The once-novel practice of carrying cargo loads externally suspended from a helicopter has evolved into a standard operating procedure, especially in combat operations. In many cases, the cargo is sling-

TABLE 13-1. COEFFICIENTS OF FRICTION

MATERIAL	COEFFICIENT OF FRICTION
WOOD SKIDS ON ANTI-SKID FLOORING	0.60
WOOD ON WOOD	0.25 TO 0.50
WOOD ON METAL	0.20 TO 0.50
METAL ON METAL	0.15 TO 0.30
TRACKED VEHICLES	0.08
GREASED SURFACES	0.05 TO 0.08
WHEELED VEHICLES	0.03

TABLE 13-2. STANDARD CARGO TIEDOWN DEVICES

DEVICE	SIZE, ft	TYPE	RATING, lb	SPECIFICATION
NET, STEEL CABLE	9X9	A-2	10,000	MIL-T-9166
NET, WEBBING	15X15	MA-2	10,000	"
NET, WEBBING	15X20	MA-3	10,000	"
CHAIN, TIEDOWN	9	C-2	10,000	MIL-T-6480
CHAIN, TIEDOWN	9	MB-1	10,000	MIL-T-25959
STRAP, TIEDOWN	15	PC-1	5,000	MIL-T-865[illegible]
STRAP, TIEDOWN	15	A-1A	5,000	MIL-T-7181
STRAP, TIEDOWN	20	GCU-1/B	5,000	MIL-T-27260

loaded from a helicopter even though it could easily fit inside the cargo compartment of the same helicopter. Numerous advantages to this method:

1. The helicopter need not land, either to pick up or to release its cargo.
2. Loading time is minimized for the hookup, and unloading is nearly instantaneous. This reduces exposure to hostile fire.
3. Oversize cargo can be carried.
4. When using single-point suspension, loading and unloading have little or no effect on longitudinal CG position, and therefore recomputation of CG location is not required.
5. Cargo can be jettisoned to lighten the helicopter load prior to an emergency landing.

Generally, external cargo suspension configurations can be classified by the number of points through which the load is attached to the helicopter: single-point, two-point, and four-point suspensions comprise the large majority of installations (see Fig. 13-7). While three-point suspensions offer some advantages, they have seldom been used because of the unavailability of centerline hard points on existing helicopters, and because few loads, with the notable exception of artillery pieces, are configured for three-point pickup.

In addition to the discussion that follows, design guidance for external load systems is provided by Ref. 7.

Single-point suspensions are by far the simplest means of carrying cargo. Loose cargo can be carried in a net or on a pallet, with a minimum of preparation. There is no need to pack and secure the cargo carefully using high restraint factors as in the case with loads carried within the cargo compartment.

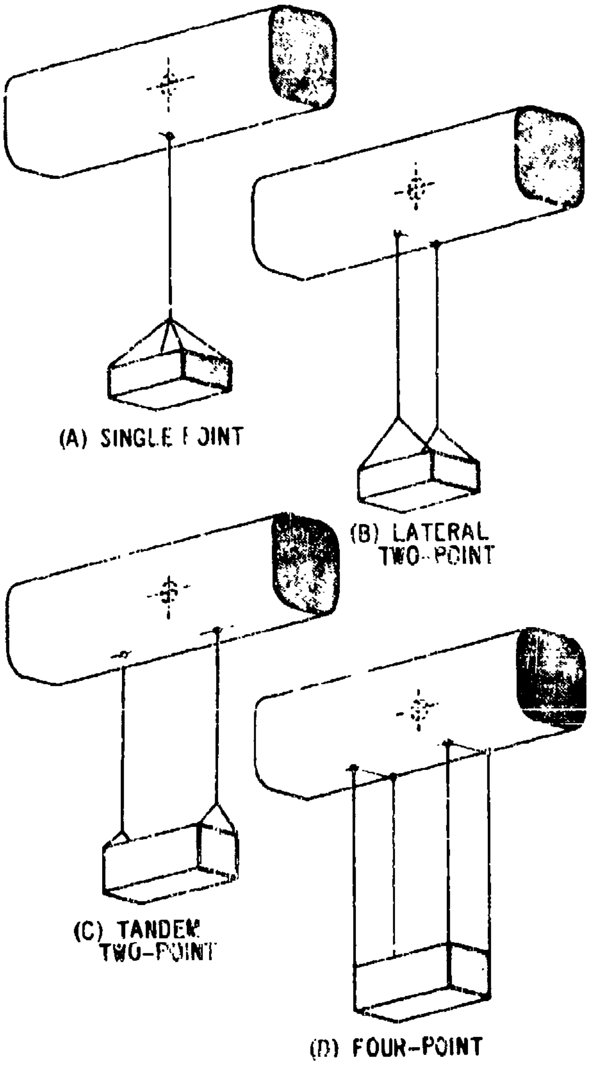
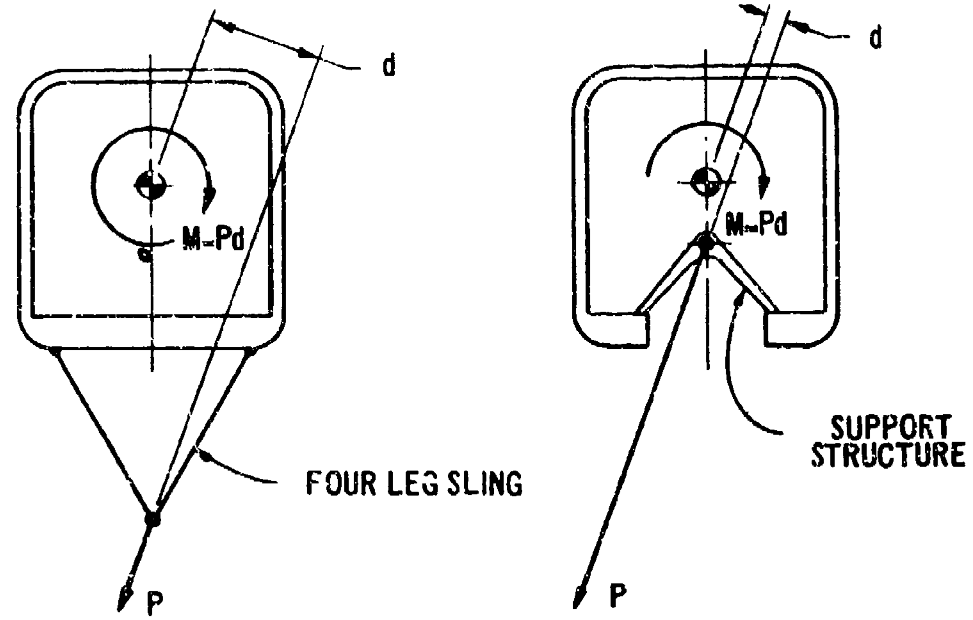


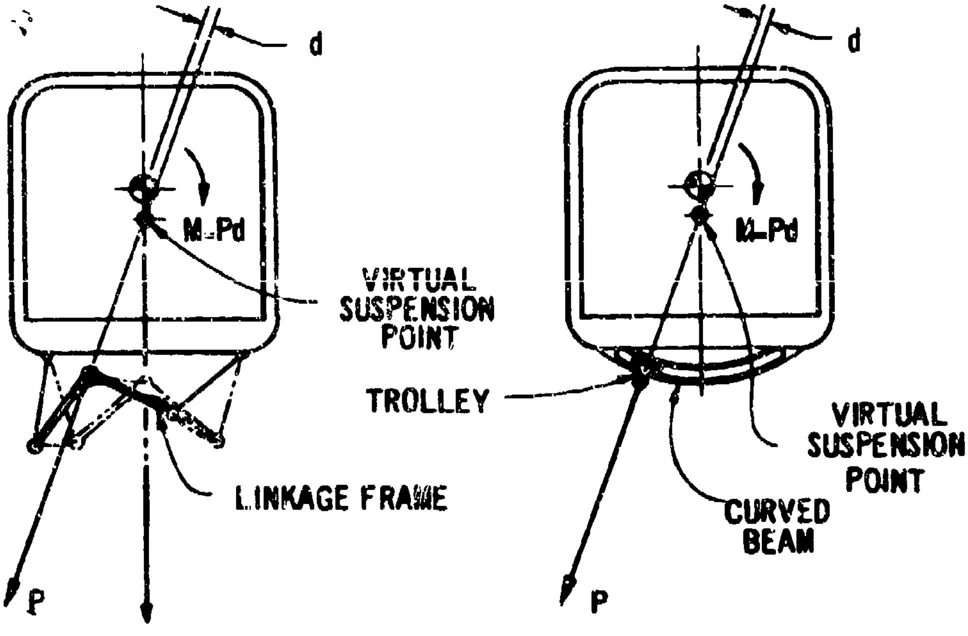
Figure 13-7. External Suspension, Basic Configuration

Most light helicopters and certain early cargo models achieve single-point capability by using a short four-member sling, the legs of which are anchored to an apex below the CG of the helicopter. The cargo is suspended from this apex. While this arrangement serves to spread the load into the airframe, allowing reduced structural weight, it has the undesirable side effect of moving the suspension point too far below the CG, with the result shown in Fig. 13-8(A).

As the load swings laterally and longitudinally, the weight vector pivots about the suspension point at the apex, imposing substantial upsetting moments upon the helicopter as it diverges widely from the


d
M=Pd
FOUR LEG SLING
P
d
M=Pd
SUPPORT STRUCTURE
P

(A) LOW SUSPENSION POINT (B) RELOCATED SUSPENSION POINT


d
M=Pd
VIRTUAL SUSPENSION POINT
LINKAGE FRAME
P
d
M=Pd
TROLLEY
VIRTUAL SUSPENSION POINT
CURVED BEAM
P

(C) FOUR BAR LINKAGE (D) CURVED BEAM AND TROLLEY

Figure 13-8. Methods of Raising the Suspension Point

location of the CG (Ref. 8). The solution to this unstable situation is to raise the suspension point to a location as close as possible to the CG of the empty helicopter. This can be done by physically rearranging the structure as shown in Fig. 13-8(B).

For most cases, including existing helicopters, it is more feasible to create a virtual suspension point by anchoring the cargo hook or pendant upon a linkage (Fig. 13-8(C)) or a trolley running on a curved track, which allows the pendant to move as though it were pivoting about a point near the CG (Fig. 13-8(D)).

The major difficulties with single-point suspensions arise from the fact that the suspension can provide only simple pendular stability to the load, and cannot provide any restraint or stability in yaw or pitch. An auxiliary line connecting the load to the helicopter would provide pitch and yaw restraint; however, this second line cannot be used because it would compromise the emergency release capability and possibly impose uncontrollable moments upon the helicopter (Ref. 9). Thus, the only means of providing stability to a single-point load is by using the aerodynamic forces generated by forward flight. A drogue parachute attached to the trailing end of a cargo load can provide the necessary restoring moment to keep an otherwise unstable load aligned with the direction of flight. However, the drogue loses its effect at very low speeds and in hovering flight, where rotor downwash can apply considerable rotational forces on certain types of loads.

There is one type of single-point suspension which offers a limited amount of incidental yaw restraint. If a hook is rigidly mounted to the airframe at the suspension point, and if the load is suspended by a short multileg sling with a ring at its apex, the ring interlocks with the hook to provide resistance to twisting. Load yawing moments tend to wind up the sling, which resists this windup with a torque proportional to yaw angle. The hook and its supporting structure must be adequate to carry this sling windup torque.

Symmetrical high-density loads can be allowed to rotate if such rotation has only a minor effect on helicopter flying qualities. Provision must be made for this rotation; a swivel must be mounted between the hook and the pendant to protect the pendant from damage due to excessive twisting.

The two-point suspension provides the yaw and pitch stability which single-point suspension lacks. The directional (yaw) restoring moment N of a two-point suspension, as a function of cable length, is expressed as

$$N = \frac{W x^2}{57.3\,L}\text{, ft-lb/deg} \qquad (13\text{-}1)$$

where

W = weight of external load, lb

x = distance between suspension attach points, ft

L = length of suspension, ft

It can be seen that shorter suspension cables generate a greater restoring moment and result in a more stable load and higher allowable airspeeds. Pitch restraint is provided by the tandem type of two-point suspensions, with the distance between the helicopter attachment points determining whether the load pitches up, remains level, or pitches down as aerodynamic drag swings it aft during flight. To avoid having a low-density load "fly" up into the helicopter due to a drag-induced pitchup, it is desirable to have the load attachment points spaced farther apart than the helicopter attachment points.

Laterally disposed two-point suspensions do not provide any pitch restraint, but do offer some roll restraint as well as yaw restraining torque.

Four-point suspensions provide simultaneous restraint in pitch and roll, and provide a slightly more effective yaw restoring moment N, as expressed by Eq. 13-2 (Ref. 10).

$$N = \frac{W(x^2 + y^2)}{57.3\,L}\text{, ft-lb/deg} \qquad (13\text{-}2)$$

where

x = longitudinal distance between cable attach points, ft

y = lateral distance between cable attach points, ft

A four-point suspension is compatible with most vehicular and containerized loads, and permits relatively high-speed flight with such loads slung close to the fuselage. The four-point suspension layout, however, has a number of inherent problem areas, some of which are unique and some of which are shared to a lesser degree by the two-point suspension.

Emergency release of multipoint suspensions requires simultaneous jettison of all cables with a high degree of reliability. If hoists are incorporated at the suspension points, these hoists must be synchronized. The presence of two or more attachment points, while providing stability to the cargo, provides a load path through which potentially uncontrollable moments could be applied to the helicopter. Hooking up multipoint suspensions to a hovering helicopter can be difficult. The operation is dangerous because of the possibility of picking up a partially unhooked load and sending the helicopter out of control. The problem of indeterminate structural performance is unique to the four-point suspension. Proper analysis of this redundant structure requires consideration of both payload and helicopter structural stiffness, as well as sling member elasticity (Ref. 11).

In the practical application of a four-point suspension, it has been found necessary to incorporate a load trimming system to equalize forces in the four cables. Without such a system, flexing of the fuselage and the payload in flight is likely to load some cables while causing others to go slack. An automatic takeup device designed to alleviate this problem is likely to be complex and expensive, but may be a necessity in future helicopters using four-point suspensions.

To determine the location and capacity of the suspension points for any new helicopter, the designer must have some knowledge of the type of loads to be carried. Refs. 9, 10, and 12 list, among them, most of the Army equipment and vehicles which can be sling-loaded. Weights and dimensions are provided. Army aircraft are included in the listed loads because of the frequent use of helicopters for aerial recovery of aircraft from otherwise inaccessible forced landing sites.

13-4.2.1 Static Loads

The rated capacity of the external load suspension system is established by preliminary design and flight tests. All components of the suspension system should be designed uniformly for this rated load. The rated load must be multiplied by a limit flight load factor of 2.5, with a safety factor of 1.5 applied to give an ultimate design load factor of 3.75. The attaching structure must be capable of sustaining this tension load applied in any direction within 30 deg of vertical. Relief from the 30-deg requirement can be obtained if it can be shown that, with the limit flight load applied at some lesser angle, the limits of control capability are reached. If ground vehicle towing capability is desired, the suspension must be analyzed for the maximum towing force times the ultimate safety factor of 1.5, applied at the extreme aftward towing angle, as established by preliminary design. Maximum towing force will be determined by pitch angle and control power limits, when the location of the cable attachment point is known.

13-4.2.2 Dynamic Loads

A phenomenon known as vertical bounce can occur when a helicopter is carrying an external load suspended from a sling or a pendant. It is a divergent vertical oscillation of the airframe/cargo system caused by resonance of the coupled airframe/cargo natural frequency with the one-per-rev (1P) vibration frequency of the helicopter. This is shown schematically in Fig. 13-9.

This simplified diagram makes no attempt to represent the fuselage structural bending modes; both the helicopter and the cargo are treated as rigid masses, and the suspension sling is shown as a spring. The natural frequency ω of this arrangement is

$$\omega = \frac{1}{2\pi}\sqrt{\frac{K(M_H + M_L)}{M_H M_L}}, \text{Hz} \qquad (13\text{-}3)$$

where

K = spring rate of suspension, lb/ft
M_H = mass of helicopter, slug
M_L = mass of external load, slug

Experience has shown that, when this natural frequency reaches close proximity to the 1P main rotor frequency, vertical bounce will occur. The lower practical threshold of vertical bounce has been defined as approximately 0.6 times the 1P frequency (Ref. 9).

With presently used sling materials and typical load weights, vertical bounce has been encountered in degrees ranging from mild crew discomfort to potentially destructive divergent airframe response. The milder cases usually are the result of exceeding the lower threshold frequency with the combination of a stiff sling and a relatively light load. The more serious divergent cases occur when the natural frequency is taken too close to the 1P frequency with a suspended load weighing close to the empty weight of the helicopter.

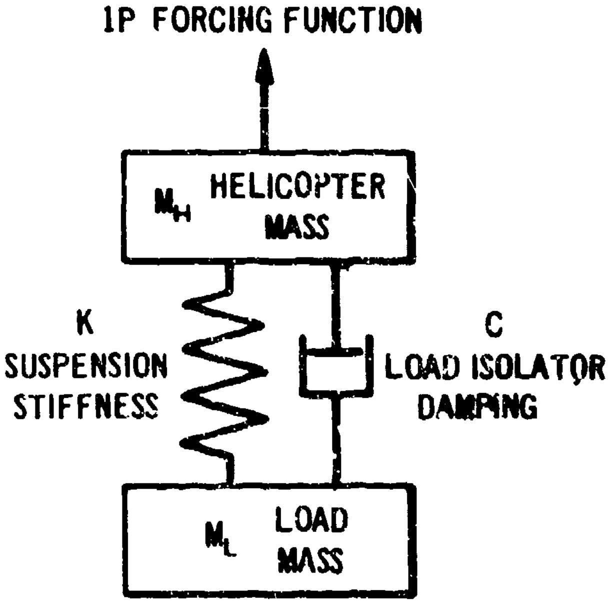


Figure 13-9. Helicopter/Load Dynamics Schematic

On the assumption that helicopter empty weight and main rotor speed are invariable, and that the helicopter is capable of carrying its rated payload externally, the only way that divergent vertical oscillations can be avoided is by controlling the stiffness K to keep the suspension system frequency below both the 1P frequency and the fuselage first bending mode.

If the characteristics of individual cargo loads are known, it is possible to control the stiffness K by tailoring the sling design to the load, by manipulating sling geometry (length), material, and cross-sectional area. This has been done for some existing helicopter load combinations by preparing curves of minimum sling length versus load for various suspension geometries and sling webbing thicknesses (Ref. 13). Where suspension length is variable in flight by the use of a winch, the stiffness of the system can be decreased by reeling out more cable whenever the onset of vertical bounce is detected.

A more satisfactory solution to the problem of divergent vertical oscillation is available to the designer (Ref. 14). This solution is to control the stiffness by inserting into the suspension system a load isolator having a soft spring rate, taking advantage of the fact that, when two springs are connected in series, the total spring rate K of the combination is always less than the stiffness of either spring, in the proportion

$$K = \frac{1}{\dfrac{1}{K_1} + \dfrac{1}{K_2}} = \frac{K_1 K_2}{K_1 + K_2}, \text{ lb/ft} \qquad (13\text{-}4)$$

where

K_1 = spring rate of first spring, lb/ft

K_2 = spring rate of second spring, lb/ft

Thus, if the load isolator by itself has a stiffness such that the external load frequency is always below the 0.6 × 1P threshold frequency, resonance with 1P frequency and coupling with first fuselage bending mode always will be avoided, regardless of the suspension sling stiffness.

If a simple spring is employed as a load isolator, the spring rate (stiffness) must be soft enough to handle the lightest load combined with the stiffest sling. A spring-type isolator designed to meet this requirement probably will exhibit excessive deflection when loaded with the maximum cargo load (times the limit load factor). The more sophisticated types of load isolators are designed to have a variable stiffness that increases with load, in such a proportion as to maintain a nearly constant suspension system natural frequency.

These nonlinear load isolators can be based on air springs or liquid springs, although studies have shown hydropneumatic load isolators to be generally superior (Ref. 11). Such an isolator usually is designed as an air-oil cylinder, with a volume of compressed air providing the spring rate, and the oil providing damping and also a means of varying the air volume to change stiffness. When the applied load is increased, an increased quantity of hydraulic fluid is metered into the cylinder by a servo valve. Provisions should be incorporated to absorb the recoil shock resulting from an inflight load release. The load limiter also provides a convenient mount for the placement of a cargo-weighing load cell.

13-4.2.3 Winches and Hooks

While light helicopters may be able to conduct external load operations satisfactorily with a simple fixed hook and pendant or multileg sling, any cargo helicopter should have the capability of hoisting its maximum rated external load while in a hover. This capability makes it possible to retrieve loads from tight spots where close hovering would be unsafe, and also enables the pilot to reel in loads in flight to minimize the kind of pendulum load oscillations which threaten controllability (Ref. 15).

The detail design of winches has many unique aspects which are beyond the realm of the helicopter designer. However, the designer should be familiar with the general characteristics required so that intelligent specifications may be written. Military helicopters used for combat transport must have a high-speed winch capable of hoisting the maximum external load at a rate of at least 60 fpm. With lighter loads, speeds in excess of 100 fpm are very desirable. A no-load deployment rate of 300 fpm or more would be useful in a combat situation. The winch speed should be smoothly variable over its entire range under load. Adequate braking capacity must be provided, as well as an automatic load holder to lock the winch in case of power failure. Hydraulic power generally is used for high-capacity aircraft winches because aircraft electrical motors have insufficient power. Both drum and capstan winches are suitable for helicopter application. Studies have shown no clear advantage for either type, when all aspects of their application are considered (Refs. 10, 11, and 16).

The same studies have concluded unanimously that, with the present state of the art, steel wire rope is the optimum tension member. Cable manufacturers' recommendations should be followed with respect to minimum allowable drum and sheave diameters (400 times wire strand diameter or an average of 18 times cable diameter). Manufacturers' recommended working loads are conservative, on the other hand, and the designer should establish high allowable loads, by test if possible. Contemporary, electrically operated hooks require a number of electrical conductors to be laid within the core of the wire rope, to be used for powering the hook-operating solenoid and conducting various indicator signals. It is suggested that the designer incorporate several spare wires within the cable so that these spare wires can be connected as replacements for conductors broken during use.

Cargo hooks must be capable of releasing by electrical command, and also should be able to be released by a backup manual mechanical system. A ground-contact release feature, which opens the hook when the load is released, also is desirable. On single-point suspensions, a swivel must be provided to isolate the hoist cable from load rotation. If an electrically operated hook is used, electrical continuity must be maintained through this swivel joint, using well-sealed slip rings.

The requirements for winches and hooks generally are similar — whether single-, two-, or four-point suspensions are used. Hooks used exclusively for multipoint suspensions do not require swivel isolators. Four-point suspension hooks must not have automatic release under relaxation. Winches on multipoint suspensions must be synchronized, either by direct mechanical interconnection or through the use of comparator servo controls. Multiple winches also should be individually controllable to permit independent use as single-point hoists and to provide a means of trimming the multipoint load in flight.

13-4.2.4 System Safety

A fault tree analysis should be made early in the design of the external cargo system to isolate potential hazards. Certain features must be provided on all external suspension systems. The normal release control, which may be electrical, must be located on both pilots' cyclic sticks. In addition, an all-mechanical means of load release should be provided as a backup to the normal release device. An emergency release control must be provided for the pilot. This switch, separate from the normal and manual controls must cause instantaneous jettisoning of the cable and hook, either by severing the cable(s) or by reliably declutching the winch drum so that the falling load strips the cable from the drum.

Two-point suspensions should incorporate a failure sensor which detects the parting of one cable and instantaneously releases the surviving cable. Four-point suspensions, in order to be man-rated for passenger pod operations, must have a means of positively locking the pod in place after hoisting, or of placing the emergency jettison system in a positively safe configuration.

Safety of ground-handling personnel should be enhanced by providing some means of attenuating static electricity discharge. The static electrical charge developed by a cargo helicopter hovering in dry and dusty conditions has enough energy to incapacitate an individual coming in contact with the suspended hook. This energy, arcing to ground, is sufficient to ignite fuel vapor or initiate explosives (Ref. 17). Two techniques are available for the elimination of static charges: active discharge (Ref. 18), which uses a high-voltage generator to null any helicopter potential detected by an onboard sensor, and passive discharge, which drains the capacitive energy to ground through a highly resistive link, at a current level which is not injurious and usually not detectable. The resistive link can be incorporated in a ground handler's gaff hook, or can be connected in series with the suspension sling. See par. 7-9.3 for additional information.

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CHAPTER 14
ARMOR, ARMAMENT, AND PROTECTIVE SUBSYSTEMS DESIGN

14-0 LIST OF SYMBOLS

A_P = presented area

A_V = vulnerable area

A_{V_i} = singly vulnerable area of component i of a system

A_{V_T} = singly vulnerable area of a system

$P_K|H$ = conditional kill probability

$(P_K|H)_i$ = conditional kill probability of component i of a system

14-1 INTRODUCTION

As described in Chapter 2, AMCP 706-201, and ADS-11, the concept of helicopter survivability is broad and complex. However, the effects of armament and protective materials can be evaluated in terms of three basic means of coping with a hostile environment (whether the hostility is man-made or natural):

1. Avoiding it
2. Neutralizing it
3. Enduring it.

Avoidance places emphasis upon vehicle performance. To avoid a specific environment, a helicopter requires the ability to fly above it, or around it, or to outrun it. Armor and armament have a negative effect upon this aspect of survivability, because they add weight and aerodynamic drag.

From a military standpoint, neutralizing the environment requires destroying the source of hostility or hazard, or otherwise rendering it harmless. This may be accomplished by strategies or tactics that do not involve the system being protected. However, a capability to correct a hostile environment may be designed into the system itself by the inclusion of standoff weapons (armament) or countermeasures.

Enduring the hostile environment places emphasis upon such areas as redundancy and separation, shielding of mission-critical components with structure or components that are less critical, concentration of critical components in a single place within the vehicle core in order to decrease the total area presented to the hazard, and addition of deadweight protection such as armor.

Armor and armament, especially the latter, must be evaluated in contexts other than survivability. For attack helicopters, the ability to survive a hostile environment, although essential for mission completion, is not in itself a true measure of the worth of the vehicles' armament. In other words, the possible trade-offs among performance, armor, and armament always affect survivability, but also may involve directly such other primary mission variables as performance (see Chapter 2, AMCP 706-201).

The major trade-offs will have been performed prior to the start of detail design. Although these trade-offs should be updated continuously throughout the design process as more complete and precise data become available, a discussion of their nature and description is beyond the scope of this chapter. However, it is essential that the designer retain a consistent, balanced approach to the overall effects of detail design decisions regarding location, installation, selection, and use of armament. These relationships are subtle and, in many instances, are difficult to quantify. As a result, much of the material contained in this chapter is qualitative, with specific relationships being included only in instances in which general physical relationships are known.

14-2 ARMAMENT SYSTEMS

Helicopter armament systems typically are individual installations of guns, guided missiles, or free-flight rockets. The number and type of installations, and the mix of guns, missiles, or rockets, are described by the governing helicopter detail specification. This chapter contains the basic helicopter design guidelines, considerations, and requirements that apply to the installation of armament systems.

14-2.1 GUNS

Guns of the number and caliber described by the governing helicopter detail specification *shall* be installed in the required positions. An ammunition supply of the quantity and type dictated by the detail specification *shall* be provided.

14-2.1.1 Types

Several types of guns are available for helicopter use. They include air-cooled, gas-operated 7.62 mm weapons; air-cooled, automatic 20 mm weapons; air-cooled electrically-operated and controlled 30 mm automatic cannons, and air-cooled weapons for firing 40 mm grenade ammunition. Typical guns are listed in Table 14-1.

Operational and maintenance details are contained in the applicable specification. US Army Armament Command (ARMCOM), Rock Island, Ill., can provide information about guns not listed in Table 14-1 and also can provide operation temperatures, lubrication requirements, power requirements, peak and steady-state recoil forces, life of gun and critical parts, and similar pertinent details on all guns available for helicopter use.

For some applications, the gun is installed on recoil adapters. This installation minimizes the effect of gun recoil forces upon the helicopter structure. In selecting and designing recoil adapters, careful consideration should be given to gun muzzle energy; gun weight, including attaching feeder and drive motor; and the response dynamics of the helicopter structure. ARMCOM can provide the latest information on these requirements.

Feed mechanisms for these weapons vary, and each type presents a special set of design problems. In some instances, the ammunition feed must start from rest and must reach the peak rate of fire within the time required for the firing of one round. This imposes high acceleration forces upon the feed train and the storage containers.

In other applications, a feed mechanism is used that extracts cartridges from a recycling conveyor belt. The integral feeder is adapted for this purpose by replacing the feeder cover with special link guides. The high acceleration forces imposed upon the belt, and cyclical deviations of the gun from a nominal rate of fire, require careful design of the ammunition feed train to insure equivalent belt tension on both sides of the gun feeder.

In other cases, the feed action introduces repetitive accelerations upon the ammunition belt by requiring the belt to start, move one cartridge pitch distance, then return to rest in the process of firing each shot. Sharp turns in the vicinity of the feeder, and belt drag conditions that will cause link stretch under these conditions, must be avoided.

In some systems an ammunition booster will need to be added at an appropriate location in the ammunition path so that the belt pull forces are alleviated. The booster commonly is driven by a fractional HP motor and must be controlled by various means to sense ammunition demand. Starting and stopping accelerations must be controlled so that feeding will be compatible with gun demand for ammunition. Care must be taken to prevent obstructions from falling between flexible chute elements.

14-2.1.2 Location

Gun location is part of the overall helicopter optimization as discussed in Chapter 2, AMCP 706-201. There are minimum requirements imposed by the nature of the weapon. The location must provide accessibility, unimpeded projectile flight paths and debris ejection paths, and the ability to jettison externally mounted gun pods; and it must be such that the vehicle can withstand gun muzzle blast effects.

Beyond these minimum requirements, the degree of optimization must be related to the overall effectiveness criteria used in evaluating the vehicle. The trade-offs will involve structural and geometrical limitations arising from the desire to optimize cost and flight performance versus the optimum location

TABLE 14-1. TYPICAL HELICOPTER GUNS

GUN DESIGNATION	CALIBER	RATE OF FIRE, spm	METHOD OF OPERATION	METHOD OF FIRING	REFERENCE
M60C MACHINE GUN	7.62 mm	600	SELF-POWERED GAS OPERATED	ELECTRIC SOLENOID	TM 9-1090-201-12
M134 AUTOMATIC GUN	7.62 mm	750–6000	EXTERNAL MOTOR	PERCUSSION (SELF-CONTAINED)	TM 9-1005-286-15
M61 AUTOMATIC GUN	20 mm	4000–7200	EXTERNAL MOTOR	ELECTRIC	TO 11W1-12-4-32
M197 AUTOMATIC GUN	20 mm	400–1500	EXTERNAL MOTOR	ELECTRIC	NAVAIR 11-95M197-1
XM140 AUTOMATIC CANNON	30 mm	405	SELF-CONTAINED ELECTRIC MOTOR	PERCUSSION (SELF-ACTUATED)	POMM 1005-285-15
XM129 AUTOMATIC GRENADE LAUNCHER	40 mm	400	EXTERNAL MOTOR	PERCUSSION (SELF-CONTAINED)	TM 1090-203-12

to provide weapon system performance and maintainability.

Parameters relating weapon system performance to location are discussed individually in the paragraphs that follow.

14-2.1.2.1 Projectile Flight Path

Gun location must be such as to avoid intersection of the extremes of the projectile flight path envelope with the helicopter structure, including the main rotor and externally-carried stores. The projectile flight path envelope is described by a circular dispersion of the fired projectile, with the circle center being coincident with the gun barrel centerline. Factors to consider in determining the dispersion envelope include gun and ammunition dispersion, aerodynamic forces acting upon the projectile, and deflections of the gun mount and helicopter structure. ARMCOM can provide details of gun and ammunition dispersion, and aerodynamic test data.

14-2.1.2.2 Blast Effects

Guns *shall* be located as far as is possible from helicopter structure to minimize the effects of muzzle blast. The aircraft skin near the muzzle and adjacent aircraft structure must be strong enough to prevent gun blast damage. Reinforcement requirements of the aircraft skin and structure are determined by the distance between gun and skin, the thickness of the skin, and the density of frames and stringers. The gun muzzle *shall* never be located near enough to canopies, radar antennas, or door frames to cause or create a hazardous condition.

For some guns, a muzzle brake can be incorporated to reshape the blast pressure field. This device distorts the blast field so that peak pressures and impulses are rotated and displaced from their normal positions relative to the gun barrel and thereby reduce recoil forces. ARMCOM should be consulted for details regarding availability of muzzle brakes and flash suppressors for particular weapons, design considerations for fitting a particular gun, and definition of muzzle pressure fields. AMCP 706-251 provides information on the design and application of muzzle brakes. Consideration also should be given to the relationship of different weapon systems to each other, such as machine guns vs rockets or missiles, both from stationary firing position (of machine guns) and during the trajectory (of rockets, etc.).

14-2.1.2.3 Debris Ejection Path

Ejected ammunition cases and links *shall not* impinge upon helicopter structure, control surfaces, rotors, or externally-carried equipments. The trajectories of the ejected debris can be determined from gun ejection velocities and the local aerodynamic conditions about the helicopter. In general, the debris ejection velocity is equivalent to or higher than ammunition feed velocity.

Debris ejection velocities can be increased by the use of accelerator mechanisms. Some accelerators use rotating brushes to capture and accelerate the debris; others use sprockets. The selection of the design technique must consider the available space, and the attitude, kinematics, and shape of the ejected debris. The accelerator must be designed for positive capture and retention or rejection of the debris, and must be located as close to the gun ejection port as is possible. Deflector plates can be placed strategically to redirect the caselink ejection path properly, particularly where cases can strike a surface perpendicular to the ejection path and bounce back into the gun mechanism.

For some applications, an ammunition feed system is used that returns gun debris to an internal storage compartment. This design is suited best for systems that employ recycling conveyors or sprockets, rather than a linked belt, to transport ammunition. The return conveyance path *shall* accommodate positive guidance of an occasional misfired cartridge as well as fired cases. The strength of the storage compartment must be sufficient to protect personnel and the vehicle from hazards caused by misfired cartridges. A ventilating system in the storage container *shall* be provided to remove residual gases in the fired cases.

14-2.1.2.4 External Gun Jettisoning

In general, for installations requiring jettison, the gun and ammunition should be located in externally-mounted pods. The locations of the gun and pod and the angle of ejection *shall* be such as to insure clearance from the helicopter, landing gear, and adjacent stores. Pod and helicopter structure *shall* be designed for compatibility with power jettisoning per MIL-A-8591.

14-2.1.2.5 Accessibility

Location of the gun must provide sufficient clearance and accessibility to allow performance of maintenance — including servicing, removal, and replacement of the gun and related accessories; and loading and unloading of ammunition. The weapon also must be accessible enough to permit such activities as diagnosis of malfunctions and the clearing of stoppages or jams, partial disassembly while in place, performance of all standard adjustments with appropriate tools, and viewing and reading of all dials and gages.

An unobstructed view from the gun barrel must be provided for boresight alignment. Electrical cables, tubing, and equipment that must be placed in the vicinity of the gun should be located to prevent damage during the removal and replacement of the gun. Minimum maintenance and turnaround times are a primary consideration.

Cables and equipment should be restrained so that they cannot be proximate to hot gun barrels or other moving parts of a weapon or system. Provisions should be made for removing hot weapons (gun barrels and adjoining components) without burning crewmen's hands. Appropriate "NO STEP" markings will be provided on the weapon turret, feed chuting, ammunition caps, and other materiel not designed to withstand stepping forces.

14-2.1.2.6 Dynamic Forces

Dynamic forces may best be determined from instrumented tests of the actual weapon turret on a mount that simulates the flexibility profile of the helicopter. Weapon systems contain a series of shock absorbers that operate at different natural frequencies, thus alternately changing the load profile during the firing burst. These shock absorbers are characterized as:

1. Weapon internal mechanism
2. Buffer (spring or hydraulic package)
3. Turret structure
4. Connecting struts between turret and helicopter
5. Helicopter framework or structure.

Since the weapon fires at various attitudes of elevation and azimuth, the torques and loads vary. The true load profile is statically indeterminate and should be obtained from ARMCOM in the form of instrumental firing data with all test conditions clearly identified as to:

1. Ammunition Lot No.
2. Weapon weight
3. Firing schedule
4. Weapon elevation/azimuth angle for each test sequence
5. Stiffness of mount base
6. Other special test conditions (temperature, etc.).

Moderate overdesign in the weapon system mounting interface should be considered in new or developmental designs to allow for normal "growth" or increase in impulse as an evolutionary factor in a system life cycle. A factor of 50% is recommended. Additionally, this consideration will provide an increased measure of capability to accommodate unpredictable adverse conditions or necessary stiffening of the shock absorbing elements in the weapon system.

14-2.1.3 Types of Installations

Guns normally are installed in the fixed, forward position in pods, or are mounted flexibly in remote-controlled turrets or upon manually-operated pintles. Guns also may be installed in the fixed, forward position without pods. The configuration that best suits the detail helicopter mission requirements should be determined and employed.

14-2.1.3.1 Pod Installation

Gun pod installations usually contain the gun, the ammunition storage and feed system, and the operating mechanisms within an aerodynamically-shaped enclosure. Size and contour of the pod envelope are selected for minimum aerodynamic drag. Construction and suspension features of the pod *shall* conform to MIL-A-8591 structural design standards for jettisonable stores. Pods normally are designed for installation on either 14- or 30-in. spaced bomb rack mounting hooks. The design of the supporting structure also should consider:

1. Location of the pod so as to permit normal servicing and maintenance of the gun, ammunition, and operating mechanisms without removal of the pod. These actions will include, as a minimum, ammunition loading and unloading, boresighting, component checkout, and normal removal/replacement of components.
2. Design of the supporting structure so as to withstand forces imposed by gun recoil and aerodynamic pressures. The structure should provide proper rigidity in order to minimize gun firing errors as a result of structural deflections.
3. Location and design of the pod to avoid aerodynamic interference among control surfaces, sensors, and adjacent stores.
4. Asymmetrical firings occurring due to failure of a matching gun pod.

14-2.1.3.2 Turret Installations

Turret-mounted guns are aimed and positioned remotely by means of a sight or a fire control system. Such guns normally require remote location of ammunition supplies, which are connected to the gun by means of flexible or rigid chutes that guide the ammunition. Turrets normally are placed just beyond the helicopter skin line, and require an aerodynamic fairing in order to reduce aerodynamic drag. The fairing design must provide clearance for the turret internal mechanisms, and must minimize aerodynamic torques against the turret drives. The fairing, or a portion of it, must be quickly removable so that crewmen can determine the "safe" condition of the weapon and/or take immediate action to render the weapon safe.

The location of the turret *shall* permit unobstructed firing throughout the turret limits of coverage. Positive structural limit stops *shall* be provided in order to prevent the turret from moving beyond the selected limits and to maintain a projectile trajectory which does not endanger the aircraft and its stores.

The type of power selected to operate the turret depends upon the available total power, and upon total helicopter weight and cost factors. Quick-disconnect couplings should be used for attachment of the power service to the turret.

The designer must provide a turret-mounting surface that has a definite planar relationship to the Weapon System Datum Plane (see par. 14-2.1.6). The Weapon System Datum Plane is a surface or reference plane in the helicopter to which all armament, sights, and fire control systems are referenced. The supporting structure must provide the rigidity necessary to meet the accuracy requirements described in the helicopter system specification and to withstand the mechanical forces and moments imposed by the gun and turret. The principal mechanical loads will result from the following conditions, applied individually or in unison:

1. Actual loads imposed by the maneuvering of the helicopter

2. Cyclical recoil forces imposed by the firing of the gun

3. Torques applied to accelerate the gun to its maximum angular velocity, or to bring the pivoting gun to rest from its maximum angular velocity of rotation

4. External forces such as those created by disregard for the "NO STEP" marking.

Accessibility of the turret must be adequate to permit maintenance and inspection of the weapon. Turret design should permit the cowling to be removed without requiring the turret to be slewed from the stowed position. The cowling should not pass through the line of fire while being removed.

14-2.1.3.3 Pintle Guns

Typically installed in helicopter doorways, pintle guns are aimed and fired manually. Ammunition containers usually are located remotely and are connected to the gun with flexible chutes. Some guns can be installed with integral ammunition containers, provided that the weight of the full ammunition containers is low.

Guns are installed on pintle mounts so that the centerline of the firing barrel is coincident with the pivot center of the mount. Pivot centers should be coincident with the CG of the gun and all attaching hardware. If it is not convenient to provide a coincident CG/pivot axis, counterbalance mechanisms can be installed to help offset the inertial forces imposed upon the gunner while pivoting the gun. Mechanical stops *shall* be provided in order to maintain a projectile trajectory which does not endanger the aircraft.

Clearance must be provided around the pintle location to allow 5th through 95th percentile size gunners to operate the gun and mount throughout the full range of gun travel. A gunner safety belt to secure the gunner *shall* be provided.

14-2.1.4 Ammunition Storage

Ammunition *shall* be installed in containers with capacities as described by the governing helicopter specification. Ammunition containers must be located as close to the gun as possible within the helicopter geometrical limitations and weight and balance restrictions. The designer must provide the accessibility and clearance necessary to:

1. Load and unload the container without removal from the helicopter

2. Remove and replace the container (The designer must consider replacement of a damaged full container.)

3. Permit normal servicing as required by the specific container design, including lubrication, disassembly of attaching hardware, and replacement of container subassemblies.

For some applications, ammunition containers have power drives. The designer *shall* provide the power service as needed. Power service should be attached using quick-disconnect couplings.

The use of preloaded containers, which significantly reduce turnaround time, should be considered. Containers should contain sufficient accesses to facilitate remedial action in event of hangups, and should be secured appropriately to tie points or mounting fixtures. When using linked ammunition, helicopter vibration should be taken into account because feed problems have been traced to this cause. Vibration causes a settling of unsupported linked ammunition, making layers of belts difficult to pull apart. Use of spacer shelves or hanging loops should be considered; also note that drum or linkless feed eliminates this problem. Shake tests of the storage and feed system therefore should be made during development.

14-2.1.5 Ammunition Feed

The chutes that carry the ammunition from the container to the gun feed mechanisms may be of either rigid or flexible design, but should be free of inside surface projections and independent of other accessories. Chutes must be attached to the feed mechanism by a quickly detachable means. When locked in place, the chutes must remain in alignment with the ammunition boxes and the gun feed mechanism throughout all adjustments of the gun. Feed chute design must accommodate the allowable twist, bend, and fan radii of the ammunition belt.

The detail design *shall* provide accessibility to the ammunition belt within the chute to permit threading the belt through the chute and inserting it into the gun feed mechanism. The feed chutes *shall* be of light weight, low friction, and long wearing materials and of a gage that will insure maintenance of original inside dimensions under ordinary operating conditions. If flexible chuting is used, an approved design must be employed.

Where the ammunition belt contacts the feed chute, the design *shall* provide relief so that the ammunition links will not drag on the chute. This can be accomplished with tracks added to the chute beads rolled into the chute, or with a clearance slot cut from the chute so that the ammunition will be supported by the case and projectile and not by the links.

14-2.1.6 Boresighting and Harmonization

The armament installation *shall* be designed for compatibility with the sights and/or sighting station equipment described by the helicopter system specification. A means *shall* be provided to boresight the gun to an accurate coincident relationship with the sight. For pivoting guns, a means must be provided to check the gun pointing angles in reference to the sight command angles.

The helicopter design must include a definite relationship between armament installations and the sight and fire control references. This is accomplished by establishment of a Weapon System Datum Plane. The installation and traverse of each weapon, and the sighting and fire control equipment *shall* be referenced to this datum. Accuracy of the relationship between the references *shall* be in accordance with the governing system specification.

Gun mounts *shall* be adjustable, and *shall* be capable of being locked in the transverse and vertical planes to provide for a minimum of ±0.25-deg gun adjustment in addition to any adjustment required to overcome aircraft manufacturing tolerances. The detail design *shall* provide for the use of standard boresight telescopes for performing the boresight operation with the guns in place. For turrets, provisions *shall* be made for checking alignment of axes to the aircraft datum planes (vertical and horizontal) through the use of the standard boresight telescope, with the turret aligned to three azimuth angles as a minimum.

14-2.2 GUIDED MISSILES

Guided missile launchers and guidance control equipment of the number and type described by the governing helicopter system specification *shall* be installed. Currently being used for helicopter applications is the TOW, a tube-launched, optically aimed, wire-guided missile. Details regarding this missile are classified and, with required justification, are available from the US Army Missile Command (MICOM). This paragraph provides helicopter design standards that can be applied to the TOW weapon system or to any other missile installation.

14-2.2.1 Location of Launcher Installations

The primary function of the launcher installation is to release the missile from the helicopter without damaging either the missile components or the helicopter. The launch mechanism should be designed so that the missile flight path (during launch) will be directed to position the missile within (1) the capture envelope required for initiation of guidance by the gunner, or (2) the flight path limitations required for target acquisition and lock-on when using a homing missile.

Helicopter missile launchers generally will be installed offset from the helicopter centerline on armament pylons or stub wings to protect the tail control surfaces and rotor system from possible immersion in the exhaust wake of the missile. Good design practices include location of the launcher on the helicopter to prevent:

1. Engine compressor stall or flameout as a result of exhaust gases entering the engine intake ducts
2. Exhaust gas impingement upon, or ignition debris collision with, the airframe and all rotor systems
3. Harmful corrosion effects as a result of deposits of missile exhaust residue within the engine or upon other components that are not accessible readily for prompt cleaning
4. Impairment of pilot's or gunner's vision by flash during firing
5. Excessive acoustic noise in the crew compartment during firing
6. Pitting or coating of the canopy by exhaust gas and debris
7. Aerodynamic interference between launchers

and control surfaces, sensors, and adjacent stores. The design and location of the launcher installation should be such as to minimize corrosive effects resulting from the exhaust particles inherent to solid propellant missiles. Proper consideration of preventive or corrective methods, including cleansing of affected parts, can reduce significantly the possibility of structural corrosion or surface damage caused by motor exhaust. In general, missile launchers should be located as far as possible from other parts of the aircraft.

14-2.2.2 Structural Clearance

Adequate structural clearance *shall* be provided to prevent interference of the missile (including fins) with any part of the helicopter (including adjacent stores) during launch of the missile. A clearance cone of 3 deg half angle, measured from the missile longitudinal centerline at the exit port, is an example.

Definition of clearance should include consideration of the aerodynamic forces acting upon the missile at launch. These forces can cause significant variations in the missile pitch and yaw motion, and in the linear displacement, during the launch phase. Sufficient ground clearance *shall* be provided to prevent launcher ground contact during normal takeoffs and landings, and during hard landings at maximum gross weight.

14-2.2.3 Blast Protection

The helicopter designer *shall* provide strength and/or surface protection for helicopter structure and exposed subsystems that is adequate to protect them from missile exhaust effects. These effects include overpressure, heat, recoil or reaction loads, erosion, and corrosion resulting from normal repetitive firing. Details of these characteristics will be available in the weapon specification

14-2.2.4 Accessibility

Maximum accessibility *shall* be provided to the launching mechanisms, tubes, detents, firing contacts, and electrical connections to facilitate loading, unloading, circuit checking, diagnosis of malfunctions, clearing of stoppages, partial disassembly while in place, viewing of all dials and gage marks, accomplishment of all adjustments with the appropriate tools, cleaning, and replacement. Minimum maintenance and turnaround times are a primary consideration. Other guidelines pertinent to this topic are contained in Chapters 11 and 13, AMCP 706-201.

14-2.2.5 Firing Circuit Testing

The designer should provide a self-contained firing circuit tester, or a single-point electrical quick-disconnect in the individual missile contact circuit which is suitable for use with an external circuit tester.

14-2.2.6 Jettisoning

The launcher installation *shall* include provisions for jettisoning the unit from the aircraft under all normal flight conditions, including undetected sideslips. Launch structure should be designed for compatibility with power jettisoning per MIL-A-8591. The angle at which the launcher is ejected *shall* be selected to provide clearance with the airframe, landing gear, and adjacent stores.

14-2.2.7 Effects of Aircraft Maneuvers

Structural design of the missile launcher installation *shall* consider the effects of loads imposed by maneuvers of both the missile and the aircraft.

14-2.2.8 Types of Installations

The launcher installation should provide for effective missile deployment in specified tactical situations associated with a particular missile configuration. Factors affecting selection of the launcher configuration are launcher size and weight, helicopter speed and altitude environment, and ground-handling and loading requirements. For helicopter applications, the launcher generally will be a fixed installation located on a wing or armament pylon, and may include either a zero or a finite launch length depending upon the missile characteristics.

14-2.2.9 Loading

The missile launcher should be designed to facilitate fast loading during ground operations. The loading process should require a minimum number of precise locating and positioning operations by the armament mechanics.

14-2.2.10 Aerodynamic Effects

Effects of local airflow conditions upon the initial missile flight path can be significant, and *shall* be considered during the launcher design task. Immediately upon release from the launcher, the missile is exposed to aerodynamic forces that tend to displace it from its intended flight path. Missile response is affected by launch velocity, guidance system operation during the launch phase, and control surface effectiveness at the launch speed.

14-2.2.11 Suspension and Retention

Suspension and retention components include the equipment used to attach the launcher to the aircraft. MIL-A-8591 contains a detailed method for calculation of suspension system interface loads. This specification is applicable to bombs and other externally-mounted stores on fixed-wing aircraft, and may be used for a helicopter missile launcher whenever the launcher is compatible with the lug and sway brace criteria contained therein. Missile launch fixtures and suspension hardware generally should be as simple, lightweight, and small as possible, compatible with maximum reliability and with minimum effects upon missile and aircraft performance. A safety lock or retention mechanism is required in the launcher to prevent inadvertent launch and to retain the missile under severe load conditions (i.e., crash loads). The retention device should be designed to interrupt the launch initiation system, as well as to restrain the missile mechanically.

14-2.2.12 Launch Initiation

The missile system should include a means of transmitting a launch initiation signal from the aircraft to the missile. The nature and complexity of this system will depend upon the type of missile to be launched. Some missiles require only the ignition of a rocket motor, while others require in-flight prelaunch checkout, initial condition inputs to the guidance unit, and multistage launch sequencing. The design should be as simple as possible, consistent with a reliable and safe launch.

14-2.2.13 Restraining Latch

A restraining latch mechanism sometimes is required to retain the missile just prior to launch. This is similar to the rocket restraining latch discussed in par. 14-2.3.7. The latch is designed to retain the missile under normal maneuver loads, but to release at a predetermined load created by the motor thrust. The mechanism may be designed as part of the suspension and retention system (par. 14-2.2.11). The restraining latch should be releasable easily by armament mechanics during ground loading and unloading operations. The latch design should contain provisions for adjusting the release load in order to compensate for wear in the mechanical components. Locking devices should be provided to prevent post-loading variations in the missile-restraining force.

14-2.2.14 Forced Ejection

Some missile systems may require a means of ejecting the missile from the launcher in such a manner as to provide separation of helicopter and missile prior to ignition of the boost motor. The necessity and mechanism for forced ejection must be determined by a dynamic and aerodynamic evaluation of the missile/helicopter system. Sources of ejection force that have been used successfully include compressed gas, mechanical springs, and explosive or propellant devices. Provision *shall* be made to prevent inadvertent operation of the ejection system, either in flight or on the ground.

14-2.3 ROCKETS

Rocket launchers of the number and type described by the governing helicopter system specification *shall* be installed. The current rocket type qualified for use on helicopters is the 2.75-in. folding fin aircraft rocket (FFAR). This rocket is available in a variety of warhead/fuze combinations to suit specific helicopter mission requirements. It is carried in and launched from the helicopter by means of tubular launchers. The rocket and some of its available launcher types are described in TB 9-1340-201. ARMCOM should be consulted for details regarding launch recoil forces, exhaust blast envelope, firing power, and other pertinent items.

14-2.3.1 Rocket Launcher Installations

The primary function of the rocket launcher is to release the rocket safely from the helicopter without disturbing the rocket from its intended flight path. The initial flight direction of unguided rockets directly influences delivery accuracy. Therefore, the launcher design should provide for accurate alignment of the launcher boreline with the helicopter aiming reference under all tactical deployment conditions. The effectiveness, safety, and maintainability requirements and considerations of pars. 14-2.2.1 through 14-2.2.7 are relevant to both rocket and missile installations. Additional interface design considerations related only to free-flight rockets are contained within this paragraph.

The current 2.75-in. FFAR launchers for helicopters consist of a fixed, forward-firing, rearward-venting, open-breech tube cluster. The individual tubes may be reusable or replaceable, or the cluster may be expendable. The launchers normally are installed with the launcher axis (boreline) parallel to the line of flight under specified flight conditions.

Other types of launcher installations have been used successfully in fixed-wing aircraft, but have not been applied to helicopters to date. They are listed here because of possible applications to helicopters of the future, and include:

1. Open-tube pod, retractable into the fuselage and extended for firing
2. Restricted-breech, rearward-venting, with constricted or deflected exhaust

3. Closed-breech, forward venting

4. Tandem tubes, side-venting through deflector doors or rear-venting through rail-lined, larger-inside-diameter tubes.

The use of closed-breech launching tubes requires provisions for adequate forward exhaust venting. The fins of the 2.75-in. FFAR are subject to damage or failure from excessive launching tube pressures, or from detonation of the forward vented exhaust gases at the launcher muzzle as the rocket leaves the launch tube.

14-2.3.2 Launch Tube Materials

Launcher tubes capable of withstanding repeated firing of the 2.75-in. FFAR have been constructed of aluminum, stainless steel, titanium, resin-impregnated Fiberglas, and phenolic-impregnated fabric-base material. Expendable or one-shot launching tubes have been constructed of aluminum, steel, and plastic-impregnated or coated paper-base material. Any materials that have been fully tested and found suitable for the purpose may be used for rocket launch tubes, provided due account is taken of availability, cost, and compatibility with other installation requirements.

14-2.3.3 Launcher Mounting

Forward-firing launchers should be mounted with the launch tubes at the optimum angle for highest system accuracy for tactics to be employed. The corresponding aircraft pitch angle also should be evaluated for one-half fuel load and maximum ammunition load at the mean combat altitude. Provision for adjustment of the launcher elevation may be required so as to account for variations in the flight attitude of the helicopter between level and diving flight. Adequate structural clearance *shall* be provided to prevent interference of the rocket (including fins) with any part of the helicopter (including adjacent stores) during rocket launch. A clearance cone of 7 deg half angle measured from the rocket longitudinal centerline of the exit port is an example.

14-2.3.4 Number of Rockets

The applicable helicopter system specification will define the number of rockets to be carried and the sequence in which they are fired. Lateral spacing of successively-fired rockets should be selected to prevent the mutual interference effects of rocket blasts, fin opening, and jostling of adjacent rockets fired from closely spaced tubes. Rockets usually can be ripple-fired with relatively short firing intervals if successive rounds are fired from tubes with adequate centerline separation distance. Some helicopter configurations using wing-mounted launchers at outboard locations may require simultaneous rocket firings (pair firing) from each side of the fuselage in order to prevent excessive reaction torques about the helicopter directional stability axis.

14-2.3.5 Load Requirements

The rocket launcher installation *shall* be designed to withstand, without permanent deformation, the acceleration and aerodynamic loads associated with the maximum helicopter maneuvering load factors, in combination with rocket blast, recoil, and hangfire loads. The installation also *shall* withstand applicable crash loads without failure (Ref. 1).

14-2.3.6 Ground Safety

The rocket installation *shall* include a ground safety cutout switch in order to prevent accidental rocket firing during ground operations. The firing circuit may be interrupted by landing gear extension, by landing skid compression under the helicopter empty weight, or by insertion of an interlock pin at each launcher by the ground crew.

A manually-operated override switch *shall* be provided to permit ground checkout of the firing circuit. The grounding path for each rocket ignition circuit *shall* be located physically as close as possible to the actual rocket firing contact. Grounding *shall* continue until the rockets are loaded or until the firing circuit is energized. The firing circuitry also must contain provisions for preventing inadvertent rocket motor ignition due to radio frequency energy.

14-2.3.7 Restraining Latches

Rockets will be restrained positively in each launching tube by means of an appropriate mechanism. The restraining mechanism must engage automatically when the rocket is loaded into the launcher, and must remain engaged if the rocket is rotated within the launcher during the loading procedures. The latch will prevent aft movement of the rocket, and will restrain the rocket from forward movement with a force equivalent to that imposed by longitudinal crash load factors. If a blast-operated detent is used, the mechanism will prevent both fore and aft movement of the nonburning rocket, and will contain a means for manual release of the individual rocket latches in order to permit rapid loading and removal of unfired rockets.

14-2.3.8 Firing Contacts

The electrical contacts will be designed to provide a low-resistance path for the electrical current flow from the helicopter firing circuits to each rocket. The contacts and supporting structure *shall* be designed to fire a minimum of 50 rockets without repair or replacement of parts. The contacts *shall* be designed to provide minimum internal blockage of the launch tube, to shed debris from the rocket exhaust, and to be accessible readily for cleaning and replacement.

14-2.3.9 Intervalometer

The rocket system *shall* include methods to allow both automatic and manual firing of rockets in a predetermined quantity, sequence, and timing interval. The requirements for rocket selection and firing sequence will be defined by the basic helicopter system specification. As a minimum, the operator control panel *shall* allow selection of the number of rockets to be fired singly, in pairs, or in ripple at a preset time interval. The timing mechanism *shall* be designed for reliability and accuracy under all firing conditions.

14-2.3.10 Launcher Fairing

If frangible or breakaway material is used to cover launcher openings or to streamline the rocket installation, the design *shall* withstand rocket blast and air loads at airspeeds up to the maximum aircraft velocity. The covers also *shall* provide protection against rain, dust, and other severe environmental conditions to which the aircraft may be subjected. The design of the fairing *shall* insure that the frangible material will break into segments small enough to prevent damage to radomes, windshields and plastic closures, engine components, aircraft structure or skin, and control surfaces. The fragments *shall not* be permitted to enter the engine intake ducts. Some frangible materials that have been used successfully are:

1. Polyester resins with inert fill-material such as chopped cellophane, short jute fibers, and short glass fibers
2. Plastic-impregnated papers
3. Foamed plastic plugs and plates.

14-2.4 SAFETY CONSIDERATIONS

This paragraph specifically covers the interface of fixed and movable weapon systems with other helicopter systems.

Proper system safety design must incorporate the applicable data and techniques related to insuring safe man-machine relationships, yet must avoid placing unrealistic or costly impositions upon hardware design. To be effective, man-machine relationships must be integrated with system safety to provide a logical and consistent continuum throughout the life span of the weapon system. This means that system safety engineering principles must be applied throughout the design and development — in the conceptual phase, design engineering, fabrication, test, installation, checkout, operations, modernization, and retrofit phases of the program.

14-2.4.1 Safety Criteria

Safety criteria for weapon systems must be compatible with both combat environments and peacetime conditions. This requirement presents broad engineering design implications since armament systems are necessitated in the first place by combat operations. While every effort must be made to achieve the maximum level of safety consistent with armament system design and operation under all conditions, this objective also must be considered with respect to the attainment of maximum tactical effectiveness. Where rigorous safety criteria impose major design complexities that may result in decreased tactical efficiency, an acceptable level of safety may have to be established. This level must be based upon a thorough engineering safety analysis and an evaluation of the overall integrated weapon systems.

A failure mode and hazardous effect analysis (FMHEA) should be conducted for the gun system/vehicle interface design. This analysis considers the effects of both subsystem failure and personnel error during both operational use and maintenance. Data from other reliability, maintainability, system safety, and human factors analyses can be used when applicable. Safety hazards are defined in MIL-STD-882 as Class I, Negligible; Class II, Marginal; Class III, Critical; and Class IV, Catastrophic. The effect analysis should identify failure modes, their resultant effect upon the helicopter and personnel, compensating provisions, and the hazard classification.

When Class III or Class IV hazards are identified, immediate action must be taken to eliminate Class IV items and to minimize Class III items, consistent with design objectives. System safety requirements are discussed further in Chapter 3, AMCP 706-203.

14-2.4.2 Fire Interrupters

Fire-interrupting devices are required on gun turrets for the following reasons:

1. To protect against firing at some part of the aircraft structure, rotors, landing gear, external stores, etc.
2. To protect against firing into the path of other firing ordnance, such as other guns, rockets, or smoke grenades
3. To interrupt the weapon firing when the error signal between the fire control system sight line and the gun line is outside specified limits

4. To interrupt the weapon firing when any other predetermined conditions are considered to be hazardous to personnel or to the vehicle.

Fire interrupters generally are electronic, and use servo system logic to determine gun position and its relationship to a potentially hazardous situation. Fire interrupt is controlled by circuitry located on specific logic circuit cards. Because fire interrupters are electronic, they may not possess the positive control found in such mechanical devices as contour followers. In this regard, for safety, particular emphasis must be placed upon failure analysis of fire interrupt devices. Fire interrupters may be employed along with positive turret travel-limiting devices for added safety.

14-2.4.3 Contour Followers

The contour follower is a positive mechanical means of limiting turret travel in those areas in which unsafe operation otherwise would result. It usually is a cam built into the turret in order to prevent firing at a wing store, for example, as the gun barrel is slewed, with the cam allowing the turret to contour follow a path around the store. The cam contour must be positioned so that the gun travel, as limited with the gun dispersion superimposed, is at a safe distance from the prohibited area. The design of the supporting structure must be such as to avoid unsafe deflection if the gun strikes the contour at maximum rotational velocity.

14-2.4.4 Burst Limiters

Overheated gun barrels on fixed or turreted weapons can create hazardous situations. Firing of machine guns for extended bursts can cause:

1. Damage to the weapon
2. Weapon malfunction, with the attendant possibility of misfire or of exploding ammunition, which could result in damage to the helicopter or injury to personnel
3. Inability to re-fire the weapon until after a considerable cool-off period.

Burst-limiting devices generally are classified as either temperature-limiting or time-limiting. A temperature-sensing device normally is located on the gun barrel. A time-limiting device can be located remotely in the weapon control circuitry. ARMCOM should be consulted for information about burst-limiting requirements for specific guns and for details about current burst-limiting devices.

14-2.4.5 Cockpit Noise

Noise resulting from weapon firing can affect hearing, produce annoyance and discomfort, reduce the level of skilled performance, and contributes to human error. Temporary hearing losses can occur as a function of high noise levels, duration of exposure, and the size of the bandwidth within which the energy is concentrated.

The human ear can withstand extremely intense sounds, such as gunfire, for a few seconds without lasting effects, but prolonged exposure to intensities of 85-90 dB or above can cause damage to the ear and resultant hearing loss. The effects of exposure are cumulative. A level of 95 dB is the limit for an 8-hr exposure, while 130 dB can be tolerated for only 10 sec. Design attention must be given to resolution of noise problems in order to achieve safe operation. The limiting of noise to safe levels can be accomplished by:

1. Direct reduction of noise at the weapon, through the use of a muzzle device
2. Damping through suitable mounting provisions
3. Attenuation or reflection of the sound by means of insulation, baffles, or ear-protection devices.

Weapons should be mounted to prevent direct noise radiation through openings in the aircraft.

The applicable military specification for crew compartment noise level is MIL-A-8806. Also, MIL-STD-1472 states that equipment *shall not* generate noise in excess of the maximum allowable levels prescribed by HEL-STD-S-1-63, AFR 160-3, BuShips Specification S-1-10, BuMedInst 6260.6A, MIL-STD-740, or MIL-A-8806, as applicable. The impact of noise upon the safety and effectiveness of personnel is described in par. 13-2.2, AMCP 706-201. Acoustical noise detectability is covered in par. 8-5.6, AMCP 706-203.

14-2.4.6 Debris Disposal

Effective disposal of fired ammunition cases, misfired rounds, and links from a firing weapon must be considered from a safety design standpoint. Debris emitted from a fixed or turreted gun can be:

1. Blown into the helicopter structure with damaging results
2. Be deflected into antennas, lights, external stores, etc.
3. Be diverted by air currents into the main or tail rotors or engine inlets, causing possible catastrophic failure
4. Cause gun jams, rendering the weapon system inoperable
5. Be deflected onto cockpit glass areas, causing personnel injury.

Certain weapons provide sufficient debris ejection velocity to preclude debris disposal problems when the weapon is installed on certain helicopters. Each installation must be analyzed individually. Helicopter velocity, gun ejection velocity, air conditions, proximity of debris to the helicopter, etc., must be considered initially to determine debris disposal. Design analyses then must be confirmed with either wind tunnel or flight testing.

Debris may be stored within the vehicle after firing. This eliminates disposal problems, but introduces other disadvantages such as retention of useless weight and undesirable gun gas in spent cases. Also, storage of debris requires that a debris area be designed into the vehicle, thereby using valuable space.

14-2.4.7 Toxic Explosive Gas Protection

The applicable criteria for toxic gas concentrations are contained in MIL-STD-800. Threshold limit values for personnel exposure are contained in Ref. 2.

Gun gases can have toxic effects on personnel if allowed to enter the crew compartment. Adequate sealing, purging, and ventilation should be considered in order to avoid this hazard. Accumulation of gun gases in mixture with the proper proportion of air also can result in explosion and catastrophic failure. Gun bay or turret areas must have adequate natural or forced air (blower) ventilation to maintain gas levels below their explosive limits. Concentration of gases above 75% of the lower explosive limit, the proportion at which burning can occur, is considered hazardous. Design of a purging system must consider hover and ground firing conditions when air in the proximity of the gun bay may virtually be stagnant. Rotor downwash may be considered as a means of gas dispersal. The purge system *shall* be designed to remain operational for some time after cessation of gunfire to avoid trapping gases. Ventilation or ram air purging systems should be designed so that air intakes are located away from the gun muzzle in order to prevent intake of gun gases along with purging air. Care must be taken not to locate any possible electrical arcing sources in the area of gun gases. Final composition of gas levels, and proof of purging design, must be determined by actual firings monitored by proper detection equipment.

14-2.4.8 Turret Master Power Switch

To prevent inadvertent operation of a remote control turret while it is being serviced, a master power switch that is accessible from the turret servicing area *shall* be provided.

14-3 PROTECTIVE SUBSYSTEMS

14-3.1 GENERAL

Helicopter survivability is defined as the ability of the vehicle to endure or withstand a man-made hostile environment in the accomplishment of its mission. Survivability can be enhanced by controlling the exterior signatures of the aircraft (such as IR signature and noise) by incorporating passive and active countermeasures to suppress acquisition, tracking, and fire control (such as radar jammers) by incorporating performance and agility and, finally, by designing the aircraft to withstand gunfire.

The appropriate mix generally is determined by the Army based on the purpose of the aircraft and postulated threat. Thus the aircraft designer's job is simplified since he is supplied with the aircraft survivability features and characteristics, which he must strive to attain. Various techniques for achieving these goals are discussed in appropriate paragraphs.

The ability of an aircraft to withstand gunfire generally is termed "vulnerability reduction" or "ballistic hardening".

Vulnerability reduction design depends upon at least three factors: (1) the threat characteristics, such as caliber and type of projectile, impact velocity, and angle of obliquity; (2) the target (helicopter component) characteristics, such as material, thickness, and operating stresses; and (3) the relationship between the behavior of the target component and that of the remainder of the helicopter. Vulnerability reduction during new design generally can be accomplished more efficiently by proper subsystem design technique than the use of armor. An optimum design is characterized by armor being limited to the crew station. Methods used to achieve vulnerability reduction can be classified as burial, concentration, duplication, separation, ballistic resistance, ballistic tolerance, and, as a last resort, armor.

Duplication, as an example, is the optimal method for protection of flight control systems. Associated problems are the need to separate to be effective and how far to extend the redundancy. Duplicate controls running side by side would not reduce overall vulnerability significantly, since a single hit might rupture both sets. Moreover, if both sets terminate at a common actuator, that actuator becomes the vulnerable element.

Shielding and concentration of components generally go together. The numbers of heavy structural members and components and subsystems that can be regarded as not mission-critical are very limited in a properly designed helicopter. The practice of concentrating critical components within a small volume, and then shielding them with less critical systems and structural members, reduces the statistical probability that any single projectile will hit them, but it also may increase the vulnerability of the helicopter to any projectile that does penetrate the critical core.

A design that buries concentrations of sensitive components inside the helicopter core, or even hides them behind a single layer of structural protection, may affect maintenance and serviceability. Analytical procedures for evaluating these trade-offs are well known, but the importance that should be assigned to the various values in the analysis is not so firmly established.

Ballistic resistance is the construction of critical components such that they are massive enough to defeat the stated threat; for example, a control rod constructed out of steel armor material. As opposed to this, ballistic tolerance is the construction of a component such that the projectile passes through but the item still functions; for example, a multipivot point bellcrank of a nonshattering composite material.

Since vulnerability reduction by inherent design is discussed under the appropriate subsystems, par. 14-3 generally concentrates on the armoring method.

14-3.2 DEVELOPMENT OF VULNERABILITY REDUCTION SYSTEMS

Designing for vulnerability reduction, irrespective of environment, involves an analytical procedure that begins with a study of effects of enemy weapons. The designer then determines how the existing or proposed helicopter system couples into, or responds when exposed to, these effects. The uncertainties in the first group of data are minor since weapon effects are usually well documented; but in the area of system coupling, they are very great, especially for systems in the early stages of design.

A ballistic vulnerability reduction program *shall* be conducted with the development program to assess effects of design concepts, and provide analysis and guidance for controlling and reducing vulnerability by the most effective means. A vulnerability analysis of the complete aircraft is conducted as outlined. The definitions, criteria, and general methodology have been standardized by triservice agreement.

14-3.2.1 Vulnerability Analysis

The vulnerability analysis is presented in the form of vulnerable area as a function of striking velocity for each threat and each category of kill. The key threats to be considered in this study area are generally 7.62 mm API, and 23 mm HEI, (low and mid-intensity threats). The striking velocities to be considered are muzzle velocity, velocity at expected engagement range, and velocity at maximum effective range unless otherwise stated.

The categories of kill which are mutually exclusive are attrition, forced landing, and mission abort. Each is defined:

1. Attrition. Damage to the helicopter which causes the helicopter to crash and become a complete loss after the terminal ballistic damage occurs.

2. Forced Landing. Damage to the helicopter which causes the pilot to land (powered or unpowered) because he receives some indication of damage (a red light, low fuel level warning, difficulty in operating controls, loss of power, etc.). The extent of damage may be such that very little repair would be required to fly the helicopter back to base; but, if the pilot continued to fly, the aircraft would be destroyed. The forced landing kill category includes a forced landing at any time after damage occurs (within specified design mission duration).

3. Mission Abort. Damage to the aircraft which causes the aircraft to be unable to complete its defined mission.

The following procedure and methodology are used in the vulnerability analysis.

1. Target Technical Description. Detailed information on the construction and operation of all the systems, subsystems, and components form the major portion of the target technical description. The description includes a tabulation of all critical components, listing their title and function along with the failure mode; and the cause and the effect of a failure on the components, subsystem, and system (see Table 14-2). All potential damage mechanisms are considered as well as secondary damage effects such as fires, explosions, and leaking fluids. Scale drawings with dimensions of the aircraft configuration with locations of its major systems and their components are used to determine the presented areas of these components for up to eighteen (18) attack directions (i.e., azimuth, elevation 0,0; 45,0; 90,0; 135,0; 180,0; 225,0; 270,0; 315,0; 0, −45; 45, −45; 90, −45; 35, −45; 180, −45; 225, −45; 270, −45; 315, −45; 0 +90; and 0, −90). Normally, the number of aspects used are restricted to the cardinal views (front, rear, left side, right side, and bottom); more views are used where required for improved accuracy. If a computer is used, these drawings can be used to determine the input data for the computer. The drawings also should provide data concerning the shielding offered to individual target components by other portions of the vehicle. Additional detailed drawings may be required for critical components. Examples of critical

TABLE 14-2. VULNERABILITY DAMAGE CRITERIA DATA SUMMARY

CRITICAL COMPONENT	FAILURE MODE & CAUSE	EFFECT OF FAILURE ON COMPONENT SUBSYSTEM AND SYSTEM
TITLE & FUNCTION		

TABLE 14-3. VULNERABILITY TABLE

SYSTEM	COMPONENT	PRESENTED AREA (A_p, ft^2)	FORWARD FLIGHT				HOVER MODE			
			Singly Vulnerable		Multiply Vulnerable		Singly Vulnerable		Multiply Vulnerable	
			Attrition	Forced Landing	Attrition	Forced Landing	Attrition	Forced Landing	Attrition	Forced Landing

components may include:

a. Engine compressor
b. Fuel control
c. Engine oil-cooler
d. Transmission oil-cooler
e. Hydraulic module and reservoir
f. Fuel cross-over valve
g. Fuel cell sumps
h. Control linkage and bell cranks
i. Actuators
j. Crew
k. Drive shafts.

2. Kill Definition. Attrition, Forced Landing, Mission Abort.

3. Kill Criteria. For convenience of analysis, the helicopter generally is divided into distinct systems (i.e., crew, propulsion, fuel, flight controls, rotor, power train, etc.). Each system contains components that are vital to the successful operation of the system. Damage to a single component is said to be singly vulnerable. Components of a set are said to be multiply vulnerable in a given kill category if damage to less than $(n-1)$ members of the set does not result in a helicopter kill, but damage to at least n members of the set does result in a helicopter kill. Although the primary vulnerability analysis is conducted for the aircraft in forward flight, the list of vital components *shall* show the level of kill for damage when the helicopter is in a hover mode of flight (see Table 14-3).

4. Probability of a Kill Given a Hit ($P_K|H$) for Each Component. The designer assigns the $P_K|H$'s to all the components and submits the list for approval to Ballistic Research Laboratories (BRL) at Aberdeen Proving Ground, Maryland. $P_K|H$'s are based on experimental firings on subject item, or like items.

5. Presented Area (A_P). Once the conditional kill probabilities for each component have been ascertained, it is necessary to reconsider the target technical description to determine component presented areas. One of the least complex and most frequently used means of obtaining presented areas of helicopter components employs a planimeter to measure them from scale drawings made for each of the principal views to be considered. Several computer programs, notably the MAGIC (used at BRL) Program and the SHOTGUN program, have been developed for obtaining presented areas, as well as shielding for the components. During this process of obtaining presented areas, it is necessary to consider the space orientation of the individual components and evaluate the masking or shielding provided to each component by the rest of the helicopter. In general, any shielding provided to a particular component by the rest of the aircraft will be reflected as a reduction in the conditional kill probabilities for that component for all or some of the velocities for a particular view or views. This reduction of conditional kill probabilities applies for that component for all or some of the velocities for a particular view or views. This reduction of conditional kill probability is not arbitrary but is based on the residual weight and velocity of the threat after it has perforated the shielding material. A component may be completely shielded for some or all of its presented area for one or more of the views considered. If this occurs, the presented area of the component for these views must be reduced correspondingly. When armor is employed to protect aircraft components, the effect of the armor is analyzed as masking. When the armor is perforated, behind-plate effects will be accounted for; i.e., number, weight, velocity, and direction of pieces. (For some new types of armor, new firing data will be required.) Data on material shielding properties are contained in the classified data on material "Thor" series of reports. Consult BRL concerning this subject

6. Vulnerable Area (A_V). After the conditional kill probabilities have been estimated for the components of the helicopter and the presented areas of these components have been determined, vulnerable areas of these components are generated. For a given threat, striking velocity, and view, the vulnerable area of a component is obtained from Eq. 14-1.

$$A_V = A_P \cdot P_K | H \qquad (14\text{-}1)$$

where

A_V = vulnerable area of component
A_P = presented area of component
$P_K|H$ = component conditional kill probability

Vulnerable areas generally are classified as one of two types, singly vulnerable area or multiply vulnerable area for singly and multiply vulnerable components. The singly vulnerable area A_{V_T} of the system is the sum of the singly vulnerable areas A_{V_i} of the components of the system.

$$A_{V_T} = \sum_{i=1}^{n} A_{V_i} = \sum_{i=1}^{n} A_{P_i} \cdot (P_K|H)_i \qquad (14\text{-}2)$$

Only a limited amount of subtotaling may be accomplished in a system of multiply vulnerable components. If, for example, vulnerable areas were computed for a multiply vulnerable set of components forming a propulsion system which contained two engines, two engine transmissions and two drive shafts linking the engine transmission to some other transmission, then the vulnerable areas for the right engine, right engine transmission and right drive shaft would be subtotaled and the vulnerable areas for the left engine transmission and left drive shaft would be subtotaled. The resulting subtotals would represent the vulnerable areas of two independent subsystems which function together as a redundant subsystem (i.e., multiply vulnerable).

14-3.2.2 Vulnerability Reduction Checklist

The partial check list that follows may be useful for insuring that features are not overlooked. Discussions of these items are located throughout this handbook.

1. Pilot(s):
   a. Seat armor
   b. Helmet
   c. Body armor
   d. Cockpit armor
2. Fuel Cells:
   a. Self-sealant
   b. Internal foam
   c. Backing board
   d. External foam
   e. Fire detection/extinguishing
   f. Hydraulic ram
   g. Blast loads
   h. Multiple tanks
3. Fuel lines:
   a. Self-sealant
   b. Shutoff/isolation valves
   c. Duplication/separation
   d. Routing behind structure
4. Engine and Power Train:
   a. Suction fuel pump
   b. Self-sealing oil reservoirs
   c. Engine and transmission operation without lubrication
   d. Damage tolerant shafting
   e. Damage tolerant gearboxes and transmissions
   f. Fire detection/extinguishing
   g. Shielded fuel control
5. Finish:
   a. Insignia or symbols that could be used as aim points absent or subdued.
   b. Matte/camouflaged paint scheme
6. Flight and Engine Controls:
   a. Redundant mechanical controls
   b. Redundant hydraulic systems
   c. Fire resistant hydraulic fluid
   d. Boost actuators jam proof and redundant
7. Shatterproof high pressure containers
8. Structure:
   a. No singly vulnerable structure
   b. Blast (internal/external)
9. Instrument Panels. Console and instruments featuring antispall characteristics
10. Canopy/Windows/Doors:
    a. Transparent material that does not produce secondary fragmentation
    b. Flat plate canopy and window for control of sunglint
11. Rotor Blades Redundant Spars
12. Electrical System Redundant.

14-3.2.3 Vulnerability Data Presentation

Total aircraft vulnerability for all threats — striking velocities, directions, kills, etc. — should be considered and presented in a manner that will facilitate evaluation of the net effects obtained by subsequent changes in component size, location, or kill probabilities and protection additions.

14-3.2.4 Aircrew Armor Configuration Development

The objective of the armor designer is to minimize the area of the crew remaining exposed to the threat. Extensive armoring, however, adversely impacts other essential crew station requirements, such as vision and emergency egress. Therefore, several alternate armor configuration concepts should be developed in accordance with MIL-STD-1288 and the effects of each upon aircrew helicopter performance should be evaluated similar to methodology shown in Fig. 14-1. The configuration that best satisfies the protection need by producing the smallest changes in system weight, crew vision, and crew motion envelopes while also providing the largest increase in crew protection can be proposed as an optimum integrated armor system. Surveys of experienced Army aviators should not be overlooked as a possibility in achieving a balanced configuration.

The armor configuration that is most widely accepted consists of an armored seat bucket, individually worn chest protector, and supplemental airframe mounted plates.

14-3.2.5 Armor Material Selection

Testing and evaluation of armor materials are generally unnecessary for the selection of optimum armor with the lowest possible weight. Refer to Chapter 2 for guidance on selection. All armor materials *shall* be qualified in accordance with USAAVSCOM specification 1560-MULTI-001, *Procurement Specification for Lightweight Aircraft Armor*.

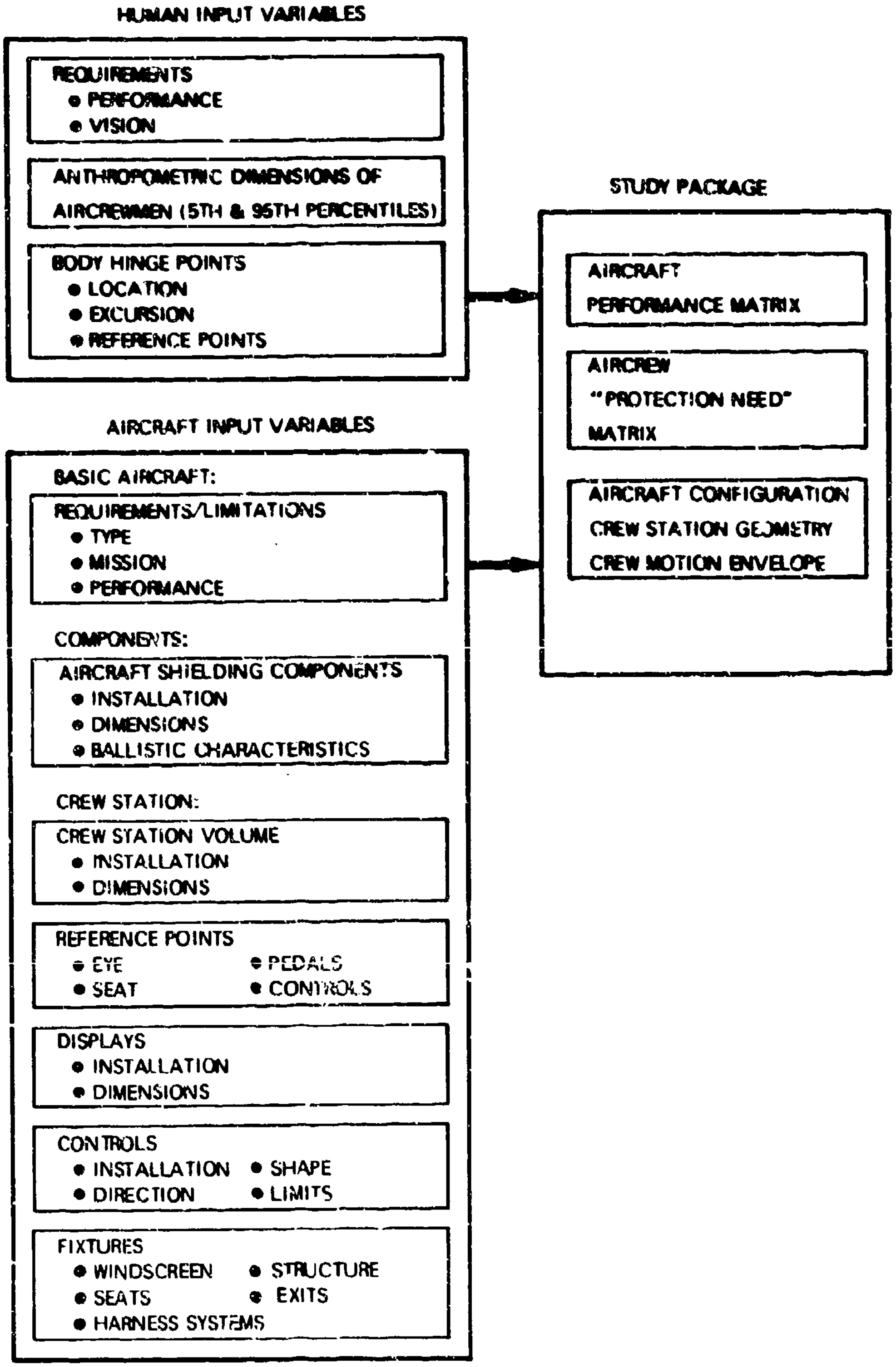

HUMAN INPUT VARIABLES
REQUIREMENTS
● PERFORMANCE
● VISION
ANTHROPOMETRIC DIMENSIONS OF AIRCREWMEN (5TH & 95TH PERCENTILES)
BODY HINGE POINTS
● LOCATION
● EXCURSION
● REFERENCE POINTS
AIRCRAFT INPUT VARIABLES
BASIC AIRCRAFT:
REQUIREMENTS/LIMITATIONS
● TYPE
● MISSION
● PERFORMANCE
COMPONENTS:
AIRCRAFT SHIELDING COMPONENTS
● INSTALLATION
● DIMENSIONS
● BALLISTIC CHARACTERISTICS
CREW STATION:
CREW STATION VOLUME
● INSTALLATION
● DIMENSIONS
REFERENCE POINTS
● EYE
● PEDALS
● SEAT
● CONTROLS
DISPLAYS
● INSTALLATION
● DIMENSIONS
CONTROLS
● INSTALLATION
● SHAPE
● DIRECTION
● LIMITS
FIXTURES
● WINDSCREEN
● STRUCTURE
● SEATS
● EXITS
● HARNESS SYSTEMS
STUDY PACKAGE
AIRCRAFT PERFORMANCE MATRIX
AIRCREW "PROTECTION NEED" MATRIX
AIRCRAFT CONFIGURATION
CREW STATION GEOMETRY
CREW MOTION ENVELOPE

Figure 14-1. Study Input Variables

14-3.3 ARMOR INSTALLATION DESIGN CONSIDERATIONS

The basic approaches to the armor type protection of helicopters from ground fire consist of:

1. Placement of armor in very close proximity to the component for which protection is required.
2. Incorporation of armor into the vehicle structure or into the component structure
3. Consideration of benefits gained (if any) from the shielding (masking) effects of helicopter structure and/or components.

These basic approaches have been divided into three principal passive protection concepts — integral armor, parasitic armor, and indigenous armor.

Integral armor consists of armor incorporated into the aircraft as a replacement for component structure (e.g., hydraulic actuator housings, which are fabricated from armor) or for existing load-carrying structure. The most obvious use of the integral armor concept is in the floor of an aircraft having relatively small crew stations and/or areas of vulnerable components. Integral armor can be expected to be heavy, and thus is limited generally to smaller installations. The reason for the inherently higher weight of integral armor is that it is generally metallic. The ceramic composites and other nonmetallic materials are not well suited for use as load-carrying members because of their lack of strength or low ductility. Also, because integral armor is by definition a permanent installation, materials having poor multiple-hit-protection capability are not suitable. Permanent or integral armor should be made from materials that have the capability to withstand multiple hits, and that can be repaired in some manner without the necessity for removing structure or components.

Parasitic armor is attached in some manner to the helicopter structure. It provides only ballistic protection, and does not function as a load-carrying member. This concept provides more flexibility in the selection of materials than do the integral and indigenous armor concepts. There are, however, potential drawbacks, including the need for sufficient structural hard-points for adequate attachment of armor; the necessity for strengthening of backup structure; interference with access doors and other items essential to the proper use of the helicopter; and the establishment of realistic tolerances to allow armor panel interchangeability in the field between helicopters of the same type. When parasitic armor is installed on the exterior of the helicopter, crashworthiness is not affected particularly. However, if the armor panels are located in the interior of the helicopter, design of the bracketry and attachments is dictated by a crash-load criterion. Convertibility to peace-time operation is an attractive feature of a parasitic armor installation.

Indigenous armor refers to the benefits, if any, that may result from shielding of the critical component by aircraft equipment or structure that normally is located, or possibly may be relocated, between the projectile and the component which requires protection.

The suggested technique is to take maximum advantage of the structure and equipment of the helicopter in providing ballistic protection for the item. This involves accounting for the indigenous protection that is inherent to the original helicopter configuration, as well as relocating equipment to improve the protection. Consideration is best given to ballistic protection in the preliminary configuration design stage. Under the impact of a projectile, certain items of equipment and structure will create fragments and splinters that, while they are not lethal individually, will form a pattern covering a large area, thus greatly increasing the probability of a personnel hit. Other items will arrest the projectile energy completely and give, thereby, 100% ballistic protection. The fact that the net effect of indigenous materials may be either positive or negative indicates the difficulty of devising a method for employing indigenous armor. Consult BRL concerning data relative to the ballistic properties of various structural materials. Refer to Volume II, Ref. 5 for the behavior of fuel cells as masking.

14-3.3.1 Aircrew Torso Armor

Service experience with seat mounted torso armor (chest protection) has been unsatisfactory and resulted in the development of the individually worn armored vest. Armored seats and restraints *shall* be designed to interface with these vests and use them as the forward protection. US Army Natick Development Center, Natick, MA, is responsible for this equipment and details may be obtained from there.

14-3.3.2 Interchangeability

Armor of identical location *shall* be interchangeable between helicopters of the same model. If armor for more than one level of protection is provided, both should be interchangeable (see MIL-I-8500).

14-3.3.3 Removability

Except for the integral armor, each component of the armor system should be capable of being removed by a maximum of two men, using tools normally found in line maintenance areas.

Parasitic armor should be removable in order to permit inspection, repair, and maintenance and to provide access to masked components. Consideration should be given to the removal problems associated with combat operations. The design of armor should take maximum advantage of the modular concept with respect to size and contour.

The maximum weight of a single piece of armor should not exceed 80 lb. Where the working area is restricted by existing structure and/or equipment, the weight should be reduced accordingly.

Transparent armor should be readily replaceable and cleanable. Removal should be possible without disturbing the fire-control system.

Each component of the armor system which requires three or more hours for removal should be considered as structural or integral. Such armor should be evaluated as part of the helicopter structure with respect to load-transmission characteristics and fatigue life.

14-3.3.4 Flying Qualities

The removal of armor, partial or total, should not adversely affect the flying qualities of the helicopter. The use of ballast should be avoided.

14-3.3.5 Immobilization

Immobilization, as related to armor, is defined as rendering any moving part of a structure immovable following ballistic attack. The three most common types of immobilization are burring, keying, and deformation.

The edges of pieces of armor may become cracked or torn as a result of projectile impacts. When two armor surfaces are very close together their torn edges may come in contact with one another. The resultant burring often causes immobilization, because the raised edges prevent or restrict movement. Keying occurs when a projectile or fragment becomes wedged between two surfaces, or when a projectile penetrates one surface and partially penetrates the other surface so as to lock or "pin" the two movable parts together. Deformation results when projectile impacts swell the metal or push it out of shape, thus jamming moving surfaces together or otherwise preventing normal operations.

Protection against, or insensitiveness to, these types of immobilization always must be considered in designing moving parts. In some cases, the use of projectile deflector strips prevents immobilization. A large clearance between moving parts also offers protection against immobilization.

14-3.3.6 Armor Material Attachment/Installation

14-3.3.6.1 Mounting of Armor Plate

Armor plate *shall* be mounted on strong, rigid structural members or on energy attenuating members that are designed to reduce the peak impact loads thereby reducing the structural requirements and hence the weight of local airframe structure. Deflection of the armor *shall* be considered in determining space allowances. Wherever possible, mountings should be on the rear side of the armor plate (remote from the anticipated direction of attack). This arrangement, which is advantageous from the strength viewpoint, also prevents damage to the mountings from gunfire.

Refer to Figs. 19 and 20, Ref. 5, for armor attachment methods.

The structure and brackets for armor support should exhibit sufficient strength to withstand the normal loads of flight, gust, blast, and landing, as well as crash loads. The strength of hinges, locks, and fasteners on doors and/or removable inspection panels should be maintained at the design level. The added weight of the armor should not stretch springs, damage hinges, or warp doors or panels to which armor is attached. The attachments should retain sufficient strength after the limit load imposed by ballistic impact to withstand flight loads without breaking loose or incurring additional deformation that would cause interference with any critical component. The bracketry may be designed to yield under limit load. When armor such as hard-faced steel must be oriented in a particular manner, attachments should be positioned to preclude improper installation. The sizing of fasteners is determined most conveniently by gunfire tests.

When practicable, the armor attachments should be designed to take the impact loads in compression or shear, but not in tension.

Armor should be attached at three or more points having sufficient strength to support the armor and to withstand normal operational loads in the event one attachment point is shot away.

The standoff space provided by the armor attachments should be such that any deformation and/or deflection of bracketry and armor will not cause interference with the functioning of the armor-critical component.

14-3.3.6.2 Installation Design

The installation of armor *shall not* preclude the rapid egress of the crew in emergency situations.

Metallic armor should have no discontinuities or rough edges that might set up stress raisers. All edges should be broken or deburred in order to prevent delayed cracking.

Ceramic composite armor should not be used for primary aircraft structure.

Shock-susceptible components should not be mounted on armor panels. After ballistic acceptance testing, no thermal processing is permissible without reverification of conformance with the ballistic acceptance limits. Deviation from this requirement *shall* be subject to review and approval by the procuring activity.

Armor should not be attached by methods that transmit the impact shock from the armor to the critical component.

In general, the following detailed installation requirements should be met:

1. Use flat plates except where a simple curved or bent shape is advantageous in gaining angular protection or weight savings.
2. All armor installations should provide space for a possible future increase of 50% in armor thickness.
3. Avoid the use of cutouts or holes in any portion of the plate for supporting or clearing miscellaneous apparatus.
4. Do not allow any cutting or burring after the final delivery from the armor manufacturer's plant since this may locally degrade the ballistic capability.

14-3.3.6.3 Bullet Splash and Spall

Bullet splash is defined as particles of the projectile formed from the impact against armor. With steel armor the splash tends to travel along the surface of the armor, much like the flow of a fluid. Bullet splash is dangerous to the eyes and bodies of the crew, and, when sizable particles are included, can cause damage to equipment. The effective methods of providing protection against bullet splash are to deflect it away, to trap it, or to turn it back along its original course by specially designed deflecting surfaces.

Any steel armor in such a position that it could direct secondary or bullet splash fragments into vital components *shall* be provided with flanges, spall shields, or splash strips (peripheral fences) to deflect these particles.

Spall is the fragmentation of the armor, either on the impact side or on the reverse, with or without complete penetration of the armor. The armor materials listed in Chapter 2 feature adequate spall resistance, with the exception of the ceramic composites. The remedial action is to overlay the exposed ceramic facing with a bonded layer of ballistic nylon cloth and, where required, to curl the backing up beside the tile at panel edges.

The splash or spall produced on ceramic composite armor tends to form a rather narrow cone centered about the projectile flight path.

REFERENCES

1. 1560-MULTI-001, *Procurement Specification for Lightweight Aircraft Armor*, USAAVSCOM, St. Louis, MO, 17 Oct 66.
2. MIL-STD-1288, *Aircrew Protection Requirements, Nonnuclear Weapons Threat*, 29 Sep 1972.
3. *American Conferene of Government Industrial Hygienists, Committee on Threshold Limit Values*, Cincinnati, OH, Revised 1966.
4. ADS-11, *Survivability/Vulnerability Program Requirements*, USAAVSCOM St. Louis, MO, 7 Sep 1972.
5. TR 7-41, *Survivability Design Guide for US Army Aircraft*, Vol I (U), Vol. II (C) USAARDL (now USAAMRDL), Eustis Directorate, Ft. Eustis, VA, Nov 1971.

CHAPTER 15

MAINTENANCE AND GROUND SUPPORT EQUIPMENT (GSE) INTERFACE

15-1 INTRODUCTION

This chapter contains basic guidelines, considerations, and requirements that must be followed by the design engineer to insure satisfactory interfaces between the helicopter and its required maintenance ground support equipment (GSE). The design guidelines presented herein are based upon Army circulars, regulations, and policy manuals related to operation and maintenenace support of supplies and equipment.

GSE interface design must consider standard items of GSE and the maintenance personnel capabilities available within the Army's system, and must be in consonance with the Army's maintenance support policy. Specific maintenance and GSE interface design provisions should use built-in-test equipment (BITE), where feasible, and must improve efficiency in the use of GSE at each maintenance level. When incorporated, these interface design considerations and requirements will minimize the time required for turn-around servicing, maintenance, and repair; thus, maintainability of the helicopter will be improved.

As used in this chapter, the term GSE includes all equipment needed to service, inspect, test, adjust, calibrate, fault isolate, measure, assemble, disassemble, handle, transport, safeguard, store, repair, overhaul, maintain, and operate the helicopter and its installed subsystems but excludes personnel equipment, office furniture and equipment, and common production tools and tooling.

15-2 DESIGN CONSIDERATIONS AND REQUIREMENTS

The requirements for GSE interfaces must be considered from the inception of the helicopter design. Reliability, maintainability, serviceability and self-test features must be designed into the helicopter and its installed subsystems to minimize the costs of maintenance facilities and manpower.

Human factors considerations, safety, and accessibility must be included in the initial design. Similarly, standardization of equipment parts must be emphasized, beginning with design inception. Designs should conform to the human engineering principles and criteria of MIL-STD-1472.

Equipment arrangements should minimize the need for removing equipment when servicing is performed. Design considerations should include analysis of the operational deployment requirements, along with related climatic and environmental factors. Helicopter and subsystem installations should be designed to permit personnel wearing heavy gloves and clothing to perform maintenance in cold climates. Instruction markings should be legible and placed so that they are viewed easily. The design should permit ground servicing by maintenance personnel with a maximum of safety and a minimum of skill. The design must assure that personnel with minimum training and average mechanical abilities can perform the required servicing, maintenance, and repair of the helicopter and installed equipment. Sharp projections that may injure personnel during operation or servicing must be eliminated. Vital components must be protected to prevent damage during servicing.

15-2.1 SAFETY

Safety is a principal consideration in the design of GSE interfaces into the helicopter system. GSE interface design *shall* conform to system safety criteria, principles, and techniques as defined in MIL-STD-882 and Chapter 3, AMCP 706-203. The objectives are maximum safety consistent with military operational requirements; control of hazards to protect personnel and equipment; and identification, elimination, or control of hazards associated with each system, assembly, or subassembly. These considerations should include, but not be limited to, the following:

1. Human factors
2. Level of training required for safe servicing and maintenance
3. Characteristics of fuels and hydraulic fluids, and their hazard levels during storage, transportation, and handling
4. Containment of electrical and radio frequency energy and appropriate warnings
5. Protection of pressure vessels and associated piping
6. Classification of hazards resulting from essential use of explosives.

Particular attention must be given to possible malfunctions that could create hazards, and appropriate design features must be incorporated to eliminate or control these hazards.

General rules to be followed in safety design are:

1. Clearances must be provided to permit the interconnection of fuel and/or oil fill lines, along with electrical connections.

2. Appropriate valves must be provided to permit safe servicing in cases where pressure is maintained, as in pneumatic and hydraulic systems.

3. Electrical connectors must be keyed to preclude incorrect interconnection, and the installation must prevent arcing and exposure of hot pins.

4. Ground receptacles must be provided to insure a common ground and to eliminate any electrical potential difference between the helicopter and GSE.

5. Guards and warnings must be provided in situations where explosive devices are installed or where high electrical potentials are present or for other potentially sudden and serious hazards.

6. Interlock switches and devices must be provided as necessary.

7. Connections should be quick disconnect type, vibration proof, and not require safetywire.

15-2.2 ACCESSIBILITY

The designer should emphasize ease of servicing, testing, removal, and replacement of all equipment. Inaccessible and complex structural arrangements must be avoided. Moreover, the designer should consider sectioning the helicopter structure for ease of transportation, handling, and replacement in the field.

Except where weight, structural integrity, or stiffness are overriding considerations, assemblies subject to periodic removal should be attached with quick-disconnect fasteners of an approved type rather than with bolts or screws. Tool clearance must be provided for installing and removing lines, nuts, bolts, and other fasteners. Where the use of tools is restricted because of remote location, temperature, or other factors, fasteners such as self-locking plate nuts or anchor nuts should be used to allow single sided wrenching.

If lubrication is required, fittings must be located where access is possible using standard Army GSE without the need for special adapters. If positional adjustments are needed, the design should permit unobstructed adjustment over the complete range of component movement.

Accessibility for the 5th to 95th maintenance personnel should be provided for items subject to preflight inspections and servicing. Similarly, emphasis should be placed upon accessibility of all components subject to normal maintenance. Such items include fill and drain plugs, filter elements, valves, switches, and other field-replaceable assemblies.

Large, quick-opening access doors and ample space should be provided for servicing of engine accessories and replacement of components. The designer also should provide for quick removal and installation of engine(s), transmission(s), rotor(s), and propeller(s).

To the extent practicable, the helicopter design should permit complete preflight inspection without the use of special stands or ladders. Integral nonskid steps, handholds, and work platforms should be incorporated to facilitate maintenance. Similarly, utility systems should have quick-access provisions so that they can be serviced without special GSE, preferably from ground level.

Arrangements that require special tools or removal of other equipment to accomplish an aviation unit level interchange (removal and replacement) should be avoided. If it is necessary to place one unit behind another, the unit requiring less-frequent access should be located to the rear. Except for protection, or other valid reasons, equipment requiring periodic inspection, service, or replacement should not be placed behind or under structural members or other items that are difficult to remove or that can be damaged readily. Equipment should be isolated from sources of fluids or dirt.

The designer should provide built-in check points to simplify the connection of fault-isolating test equipment. Equipment arrangements should permit service and maintenance personnel to accomplish their tasks without working in awkward positions.

Maintenance at the intermediate support and depot levels will involve the use of automatic test equipment (ATE) when this is feasible and cost-effective. The helicopter system equipment and components must contain the necessary test points to interface with ATE. End items of equipment should be designed to have the test points required to permit performance evaluation and diagnostic tests — consistent with the policy of returning the helicopter to operational status by means of maximum replacement of modules — limited piece-part replacement, and repair of designated direct exchange (DX) modules by replacement of external parts or use of authorized repair kits. In turn, modules should have sufficient test points to permit performance evaluation and diagnostic tests in accordance with Army policies and practices for depot repairs.

15-2.3 STANDARDIZATION

Ground support equipment interface designs should consider two levels of standardization:

1. Equipment level. Standard GSE Items (MIL-HDBK-300 and DA Pamphlet 700-20).

2. Parts and Materials level. Military Standard (MS, AN, AND, etc.), Military Specification Qualified Products Lists (QPL).

Such standards should be used in preference to special commercial parts or designs serving the same or basically the same function. To the extent that performance is not compromised, helicopter design should permit servicing and maintenance at all levels with equipment standardized for Army use. Electrical connectors, fill plugs, fittings, and other interfacing items should permit connection of standard GSE without the use of special adapters.

Where items require modification in order to perform the required function, the designer should specify tools and tooling that will comply with Governmental specifications and documents relating to materials, processes, equivalent tolerances, and size. The designer should eliminate, where possible, the requirements for new or special tools for use on new helicopter systems.

The designer should reflect decisions of Optimum Repair Level Analysis, based on Army maintenance/support policy in the maintenance and GSE interface design. Equipment should be designed for testing at the indicated levels, giving consideration to the interface with standardized Army test equipment.

The design of individual black boxes should permit their serviceability on the helicopter to be determined by means of self-contained, go/no-go test circuitry. The circuit loops of multi black-box subsystems should be contained in a single box. This will eliminate the need for calibration adjustments at the helicopter level and will permit the use of simple, light-weight, go/no-go circuits.

15-2.4 HUMAN ENGINEERING

Maintenance personnel having a minimum of training and relatively low skill levels, and sometimes working under adverse climatic and environmental conditions, are employed to provide the needed service, maintenance, and repair of the helicopter and installed systems. Therefore, designs must be based upon the proper allocation of man-equipment performance for system operation, maintenance, and control. Thus, designs should allocate functions to optimize performance, minimize operational constraints, and minimize operating and maintenance costs. Human Factors Engineering (HFE) design criteria principles and practices may be found in MIL-STD-1472 and MIL-H-46855. Goals to achieve include:

1. Achieve satisfactory performance by operator, control, and maintenance personnel.
2. Reduce skill requirements and training time.
3. Increase the reliability of personnel-equipment combinations.
4. Foster design standardization within and among systems.

Human engineering specialists should interpret the design to define HFE problem areas and provide solutions.

Populations for design should be specified. During the iterative process of design, existing critical maintenance tasks *shall* be defined and, when possible, eliminated.

AFSC Personnel Subsystems 1-3 provide specific criteria for human engineering design requirements.

15-2.5 INSPECTION, TEST, AND DIAGNOSTIC SYSTEM

The need for increased tactical mobility requires efficient methods for inspection, test, diagnosis, and prognosis in the tactical operating environment. A design objective, when requested by the procuring activity, is to provide an automatic inspection, test, and diagnostic system capable of diagnosing malfunctions automatically, warning of impending mechanical failures, minimizing manual inspections, and permitting helicopter components to be changed on the basis of condition rather than of time. This objective may be modified by state-of-the-art, mission, and weight/volume restrictions.

The automatic inspection, test, and diagnostic system should serve the following elements:

1. Engine(s) and accessories
2. Fuel subsystem
3. Oil subsystem
4. Rotor group, including main and tail rotor transmissions, propellers
5. Flight control subsystem
6. Electrical subsystem
7. Essential avionics
8. Hydraulic subsystem
9. Airframe (critical stress area)
10. Armament subsystem.

Appropriate sensors should be installed permanently in the helicopter and in major system components, as applicable. Sensor media may include, but are not limited to, temperature, pressure, vibration, acceleration forces, liquid flow, electrical continuity, electronic characteristics, and vapor detection. Sensor outputs should terminate in a quick-disconnect outlet in the vicinity of the ground power connector terminal. The sensor outlet will interface with GSE capable of processing sensor data outputs automatically.

15-3 PROPULSION SUBSYSTEM INTERFACES

15-3.1 GENERAL

A propulsion system consists of the engine or engines, air induction subsystem, exhaust subsystem, fuel and lubrication subsystems, starting subsystem, controls, transmission subsystem, and APU (if applicable). Interface design considerations outlined in par. 15-2 *shall* apply. Specific design attention must be given to maintainability objectives and must include quick-change capabilities and accessibility for visual inspections, maintenance, and servicing. Additional propulsion system design criteria are available in Chapter 3 and in Chapter 8, AMCP 706-201.

15-3.2 INTERCHANGEABILITY/QUICK-CHANGE

Only engine components and accessories *shall* be mounted on the engine. Design objectives include component and accessory arrangements that facilitate interchangeability. Engine components and accessories that vary with installations will be removable in kit form to achieve interchangeability at minimum maintenance cost. Interchangeability requirements are outlined in MIL-I-8500.

Helicopter operational availability is sensitive to engine change requirements. Therefore, engine accessory packs or modules should be designed for attachment to the engine to form a complete engine assembly. The design of the helicopter must allow for installation and removal of the engine assembly as a single unit. To reduce life-cycle maintenance costs, ease of removal and reinstallation of the engine assembly from the propulsion system must be a principal design objective. Quick-change capability should be achieved with a minimum of special tools.

15-3.3 CONNECTORS AND DISCONNECT POINTS

Approved, standard, quick-disconnect mechanical and electrical couplings *shall* be used as required.

Automatic shutoff couplings in accordance with MIL-C-7413 *shall* be employed with fluid lines to reduce spillage and contamination during propulsion-system maintenance. For ease of alignment and minimum quick-change times, MS 28741 flexible hose, or the equivalent, should be used for fluid lines between the propulsion system and the firewall. MIL-H-25579 applies to hoses for high-temperature locations. The number of connector and coupling types *shall* be minimized. Selection of electrical connectors is described in MIL-STD-1353.

15-3.4 INSPECTION AND TEST POINTS

Sensors incorporated within the propulsion system will provide for real-time readout of engine(s) and subsystem(s) condition, performance, and efficiency. Sensors *shall* be provided at additional points as required to accomplish diagnostic testing compatible with the overall helicopter test and diagnostic system (par. 15-2.5).

15-3.5 OIL, FUEL, AND LUBRICATION

Oil system drain valves *shall* allow gravity drainage clear of the helicopter. In addition, the valves will be self-locking to preclude accidental loss of fluid.

The design of fuel-servicing areas *shall* permit easy and thorough cleaning. Areas shall be marked clearly "for fuel only" and designated as to type or types of fuel as appropriate. Filler connection designs will preclude the accidental connection of water hoses with those carrying other fluids. All fuel tanks must be capable of being drained completely and purged while on the ground.

Permanently lubricated hardware will be used wherever possible. Required grease fittings should be designed in accordance with MIL-F-3541, MS 15000, 15001, and 15002.

15-3.6 GROUNDING

A static electricity discharge path to ground must be provided through the landing gear or, for wheeled landing gear, an external ground wire. An MS 33645 receptacle for grounding the refueling nozzle should be installed not more than 42 in. nor less than 12 in. from the filler opening for gravity fueling systems only. Provisions must be made to ground the airframe through skids or other means prior to any servicing, refueling/rearming, etc.

15-3.7 STARTING

A self-contained engine-starting system is required for most Army helicopters.

External electrical power for ground starting will be furnished through connectors and receptacles in accordance with MIL-C-7974. The turbine jet-starting receptacle, if installed, will comply with MS 25018 if a split bus system is used. The standard connection for DC is MS 3506; for AC, 400 Hz 120/208 V 3-phase, MS 90362 will be used.

15-3.8 GROUND HEATERS

The designer *shall* provide a 12-in. opening for a ground heater duct (a 6-in. opening may be used on engines below 350 hp) in the accessory section, forward of the firewall and close to high-temperature-demanding units such as gearboxes and oil-system components. A heater duct opening behind the firewall also is desirable to facilitate cold-weather maintenance and servicing.

Hot air should not be directed onto electrical harnesses or other equipment that could be affected adversely by high temperatures. Openings should be marked "Ground Heater Duct".

15-3.9 ENGINE WASH

The designer *shall* provide access for connecting a ground cart to the engine water wash system.

15-4 TRANSMISSIONS AND DRIVES

General requirements for transmissions and drives are presented in Chapter 4 and MIL-T-5955. Installation design should provide for ease of replacement, repair, and servicing, and should avoid requirements for special tools and fittings. Standard eyelet fittings should be provided for attachment of slings for hoisting components such as gearboxes and transmissions.

Easy-to-read indices, keys, or other markers should be included on gears and shafts that require radial alignment during assembly and installation. Major components and assemblies should be interchangeable in accordance with MIL-I-8500.

Accessories and accessory drives *shall* be located for ease of removal and replacement. The design engineer should make maximum use of standard fittings, avoiding adapters to the maximum possible extent.

Sensors compatible with the helicopter inspection, test, and diagnostic system *shall* be incorporated (see par. 15-2.5).

Oil systems *shall* be provided with a simple, visual method of verifying oil level from the ground. Sumps should be equipped with self-locking drain valves and an adequate oil drain path clear of the helicopter structure. The oil filler cap should have positive-lock features, and should be in a clean servicing area. Consideration should be given to installation of a chain or cable to prevent loss of filler caps following removal. Required grease fittings *shall* be designed in accordance with MIL-F-3541.

15-5 ROTORS AND PROPELLERS

Quick-removal features should be provided on the cowling, spinners, and protective housings. Rotor and propeller controls should be designed with quick-disconnects to facilitate connection of control-circuit testers.

Each helicopter rotor blade *shall* be identified by serial number and marked in accordance with Army Technical Bulletin TB 746-93-2 and *shall* be interchangeable within the rotor system. Rotor blade tip markings or equivalent *shall* be provided in order to provide for ground-checking of blade alignments. The designer *shall* provide for sling lift of individual rotor blades and the rotor hub.

Blade tracking and balancing techniques should be simple so as to eliminate the need for a maintenance test flight after tracking and balancing.

The propeller assembly design *shall* provide for the use of standard tools and tool types. Also, attachment points *shall* be provided for use of the combination propeller wrench and lifting device for hoisting the complete assembly. Blades will be numbered serially and will be individually interchangeable. The design should permit blade removal, lifting, handling, and installation to be accomplished with the individual blade in a horizontal plane.

15-6 FLIGHT CONTROLS

The flight control subsystem as defined herein includes the primary and secondary flight control systems (if applicable), the rotating systems, the nonrotating systems, and the trim systems.

15-6.1 ROTATING SYSTEMS

The components of the rotating portion of the flight control system should be capable of being replaced without disturbing the rigging. A plain or tapped hole in a relatively rigid member, such as a structural beam, should be provided so that control surface rigging gages can be attached by a bolt or screw. The location must be accessible readily so that maintenance/support personnel can perform the necessary inspections and adjustments. Care must be taken to provide the work space necessary for the adjustment of pivot shafts, push-pull tubes, and bell cranks. When hydraulic or electrical boost devices are used, sufficient access must be provided to allow for inspection, servicing, and maintenance.

If autopilots and/or stability augmentation systems are used, self-testing provisions must be included to the extent feasible. Where self-test is not feasible, test connectors for isolating faults of a failed component should be provided so that test equipment can be attached externally.

15-6.2 NONROTATING SYSTEMS

Par. 15-6.1 applies equally to nonrotating systems. When cables and pulleys are used in lieu of bell cranks and push-pull tubes, a means of checking and setting the cable tensions to specified values must be provided. In all cases, wear points, such as bell-crank bearings must have access for inspection and correction.

15-6.3 TRIM SYSTEMS

Interfaces for cable and pulley, or bell crank and push-pull tube, adjustments must be provided so that trim surface free play can be maintained within specified limits.

15-7 ELECTRICAL SUBSYSTEMS

External power receptacles *shall* be provided for the electrical subsystem. These receptacles *shall* conform to MIL-C-7974 standards for ground power units. Provisions for grounding the helicopter and external power unit will be included in the design.

A minimum variety of electrical connectors should be used. Connector selection will follow instructions contained in MIL-STD-1353. Provisions must be made to prevent incorrect connections.

Provisions for lubricating and cooling generators and electric motors will be included in the design. Accessibility to lubricating points, as well as for the interchange, of generator and motor brushes will be provided.

The battery should be accessible easily with no special G. E required for its removal and installation. If nickel-cadmium batteries are used, the designer *shall* insure that water cannot be added to the battery without first removing the battery from the helicopter. (This procedure assures that water will be added only after it has been ascertained that the battery is fully charged.)

The voltage regulator should be accessible to allow for interconnecting the required test equipment and adjusting the regulator to the prescribed voltage level.

All fuses and circuit breakers *shall* be visible and readily accessible.

The internal and external lighting installation should provide easy access for installation of bulbs or tubes.

Wiring runs must be designed to allow easy access for the maintenance of wiring. Similar wiring connections to identical types of equipment on a given helicopter should be used to avoid errors in wiring during installation, replacement, and maintenance. If there is a possibility of connecting a GSE cable connector to the incorrect cable, the cable connector keying should preclude such interconnection. Common bonding posts or connectors should be provided for connections between the helicopter wiring and the GSE.

The basic design considerations for the electrical subsystem are described in Chapter 7.

15-8 AVIONIC SUBSYSTEMS

Design considerations and requirements for avionic subsystems are described in Chapter 8. Only essential VFR night communication/navigation avionics are considered in this paragraph.

15-8.1 COMMUNICATION SYSTEMS

Installations for the intercommunication set, communication radios, signal light, and flare launcher must permit ready removal and replacement of components.

When self-test provisions are included, test results must be visible to maintenance personnel. Adjustment controls must be readily accessible.

For equipment that does not include self-test features, the design should permit external test equipment to be connected readily. Plug-in connectors should be used throughout the installation in order to avoid breaking circuits for test and adjustment purposes.

Break points should be provided close to the transmitter and antenna equipment. The designer should consider the use of directional couplers to simplify maintenance. When the directional couplers are used, calibration indicators should be visible without removal of the units.

Chassis slides, runners, and tilting mechanisms may be used to facilitate accessibility of avionic equipment. When used, these mechanisms should be equipped with devices for positive locking in both extended and retracted position. Handles should be provided in order to simplify handling and removal. Withdrawal should be in the direction of available space. Cable lengths should be adequate to permit slide operation. Adequate space must be available for the installation and removal of mounting units and screws. Screws and bolts requiring access for maintenance should not be buried under cables.

Installation and test requirements for intercommunication systems and communication radio sets are contained in the applicable commodity specifications and data.

15-8.2 NAVIGATION SYSTEMS

The GSE interface considerations and requirements applicable to the communication system also apply to the navigation system. In addition, considerations must be given to the necessity for aligning the direction finder antennas and indicators during maintenance. Similarly, the gyrosyn compass system alignment requirements must be considered in the installation design.

Specific installation and test requirements for the navigation systems are contained in individual commodity specifications and data.

15-9 HYDRAULIC AND PNEUMATIC SUBSYSTEMS

The hydraulic and pneumatic subsystem design requirements are discussed in Chapter 9.

15-9.1 HYDRAULIC SUBSYSTEM

The hydraulic subsystem requires checkout connections for interfacing with ground test stands in accordance with MIL-A-5440 and MIL-C-25427. The system design should include a means for bleeding and replenishing hydraulic fluid.

The hydraulic system relates more closely to GSE than any other helicopter system in that the hydraulic fluid is transposed between the two in the accomplishment of the basic functions of filling, bleeding, and filtering.

15-9.2 PNEUMATIC SUBSYSTEM

Pneumatic subsystem ducting and coupling must be adequate for ground test purposes. The ground coupling should be included and appropriately marked. Where ram air turbines are used, an interface should be provided for driving the turbine by means of a pneumatic power source during ground testing. MS 28889 high-pressure valves should be provided to interface the high-pressure pneumatic systems with the GSE.

The designer should consider using engine pneumatic starters, if the helicopter is so equipped, as the source of compressed air for the actuation of airborne components. By tapping into the starter duct, the number of ground couplings can be minimized.

15-10 INSTRUMENTATION SUBSYSTEMS

The mounting of instruments and subsystem components *shall* permit rapid and easy inspection, adjustment, removal, installation, and diagnostic testing. Sufficient clearance should be provided for removal of covers, inspection plates, and removable modules. Items requiring more frequent adjustments or inspections should be more accessible than those requiring less frequent servicing.

The design considerations and requirements for the helicopter instrumentation subsystem are described in Chapter 10.

15-10.1 FLIGHT INSTRUMENTS

Pressure-actuated flight instruments *shall* be installed in accordance with MIL-P-26292, and the specified provisions for calibration *shall* be included.

An altimeter correction card must be provided when the combined static pressure system error and altimeter instrument error exceeds ± 150 ft.

When servo-operated instruments are used, necessary interfaces for electrical test equipment should be included.

15-10.2 NAVIGATION INSTRUMENTS

The pilot's compass installation must include provisions for swinging the compass in accordance with MIL-STD-765. Compass correction cards for each indicator in the helicopter will be provided. Radio navigation instruments are discussed in Chapter 8.

15-10.3 AERIAL VEHICLE SUBSYSTEM INSTRUMENTATION

The design requirements for the interface of the aerial vehicle subsystem instrumentation with the helicopter and the GSE are described in Chapter 10.

15-11 AIRFRAME STRUCTURE

The structural design engineer must include design provisions for jacking, mooring, lifting, loading, and transporting the helicopter by all modes of transport to include helicopter external airlift.

Lift points or sling positions for lifting of the complete crash/battle-damaged helicopter should be provided. Lift points must be designed to meet the g-loads specified in AR 70-9 and MIL-A-8421F.

Sectioning should be employed to separate the body and empennage, the body and wings, or the body itself into constituent sections. Each section should include clearly identified points for lifting and load bearing. Hoisting eyelets must be provided if hoisting is the prescribed method of lifting.

Floor obstructions of any kind should be avoided. If these are unavoidable, however, removable flooring — of sufficient strength to preclude deformation in normal use — must be provided to permit inspection, repair, and replacement of underlying structure or equipment not otherwise accessible.

All transparent areas should consist of easily-replaceable panels. Removal and replacement should be possible without the removal of other equipment.

Bearings and universal joints must be equipped with the appropriate provisions for lubrication, inspection, and replacement. Cowlings will be designed for ease of opening or removal.

Fittings as required for mating with towbars and towing vehicles must be provided. Lugs and rings suitable for coupling with hooks used on towlines, and standard methods of towing, also should be provided. Jig points for measuring and leveling in accordance with MIL-M-6756 shall be included in the airframe design.

Jacking facilities conforming to MIL-STD-809 *shall* be included. The designer *shall* assure that each landing gear can be jacked separately without interference between the jack and the landing gear system. The wheels must be removable without requiring the removal of struts or any part of the landing gear structure.

Towing provisions will be in accordance with MIL-STD-805.

The airframe structure must include mooring provisions consisting of lugs or rings for attachment of mooring ropes, cables, or lines. Where the fittings are recessed, they must have sufficient clearance space for easy extension from the recessed position. The word "moor" will appear on adjacent exterior surfaces. If detachable fittings are used, the design should provide for their storage in the baggage or tool compartment. If the landing gear is not used for restraint during runup of the engine(s), fittings to withstand the maximum load imposed during this ground operation *shall* be provided. When installed, these fittings will be marked with the statement: "Attach restraining harness here during ground runup of engine(s)".

Design requirements for airframe structural design are discussed in Chapter 11.

15-12 LANDING GEAR SUBSYSTEM

All units of the landing gear should be accessible for inspection, servicing, lubrication, and replacement. All air valves should permit easy servicing

When retractable gear is used, easily removable covers should be installed on all exposed equipment within the wells. Mud guards and scrapers should be cleanable easily without removal, and readily removable for tire servicing.

The attachment points for the landing gear must be designed to permit easy installation and removal. Retractable systems should include locks that require no adjustment, and are accessible for inspection without requiring disassembly of the actuating members of the retracting system.

Each gear in a retractable system should have a manually installed, lightweight, quick-release, ground-safety lock or pin. The design of this device should eliminate the possibility of incorrect installation. The gear should be designed so as not to retract or be damaged if manual unlocking is not accomplished before flight. A red warning streamer should be included with the lock pin to indicate when the lock is in place.

A tiedown lug should be provided on each main landing gear leg. These lugs should permit compression of the shock absorber to a position beyond the normal static deflection point when the helicopter is moored.

The landing gear design should minimize the requirement for special servicing tools, equipment, or fittings. Jack pads should be marked appropriately.

If the landing gear is not to be used as a restraining member during ground runup of engine(s), it *shall* be marked conspicuously, in accordance with MIL-M-25047, with the statement: "Do not use for restraining helicopter during ground runup of engine(s)".

Landing gear design considerations are discussed in Chapter 12.

15-13 CREW STATIONS

In general, all design provisions included for aircrew actuation of controls also are necessary for ground servicing. Additionally, sufficient clearances are required for operation and maintenance. Internal cargo compartments should have all protrusions marked conspicuously to avoid possible damage during loading and unloading. Cargo compartment doors should open easily, have a positive means of remaining open, and provide minimum interference with loading and unloading operations.

Where external cargo-carrying provisions are included, their design should facilitate loading and unloading operations. Designs for internal and external cargo-handling capabilities should permit standard material-handling equipment to be used.

The design requirements for helicopter crew stations, furnishings, and equipment are discussed in Chapter 13.

15-14 ARMAMENT, ARMOR, AND PROTECTIVE SYSTEMS

Interface design requirements for these mission-essential systems are contained in Chapter 14. GSE considerations are included in the specific commodity specifications and requirements, and are not included herein.

CHAPTER 16
STANDARD PARTS

16-0 LIST OF SYMBOLS

B = laminate length, in
C = basic dynamic capacity, lb
C_1 = load rating for life L_1 and speed N_1, lb
e = life exponent, 3 for ball bearings and 10/3 for roller bearings
ΔIC = change in interior clearance, dimensionless
K_s = nonreversing load factor, dimensionless
K_v = reversing load factor, dimensionless
L = total bearing life, hr
L_{10} = B-10 life, hr
L'_{10} = B-10 life, revolutions
L_i = calculated bearing life at load P_i and speed N_i, hr
L_w = laminate width, in.
L_1 = life, hr
M_1 = ratio of shaft internal diameter to bearing bore, dimensionless
M_2 = ratio of bearing bore to outer diameter of inner ring, dimensionless
M_3 = ratio of inner diameter of bearing outer ring to bearing outer ring, dimensionless
M_4 = ratio of bearing outer diameter to the housing outer diameter, dimensionless
N = rotational speed, rpm
N_i = speed imposed on bearing for fraction of time t_i, rpm
N_1 = rotational speed at which the capacity or equivalent radial load is determined, rpm
P = equivalent radial load, lb
P_{eq} = equivalent steady load, lb
P_i = load imposed on bearing for fraction of time t_i, lb
P_{N_1} = prorated load for speed N_1, lb
P_s = steady load, lb
P_v = vibratory load, lb
R = radial load, lb
SF = shape factor, dimensionless
T = thrust load, lb
t = laminate thickness, in.
t_i = fraction of time, dimensionless
V = rotation factor, dimensionless
X = radial factor, dimensionless
Y = thrust factor, dimensionless

16-1 INTRODUCTION

Standard parts, for the purposes of this handbook, are defined as those items normally used as purchased with no change or modifications; and manufactured to meet industry, association, or Governmental specifications as to size, materials, mechanical properties, performance, etc.

The standard parts discussed in this chapter are fasteners, bearings, electrical fittings, pipe and tube fittings, control pulleys, push-pull controls, flexible shafts, cables, and wires. The applications for and the limitations applicable to each of these are discussed. Further discussion of standard parts is found in AMCP 706-100.

16-2 FASTENERS

16-2.1 GENERAL

Fasteners are available in many types and sizes; however, fasteners for use in the design and construction of aerospace mechanical systems *shall* be selected in accordance with MIL-STD-1515. Generally, fasteners can be classified as either threaded or nonthreaded, and further as either reusable or nonreusable. Threaded fasteners include screws, bolts, and related hardware such as nuts and washers. Nonthreaded fasteners include rivets, pins, quick-release fasteners, retaining rings, clamps, and grommets.

16-2.2 THREADED FASTENERS

16-2.2.1 Screws

MIL-HDBK-5 contains allowable design loads for all structural screws. Installation of structural screws should be performed in accordance with Chapter 4, AFSC DH 1-2. Screws should be torqued to the maximum practicable preload, compatible with the applicable torquing method.

Screw threads for structural fasteners should conform to MIL-S-7742 or MIL-S-8879. MIL-S-8879 threads *shall* be used on all materials with a minimum ultimate tensile strength in excess of 150 ksi or with a minimum hardness greater than Rockwell C32 or equivalent. These threads also *shall* be used in applications where the operational temperatures will exceed 450°F; in applications that require the consideration of fatigue strength; for all bolts and screws of 0.164-in. diameter and larger; for high-temperature internal threads in excess of 900°F; and for threaded holes (other than nuts). MIL-S-7742 threads, both internal and external, may be used for fasteners smaller than 0.164 in. diameter, and for electrical connectors.

Screws used on helicopters should be restricted to two types of screw heads: pan or countersunk. In countersunk applications, a head angle of 100 deg should be used wherever possible; otherwise, a head angle of 82 deg should be used. Self-tapping screws should not be used in the primary structure. They may be employed, primarily in nonstructural applications, when the use of bolts or rivets is not practical. The installation and usage of tapping screws *shall* comply with the requirements of MS 33749.

Screws normally are reusable, and are replaced only when either the recess in the head or the threads have been damaged.

16-2.2.2 Bolts

The installation or removal of bolts normally is a two-handed operation. Therefore, it is slower and more difficult than is an assembly using screws. Another disadvantage is the many loose parts — such as nuts, washers, and cotter pins — required in conjunction with this type of installation. Special care must be taken that parts are not dropped, later to find their way into the engine inlet or to jam a moving assembly.

In general, bolts should be no longer than necessary. When tightened, the bolt should extend at least two threads beyond the nut. Hexagonal head bolts are preferred, and left-hand threads should be avoided when possible. Self-locking bolts may be used in tapped holes when one surface is inaccessible, or when there is a requirement that one surface be smooth, and when temperatures do not exceed 250°F. Bolts are more readily replaceable than studs.

Allowable design bolt loads should comply with MIL-HDBK-5. Installation and preload torque requirements are described in Chapter 4, AFSC DH 1-2. Specifications covering approved standard bolts are included in MIL-STD-1515.

Bolts *shall* be installed in such a way as to minimize the possibility of loss of the bolt due to loss of the nut. In control systems, and other applications (primarily in dynamic systems) where loss of a bolt could cause a catastrophic failure, self-retaining bolts *shall* be used, or two independent means of locking or safetying *shall* be required. Bolts *shall* be installed with heads forward or uppermost, taking into consideration ease of maintenance and replaceability.

Bolts normally are reusable, being replaced only when the head or the thread has been damaged during removal or replacement.

16-2.2.3 Nuts

Nuts can be subdivided into such general categories as locking or nonlocking, and fixed or nonfixed. Nuts *shall* be selected in accordance with MIL-STD-1515.

Self-locking nuts can be used independently or in conjunction with such devices as cotter pins, safety wiring, lock washers, locking compound, or self-locking bolts, as a means of keeping the nut tight on the bolt. Self-locking nuts should meet the requirements of MIL-N-25027 and *shall* be subject to the design and usage limitations of MS 33588.

Fixed nuts are affixed rigidly to the helicopter chassis by riveting, welding, clinching, or staking; and are used specifically in cases where the thinness of the metal prohibits tapping, or where limited space results in inaccessibility. Fixed nuts also have an advantage over nonfixed in that assembly and repair become one-handed operations.

Nonfixed nuts preferably should be of the hexagonal-head type. Wing or knurled nuts, which require no tools, *shall* be used only for low-tension, nonstructural applications; wing nuts are the easier to install. Nonfixed nuts, unless safetied, must not be used where fallen nuts can damage equipment. Self-wrenching nuts may be used in areas where insufficient space is provided for maintenance and use of tools. Self-sealing nuts are required for fastening equipment to fluid tanks in order to prevent leakage.

Nuts are reusable until the threads, or, when applicable, the locking provisions, are damaged during removal and reinstallation.

16-2.2.4 Washers

In general, washers are used under nuts to prevent injury to surfaces upon tightening the fastener, and to reduce the stress on the joint by increasing the bearing area. Spacer washers may be required in order to prevent loading of bolt threads in bearing. The intended use and the temperature limitation should be considered when choosing a washer. Washers *shall* be selected in accordance with MIL-STD-1515. Dissimilar metals should not be used together (e.g., steel washer with aluminum bolt) when normal methods of protection against corrosion, such as primer, may be damaged during the assembly of the joint.

Lock washers can be used to prevent rotation of the bolt and nut in nonstructural applications, but are not preferred. Preload-indicating washers may be used to gage bolt preloads, but must be replaced each time the bolt or nut is reinstalled.

16-2.3 NONTHREADED FASTENERS

16-2.3.1 Rivets

Allowable design loads for rivets are given in MIL-HDBK-5. Solid rivets of specific types and materials are covered by MS 20426, MS 20427, MS 20470, MS 20613, and MS 20615. Additional rivet standards, applicable to solid, tubular, and blind rivets (both structural and nonstructural) are listed in Chapter 4, AFSC DH 1-2.

Rivets are a permanent type of fastener: they must be destroyed in the process of their removal, and often they cannot be replaced by another of the same size. Therefore, rivets *shall not* be used in any application where disassembly is expected to be necessary during the normal life of the helicopter.

Rivets *shall* be used in applications where they are subjected primarily to shear. To prevent ripping, the diameters of the heads of countersunk rivets *shall* be larger than the thickness of the thinnest of the pieces they fasten. Requirements for countersinking are contained in Chapter 4, AFSC DH 1-2.

In a case where it is impossible or impracticable to reach the back of the joint to buck solid rivets, blind rivets may be used. Blind rivets may be structural or nonstructural. Mechanical expansion generally is accomplished by withdrawal of an appropriately shaped spindle through the hollow center of the rivet. Standards for structural and nonstructural blind rivets are given in MIL-STD-1515.

In installation of rivets, the distance from the center of the rivet hole to the edge of the sheet depends largely upon the stress analysis of the joint. Rivets *shall* be located so that the edge distance is not less than 1.5 times the rivet shank diameter, or greater than 2.5 times the rivet shank diameter on a lap joint. The design allowable load data for countersunk joints in MIL-HDBK-5 is based upon an edge distance of two diameters. If lesser edge distances are used, the allowable loads *shall* be substantiated by test data. The head angle of countersunk rivets *shall* be 100 ±1 deg.

16-2.3.2 Pins

In tie rods and on secondary controls that are not subjected to continuous operation, clevis pins may be used. In these usages, the reversal of stresses and the chances of loosening are slight. Clevis pins *shall not* be used where tight joints are required.

Taper pins should be used in all permanent connections where the absence of play is essential. They can be locked safely with castellated or self-locking nuts.

Spring pins should conform to MS 16562, and they may be used within the design limitations given in MS 33547.

The use of swaged, collar-headed straight pins should conform to MS 25420, and that of flathead pins to MS 20392. The standards for positive-locking, quick-release pins are given in MS 17984 through 17990.

16-2.3.3 Quick-release Fasteners

Quick-release fasteners are classified as rotary-operated, lever-activated, slide action, or push-button.

These types of fasteners are relatively easy and fast to use, do not always require special tooling, and are recommended for securing plug-in components, along with small components and covers. Quick-release fasteners also are known as cowl or panel fasteners.

Specific requirements for low-strength, quick-release panel fasteners are given in MIL-F-5591. The disadvantages of these fasteners are that their holding power is limited and many types cannot be used where a smooth surface is required. Their advantage is that they can be attached and released easily, with a maximum of one complete turn, and usually without the use of hand tools.

High-strength, quick-opening rotary fasteners, in accordance with MS 17731 (countersunk) and MS 17732 (protruding head), may be used as structural panel fasteners.

16-2.3.4 Turnbuckles and Terminals

In helicopter applications, turnbuckles are used primarily in control cable installations, and their use should be in accordance with the system specification for the specific helicopter model. In most cases, the desired end fitting (except a threaded end fitting) is swaged onto the cable. Swaging is discussed in par. 17-4.

Turnbuckle terminals may be of either fork, eye, or swaged configuration. Following installation and adjustment of cable tension, turnbuckles must be safetied to prevent loss of tension. MIL-T-5685 describes one type of turnbuckle for aircraft application, while a positive-safetying type of turnbuckle is described by MIL-T-8878. General-purpose turnbuckle bodies (MS 27954) *shall* be safety-wired in accordance with MS 33591. Clip-lock turnbuckle bodies (MS 21251) *shall* be locked in accordance with MS 33736.

16-2.3.5 Retaining Rings

Retaining rings, or snap rings — selected in accordance with MIL-STD-1515 — may be used to retain bearings or seals, and, in limited applications, to retain pins or bolts. The application of retaining rings to bearings and seals is discussed further in par. 16-3.7.2.

Both external and internal retaining rings are available as tapered or reduced cross section types. The reduced cross section type has a lower load capability and, therefore, is used only in locations where the load is not critical and where the retaining ring is not required to maintain a tight installation. When a tapered cross section retaining ring is used to maintain a tight joint, both the location and the width of the snap ring groove are critical.

The external ring is installed in an appropriately located groove in the shaft or pin. Internal retaining rings require that a groove be cut in the housing bore.

The primary types and configurations of external and internal retaining rings are described in MIL-STD-1515. External retaining rings used as primary fasteners *shall* be safety-wired in accordance with MS 33540.

16-2.3.6 Clamps and Grommets

Clamps should be used for holding wires, tubing, or hoses that are to be removed frequently. Hinge clamps are preferred for mounting tubing or wiring on the face of a panel, thus facilitating maintenance by supporting the weight of the tubing or wiring. For large, plug-in assemblies, positive-locking clamps should be used.

Grommets *shall* be used wherever necessary to protect cables, tubes, hose, and wiring from chafing against the edges of holes in bulkheads, frames, panels, or other structure through which they must run. Grommets are available in a variety of materials, making it possible to select the correct material for each application. Grommets are covered by MIL-G-3036, -16491, -17594, and -22529.

16-2.3.7 Self-retaining Fasteners

Self-retaining fasteners are used as a safety lock in areas where a serious hazard would exist should a bolt be lost, or should a joint be broken following loss or failure of the threaded connection. Their use in helicopters is applicable particularly to control systems and dynamic systems.

Self-retaining fasteners are not to be confused with self-locking bolts. When the self-retaining fastener is inserted into the holes of two surfaces to be joined, mechanical safetying prevents it from being removed readily. Thus, a semipermanent joint, which can be opened through a specific action on the part of the user, is obtained even before the nut is attached to the bolt. Vibration, for example, is not sufficient force to allow this type of fastener to fall free.

Self-retaining fasteners of the impedance type are covered in MIL-B-83050, and self-retaining, positive-locking bolts are described in MIL-B-23964.

16-3 BEARINGS

16-3.1 GENERAL

Design and selection of bearings for helicopter applications demand the consideration of several factors: life requirements, loads and speeds imposed upon the bearing, available space envelope, and environmental conditions.

Determination of the allowable space envelope logically is the first step in the design of a bearing system. Maximum and minimum values should be obtained for the outer diameter and width, together with the preferred values within these ranges.

The next, and most significant, design consideration is the required operating life of the bearing. The life requirements are dependent upon the projected component service life, the time between overhauls (TBO), and the level of reliability that must be maintained. Because bearing failures, and, hence, bearing lives, follow a definite statistical distribution, bearing lives are calculated for a given survival rate. The bearing design life usually is given in terms of B-10 life, which represents the operating time that will be exceeded by 90% of the bearings under given conditions of load and speed. In helicopter applications, critical bearings are designed for a 3000-hr minimum life, with the actual value depending upon the helicopter system specification requirements. In determining the required B-10 life, the designer must realize that the equations used for life calculations are based largely on data from fatigue tests of bearings run under controlled and nearly ideal conditions. These equations do not take into account such adverse operating conditions as severe thermal or contaminating environments, lubricant deterioration, or shaft misalignment. Where the potential for such conditions exists, the use of a higher calculated bearing life is appropriate.

The loads and speed imposed upon a bearing are major considerations in determining the type and size of bearing to be used. Purely radial loads may be carried by cylindrical roller bearings. Combined radial and thrust loads require the use of ball bearings or tapered or spherical roller bearings, each of which may be used either alone or in combination with cylindrical roller bearings. For applications in which only thrust loads are present, special thrust bearings are available in both ball and roller types. For high-speed applications, ball bearings or cylindrical roller bearings should be given first consideration since their speed capabilities are significantly greater than those of tapered or spherical roller bearings. Where the motion between two members is oscillatory rather than rotary, consideration should be given to sliding spherical or journal bearings or to laminated elastomeric bearings.

Environmental conditions that must be considered include operating temperature, possibility of contamination, and corrosive atmosphere. Knowledge of the operating temperatures to be encountered will aid in the definition of the bearing materials to be used. Bearing steels usually can be stabilized thermally for a particular range of operating temperatures. Environments that can cause contamination or corrosion may require that special sealing devices be incorporated into the bearing. Operation in highly corrosive atmospheres may require the use of corrosion-resistant bearing steels.

Once the design requirements are defined and the possible bearing configurations selected, the designer should perform the necessary bearing life calculations. For rotating, rolling-element bearings, the basic B-10 life equation is:

$$L_{10} = \frac{16{,}667}{N}\left(\frac{C}{P}\right)^{e} \text{ , hr} \qquad (16\text{-}1)$$

or

$$L'_{10} = 10^{6}\left(\frac{C}{P}\right)^{e} \text{ , rev} \qquad (16\text{-}2)$$

where

L_{10} = B-10 life, hr
L'_{10} = B-10 life, revolutions
C = basic dynamic capacity, lb
P = equivalent radial load, lb
N = rotational speed, rpm
e = life exponent, 3 for ball bearings and 10/3 for roller bearings

The basic dynamic capacity C is defined as the radial load that a bearing will endure for a B-10 life of one million revolutions of the inner ring, with the outer ring stationary. This capacity can be defined equivalently as the radial load that will yield a B-10 life of 500 hr at a shaft speed of 33-1/3 rpm. The basic dynamic capacities of their standard bearings are available from the bearing manufacturers. For new designs, the Anti-Friction Bearing Manufacturers Association (AFBMA) has established procedures for calculating the capacity based upon internal bearing geometry (Refs. 1 and 2). When comparing catalog capacities quoted by different manufacturers, the designer must make certain that each capacity is defined similarly in terms of life and speed. Load ratings based upon lives and speeds other than 500 hr and 33-1/3 rpm can be converted to basic dynamic capacity C by:

$$C = C_1\left(\frac{L_1 N_1}{16{,}667}\right)^{1/e} \text{ , lb} \qquad (16\text{-}3)$$

where

C_1 = load rating for life L_1 and speed N_1, lb
L_1 = life, hr
N_1 = speed, rpm

The equivalent radial load P is defined as that radial load that yields a bearing life equal to the life resulting from the combination of radial and thrust loads actually imposed upon the bearing. It is calculated by the relation:

$$P = XVR + YT \text{ , lb} \qquad (16\text{-}4)$$

where

R = radial load, lb
T = thrust load, lb
X, Y = radial and thrust factors, respectively, dimensionless
V = rotation factor, dimensionless

The radial and thrust factors X and Y are based upon internal bearing geometry and are given in Refs. 1 and 2. The rotation factor V is equal to 1.0 if the inner ring of the bearing is rotating with respect to the radial load, and 1.2 if the outer ring is rotating with respect to the radial load.

In most helicopter applications, the loads and speeds vary over a predictable spectrum. In such instances, the equivalent radial load P can be expressed as a prorated load for a given speed P_{N_1} and is determined by:

$$P_{N_1} = \left[\sum_{i=1}^{n} t_i (P_i)^{e} \frac{N_i}{N_1}\right]^{1/e} \text{ , lb} \qquad (16\text{-}5)$$

where

P_{N_1} = prorated load for speed N_1, lb
N_1 = rotational speed at which the equivalent radial load is determined, rpm
P_i = load imposed on bearing for fraction of time t_i, lb
N_i = speed imposed on bearing for fraction of time t_i, rpm
t_i = fraction of time, dimensionless (the sum of fractions must be exactly 1)

An alternate method of calculation is to determine the bearing life for each condition of load and speed by using Eq. 16-1, and then to calculate a total bearing life by:

$$L = \frac{1}{\sum_{i=1}^{n} \frac{t_i}{L_i}}\ , \text{hr} \qquad (16\text{-}6)$$

where

L = total bearing life, hr

L_i = calculated bearing life at load P_i and speed N_i, hr

Bearing life at high speed may be reduced significantly by the effect of centrifugal loads imposed by the rolling elements against the outer race. The calculation of the internal bearing stresses and bearing life is a lengthy procedure that is covered adequately in Ref. 3. However, for the vast majority of applications, the centrifugal effects can be neglected, and, therefore, Eqs. 16-1 through 16-6 are valid. The effect of centrifugal force, however, must be taken into account in the case of higher rotational speeds, such as those found in some engine reduction gearbox bearings.

Usually, the life of a particular bearing can be increased by a significant factor simply by using bearing materials that offer greater uniformity and better fatigue life than does standard steel. Table 16-1 shows approximate life adjustment factors for several superior bearing steels. Consult Ref. 13 for a complete discussion of bearing life. The calculated B-10 life should be multiplied by these factors in order to determine the actual B-10 life to be expected from these materials. Although SAE 52100 air-melt steel is used as the basis for the AFBMA ratings, vacuum-degassed material now is considered to be the standard steel throughout the bearing industry.

TABLE 16-1. LIFE FACTORS FOR ANTIFRICTION BEARING MATERIALS

MATERIAL	LIFE FACTOR	
	BALL BEARINGS	ROLLER BEARINGS
SAE 52100 AIR MELT STEEL	1	1
SAE 52100 VACUUM-DEGASSED STEEL	3	2
SAE 52100 CVM	4	3
M-50 CVM	≥5	≥5

In many helicopter bearing applications, the load imposed upon the bearing is vibratory or is a combination of steady and vibratory loads. Such conditions of loading may be converted to an equivalent steady load for the purpose of bearing life calculations. For conditions in which the steady load is greater than the amplitude of the vibratory load, the method illustrated in Fig. 16-1 is used. The ratio of vibratory to steady load is calculated, and the corresponding nonreversing load factor K_s from Fig. 16-1 is multiplied by the actual steady load to yield the equivalent steady load. A similar procedure is used for the case of a reversing load condition (amplitude of the vibratory load greater than the steady load). The reversing load factor K_r from Fig. 16-2 then is multiplied by the imposed vibratory load to yield the equivalent steady load used in the life calculation.

Antifriction bearings are manufactured in various tolerance ranges or classes of precision. These classes have been standardized by the Annular Bearing

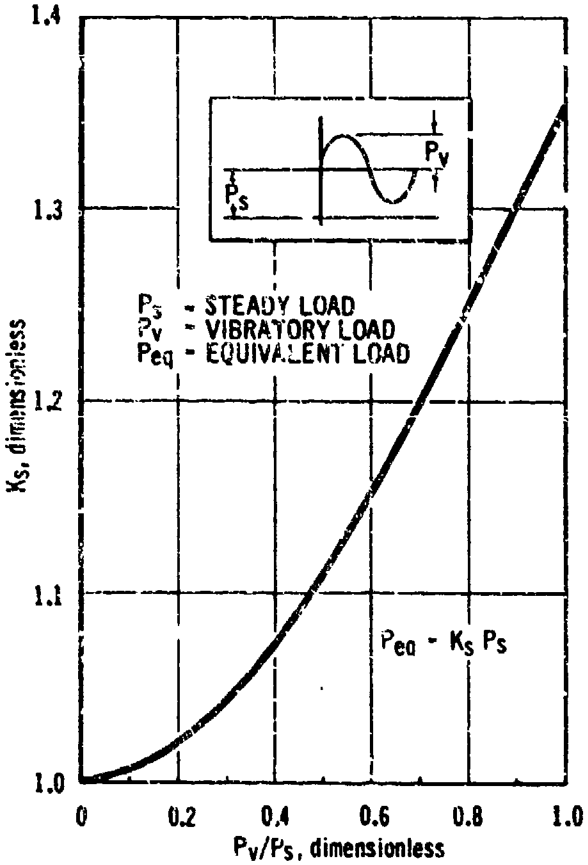


Figure 16-1. Equivalent Steady Load for Combination of Steady and Vibratory Loads (Nonreversing)

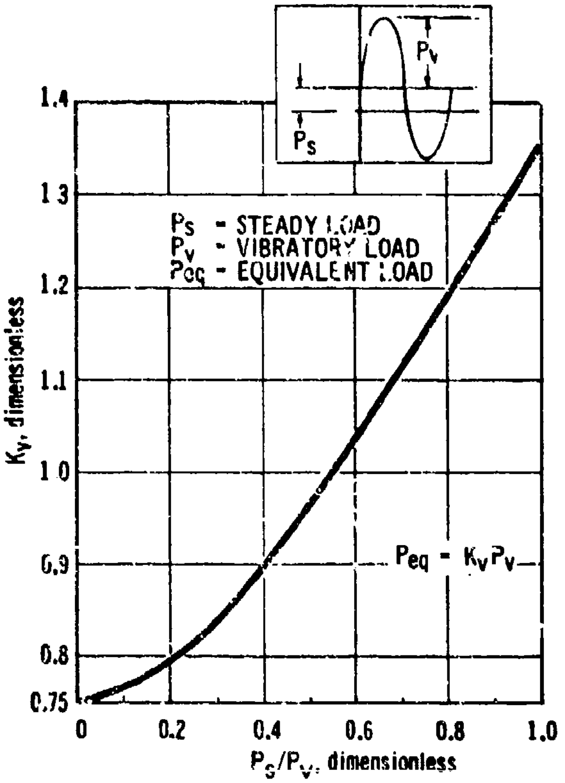


Figure 16-2. Equivalent Steady Load for Combination of Steady and Vibratory Loads (Reversing)

Engineering Committee (ABEC) and the Roller Bearing Engineering Committee (RBEC), and are designated ABEC-1, -3, -5, -7, and -9, for ball bearings; RBEC-1 and -5 for cylindrical and spherical roller bearings; and Class 4, 2, 3, 0, and 00 for tapered roller bearings. The numerical symbols are listed in order of increasing precision for each bearing type. They define the tolerances on bearing bore, outside diameter, width, radial and side runout, and also are indicative of overall bearing quality. The use of the higher precision bearing grades represents a substantial increase in cost. For most helicopter applications, the middle grades — designated ABEC-3 or -5, or Class 2 — should receive first consideration. Only in those cases requiring high-speed operation, extremely precise shaft location, low runout, or high reliability should ABEC-7 or -9, and Class 3, 0, or 00 grades be chosen. The increased costs resulting from the use of the higher-precision bearings are presented in Table 16-2.

Interference fits are used between the bearing and the shaft or housing in order to prevent creeping of the bearing rings and subsequent fretting and wear of the mating surfaces. Unless the interference fit between the mating members is greater than zero under all operating conditions, the bearing ring that rotates with respect to the load will creep. In addition, lighter press fits often are specified for the nonrotating ring in order to preclude creeping due to transmitted vibration.

TABLE 16-2. COST VS TOLERANCE CLASS FOR ANTIFRICTION BEARINGS

BEARING TYPE	GRADE	COST FACTOR
BALL	1 3 5 7	1 1.5 3 3.2
ROLLER (CYLINDRICAL AND SPHERICAL)	1 5	1 5–6
ROLLER (TAPERED)	4 2 3 0 00	1 1 2.5 5 18

The interference fit required to prevent creeping is dependent upon radial load, support stiffness, surface finish, and operating temperature. Ref. 4 gives a method of calculating the required shaft fit that considers these factors. However, this analysis assumes that the shaft is solid, while most helicopter power train applications use hollow shafts. A good approximation of the required fit for hollow steel shafts as compared to solid shafts is given by:

$$\frac{\text{Hollow Shaft Fit}}{\text{Solid Shaft Fit}} = \frac{K_1 + K_2}{1 + K_2}, \text{ d'less} \quad (16\text{-}7)$$

where

$$K_1 = \frac{1 + M_1^2}{1 - M_1^2}, \quad \text{d'less} \quad (16\text{-}8)$$

$$K_2 = \frac{1 + M_2^2}{1 - M_2^2}, \quad \text{d'less} \quad (16\text{-}9)$$

M_1 = ratio of shaft internal diameter to bearing bore, dimensionless

M_2 = ratio of bearing bore to outer diameter of inner ring, dimensionless

Standard bearings are manufactured to provide a small internal running clearance when mounted with the fits recommended in the manufacturer's catalog. When heavier fits (less clearance, or greater interference) are needed to prevent creeping, a bearing with increased internal clearance must be used in order to avoid preloading the bearing radially. The change in internal clearance due to interference fits for both ball and roller bearings can be calculated for steel shafts or housings by the following equations:

$$\frac{\Delta IC}{\text{Shaft Fit}} = \frac{K_2 - 1}{M_2(K_1 + K_2)}, \text{ d'less} \quad (16\text{-}10)$$

$$\frac{\Delta IC}{\text{Housing Fit}} = \frac{M_3(K_3 - 1)}{K_3 + K_4}, \text{ d'less} \quad (16\text{-}11)$$

where K_1, K_2, M_1, and M_2 are as previously defined, and ΔIC = change in interior clearance, dimensionless, and:

$$K_3 = \frac{1 + M_3^2}{1 - M_3^2}, \text{ d'less} \quad (16\text{-}12)$$

$$K_4 = \frac{1 + M_4^2}{1 - M_4^2}, \text{ d'less} \quad (16\text{-}13)$$

where

M_3 = ratio of inner diameter of bearing outer ring to bearing outer diameter, dimensionless

M_4 = ratio of bearing outer diameter to the housing outer diameter, dimensionless

The changes in internal clearance due to shaft and housing fits are added. The lowest range of initial clearances that will give a positive running clearance then should be selected.

Additional information on the selection, design, and installation of both rolling element and sliding bearings can be found in Ref. 5.

Bearings subjected to high loads or speeds must be provided with sufficient quantities of lubricant to maintain a satisfactory heat balance and avoid thermal damage to the contact surfaces. Grease lubrication or splash-type oil lubrication is satisfactory for bearings operating at moderate speeds in areas where the heat rejection characteristics of the bearing housing are favorable. For higher speeds — or in potentially high-temperature areas — jet, splash, or mist oil lubrication usually is necessary, often with the incorporation of an oil cooler.

For heavy loads, the use of high-viscosity lubricants, or of lubriants formulated with extreme pressure additives, often is necessary. High-temperature applications, such as in engines or in high-speed engine reduction gearboxes, may require the use of synthetic-base lubricants of the ester type. These synthetic lubricants also provide the advantage of good low-temperature performance. The characteristics of many types of applicable lubricants, as well as descriptions of lubrication systems, are given in Ref. 6.

16-3.2 BALL BEARINGS

The type of bearing most widely used in aircraft applications is the ball bearing. Ball bearings can be classified in three categories: radial, angular contact, and thrust.

16-3.2.1 Radial Ball Bearings

The most common configuration is the single-row, deep-groove ball bearing. It is capable of supporting both radial loads and light thrust loads in either direction while operating at high speeds. While most radial ball bearings are produced to ABEC-1 tolerances, higher precision bearings, such as ABEC-3, -5, -7, or -9, are available for very high-speed applications in those areas where shaft location and runout are critical. The bearings in the higher grades usually contain better-quality retainers, typically of machined bronze. In addition, these retainers are guided by one of the bearing races, rather than being positioned by the balls, thus further improving the high-speed performance of the bearings. Prelubricated radial ball bearings equipped with shields or seals are available for grease-lubricated applications.

Because the radial ball bearing is assembled by radially displacing the inner and outer rings and then packing the balls into the resulting annular space, the number of balls in a given bearing is limited. The number of balls in the bearing, and hence the load capacity, can be increased by the use of a filling notch, a counterbored inner or outer ring, a circumferentially split inner or outer ring, or a fractured outer ring. These features yield bearings having calculated capacities that are substantially greater than those of normal, deep-groove bearings of the same size. However, each of these techniques has its limitations. A filling notch limits the ability of a bearing to support thrust loads, because the balls contact the notch. Counterboring a bearing ring results in a bearing that can support thrust loads only in one direction; therefore, such bearings generally are used in pairs (see par. 16-3.2.2). Bearings with a circumferentially split ring should be used only in applications having a thrust load sufficient to prevent the balls from riding on the split. Lastly, bearings with a fractured outer ring should not be used to support thrust load since this type of loading tends to spread the outer ring apart at the fracture.

Double-row bearings are available for applications requiring higher radial-load capacity within a limited space envelope. Two rows of balls are held between single-piece, double-grooved inner and outer rings. This configuration provides approximately a 50% increase in radial load capacity over the single-row bearing, and is available in both deep-groove and maximum-capacity types.

For applications where misalignment is a factor, self-aligning bearings are available in two types. In one type, the outer race is ground as a spherical surface, yielding a standard space envelope, but with reduced capacity. The other type employs a separate spherical outer ring. While the capacity of this design is not reduced significantly, it has a larger outside diameter for a given basic size.

16-3.2.2 Angular Contact Bearings

Angular contact bearings provide increased radial and thrust capacity, but limit the supportable thrust load to only one direction. In this type of bearing, one shoulder of the outer ring is removed almost completely. The remaining small shoulder serves to hold the bearing together, but cannot support thrust loads. The bearing is assembled by heating the outer ring and then installing the inner ring, balls, and retainer as a preassembled unit. This construction permits the use of a maximum ball complement and a one-piece, machined retainer, thus yielding both high capacity and good high-speed capability.

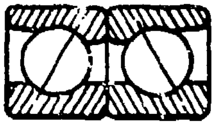
Angular contact bearings can be made with a wide range of contact angles. As the contact angle increases, thrust load capacity increases and radial load capacity decreases. At high speeds, a bearing having a high contact angle will experience a large amount of ball spinning, with resultant heat generation. Therefore, for high-speed angular contact ball bearings, the contact angle should be kept as low as possible.

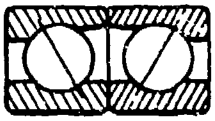
For applications in which thrust loads must be carried in both directions, angular contact bearings often are mounted in duplexed pairs. When a pair of these bearings is mounted with like faces together, they become preloaded. Two bearings are preloaded if all of their internal looseness is removed when their inner and outer rings are clamped together. Preload usually is built into a ball bearing by grinding the outer-ring thrust face flush with the inner ring while the bearing is loaded axially with a given gage load. The thrust face of the outer ring is marked with an identifying symbol.

When two such angular contact bearings are mounted with their outer-ring thrust faces together, as shown in Fig. 16-3(A), the bearing set is known as a duplexed pair mounted back-to-back with a preload equal to the gage load. Higher preloads are used to ensure that the nonthrust-carrying bearing does not become unloaded completely, which can result in ball skidding. Back-to-back duplex mountings are capable of carrying combined radial and thrust loads, reversing thrust loads, and moment loading. They provide a rigid mount for the shaft because the lines of contact intersect the bearing axis outside the bearing envelope. They also provide precise location of the shaft since all internal looseness is removed.

If a pair of angular contact bearings is mounted with the nonthrust faces of their outer rings together, as shown in Fig. 16-3(B), the bearing set is known as a duplexed pair mounted face-to-face. This type of mounting also can take combined radial and thrust loads, along with reversing thrust loads. It does not provide the rigidity and moment-carrying ability of the back-to-back mounting, but will tolerate small amounts of misalignment.

If a pair of angular contact bearings is mounted with the outer-ring thrust face of one bearing against the nonthrust face of the outer ring of the other bearing, the bearing set is known as a tandem pair, as illustrated in Fig. 16-3(C). This type of mounting is


(A) BACK-TO-BACK


(B) FACE-TO-FACE

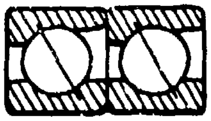

(C) TANDEM

Figure 16-3. Mounting of Duplexed Ball Bearings

used to carry heavy thrust loads in one direction, with the bearings sharing the load equally. However, tandem mounting does not remove all of the internal looseness from the bearings, and, therefore, permits some shaft float.

16-3.2.3 Thrust Ball Bearings

Thrust ball bearings are available for applications in which pure thrust loads are to be supported at moderate speeds. These bearings afford a very high thrust capacity, but provide no radial support for the shaft. Thrust ball bearings are quite limited in speed capability due to spinning in the ball-to-race contacts, and they also are sensitive to misalignment. Because even small amounts of misalignment can result in high internal contact stresses, a high degree of perpendicularity must be maintained between the bearing races and the axis of the shaft.

Thrust ball bearings are made for applications requiring thrust-supporting capabilities in one or two directions. The various configurations available are shown in such catalogs of manufacturers as Ref. 7.

16-3.3 ROLLER BEARINGS

Roller bearings are used most frequently in applications requiring high load capacity for a given space envelope. The various types of roller bearings include cylindrical, needle, spherical, and tapered roller configurations. Needle roller bearings have a rolling element length-to-diameter ratio significantly greater than that encountered in typical cylindrical roller bearings. Spherical roller bearings use concave or convex rolling elements in order to permit operation with misalignment between the shaft and the housing. Both spherical and tapered roller bearings can support combined radial and thrust loads Their limiting speeds are lower than those of cylindrical roller bearings because of sliding between the rollers and the guiding ribs.

16-3.3.1 Cylindrical Roller Bearings

Cylindrical roller bearings are used typically in applications in which a purely radial load is to be supported. In most cases, the rollers are crowned in order to prevent end loading and to compensate for small amounts of misalignment.

Cylindrical roller bearings are manufactured in a standard grade designated RBEC-1 for most commercial applications. Precision-grade RBEC-5 bearings are used for critical helicopter applications where high-speed capabilities, and very precise location and alignment of the shaft are essential.

Cylindrical roller bearings are constructed with the rollers running between two flanges on either the inner or the outer ring. In some cases, one or two additional shoulders are used in order to limit axial motion and to allow the bearing to support light thrust loads. Such configurations are known as locating types of bearings and are designated as one- or two-directional, depending upon whether one or two shoulders are used. One-directional locating bearings have separable rings, and incorporate a single shoulder to prevent axial movement of the shaft in one direction. Two-directional locating bearings use two race shoulders to provide shaft location and light thrust load capability in both axial directions. Locating roller bearings have design capabilities similar to those for non-locating bearings, except for a slightly lower limiting speed due to sliding of the roller ends on the face of the shoulder.

Cylindrical roller bearings afford the highest speed capabilities of all roller bearing types. In utilizing these capabilities, the designer must be aware of the special problems associated with high-speed operation. Operation at very high speeds, usually while carrying rather low radial loads, can result in roller skidding, contributing, in turn, to early bearing failure. In order to prevent skidding, several techniques can be employed. The simplest and most reliable method is to insure that a radial load sufficient to maintain rolling contact always is present. Because this is not always possible, particularly when the only load imposed upon the shaft is torque, specialized design features — such as reduced internal clearance, out-of-round outer races, or the incorporation of two or more preloaded hollow rollers — have been introduced to help maintain rolling contact and a constant retainer speed. However, bearings incorporating such features usually require extensive development and testing.

With high-speed roller bearings, in which the retainer is guided by a shoulder on the outer ring, the radial growth of the retainer due to increased temperature must be considered. In order to avoid interference between the retainer and the outer ring, the initial retainer clearance must be large enough to insure a running clearance at the maximum operating temperature of the bearing.

16-3.3.2 Needle Bearings

Needle bearings comprise a special class of cylindrical roller bearings in which the rolling elements are long in relation to their diameter. Such a design has the advantage of a very high load capacity for a given radial section. Needle bearings are manufactured in both full-complement and retainer types. The full-complement types provide high radial capacity since the maximum number of rolling elements is used. Retainer types sacrifice some load capacity, but have the highest speed capabilities of any of the needle bearing types due to the roller guidance and spacing provided by the retainer. Because of the high length-to-diameter ratio of the rollers, these bearings are susceptible to roller skewing and roller end loading. Therefore, they are limited to lower speeds than are standard cylindrical roller bearings. Limiting speeds, as well as load capacities for the various types of needle bearings, are given in such manufacturer's catalogs as Ref. 8.

Needle roller bearings are produced in several configurations for different applications. The most common type is the drawn-cup bearing, which consists of a drawn, case-hardened cup surrounding hardened and ground rollers. The cup acts as the bearing outer race and incorporates a lip at each end to provide roller retention. The shaft, when properly hardened and finished, may serve as the inner race, or a separate inner race may be provided. Drawn-cup bearings are manufactured in both full-complement and retainer types.

Caged roller assemblies, consisting of a complement of hardened, crowned needle rollers and a steel retainer, also are available. The inner and outer races are designed separately, using clearances and finishes recommended by the bearing manufacturer. The crowned rollers provide fairly even stress distribution along the roller length, and increase the misalignment capability of the bearing.

Needle thrust bearings use cylindrical rollers arranged with their axes positioned radially with respect to the axis of the bearing and held in a flat, machined retainer. The rollers run on flat races that are hardened and ground. Care must be taken to locate the race surfaces perpendicular to the axis of rotation in order to prevent end loading of the rollers. Because the rolling elements of needle thrust bearings are cylindrical rather than tapered, some sliding always occurs between the rollers and the races. Because of this condition, a generous lubrication film should be provided in order to prevent excessive heat generation.

16-3.3.3 Spherical Roller Bearings

Spherical roller bearings are used to accommodate shaft misalignment. They are made with either concave or convex spherical rollers in both single- and double-row configurations. One raceway is machined and ground to conform to the roller. In the case of convex rollers, guide flanges are provided on this race. The other raceway, typically the outer, is ground with a continuous spherical surface. Single-row spherical bearings can carry high radial loads while supporting only very light thrust loads. Double-row spherical bearings also have high radial load capacities and can support thrust loads much higher than those permitted by single-row designs. In addition, double-row bearings can be made with asymmetric rollers, which reduce the roller skewing tendency, and thus permit the bearing to operate at higher speeds. Speed limitations and load capacities for typical spherical roller bearings are presented in Ref. 9.

In addition to the usual single- and double-row designs, spherical roller bearings are available in a special thrust bearing configuration. This type of bearing has a high thrust-load capacity and can be manufactured with either symmetric or asymmetric rolling elements.

Although spherical roller bearings do not possess the high-speed capabilities of cylindrical roller bearings or ball bearings, they afford a combination of high load and misalignment capacity that cannot be equaled by any other bearing type. These unique attributes are useful in many helicopter applications.

16-3.3.4 Tapered Roller Bearings

In helicopter applications in which both radial and thrust loads are high, as on a bevel gear shaft, tapered roller bearings should receive first consideration. Of all antifriction bearings, tapered roller bearings offer the best combination of radial and thrust capacities. The raceways and rollers of a tapered roller bearing are conical in shape. If the lines of contact between the rollers and each race are extended, they meet at a common point on the axis of the bearing. This geometry results in virtually pure rolling at all points of contact. Because the loads imposed upon each roller by the inner and outer races (referred to as the cone and cup, respectively) are normal to the line of contact, the roller experiences a net thrust load toward its larger-diameter end. This load is reacted by a shoulder on the inner ring (or cone) called the cone back rib. Sliding always is present between the roller and the cone back rib, resulting in the significant disadvantage of tapered roller bearings — their limited speed capability. At the present state of technology, tapered roller bearings should not be operated with a relative velocity greater than 7000 fpm between the roller and the cone back rib. However, this figure is arbitrary and will be increased as more-advanced taper bearing designs are produced.

Tapered roller bearings often are mounted in pairs or in a double-row configuration with either a one-piece dual cup or a cone. Either the pair-mounted or the double-row design can be adjusted during installation to preload the bearings axially. This preload is used to insure positive axial location of the shaft, and to prevent roller skidding. In helicopter power transmission applications, preload usually is adjusted by means of a hardened steel spacer mounted between the two bearing cones. This spacer is ground to a thickness that results in a predetermined rotational drag on the bearing assembly. Because of bearing manufacturing tolerances, the final grinding of the preload spacer is a trial-and-error procedure, and an oversized spacer should be provided with the assembly in order to insure that the proper preload can be achieved.

16-3.4 AIRFRAME BEARINGS

Several series of ball and roller bearings are manufactured specifically for airframe control applications. These bearings possess high static-load capacity, high tolerance to misalignment, and typically include integral shields or seals suitable for grease lubrication. These bearings are tailored for control applications in which they must support heavy combinations of steady and vibratory loads under conditions of variable oscillatory motion.

Ball bearings for airframe control applications are of the full-complement type which provide high-load capacity within a limited space envelope. Retainers are not required since high speeds are not a factor. These airframe bearings are produced in both annular and rod end configurations in accordance with applicable Military Specifications. Annular ball bearings for use in control applications are covered by MIL-B-7949, and the applicable standards under the specification are presented in Table 16-3. The requirements for ball bearing rod ends are defined in MIL-B-6039 and the applicable standards for this type of bearing are shown in Table 16-4.

TABLE 16-3. STANDARDS FOR AIRFRAME CONTROL ANNULAR BALL BEARINGS

STANDARD	TYPE	(PREVIOUSLY)
MS 27640	KP SERIES	(MS 20200)
MS 27641	KP-A SERIES	(MS 20201)
MS 27642	KP-B SERIES	(MS 20202)
MS 27643	DSP SERIES	(MS 20206)
MS 27644	DPP SERIES	(MS 20207)
MS 27645	KSP SERIES	(MS 28261)
MS 27646	B500DD SERIES	
MS 27647	DW, GDW SERIES	
MS 27648	KP-BS SERIES	
MS 27649	AW-AK SERIES	

In addition, two groups of rod ends referred to as the Balanced Design Series are produced to meet the requirements of NAS661. The applicable standards pertaining to this specification are NAS659 and NAS660.

Spherical roller bearings also are used for airframe applications where especially heavy loads must be carried. They provide good self-aligning capabilities and may be used for rotating shafts as well as in oscillatory applications. Such bearings are described in MIL-B-8914. The standards applicable to that specification are presented in Table 16-5.

16-3.5 SLIDING BEARINGS

Various types of sliding bearings are used in helicopter applications. The control systems of several current aircraft employ spherical bearings of both the grease-lubricated and the self-lubricating types. A typical spherical bearing is shown in Fig. 16-4. Spherical bearings also are available in rod ends

TABLE 16-4. STANDARDS FOR AIRFRAME CONTROL ROD END BEARINGS

STANDARD	TYPE
MS 21150	SOLID SHANK
MS 21151	EXTERNAL THREAD
MS 21152	HOLLOW SHANK
MS 21153	INTERNAL THREAD

TABLE 16-5. STANDARDS FOR SPHERICAL ROLLER AIRFRAME BEARINGS

STANDARD	TYPE
MS 28912	ANNULAR, SINGLE ROW
MS 28913	ANNULAR, DOUBLE ROW
MS 28914	ANNULAR, DOUBLE ROW, WIDE INNER RACE
MS 28915	TORQUE TUBE

for installation in control rod assemblies. Journal bearings are used in areas, such as rotor head scissor assemblies, where the loads and speeds do not require antifriction bearings. Sliding bearings are not subject to fatigue failures of the type that occur in rolling-element bearings, and thus they afford an increased degree of reliability. However, sliding bearings exhibit wear characteristics that are dependent upon materials, lubrication, load, speed, and environment.

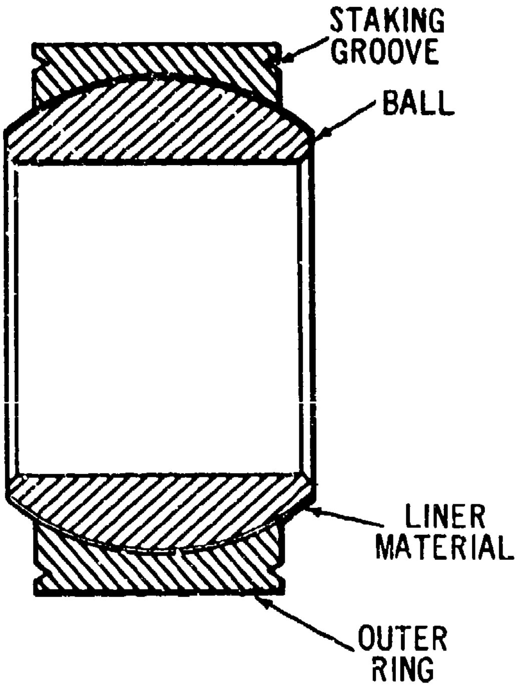


Figure 16-4. Spherical Aircraft Bearing

Grease-lubricated sliding bearings employ either steel-on-steel members or sintered bronze running on steel. Both material combinations are highly resistant to wear if they are lubricated properly and frequently. In many helicopter applications (oscillating motions), however, daily lubrication of such bearings is required in order to insure satisfactory operation.

Several self-lubricating materials are used currently as liners of inserts on the sliding surfaces of these bearings. The most commonly used liner materials are Teflon fabric and carbon-graphite.

Teflon fabric is relatively easy to manufacture and can be bonded in both spherical and journal bearings. Teflon-lined bearings are relatively low in cost and, if properly protected, can provide very good service However, the liner material is subject to deterioration as a result of exposure to water, dust, and oil environments. Teflon-lined bearings also are limited by the pressures and velocities that can be imposed upon them. Table 16-6 gives limiting values of pressure and velocity for Teflon fabric, as well as for other commonly used bearing materials. A PV factor (the product of pressure and velocity) also is given in the table; this factor frequently is used as a parameter for initial design. It should be emphasized that the values given in Table 16-6 are approximate and should be used only as a guide. The use of any self-lubricating sliding bearing in a critical component requires that carefully controlled qualification testing be performed. Unlike life calculations for antifriction bearings, those for Teflon-lined bearings, as given in most manufacturers' catalogs, can serve only as a rough approximation and must be modified by the results of qualification testing and service experience.

TABLE 16-6. PROPERTIES OF SLIDING BEARING MATERIALS FOR AIRFRAME USE

MATERIAL	LIMIT STATIC PRESSURE OVER PROJECTED AREA P, psi	LIMIT SPEED V, fpm	LIMIT PV (PRODUCT OF STRESS AND SPEED) psi x fpm
SINTERED BRONZE	8500	1200	25,000
TEFLON FABRIC	60,000	200	5000-15,000
CARBON-GRAPHITE:			
PLAIN	200-500	200-500	15,000
RESIN-IMPREGNATED	500-1000	500-1500	12,000

Carbon-graphite also has been used for self-lubricating bearings. This material combines the lubricity and low friction of graphite with the good compressive-strength characteristics of carbon. It is slightly more resistant to adverse environments than is Teflon and has demonstrated a wear life equivalent to that of a good Teflon fabric. Its disadvantages include brittleness, which may be a factor if shock loads are present, and the relatively high cost of finished bearings, which is related to the poor machinability of the material.

The Military Specifications and Standards that define the requirements for sliding bearings for airframe use are presented in Table 16-7.

16-3.6 LAMINATED ELASTOMERIC BEARINGS

The rotor heads in most helicopters employ ball, tapered roller, and needle bearings operating with oscillatory motion. Such bearings represent a substantial proportion of the rotor system weight, and require lubrication and sealing. Laminated elastomer bearings, on the other hand, require no lubrication and are lighter than conventional antifriction bearings for a given load capacity. They also permit the type of oscillatory motion and loading present in most rotor applications.

These bearings consist of thin, metal laminates alternated with thin sheets of natural rubber. The rubber is bonded to the metal by a process similar to that employed in the manufacture of lip seals.

Four basic types of laminated elastomeric bearings are made. Radial bearings are able to support a radial load only, and permit oscillation about one axis. Similarly, thrust bearings support only thrust load, and also permit motion about one axis. Conical elastomeric bearings are analogous to tapered roller bearings; they are capable of supporting both radial and thrust loads, and permit single-axis oscillation. Spherical elastomeric bearings are the most suitable for rotor systems. These bearings employ spherically shaped laminates, are able to support combined loads, and permit oscillatory motion in any plane.

The laminated construction greatly increases the stiffness of the elastomeric structure in the direction normal to the laminations, while maintaining virtually the same deflection characteristics in the plane of the laminates as would be found in a solid block of rubber. This permits laminated elastomeric bearings to support high loads while permitting the degree of motion necessary in rotor system components.

In the design of a laminated bearing, the most important parameter is known as the shape factor. This factor determines the load and deflection capabilities of the laminated structure, and is defined as the loaded area divided by the force-free area. For a single rectangular rubber laminate, the shape factor SF is given by:

$$SF = \frac{L_w B}{2t(L_w + B)}, \text{ d'less} \qquad (16\text{-}14)$$

where

L_w = laminate width, in.

B = laminate length, in.

t = laminate thickness, in.

The shape factor for a laminated elastomeric bearing is calculated as the shape factor for a single rubber laminate multiplied by the number of

TABLE 16-7. SPECIFICATIONS AND STANDARDS FOR SELF-LUBRICATING SLIDE BEARINGS

SPECIFICATION	STANDARD	SPECIFICATION	STANDARD	TYPE
MIL-B-8942	MS 21230	MIL-B-81820	MS 14103	WIDE ANNULAR, GROOVED
MIL-B-8942	MS 21231	MIL-B-81820	MS 14102	WIDE ANNULAR, NONGROOVED
MIL-B-8942	MS 21232	MIL-B-81820	MS 14101	NARROW ANNULAR, GROOVED
MIL-B-8942	MS 21233	MIL-B-81820	MS 14104	NARROW ANNULAR, NONGROOVED
MIL-B-8943	MS 21240	MIL-B-81934	MS 21240	SLEEVE (JOURNAL), PLAIN
MIL-B-8943	MS 21241	MIL-B-81934	MS 21241	SLEEVE (JOURNAL), FLANGED
MIL-B-8948	MS 21242	MIL-B-81935	MS 21242	ROD END - MALE
MIL-B-8948	MS 21243	MIL-B-81935	MS 21243	ROD END - FEMALE

laminates used. For helicopter applications, shape factors of 30 to 40 have been found to yield the best results. Additional information on the design of this type of bearing can be found in Ref. 10.

16-3.7 BEARING SEALS AND RETAINERS

16-3.7.1 Seals

Several types of sealing devices are employed in helicopter power trains and rotor systems. The most commonly used type is the radial lip seal. In this configuration, a V-shaped sealing lip, which contacts the rotating or oscillating shaft, is held within or bonded to an outer case, which in turn is pressed into the component housing. The sealing lip typically is fabricated from either natural or synthetic rubber. For oil-lubricated applications, the sealing lip is forced against the shaft by means of a circumferential garter spring. In grease-lubricated components, this spring usually is omitted.

Radial lip seals are fairly inexpensive to manufacture, and provide reliable operation at surface speeds up to approximately 3500 fpm. For higher speeds, a hydrodynamic lip seal should receive first consideration. In this type of seal, the interface pressure between the shaft and the seal is low, and a hydrodynamic oil film is maintained at the interface. Helical grooves, indentations, or ribs are molded into the air side of the sealing lip and are effective in directing the oil flow back into the area to be sealed. Hydrodynamic lip seals can be operated at surface speeds of up to 10,000 fpm because of the reduced contact force and the maintenance of an oil film that lubricates the sealing lip.

For all lip seals, the compatibility of the seal material with the fluid to be sealed must be taken into account. Many of the current lubricants, particularly those of the synthetic ester type, have a detrimental effect on certain seal materials. Materials such as nitrile rubber and fluoroelastomers are available for use with this class of lubricants. Their use results in increased seal cost, but often is necessary in order to prevent deterioration of the seal lip and resultant leakage.

For applications involving high operating temperatures or shaft speeds, carbon-face seals generally are used. These seals consist of a carbon sealing face, called a nosepiece, bearing against hardened steel mating rings, which must be extremely flat. Such seals have been used successfully at speeds and temperatures typical of turbine engine operation. Because of the very precise tolerances required in the manufacture of all parts of carbon-face seals, their cost is many times that of a similarly sized lip seal. Therefore, face seals should be restricted to applications where surface speeds or temperatures prohibit operation with a lip seal.

Clearance seals — such as labyrinth and ring seals — also are used for high-speed operation. They provide satisfactory sealing when the shaft is rotating, but permit leakage under static conditions. Therefore, these seals should be used only in applications in which a static head of oil is not present in the seal area when the component is not in operation.

For applications in which small oscillatory motions are present, such as in rotor head hinges, diaphragm seals have been used successfully. These seals employ an elastomeric membrane that spans the gap between the oscillating and stationary membranes of the seal. The membrane undergoes torsional deflection under the oscillatory motion. As long as the elastomer and the metal-to-elastomer bonds remain intact, the diaphragm seal is able to operate without leakage.

Additional information on various available sealing devices can be found in Ref. 11.

16-3.7.2 Bearing Retention

Several methods are used to retain bearing rings on shafts or in housings. They include lock nuts, snap rings, retention clips, and staking. A lock nut is used to retain the inner ring of a bearing on a shaft. Together with the diametral interference fit, the lock nut prevents the bearing from creeping under load, and also provides axial location for the bearing. The lock nut may be secured with a tab washer. Tabs inserted into slots on the nut lock the washer to the nut, while serrations on the bore of the washer fit into similar serrations on the thread area of the shaft, thus preventing the assembly from turning.

For lightly loaded applications, in which creeping is not a factor, a bearing outer ring may be retained by a snap ring. The snap ring is installed in a groove in the bearing housing, and bears against the face of the bearing outer ring. However, snap rings provide no circumferential retention for the bearing and, therefore, should not be used where the outer ring is rotating with respect to the radial load. For such applications, retention clips normally are employed. These clips are bolted to the bearing housing, and fit into slots that are machined into the face of the bearing outer ring. The clips thus can provide both axial and circumferential retention. However, bearing clips should not be used as the primary method for prevention of creeping. Proper selection of the interference fit will insure that the bearing does not turn in the housing under load.

Spherical control bearings commonly are retained in a control rod by staking their outer rings. This method uses a circumferential, V-shaped groove machined into the bearing outer ring, as shown in the detail of Fig. 16-4. After the bearing is pressed into the housing, a special tool is used to roll the portion of the outer ring outboard of the groove over a chamber in the base of the housing. This provides positive axial retention of the bearing, and permits the bearing to be removed from the assembly by applying a heavy axial load. Because loading of these bearings is primarily radial in direction, the staking procedure has been found to be satisfactory for most helicopter applications of spherical control bearings.

16-4 ELECTRICAL FITTINGS

16-4.1 GENERAL

The design of electrical systems is discussed in Chapter 7. This paragraph discusses the fittings used in these electrical systems.

Electrical fittings should be selected for reliability and ease of maintenance. Therefore, several design principles should be observed:

1. Mounting hardware should be connected permanently to the part being mounted.
2. Features should be provided to prevent incorrect assembly.
3. Right- and left-hand parts either should be identical or should be incapable of being interchanged.
4. Components should be protected against incorrect use of attachments.
5. Universal mounting features should be incorporated where possible.
6. The use of dissimilar metals in intimate contact *shall* be avoided (see MIL-STD-889 and MIL-STD-454, Requirement 16, for definitions of dissimilar metals). An interposing compatible material *shall* be used if combinations must be assembled.
7. The number of wire and cable junctions should be minimized, and only approved devices *shall* be used where junctions are required.
8. The use of identical connectors in adjacent locations should be avoided.
9. Junctions should be accessible for inspection and maintenance.
10. Terminals and junctions should be spaced a sufficient distance apart to prevent arcing and detrimental current leakage between circuits.

Because of the number and variety of switches, terminal blocks, connectors, terminals, and insulating materials available, manufacturers' data and the referenced Military Specifications should be consulted for additional information not presented in this chapter.

Marking requirements of selected electrical and electronic parts are defined in MIL-STD-1285. MIL-E-7080 contains the general requirements for airborne electrical equipment.

16-4.2 CONNECTORS AND CABLE ADAPTERS

Connectors are used for joining a cable to other cables or for joining cables to equipment in cases where frequent disconnect is required. The most commonly used connectors have been the circular and the rack-and-panel types. MIL-W-5088 does not state a requirement for a specific type of connector, but requires that selection and use *shall* be in accordance with MIL-STD-1353. Although there are more than 30 Military Specifications dealing with connectors, the designer should reduce the quantity of connector variations and limit the selection to those which lend themselves to common termination methods; i.e., common contacts, common back hardware, and common assembly tools. This discussion is limited to those connectors most commonly used in helicopters. Table 16-8 provides a ready reference to these types, along with their general descriptions. Table 16-9 lists additional specifications to be used where special requirements exist.

Connector is a generic term used to denote both an electrical plug and a receptacle. A plug is a connector that normally is attached to a free-swinging electrical cable. A receptacle is a connector that normally is attached rigidly to, or is an integral part of, a supporting surface. Each connector *shall* be selected to make the "live" or "hot" side of the connector the socket member to minimize possible shorting when the junction is disconnected. The dead side of the circuit is the pin member. Therefore, depending upon the individual circuit, a plug may contain either pins or sockets. The mating receptacle, of course, will contain the opposite.

16-4.2.1 Connector Selection

A multitude of connector designs, with specific capabilities, are available. Moreover, many of the features of these designs overlap. Therefore, the requirements pertinent to the task for which the connector is intended should be identified and then compared with the features of those available.

Information necessary for the selection of the proper connector includes the following:

1. Specific types of connectors, if any, designated in the contract
2. Applicable environmental conditions
3. Maximum voltage and current for each circuit
4. Number of circuits to be accommodated plus spare contacts

TABLE 16-8. MILITARY SPECIFICATIONS AND STANDARDS FOR CONNECTORS FOR AIRCRAFT

SPECIFICATIONS	TYPE	STANDARDS	DESCRIPTION
MIL-C-5015	CIRCULAR	SEE SPEC. SUPPS.	CONNECTORS, ELECTRICAL, "AN" TYPE
MIL-C-26500	CIRCULAR	SEE SPEC. SUPPS.	CONNECTORS, GENERAL PURPOSE, ELECTRICAL, MINIATURE, CIRCULAR, ENVIRONMENT RESISTING, ESTABLISHED RELIABILITY
MIL-C-38999	CIRCULAR	SEE SPEC. SUPPS.	CONNECTORS, ELECTRICAL, CIRCULAR, MINIATURE, HIGH DENSITY, QUICK DISCONNECT, ENVIRONMENTAL RESISTING, REMOVABLE CRIMP TYPE CONTACT, RELIABILITY ASSURANCE PROGRAM
MIL-C-83723	CIRCULAR	SEE SPEC. SUPPS.	CONNECTORS, ELECTRICAL, CIRCULAR, ENVIRONMENT RESISTING, GENERAL SPECIFICATION FOR
MIL-C-24308	RACK AND PANEL	SEE SPEC. SUPPS.	CONNECTORS, ELECTRICAL, RECTANGULAR, MINIATURE POLARIZED SHELL, RACK AND PANEL, GENERAL SPECIFICATION FOR
MIL-C-26518	RACK AND PANEL	SEE SPEC. SUPPS.	CONNECTORS, ELECTRICAL, MINIATURE, RACK AND PANEL, ENVIRONMENT RESISTING, 200°C AMBIENT TEMPERATURE
MIL-C-28748	RACK AND PANEL	SEE SPEC. SUPPS.	CONNECTORS, ELECTRICAL, RECTANGULAR, RACK AND PANEL, SOLDER TYPE AND CRIMP TYPE CONTACTS, GENERAL SPECIFICATION FOR
MIL-C-55544	FLAT FLEX CABLE	SEE SPEC. SUPPS.	CONNECTORS, ELECTRICAL, ENVIRONMENT RESISTING, FOR USE WITH FLEXIBLE, FLAT CONDUCTOR CABLE, GENERAL SPECIFICATION FOR SEE MIL-C-55543
MIL-C-55302	PRINTED CIRCUIT	SEE SPEC. SUPPS.	CONNECTOR, SOCKET, STRAIGHT THROUGH, FOR MULTILAYERED PRINTED WIRING BOARDS

5. Type of attachment required
6. Wire size, material, construction, and other characteristics
7. Type of coupling required
8. Special mounting provisions required.

The requirements of MIL-W-5088 also should be considered. These cover safety-wiring of coupling nuts, use of noninterchangeable connectors in adjacent locations, drainage provisions, insulation, adapters, and sealing requirements. When conditions permit a choice, the crimp style of attachment of wire terminations is preferred. The use of identical connectors in adjacent locations *shall* be avoided. Difference in size or insert arrangement is preferred.

TABLE 16-9. OTHER MILITARY SPECIFICATIONS AND STANDARDS FOR CONNECTORS

SPECIFICATIONS	TYPE
MIL-C-10544	CIRCULAR
MIL-C-12520	CIRCULAR
MIL-C-22249	CIRCULAR
MIL-C-22539	CIRCULAR
MIL-C-22992	CIRCULAR
MIL-C-24217	CIRCULAR
MIL-C-26482	CIRCULAR
MIL-C-27599	CIRCULAR
MIL-C-27699	CIRCULAR
MIL-C-55116	CIRCULAR
MIL-C-55181	CIRCULAR
MIL-C-55243	CIRCULAR
MIL-C-81511	CIRCULAR
MIL-C-81582	CIRCULAR
MIL-C-81703	CIRCULAR
MIL-C-21617	RACK AND PANEL

Where identical connectors are used in adjacent locations, wires and cables *shall* be so routed and supported that improper connections cannot be made. Adjacent connectors using the same insert arrangement *shall* be selected to take advantage of alternate insert positions or alternate shell keying positions. If this requirement cannot be met, color coded sleeves having the identification of the associated receptacles *shall* be attached to the wires or cables near the plugs. The receptacles *shall* be color coded by a colored band on the mounting structure.

16-4.2.2 Circular Connectors

Circular connectors are the most popular st[illegible] used for general aircraft wiring. This type of connector is shown in Fig. 16-5(A), and Table 16-8 contains a listing of these connectors and their descriptions.

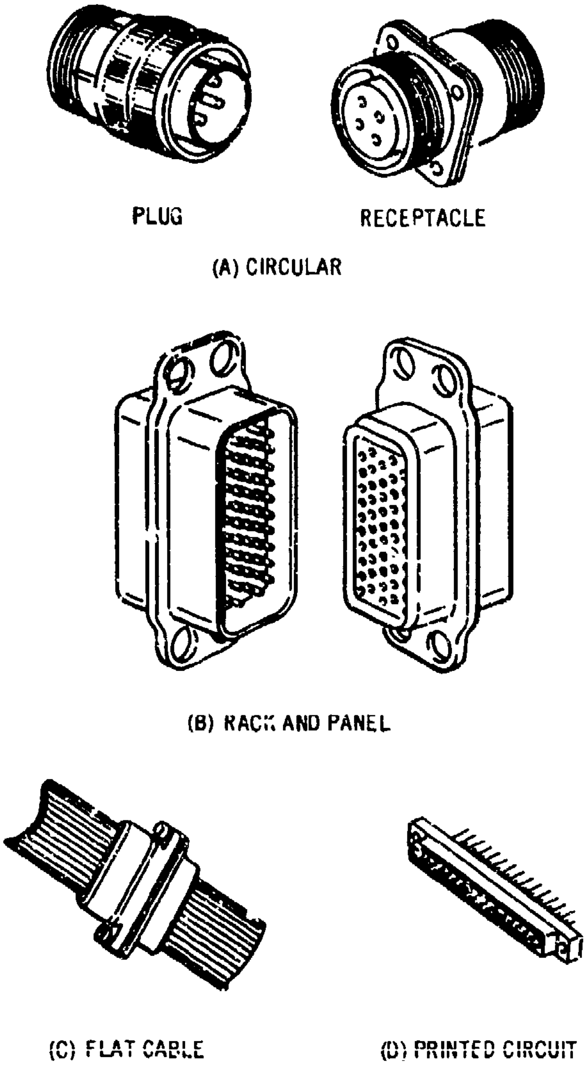


Figure 16-5. Common Types of Connectors

Standard MS connectors are available with from 1 to more than 100 contacts, and in 15 different insert diameters ranging from 0.250 to 2.550 in. Connector sizes are based on the diameter of the receptacle shell, stated in sixteenths of an inch. Standard contact sizes available range from 0 through 22. Either solder- or crimp-style terminations may be obtained; however, unless otherwise approved by the procuring activity, crimp-style terminations should be used. Standard pin arrangements for cylindrical connectors conforming to MIL-C-5015 are contained in MS 33680 through MS 33690. Pin arrangements of other types of connectors can be found in pertinent MS standards.

16-4.2.2.1 Termination Seals

Environment resistant connectors having wire sealing grommets *shall* be used whenever possible; however, potting may be used when no connector having a sealing grommet is suitable for the application. Sealing materials should meet the applicable environmental requirements and should be selected from either MIL-S-8516 or MIL-S-23586.

16-4.2.2.2 Cable Adapters

A cable clamp often is used at the end of the circular connector to support the cable or wires and to prevent twisting and pulling of the contact terminations. Avoiding this motion also helps to reduce the transmission of moisture along the wire. For applicable cable clamps and other accessory hardware, consult the supplement to the connector specification.

16-4.2.2.3 Connector Couplings

Three basic styles of circular connector couplings are used: threaded, bayonet, and push-pull. The bayonet and push-pull styles offer quick-disconnect features. Other features, or a combination of features, are sometimes used. For example, a pull-to-breakaway action may be incorporated with the basic coupling.

16-4.2.3 Rack and Panel Connectors

Rack and panel connectors are used to connect a cable to a fixed receptacle, a cable to a cable, or a module to a backplate. The common configuration is rectangular or some variation of a rectangular shape (see Fig. 16-5(B)). These connectors are available in many sizes and shapes conforming to Military Specifications, and also in commercial configurations. Refer to Tables 16-8 and 16-9 for listings of the military types, and to the manufacturers' and distributers' catalogs for the commercial types.

16-4.2.4 Flat Conductor Cable Connectors

Where flexible, flat conductor cable is used, connectors of the MIL-C-55544 type (Fig. 16-5(C)) should be considered. These connectors are suitable for connecting flat cable to flat cable, to round wires, or to printed circuit boards.

16-4.2.5 Printed Wiring Board Connectors

Printed wiring board or printed circuit connectors (Fig. 16-5(D)) normally are not encountered in aircraft wiring. They are used for connecting printed wiring boards to conventional wiring, printed wiring boards to each other, or printed wiring boards to a backplate. Connectors in accordance with MIL-C-55302 are preferred for printed wiring board applications.

16-4.3 TERMINALS

Common types of terminals currently in use include lug, eyelet, and notched. Their installation may be of the dip-solder, solderless wrap, taper tab, taper socket, or crimp style. Unless otherwise specified in the contractual documents or approved by the procuring activity, wire and cable terminals should be of the crimp style.

Crimp terminals allow direct contact of wire and terminal to be accomplished by deformation. They can be installed quickly with uniform and reliable quality, even by newly trained personnel. The crimp can be accomplished with hand tools or with automatic equipment and can be inspected easily. The crimp terminal may be used in high-temperature applications where other types would be unacceptable. The connection is electrically sound and mechanically strong. However, reuse of crimp joints is almost impossible. A common practice is to provide extra wire length in new harness assemblies so that a damaged or defective terminal may be replaced by cutting off the old one and crimping on a new one. AMCP 706-125 provides a detailed discussion of the requirements and characteristics of various types of terminations. MIL-STD-195 defines the marking of connections for electrical assemblies.

For copper wire, MS 25036 terminals (Type II, insulated) should be used — except in applications requiring conductor temperatures above 105°C. For such applications, uninsulated terminals (Type I), conforming to MS 20659 and to the requirements of MIL-T-7928, should be used. For aluminum cable, terminals conforming to MIL-T-7099 should be used. Table 16-10 contains additional standards.

16-4.4 TERMINAL BOARDS

Where wire or cable junctions require infrequent disconnection, or where it is necessary to join two or more wires or cables to a common point, terminal boards should be used.

Terminal boards should conform to MS 27212, and covers conforming to MS 18029 should be used with them. These boards have molded barriers and molded-in studs. They may be purchased in standard lengths and cut to the length (number of studs) required for each installation. Terminal board identification should be in accordance with the paragraph titled "Junctions" in MIL-W-5088.

Busses connecting the terminal studs should conform to MS 25126.

TABLE 16-10. MILITARY SPECIFICATIONS AND STANDARDS FOR CRIMP-STYLE TERMINALS

SPECIFICATIONS	TYPE	CLASS	STANDARDS	WIRE
MIL-T-7928	I	1	MS20659, MS21003, 4, 5, 6, 7, 8, 9, 10, 11, 12, 13, 14, AND 15	COPPER
MIL-T-7928	II	1	MS17143, MS25036	COPPER
MIL-T-7099		1	MS25435, 6, 7, 8 AND 9	ALUMINUM
MIL-T-21608	FERRULE	1	MS18121, MS21980, MS21981	SHIELDED CABLE
MIL-T-38732	SPLICE		MS27429	COPPER

16-5 ELECTRICAL SWITCHES

16-5.1 GENERAL

Military switches must be designed to obtain maximum power capacity within a limited size and space. Electrically, the most important considerations involve contact load rating and arrangement. These factors also may dictate the type of switch to be used in a given application. Other considerations include desired switching sequence, insulation resistance, radio interference, environmental aspects, and any safety features that might be required.

The load-carrying requirements for each switch must be analyzed thoroughly. Such factors as voltage, steady current, surge current, frequency, contact heat dissipation, types of loads, and life requirements must be considered.

The most commonly used switches are toggle, push-button, slide, and rotary. Each one has particular advantages, which must be weighed for the design task under consideration. MIL-STD-1132 contains selection and installation requirements for switches.

Where inadvertent actuation of a switch might produce serious consequences, a switch guard *shall* be installed. The acceptable types of guard generally fall into either the fixed or the hinged category.

Fixed guards are channel-shaped, metal members that require a finger to be inserted into the area between the channel legs to actuate the switch. See MS 24417 and MS 25221 for this configuration.

Hinged guards have a cover that swings down over the switch. This cover must be raised and rotated out of the way before the switch can be actuated. Two types of hinged guards specified in MIL-G-7703 are Type A switch guards, which have more than one maintained position, and Type B switch guards, which are spring-loaded to the closed position. Military Standards describing switch guards include MS25024, -25214, 25223, 25224, -25225, and -25452.

Table 16-11 lists specifications and standards applicable to the several types of switches.

16-5.1.1 Toggle Switches

Two types of toggle switch action are common: momentary and maintained. These switches generally are actuated by a toggle lever (often shaped in the form of a bat), and have a snap type of action. They usually have either two or three positions: either side of the center and the center position. Other arrangements are available as special configurations. The switching action may be one of many standard types, such as rocker-contact, positive-action toggle link, or others. Toggle switches are relatively low in cost, and are used widely.

16-5.1.2 Push-button Switches

Push-button switches are classified as momentary-action, maintained-action, and sequential-action

TABLE 16-11. MILITARY SPECIFICATIONS AND STANDARDS FOR SWITCHES

SPECIFICATION	TYPE	STANDARDS	DESCRIPTION
MIL-S-3786	ROTARY	SEE SPECIFICATION SUPPLEMENTS	SWITCH, ROTARY, GENERAL SPECIFICATION LOW POWER, AC, DC, FOR ELECTRONIC AND COMMUNICATION EQUIPMENT
MIL-S-3950	TOGGLE	SEE SPEC. SUPPS.	SWITCH, TOGGLE, GENERAL SPECIFICATION SEALED AND UNSEALED, AC AND DC
MIL-S-6743	PUSH BUTTON AND LIMIT		REPLACED BY MIL-S-8805
MIL-S-6744	PUSH BUTTON AND LIMIT		REPLACED BY MIL-S-8805
MIL-S-6745	TOGGLE		
MIL-S-6746	ROTARY	AN3212, AN3213, AN3214	SWITCHES, ROTARY, SHIELDED AIRCRAFT IGNITION SINGLE, TWO AND FOUR ENGINE
MIL-S-6807	ROTARY	MS21994 AND 5 MS25002, MS90547	SWITCH, ROTARY, SELECTOR POWER FOR USE IN POWER CIRCUITS
MIL-S-8805	SENSITIVE AND PUSH	SEE SPEC. SUPPS	SWITCHES AND SWITCH ASSEM. SENSITIVE AND PUSH (SNAP ACTION), GENERAL SPECIFICATION SINGLE PHASE, AC, DC
MIL-S-8834	TOGGLE	MS21026 AND 7, MS24612, 13, AND 14, 55 AND 6, MS25306, 7, AND 8, 10 AND 11	SWITCHES, TOGGLE, POSITIVE BREAK, AIRCRAFT, GENERAL SPECIFICATION FOR SEALED, AC, DC
MIL-S-9419	TOGGLE	MS28939, SEE PAR. 3 OF THE SPECIFICATION	SWITCH, TOGGLE, MOMENTARY, FOUR-POSITION ON, CENTER OFF MINIMUM ENVELOPE, FREE RETURN
MIL-S-55433	DRY-REED	SEE SPEC. SUPPS.	SWITCH CAPSULES, DRY-REED TYPE HERMETICALLY SEALED IN GLASS MAY BE MAGNETICALLY OPERATED, FOR USE IN COMMUNICATION, ELECTRICAL, ELECTRONIC EQUIPMENT

types. A pushing motion in line with the button normally is required; however, a modification of the maintained-action type requires a pull-to-operate motion.

Military push-button switch requirements are defined in MIL-S-8805. The switches described are available in five enclosure designs, four temperature characteristics, two shock types, and three vibration grades. The specification covers both sensitive and push-type switches. Sensitive switches are intended for a nonhand-operated mode, while manual (push) switches are intended for hand operation. The Supplement to MIL-S-8805 contains a list of specification sheets by switch title and a list of superseded documents (MS, AN, and JAN standards and specifications).

16-5.1.3 Rotary Switches

Rotary switches are used to control a number of circuits in consecutive steps. These switches are available in single or multiple wafers, allowing a large variation in the configurations and making the switches adaptable to many tasks.

Detents are provided on these switches by the use of steel balls carried in a detent plate. The detents cause a requirement for a relatively high torque to turn the knob, and the switching takes place with a snap action at maximum force and velocity. This action reduces the arcing time between contacts and promotes longer contact life.

Most rotary selector switches are used in electronic circuitry where currents are low. The use of the sliding type of contacts on these switches provides the necessary self-cleaning required for low contact resistance. Rotary switches should be installed with the extreme counterclockwise position as the OFF position. MIL-S-3786, MIL-S-6746, and MIL-G-7090 define the requirements for rotary switches.

16-6 PIPE AND TUBING FITTINGS

16-6.1 GENERAL

Aircraft hydraulic systems *shall* be designed and installed in accordance with MIL-H-5440 unless otherwise directed by the procuring activity. Hydraulic system requirements are discussed in Chapter 9 and the requirements for fuel and lubrication systems are described in Chapter 3. This paragraph discusses the fittings used in the installation of hydraulic systems and in the installation of fuel and lubrication systems.

Since each joint represents a possible leakage and trouble spot, the installation of piping systems must be accomplished carefully. The number of connections used should be reduced to a minimum in order to assure maximum safety and system efficiency with minimum installation and maintenance costs.

16-6.2 TYPES OF FITTINGS

Many types of pipe and tube fittings are available. For aircraft systems, those generally used are tapered pipe thread, straight thread using a gasket seal, flared tube, flareless tube, thin-wall tube, quick disconnect, and permanent. Solder-type tube fittings conforming to MIL-F-6001 are inactive for new design, except for oxygen systems and engine primer lines.

16-6.2.1 Tapered Pipe Threads

Tapered pipe thread fittings provide reasonably leakproof connections, but should be used only for permanent attachments or closures. Fig. 16-6 illustrates this type of fitting and the related specifications and standards. High-quality workmanship and machining to close tolerances will insure a good seal with this type of fitting. However, it cannot be used for directional adjustment because it must be tightened properly to prevent leaks, and there is a

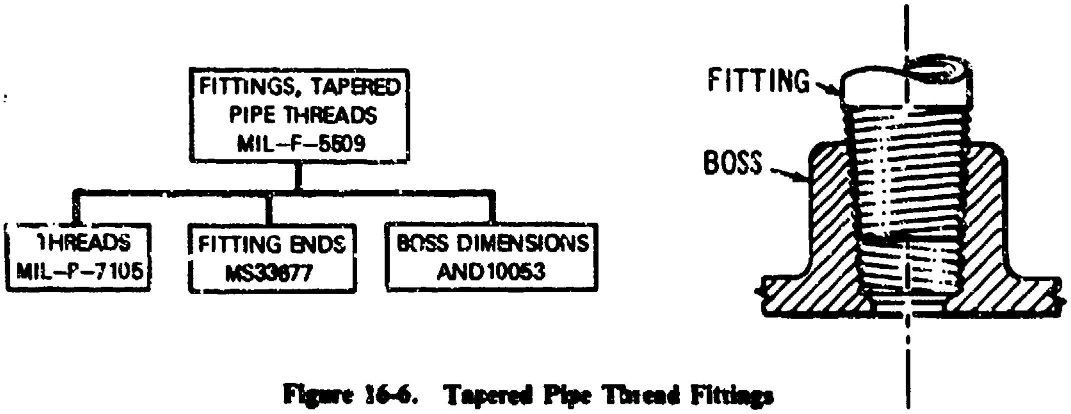


Figure 16-6. Tapered Pipe Thread Fittings

danger of damaging the component housing if the fitting is overtightened. This type of fitting is inactive for new aircraft design, except for oxygen systems, and should be avoided where possible

16-6.2.2 Straight Thread Fittings

Straight thread types are preferred for installation of fittings into the boss of a component, as illustrated in Fig. 16-7. Standard internal thread dimensions for bosses are shown in MS 33649. The tubing end fittings should conform to MS 33514 and MS 33515 (flareless tube style); or MS 33656 and MS 33657 (flared tube style); or MS 24385 and MS 24386 (flared tube precision style). Installation instructions are given in MS 33566 and AND 10064.

Straight thread fitting installations are usable for nominal operating pressures up to and including 3000 psi but are inactive for hydraulic systems in new aircraft design. For the universal elbow fitting installation, refer to AND 10080.

MS 28778 or MS 29512 gaskets should be used for producing the seal between straight thread fittings and components. As the pressure increases, the gasket increases the sealing effect.

This style of fitting requires less torque to insure a good seal than is necessary for pipe threads, and it may be disassembled and reassembled without deterioration of the joint quality. Choice of the proper thread style (MS 33566 and AND 10064) allows the fitting to be positioned without impairing the sealing properties.

Installations of this type of fitting should be made carefully in order to minimize service problems. Extreme caution should be used when a gasket is installed on the fitting. The gasket should not be pushed over the threads or other areas that might nick the gasket surface. A plastic or metal thread-protector or "thimble" should be used during installation of the gasket.

Specifications for installation, gasket selection and lubrication, fitting torque, and positioning should be in accordance with MS 33566 and AND 10064.

16-6.2.3 Flared Tube Fittings

Flared tube fittings are of the expanding type. The end of the tube is flared to an angle of 37 deg in the mating sleeve. The sleeve and flared tube end then are clamped between the fitting and the nut to form the fluid seal. Fittings of this type are shown in Fig. 16-8 and should conform to MIL-F-5509. The styles of fittings available are listed in AND 10059.

The external male fitting end should conform to the dimensions listed in MS 33656 or MS 33657 for flared tube connections, or to MS 24385 or MS 24386 for the precision type of flared tube connections. For 3000-psi systems, components smaller than 0.5-in. tube size should be made of steel or aluminum alloy. Aluminum alloy fittings should be used with caution in installations where repeated disassembly and reassembly could damage the threaded fitting end.

The tubing end should be flared in accordance with MS 33584, except for aluminum alloy tubing of 0.388 in. or less (outside diameter); this, instead, should be double flared in accordance with MS 33583.

These fittings are intended for use in aircraft hydraulic and pneumatic systems. This type of fitting is

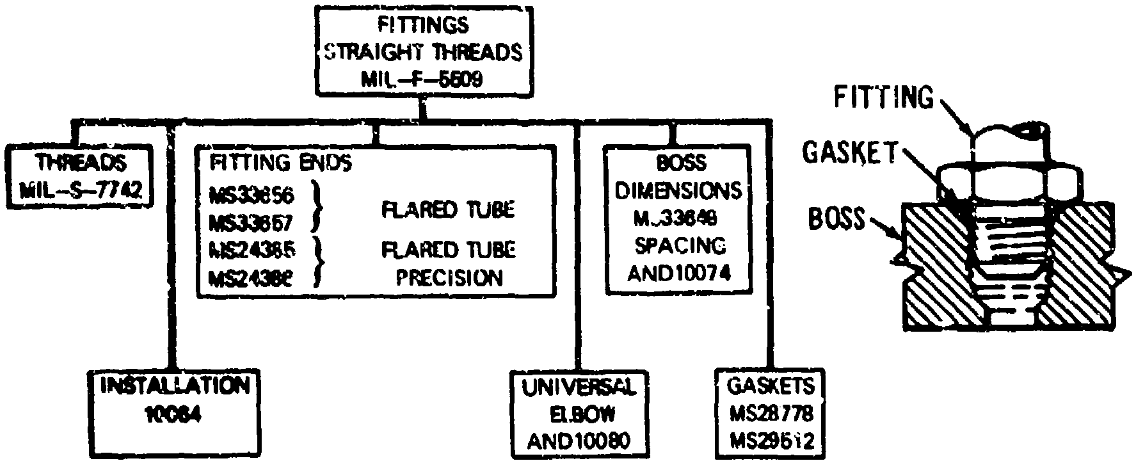


Figure 16-7. Straight Thread Fittings

inactive for hydraulic systems in new aircraft design. If a flared tube fitting connection is made improperly, the joint will leak. However, line pressure is not likely to cause the tube to blow out of the joint. For installation information concerning flared tube fittings, see AND 10064.

The tubing layout and routing should be designed in accordance with MIL-H-5440 or MIL-P-5518. The tubing should be formed so that no strain is placed upon the fitting when the connection is tightened. Sufficient straight length should be allowed between the end of a bend and the flared portion of the tube for installation of the nut and sleeve. Tube cutoff must be square, and all burrs must be removed before the nut and sleeve are placed on the tube and the flare is made. The finished flare *shall* be inspected for pock marks, splits, inadequate squareness or concentricity, and other defects. AND 10064 defines acceptable values for nut torque for assembly of the joint.

16-6.2.4 Flareless Tube Fittings

Flareless tube fittings are of the compression type, as shown in Fig. 16-9. The tube does not require a flaring operation. As the nut is tightened, the sleeve is compressed by the fitting body and deflects into the outer surface of the tubing wall to create an interference seal between sleeve and fitting and sleeve and tube.

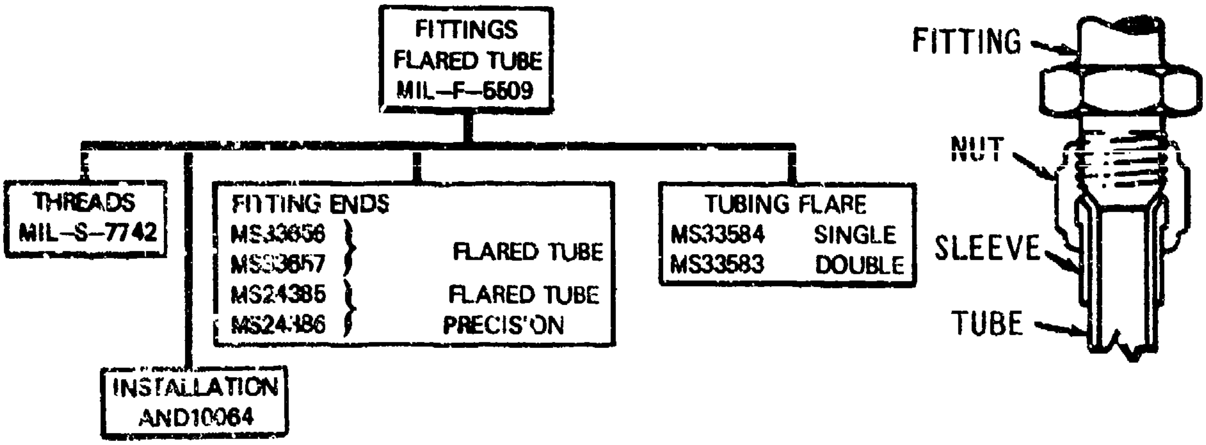

FITTINGS
FLARED TUBE
MIL-F-5509
THREADS
MIL-S-7742
FITTING ENDS
MS33656
MS33657
FLARED TUBE
MS24385
MS24386
FLARED TUBE
PRECISION
TUBING FLARE
MS33584 SINGLE
MS33583 DOUBLE
INSTALLATION
AND10064
FITTING
NUT
SLEEVE
TUBE

Figure 16-8. Flared Tube Fittings

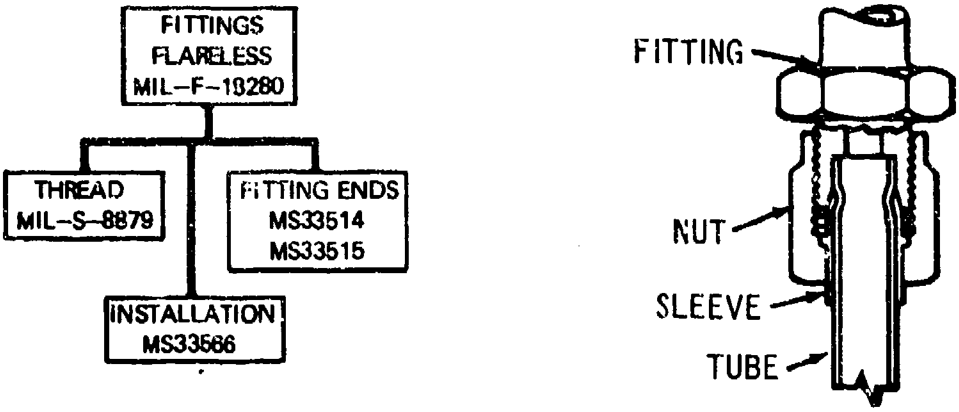

FITTINGS
FLARELESS
MIL-F-18280
THREAD
MIL-S-8879
FITTING ENDS
MS33514
MS33515
INSTALLATION
MS33566
FITTING
NUT
SLEEVE
TUBE

Figure 16-9. Flareless Tube Fittings

Fittings of this type are intended for use in aeronautical fluid systems in accordance with MIL-H-5440 and MIL-P-5518. They should conform to MIL-F-18280, and the installation instructions of MS 33566 should be followed.

The external male fitting end for flareless tubing should conform to the dimensions of MS 33514 or MS 33515. For 3000-psi systems, tubing smaller than 0.5-in. used on the pressure or actuating parts at the circuit should be made of steel. Aluminum or steel may be used for return lines. Tubes measuring more than 0.5-in. may be of steel or aluminum alloy. Where repeated disassembly and reassembly could damage the threaded fitting end, caution must be used if the fitting is of aluminum.

The tubing layout and installation should be designed so that no bending or springing of the tube is required for alignment with the fitting.

For flareless tube fittings, a tube of less-ductile material — such as high-strength, thin-wall tubing — may be used.

When flareless fittings are used, an improperly constructed joint may cause the tube to be blown out of the fitting when pressure is applied to the line. This could result in the complete loss of the hydraulic fluid.

The tube should be cut square, and internal and external burrs should be removed. The nut and sleeve must be slipped on the tube and the tubing then held against the shoulder of the fitting while the nut and sleeve are being tightened. The nut should be torqued in accordance with the instructions of MS 33565.

16-6.2.5 Thin Wall Tube Connectors

Installations involving large volumes of fluid, such as pneumatic systems or power plant fuel systems, should use relatively large, thin-wall tubing; standard tubes and tube fittings could result in unacceptably high weights and pressure losses. Fuel systems *shall* be designed to the requirements of MIL-F-38363.

For fuel systems where operating pressures to 125 psi are present, couplings conforming to MIL-C-22263 should be considered. Type I couplings connect two tubes with MS 33660 Type A rolled bead ends; Type II couplings connect two tubes with MS 33658 machined fitting ends, two MS 33660 Type A rolled bead ends, or one MS 33658 machined and one MS 33660 Type A rolled bead; and Type III couplings connect two straight end tubes. Standard coupling sizes are from 1.00 to 8.00 in. This type of coupling allows expansion and contraction, small angular and radial misalignments, and smooth flow. Fuel hose coupling requirements are given in MIL-H-7061 and MIL-H-58089.

16-6.2.6 Quick-disconnect Couplings

For fuel and oil lines, automatic shutoff, quick-disconnect couplings should be used and should meet the requirements of MIL-C-7413 for Type I fuel line couplings or Type II oil line couplings. Reference should be made to the specification for classes and other data.

For the hydraulic, self-sealing, quick-disconnect couplings used in Type I and Type II hydraulic systems, detail requirements are covered by MIL-C-25427. Two classes of couplings are designated: Class 600 with a rated pressure of 600 psi, furnished only in 1.25-in. tube size; and Class 3000 with a rated pressure of 3000 psi, furnished in 0.25- through 1.0-in. sizes.

For protection against post-crash fires, all flammable fluid systems *shall* include automatic-shutoff, breakaway fittings. These fittings *shall* be designed to break away at the specified crash load factor, and to shut off and thus prevent spillage of flammable fluids. Such components are discussed in Chapter 8, AMCP 706-201, and in Ref. 12, and in MIL-STD-1290.

16-6.2.7 Permanent Fittings

MIL-H-5440 requires the use of permanently joined tube fittings except for production breaks and component removal. Permanent connections provide a higher degree of reliability than either flare or flareless fittings and can result in substantial weight savings for the system. The four methods in current use are swaging, brazing, welding, and cryogenic. However, standards do not exist for these components. Usage, installation, procedures, and test requirements are proprietary.

16-7 CONTROL PULLEYS

16-7.1 GENERAL

Selection of the proper pulley for a cable system is important in achieving long cable life. Some of the earliest control systems for aircraft were of the cable type, using pulleys when a directional change was required. These systems gave long, trouble-free service when properly installed.

MIL-F-9490 requires that approved Military Standard pulleys in accordance with MIL-P-7034 be used in flight control systems. The latter specification covers the requirements for single-groove, antifriction-bearing pulleys of two types and three classes: Type I-Nonmetallic and Type II-Metallic; and Class 1-Secondary control, Class 2-Flight control, and Class 3-Heavy-duty control. For standard configurations, see MS 20219, MS 20220, MS 20221, and MS 24566. Performance and strength requirements and other data are given by MIL-P-7034.

16-7.2 PULLEY SELECTION

The selection or design of the proper pulley for achieving optimum performance should be based upon several guidelines. A pulley of the largest feasible diameter should be used, and the groove radius and the pulley strength *shall* be appropriate for the size of cable being used.

16-7.2.1 Pulley Diameter

The diameter of the pulley has a primary influence upon cable life. An increase in cable life of 10-15 times can be achieved by doubling the size of a pulley from an initial pulley-to-cable-diameter ratio of less than eight. Smaller, but still significant, improvements are available as the diameter ratio increases. However, increasing the pulley diameter results in increased weight and space requirements, so the improved cable life must be evaluated against these factors. Nevertheless, pulleys of less than 30 times the cable diameter should not be used under normal circumstances; those with a larger diameter ratio should be used where possible.

16-7.2.2 Pulley Groove

The radius of the pulley groove also is important. Using a cable in a pulley where the groove radius is too small causes a wedging action, possibly resulting in distortion of the cable. When the groove radius is too large, insufficient contact exists, resulting in deformation of the groove tread and distortion of the cable, and, hence, the possibility of premature failure. The radius of the pulley should equal one-half the cable diameter plus approximately 0.015 to 0.030 in. for cable diameters to 0.388 in., and 0.060 in. for cable diameters to 0.5 in. The contact between the cable and the pulley groove should be equal to approximately one-third the cable circumference.

16-7.2.3 Pulley Strength

The strength of a pulley may be limited in several ways, depending upon the material from which it is made. Consideration must be given to the buckling or splitting strength of the sheave, the checking and shearing strength of the flange, and the strength of the bond between the pulley and its bearing. MIL-P-7034 and the MS standards provide additional information on the strengths of standard pulleys.

16-7.2.4 Pulley Performance

Sufficient wrap angle of the cable on the sheave should be provided to overcome the static friction torque of the bearing. Very small wrap angles should be avoided in order to prevent sliding between the cable and the pulley instead of pulley rotation.

Standard pulleys use ball bearings that are grease-lubricated and sealed. The seals are capable of withstanding temperatures of −55° to 121°C. The bearings are installed in the pulley, and the assembly then is checked for wobble and eccentricity within the limits specified in MIL-P-7034. The design of brackets and the placement of guards should allow clearance for these wobble variations to avoid rubbing or other interference with the pulley mounting bracket.

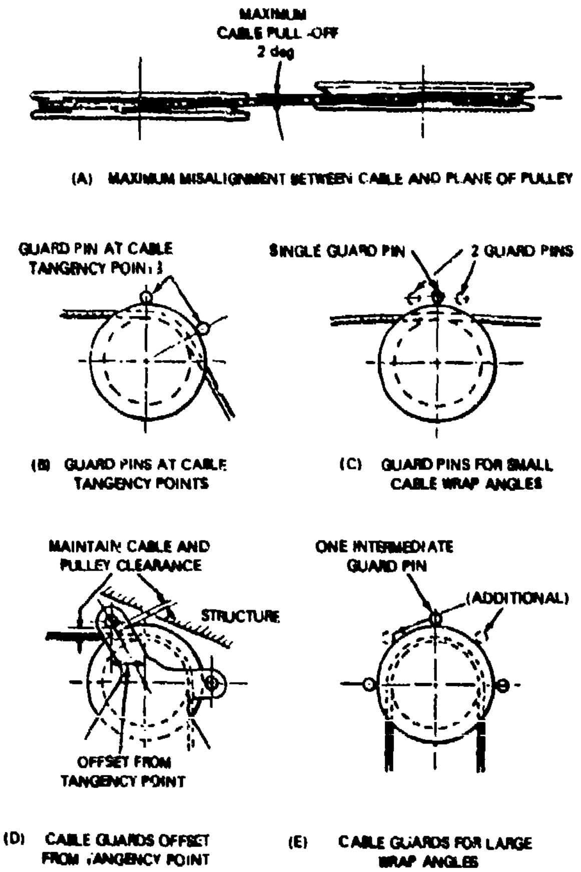
16-7.2.5 Nonmetallic Pulleys

Nonmetallic pulleys (Type I per MIL-P-7034) should be fabricated of a material that meets the nonafterglow requirements of the specification. In addition, this material should meet the requirements for fungous resistance, should be noncorrosive to tin- or zinc-coated carbon steel cables, and should meet the other qualification tests of the specification.

16-7.3 PULLEY INSTALLATION

Pulley installations *shall* be designed to insure that the cable alignment (the angle of direction of change with the plane of the pulley) does not exceed 2 deg, as shown in Fig. 16-10(A). The effect of cable sagging, where long runs are used, should be considered when determining the cable position. If necessary, fairleads and rubstrips should be used near the pulley to limit this misalignment. The design of the pulley mounting bracket should be such that the deflection of the pulley under load does not result in a misalignment angle in excess of the 2-deg maximum. The slack-cable side of a system under load should be considered to insure that sagging, binding, and excessive misalignment do not occur.

The design of the pulley brackets should be such that there is room to thread the cable ends without the removal of the pulley. Spacers that may be required between the pulley, and the mounting boss should be made integral with the bracket wherever possible. To the maximum possible extent, the use of thin shims or washers between the pulley and the sides of the brackets should be avoided. These items are difficult to in all and are lost or omitted easily


MAXIMUM CABLE PULL-OFF 2 deg
GUARD PIN AT CABLE TANGENCY POINTS
SINGLE GUARD PIN
2 GUARD PINS
MAINTAIN CABLE AND PULLEY CLEARANCE
STRUCTURE
OFFSET FROM TANGENCY POINT
ONE INTERMEDIATE GUARD PIN
(ADDITIONAL)

(A) MAXIMUM MISALIGNMENT BETWEEN CABLE AND PLANE OF PULLEY

(B) GUARD PINS AT CABLE TANGENCY POINTS

(C) GUARD PINS FOR SMALL CABLE WRAP ANGLES

(D) CABLE GUARDS OFFSET FROM TANGENCY POINT

(E) CABLE GUARDS FOR LARGE WRAP ANGLES

Figure 16-10. Cable Alignment and Pulley Guard Location

during routine maintenance; their omission could cause the pulley to be misaligned, or induce undesirable stresses on the bracket which would result in premature failure.

Eccentric loading of flat sheets, tension in welds, and the bending of mounting flanges should be avoided. Moreover, adequate back-up structure should be used to provide a rigid support of the bracket base.

For long cable life, pulleys in the same cable run should not be installed closer together than the maximum cable travel. In this way, no portion of the cable will pass over more than one pulley. In any case, the pulley should be arranged and located so that no portion of the cable is made to reverse direction in bending when passing over two or more pulleys.

16-7.4 PULLEY GUARDS

Guards or guard pins should be installed at the approximate point of tangency of the cable to the pulleys (Fig. 16-10(B)). For very small wrap angles, where the use of two guard pins is impractical, one pin may be used (Fig. 16-10(C)). When two or more pins are used, they may be offset from the point of tangency in order to obtain the space necessary for their installation (Fig. 16-10(D)). Where the wrap angle is more than 90 deg, intermediate guard pins should be installed (Fig. 16-10(E)).

To avoid possible binding due to relative deflections, the support for guard pins should come from the same bracket that supports the pulley. The gap between the guard and the pulley should be as small as possible, yet sufficient to allow for the tolerance variations, including wobble and eccentricities. The gap should not be so large as to permit the cable to become wedged between the pulley flange and the guard. The recommended maximum gap is one-half the cable diameter.

Spring guards should not be used in primary flight control systems. If used in secondary control systems, some method of retention in addition to their own spring effect should be considered. If used, spring pins should be installed in accordance with MS 33547.

16-8 PUSH-PULL CONTROLS AND FLEXIBLE SHAFTS

16-8.1 GENERAL

Although push-pull controls and flexible shafts are similar in appearance, they differ considerably in construction and application. Basically, both devices consist of a flexible core that operates inside a casing. The casing supports and acts as a bearing surface for the core. The push-pull control is used to transmit linear motion by tension or compression of the core. The flexible shaft is used to transmit power or rotary motion, usually along a curved path, between two components. These two devices are designed specifically to perform their individual functions, and they should not be interchanged.

16-8.2 PUSH-PULL CONTROLS

The basic components of the push-pull control are the inner core (usually flexible), the outer tubular casing or conduit (rigid or flexible), and the end fittings, as required.

The requirements for this type of control are defined in MIL-C-7958. Push-pull controls are designated by grades. Grade A controls are made of specially selected components that are individually tested in order to insure that backlash and operating forces are reduced to the minimum value. Grade B controls are made to high-quality commercial standards for applications not requiring Grade A controls. Many types of Grade B controls are available, with a variety of characteristics and styles of end fittings to meet special requirements. Applicable data are provided in manufacturers' catalogs.

The advantages of the push-pull type of controls include:

1. Ease of design and installation because they are readily routed around obstructions
2. Lower weight and space requirements than pulleys, bell cranks, and brackets
3. Corrosion-resistance and permanently lubricated construction, resulting in ease of maintenance
4. A variety of end fittings that adapt readily to desired control configurations
5. Ready accommodation of motion between the input and output anchor points
6. Low cost.

However, for a satisfactory design using push-pull controls, the limitations of this type of system must be evaluated and determined to be acceptable. Such consideration should include:

1. Lost motion
2. Increased system friction
3. Disassembly required to inspect the sliding core
4. Temperature limitations, particularly for assemblies using nonmetallic seals
5. The necessity for the outer housing to be anchored rigidly not only at the end fittings but at regular intervals along its length (to control friction and lost motion)
6. Possible weight disadvantage for long control systems with acceptable efficiency.

16-8.2.1 Control Travel

The control motion available (stroke) from a particular push-pull control depends upon its design and the end fittings used. If no end fittings are attached to the movable core, the stroke is dependent simply upon the amount of core that extends past the fixed outer housing. Many styles of end fittings are available in standard configurations that will accommodate control strokes of from 1 to 5 in., and may be custom tailored.

16-8.2.2 Control Loads

The allowable push load that may be imposed upon a push-pull control is dependent upon the maximum stroke of the control. As the stroke increases, the column length of the unsupported end also increases, and a corresponding reduction in compression loading is required. The allowable pull load usually is equal to the maximum rated capacity of the control, and is independent of the stroke.

As the number and sharpness of the bends and the length of the control increase, so will the internal friction. To achieve the desired output load, the input load must increase accordingly to compensate for these friction losses. This factor must be considered when selecting a type of control. The possible increase in load, due to an increase in friction, that may occur during the life of the equipment also should be considered to insure continued satisfactory operation.

16-8.2.3 Core Configurations

The inner core sliding member may have a number of configurations. The simplest form is a single member of high-tensile-strength spring wire. This type of control generally is used for light loads, and in installations having a small number of bends of generous radii (Fig. 16-11(A)).

Cores of more sophisticated construction should be used for controls requiring higher load capabilities, more flexibility, and minimum backlash. These may be single or multiple strands of wire wrapped with outer armor to provide strong but flexible members (Fig. 16-11(B)). Others may consist of components that include an action element, such as a thin, flexible member supported by balls that roll or slide in raceways inside the conduit (Fig. 16-11(C)). These units may be custom-assembled for minimum friction losses and backlash, and high load-carrying capabilities.

16-8.2.4 Conduit

The outer tubular casing or conduit of a push-pull control may be either flexible or rigid and generally is clamped to the basic structure at frequent intervals. It must be anchored firmly at the ends in order to achieve the desired control motion. Complete assemblies may consist of sections of both rigid and flexible conduit.

The flexible outer casing usually is a built-up member. Typical flexible construction contains an inner liner of hard steel or plastic to provide a good bearing surface for the core. Around this liner is a structure of outer windings to provide longitudinal compressive and tensile strength, and to maintain the desired flexibility of the control. This outer member is sealed in order to prevent moisture and other foreign matter from entering. This seal may be in the form of a plastic or rubber outer jacket, or of packing between the windings.

Rigid conduit may be built up in a manner similar to the flexible conduit, except that it uses rigid-metal tubing as the structural member. This tubing is bent

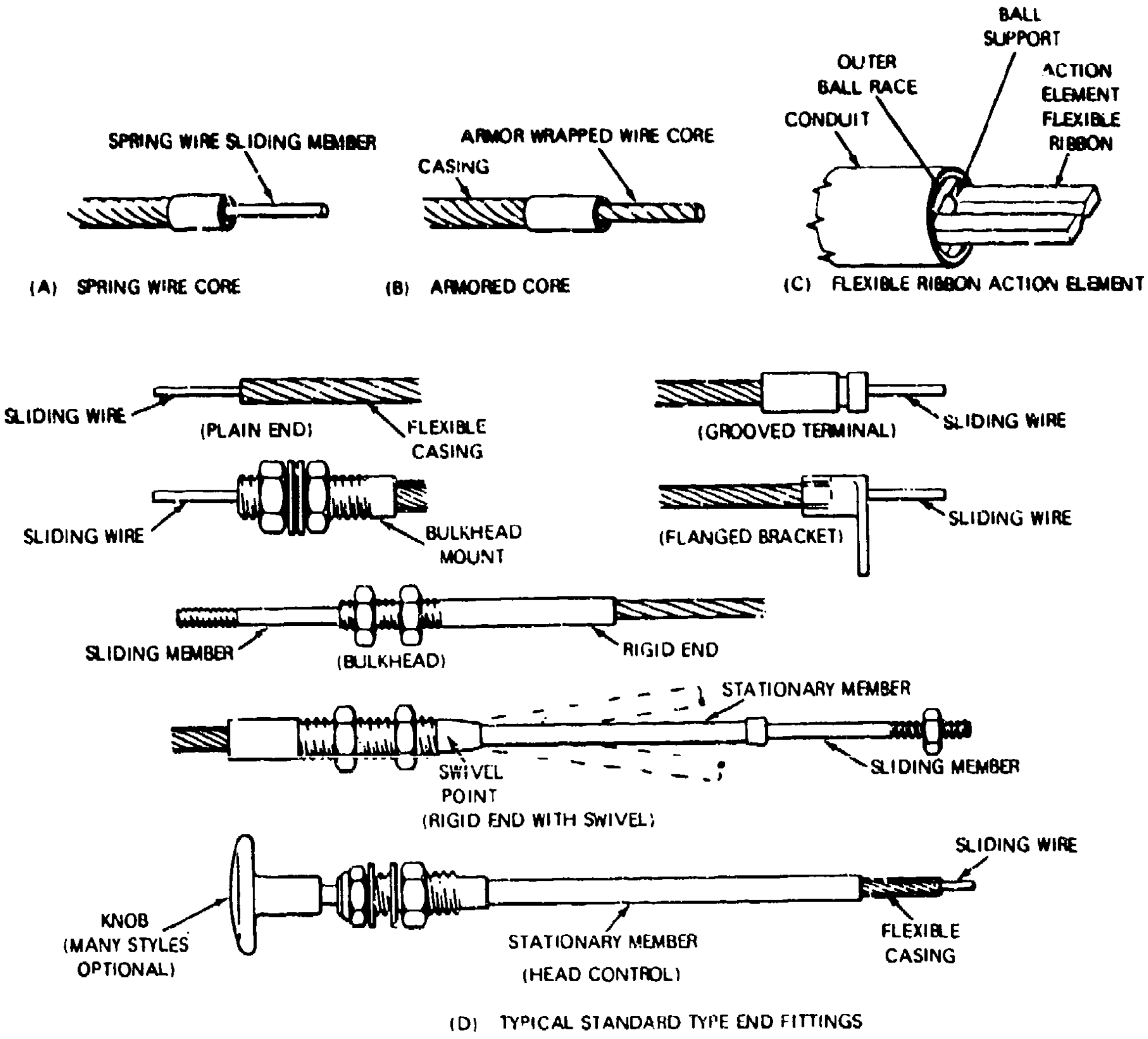

SPRING WIRE SLIDING MEMBER
ARMOR WRAPPED WIRE CORE
CASING
BALL SUPPORT
OUTER BALL RACE
CONDUIT
ACTION ELEMENT FLEXIBLE RIBBON
(A) SPRING WIRE CORE
(B) ARMORED CORE
(C) FLEXIBLE RIBBON ACTION ELEMENT
SLIDING WIRE
(PLAIN END)
FLEXIBLE CASING
(GROOVED TERMINAL)
SLIDING WIRE
SLIDING WIRE
BULKHEAD MOUNT
(FLANGED BRACKET)
SLIDING WIRE
SLIDING MEMBER
(BULKHEAD)
RIGID END
STATIONARY MEMBER
SWIVEL POINT
SLIDING MEMBER
(RIGID END WITH SWIVEL)
SLIDING WIRE
KNOB (MANY STYLES OPTIONAL)
STATIONARY MEMBER
(HEAD CONTROL)
FLEXIBLE CASING
(D) TYPICAL STANDARD TYPE END FITTINGS

Figure 16-11. Push-pull Cables and End Fittings

carefully to the desired shape and fitted to the installation.

16-8.2.5 End Fittings

End fittings for push-pull controls are available in a multitude of styles and sizes. Many configurations have a self-aligning capability, either by flexing the outer casing or by using a slider mounted in a swivel. A seal is incorporated into the end fitting to retain lubricants and to exclude foreign matter. Fig. 16-11(D) illustrates some typical types of ends.

16-8.3 FLEXIBLE SHAFTS

There are two basic types of flexible shafts. One is used to transmit power, and the other is used for control of equipment. The construction of these two types is quite different, and they should not be interchanged.

Power-drive shafts are constructed to transmit the maximum feasible torque. They generally are constructed to rotate only in one direction, and in small sizes that can be driven at continuous speeds of up to 20,000 rpm. The casing generally does not fit closely on the shaft, and therefore the unit can be disassembled for lubrication and inspection.

Remote-control shafts can be rotated in either direction. They are built to provide maximum accuracy and generally are operated at low speeds.

Since they normally do not require periodic lubrication or inspection, the contruction does not allow ready disassembly.

The main elements of a flexible shaft are casing, end fittings, and shaft end fittings.

16-8.3.1 Torque Capacity

The torque capacity of a flexible shaft is reduced as the minimum bend radius of the shaft is reduced. The flexible shaft undergoes the most severe type of flexural stress when it operates while it is bent — two phase reversals occur during each revolution. As the bend radius decreases, the cable strands also undergo more rubbing against each other. The minimum bend radius for a given shaft is dependent upon shaft size, number of layers, types of material, desired life, and the required torque output.

16-8.3.2 Flexible Power Shafts

Flexible power shafts are designed to rotate in one direction only. The torque capacity of a power-drive core in the unwind direction may be only 50% of that in the windup direction.

Flexible shaft cores of the same diameter may have significantly different characteristics. The core usually is made up of a single straight wire wrapped with additional layers of wire, each of which is wound on the preceding layer. Successive layers alternate in pitch direction. Cores may vary according to the number of wires per layer, the number of layers of wire, the diameter of wires, the wire material, the spacing between the wires, and the type of construction. These variables, in turn, affect such core characteristics as torque capacity, transverse stiffness, minimum radius of curvature, efficiency, deflection factor, core life, and cost.

When selecting a power-drive shaft, the following factors must be considered:

1. Maximum torque that the shaft must transmit
2. Operating speed ranges:
   a. Normal, 1750 to 3600 rpm
   b. Special, to 10,000 rpm
   c. Special small size, to 20,000 rpm (gearing or other means should be used to increase the shaft speed and reduce the torque)
3. Maximum torque capacity of a possible shaft configuration and effect of operating radius
4. Direction of rotation, preferably such that the outer layer of wires tends to tighten
5. Normal design for 100 million cycles at rated speed and torque capacity
6. Standard rating conditions, and environmental conditions such as elevated temperature and corrosive atmosphere for special material considerations
7. Required safety factor, possibility of shock loading, and starting overloads.

Some of the advantages of flexible power shafts over other types of torque-transmitting devices include:

1. Power to equipment can be transmitted at odd angles relative to the drive shaft.
2. Installation misalignments are accommodated easily, allowing more flexibility in the location of equipment.
3. Driven equipment need not remain in the same relative position to the driver during operation.
4. Torsional fluctuations can be absorbed.
5. Cost is relatively low.
6. Rotating elements are enclosed, thereby eliminating a safety hazard.

16-8.3.3 Flexible Control Shafts

Flexible control shafts are designed to minimize the overall deflection and thereby provide the required accuracy. They usually can be rotated in either direction at speeds of less than 100 rpm. Intermittent operation to 3000 rpm, with a low (less than 5 min) duty cycle followed by adequate rest periods, may be accommodated.

These control assemblies usually are designed with a casing that closely fits the core. The end fittings may be attached permanently. They usually do not require lubrication. If required, cores can be designed so that there is nearly equal deflection in either direction of rotation.

When selecting a flexible control, the following factors should be considered:

1. Maximum torque to be transmitted
2. Permissible deflection, deg
3. Shaft length, in.
4. Radius of smallest bend, in.
5. End connections for both shaft and casing
6. Size of core diameter.

Requirements for remote-control flexible shafts with a steel core and casing are presented in MIL-S-3857. Units built to conform to this specification are intended for use in either clockwise or counterclockwise directions. The load capacity and deflection characteristics differ, however, depending upon whether the operation is in the winding or unwinding direction for the outer layer of the core. The requirements of this specification should be reviewed for applicability to the requirement at hand. For installations and configurations requiring special characteristics, manufacturers' data should be consulted.

16-9 CABLES AND WIRES (STRUCTURAL)

16-9.1 GENERAL

Cables may be used as structural members in special applications. They are of light weight, and they may be made flexible for ease of stowing and handling. Their use as tension members — for operating controls, slings, and hoists, and as part of machinery — has resulted in a number of types of construction. Data concerning the design and use of electrical cables and wires can be found in AMCP 706-125.

16-9.2 PREFORMED WIRE STRAND AND CABLE

When a wire is preformed, it is helically formed into the shape that it will assume in the finished cable. This relieves the internal stresses of the wire and increases the useful life of cable that is subjected to repeated bendings. The total stress, which is the sum of the internal stresses and the bending stresses, is reduced by the amount of the internal stresses. Other advantages of preformed wire cables are:

1. It can be cut without seizing.
2. It is easier to handle, has less tendency to loop or kink, and is more tractable.
3. It can be used with swaged terminals.
4. It has little or no tendency to rotate, and will run true over pulleys, helping to reduce wear of the pulleys.

16-9.3 TYPES OF CABLE CONSTRUCTION

Cable is made by stranding many fine wires, which can be of various materials. However, for aircraft applications, corrosion-resistant material, as specified in MIL-W-5424 or MIL-W-5693, should be used. These specifications cover wire strand and cables of many types, as described in Table 16-12. Coated cables per MIL-W-83343 also may be used.

The most popular type of aircraft cable is the seven-strand construction. Each strand consists of a number of individual wires. It has been determined empirically that this type of construction, with six outer strands, supplies sufficient roundness to contact sheave grooves with enough outer surface to afford all of the bearing surface required, and provides enough contact points to prevent abrasion from being concentrated on too few exposed surfaces. MIL-W-5424 specifies other properties of the different types of cable construction.

16-9.4 CABLE SELECTION

When selecting the cable for a specific application, all pertinent factors must be considered. The requirements must be matched carefully with the particular cable properties. A typical comparison that might be made would balance the following:

1. Cable strength and maximum load
2. Cable stretch and allowable deflection
3. Operating characteristics and system friction (if applicable)
4. Wire material and environment
5. Cable construction, life, and abrasion.

TABLE 16-12. MILITARY SPECIFICATIONS FOR CABLES

SPECIFICATION	TYPE
MIL-W-5693	TYPE I 1x7 – WITH WIRE CENTER------------NONFLEXIBLE
MIL-W-5693	TYPE II 1x19–WITH WIRE CENTER------------NONFLEXIBLE
MIL-W-5424	3x7 – WITHOUT A CORE---------------FLEXIBLE
MIL-W-5424	7x7 – SIX OUTER STRANDS OF SEVEN WIRES AROUND A CORE STRAND OF SEVEN WIRES--------------FLEXIBLE 7x19–SIX OUTER STRANDS OF 19 WIRES AROUND A CORE STRAND OF 19 WIRES----------------------FLEXIBLE
MIL-W-5424	6x19–(IWRC) SIX OUTER STRANDS OF 19 WIRES AROUND A 7x7 INDEPENDENT WIRE ROPE CORE--------------FLEXIBLE

16-9.4.1 Cable Strength

The breaking strength of a cable is determined by the net metallic cross-sectional area and its material properties. The net metallic cross-sectional area equals the sum of the cross-sectional areas of all of the individual wires, and, therefore, varies, for a given nominal cable.

16-9.4.2 Cable Deflection

The total cable deflection may result from either constructional stretch or elastic stretch. The maximum stretch allowed by MIL-W-5424 is 2% when the cable is loaded to 60% of its breaking strength.

Constructional stretch varies from cable to cable, and results from the small spaces present between the wires and the strands, and between the strands and the core after fabrication. Cables used in control systems, and where large initial deflections cannot be tolerated, are proof-loaded to remove this stretch. The cable should be loaded to a minimum of 60% of its breaking strength in accordance with MIL-C-5688. This procedure also is used for proof-testing cable assemblies.

Elastic stretch results from the elongation of the individual wires as the load is applied. As the load is released, the wires return to their original length, provided the elastic limit of the material has not been exceeded. The elastic stretch is determined from the product of the load times the cable length divided by the product of the metallic cross-sectional area and the modulus of elasticity.

16-9.4.3 Operating Characteristics

The operating characteristics of a cable are dependent upon the type of construction and the installation. A friction-preventive compound may be applied during fabrication to reduce the friction of the cable when it is bent. The number of bends, the radius around which the cable is bent, and the pulley configuration in which the cable operates all affect the operating characteristics.

16-9.4.4 Wire Material

The composition of wire used in the fabrication of corrosion-resistant steel wire rope is given in MIL-W-5424 and MIL-W-5693. The physical properties are determined by the manufacturer in order to meet the requirements of the Military Specification. Many other types of materials are available, but their use should be considered only with the approval of the procuring activity. Other types of cable include galvanized carbon steel (MIL-W-6940 and MIL-W-1511) and nonmagnetic corrosion-resisting cables (MIL-C-18375).

16-9.4.5 Cable Construction

The construction of the cable can take many forms to obtain the results desired, as discussed in par. 16-8-3. The type of construction used should be matched with the operating requirements for the cable. A system might be designed using segments of cables with different construction. Aircraft cable measuring 7 by 19 (7 strands of 19 wires each) is preferred for aircraft controls because of its high strength, good flexibility, and bending fatigue resistance — which allows it to be operated over relatively small sheaves.

The 7 by 7 (7 strands of 7 wires) aircraft cable is not as flexible as the 7 by 19. Since each strand is made up of only seven wires, each wire is larger in diameter. Therefore this type of construction has more ability to withstand abrasion than does 7 by 19.

The 1 by 19 aircraft strand is considered nonflexible. It should be used only as a straight run section because its minimum stretch results in increased rigidity in control systems. It also is used for bracing, etc., where its compact structure, high strength, and minimum stretch all provide added benefits.

Other aircraft and commercial cable types are available. Some are designed for high energy absorption — as in launching and arresting systems, towing, etc. Other configurations include plastic-coated cable, particularly desirable for use as a handhold or guard rail; and as complex assemblies used for communications, where electrical conductors with insulation are provided inside the load-carrying outer members.

16-9.5 SAFETY WIRE AND COTTER PINS

Pins always should be made safe. In main structural members, safety can be achieved by drilling a hole and using a cotter pin or safety wire. As an additional safety measure, bolts and pins should be inserted with their large end, or head, uppermost in order to reduce the possibility of their falling out should they not be properly safetied.

In general, safety wire should be used only where self-locking fasteners or cotter pins are not adequate to withstand the expected vibration or stress. Safety wire should be attached so that it can be removed in accordance with MS 20995. Inconel (uncoated) and Monel (uncoated) wire should be used for all general lock-wiring purposes. Copper wire that has been cadmium-plated and dyed yellow should be used for shear and seal wiring in order to allow operation or actuation of emergency devices. Aluminum alloy

ALCLAD 5056, anodized and dyed blue in accordance with FED-STD-595, should be used exclusively for safety wiring of magnesium parts. All of these materials can be identified visually by their color.

The wire should measure 0.032 in. in diameter as a minimum for general purposes. The double-twist method of safety-wiring should be considered as standard. The single-wire method may be used in a closely spaced, closed geometrical pattern, on parts in electrical systems, and in other applications that make the single-wire method more suitable.

Parts *shall* be safety-wired in such a manner that the lock wire is put in tension when the part loosens. The lock wire always should be twisted so that the loop around the head stays down and does not come up over the bolt head, leaving a slack loop. A pigtail of 0.25 to 0.5 in., or about 3 to 6 twists, should be left at the end of the wiring. The pigtail should be bent backward or under to prevent snagging. Fig. 16-12 illustrates various lock wire applications. The use of safety wire and cotter pins *shall* be in accordance with MS 33540.

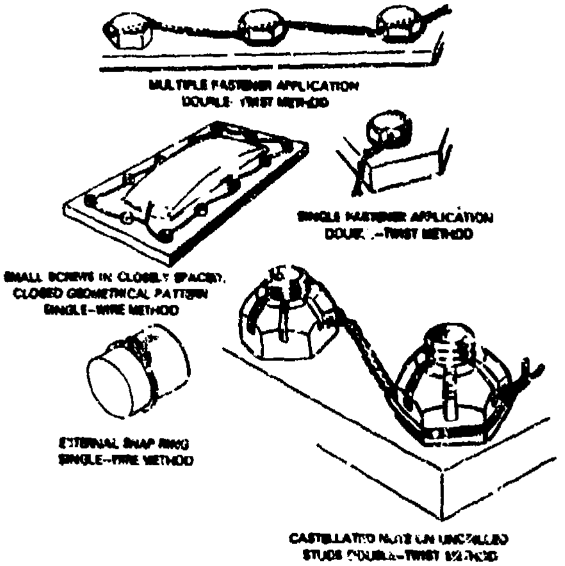

MULTIPLE FASTENER APPLICATION
DOUBLE-TWIST METHOD
SINGLE FASTENER APPLICATION
DOUBLE-TWIST METHOD
SMALL SCREWS IN CLOSELY SPACED, CLOSED GEOMETRICAL PATTERN
SINGLE-WIRE METHOD
EXTERNAL SNAP RING
SINGLE-WIRE METHOD
CASTELLATED NUTS ON UNDRILLED STUDS DOUBLE-TWIST METHOD

Figure 16-12. Right-handed Thread Application of Safety Wire

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CHAPTER 17
PROCESSES

17-1 INTRODUCTION

This chapter discusses the processes used in the manufacturing and assembly phases of helicopter construction. It includes the basic metalworking processes — casting, forging, extrusion, and sheet-metal forming. Machine shop practices and basic types of machine tools also are discussed. The methods of joining — such as welding, brazing, and soldering — are discussed, as are the processes of mechanical fastening, bonding, swaging, and cable splicing. Various types of heat treatment are described, including stress relieving, conventional quenching and tempering, aging, and surface hardening. Work hardening techniques, such as shot-peening and burnishing, are addressed, together with the types of materials and parts whose fatigue-resistance improves as a result of processing. The tooling requirements needed to produce a given type of helicopter structure or part are reviewed.

Vendors should be used as additional information sources because of the continuing development of new materials and new processes that alter the cost and performance relationships between alternative manufacturing processes.

17-2 METALWORKING

17-2.1 GENERAL

This paragraph discusses metal-forming processes and their applications to the construction of helicopters, as well as some of the parameters governing the choice of one process among many for the production of a particular part. More comprehensive discussions, as well as detailed design data, are found in other documents such as AMCP 706-100, MIL-HDBK-5, -693, -694, -697, -698, and -723. Also, Ref. 1 is an important source for design data and metallurgical details.

The primary metal fabrication processes are casting, forging, extrusion, and sheet-metal forming. The choice of the appropriate process depends upon the size and complexity of the part, the nature and magnitude of the stresses to which it will be subjected, the material from which it will be made, and the relative costs of fabrication. In general, large, fairly complex parts — which require high rigidity, are not subject to excessive stress or impact, and for which somewhat greater weight can be tolerated — can be produced more economically by casting. A part that will be subject to high stress and in which toughness also is required — and where high strength, lighter weight, and better finish also are necessary — might be formed better by forging. Extrusion forming may be preferable where high-strength, close-tolerance parts having a continuous contour — e.g., rails, tubing, and beams -- are required. Large-area, thin-wall, deep-draw configurations more often will be formed from sheet metal. In most cases, some machining, joining, and finishing will be involved. Only a detailed analysis, with a careful evaluation of the several process and material costs, will provide a sound basis for selecting the metal-forming process for a particular part.

Forgings, castings, extrusions, and sheet-metal parts will constitute the major portion of any helicopter. The parts produced by these processes range from the smallest and most precise instrumentation to the largest castings for gearbox housings.

The materials employed in these processes include iron, steels, high-performance alloys, copper, magnesium, aluminum, and titanium.

17-2.2 CASTING

Metal castings are formed by pouring molten metal into a prepared cavity and allowing it to solidify. The casting processes most commonly used in the manufacture of helicopter components are sand, investment, permanent mold, and centrifugal. Casting is selected over alternative methods primarily on the basis of cost. The strength of castings generally is lower than that of wrought alloys. However, the structural properties of castings are the same in all directions, and, therefore, in case of symmetrical loadings a casting may be the most efficient design. Castings should not be employed when the predominant loadings are not steady, i.e., when the loads either are alternating or involve impact, because castings do not have the toughness of wrought alloys.

The possibility of the inclusion of sand or other impurities, blow holes, or other invisible flaws results in the need for careful quality control of those castings used in critical applications. MIL-C-6021 provides the standards for classification of castings on the basis of the hazard following their failure, and also specifies applicable quality control requirements. Allowable strength of castings is discussed in Chapter 4, AMCP 706-201, and test and demonstration requirements are given in Chapter 9, AMCP 706-203.

17-2.2.1 Sand Castings

Sand casting consists of forming a mold from sand compressed around a suitable pattern of the part to be made, removing the pattern, and pouring in molten metal to replace the pattern. The advantages of this process are: almost any metal may be used, there is almost no limitation to the size or shape of the part, and the process is relatively low in cost. In addition, extreme complexity is possible, tool costs are low, and the process provides the most direct route from pattern to mold. There are also disadvantages:

1. Sand castings have rough surfaces.
2. Close tolerances are quite difficult to achieve.
3. Some alloys develop defects.
4. Long, thin projections are not practicable.
5. Some machining usually is necessary.

17-2.2.2 Investment Castings

Investment casting uses patterns of wax made in a split mold. These patterns can be combined into complex assemblies. The assembly then is "invested" by coating, first with a fine slurry of refractory powder and binder material and then with progressively coarser layers of sand. When the resulting mold is baked, the wax is melted out (lost), leaving a smooth-walled cavity into which the metal is cast.

This process has the advantages of high dimensional accuracy, excellent surface finish, and nearly unlimited intricacy. Moreover, almost any metal may be used. However, the size of the part that can be formed is limited, the labor cost is high, and expensive patterns and molds are required.

17-2.2.3 Permanent Mold Castings

In the sand and investment casting processes, the mold is destroyed during the removal of each casting. In permanent mold casting, the molds are made of metal, usually cast iron, die steels, graphite, copper, or aluminum. The permanent mold is machined for dimensional tolerance and draft angles. Vent plugs are inserted into the cavity to allow gases to escape when the molten metal is poured. The process is readily automated, with many molds on a turntable using a fixed pouring, cooling, and ejection cycle.

The process has the advantages of good surface finish and grain structure, high dimensional accuracy, rapid production rate, low scrap rate, and low porosity. Disadvantages include high initial mold cost; limitations on the shape, size and intricacy of castings; and process limitations for metals with low melting points.

17-2.2.4 Centrifugal Castings

In sand and investment casting, the mold is filled with metal simply by the force of gravity. In centrifugal and centrifuge casting, a centrifugal force 60-75 times the force of gravity exerts pressure on the metal by spinning the mold while the molten metal is charged to it. The mold even may be rotated about two axes. Centrifugal force helps fill the mold completely. Gases and impurities are concentrated near the center of rotation, and gates and risers are kept to a minimum.

This method is particularly adaptable to small, intricate castings that otherwise would be difficult to gate. A good surface finish is obtained. However, tooling costs for centrifugal casting are high and the castings must be symmetrical. Alloys of separable metals may not be distributed evenly.

17-2.3 FORGING

The forging of a metal part involves heating a metal blank to a plastic state and hammering it into shape. In this process, great strength is imparted because of the beneficial grain flow that takes place as a result of the kneading action on the metal. The grain flow is changed to follow the contour of the part, resulting in a tough, fibrous structure. Usually, the shaping takes place between closed dies that determine the contour of the forged part. Forgings have high strength, high impact strength, and great resistance to fatigue, and are used for applications in shafts, axles, springs, gears, rotor hubs, and similar moving parts. In the design of forgings, it is necessary to consider the lower level of strength normal to the grain flow. This consideration may affect the orientation of the grain flow as well as the cross section of the part. The cost of forgings is substantially higher than that of castings.

17-2.4 EXTRUSION

Extruding is a process in which a billet or slug of metal is pressed by a ram until the pressure inside the work piece reaches the flow state of the material. The material then is squeezed through a die that contains an orifice of the desired shape. Because of the high reduction ratio, the metal has excellent transverse flow lines. This provides greater strength in the longitudinal direction and lower strength in the transverse direction. The nonferrous alloys of aluminum, magnesium, and copper are used most commonly for extrusions, but some steel alloys are extrudable.

Very complex shapes are possible at a cost much less than that of machining. Extruded shapes often can replace weldments and parts previously machined from bar stock. The cost of extrusion dies is relatively low, making short runs practicable. Extruded tracks and rails, beams, bars and tubes, and stringers find ready application in helicopter structures.

17-2.5 SHEET-METAL FORMING

In addition to large surface areas, such as fuselage shells and skins, a multitude of smaller parts are made by sheet-metal forming. These include straps, brackets, clamps, cups, pressure housings, headers, inserts, grommets, stringers, and conduits. Sheet thickness normally is 0.013 to 0.141 in., but thicker blanks may be used for various draw-forming operations. Sheet materials from which metal parts for helicopters are formed include aluminum, titanium, and stainless steel.

17-2.5.1 Machine Forming

Many sheet-metal parts are made by machine-forming operations that start with sheet-metal blanks. The operations take place within dies, which are mounted on various machines designed to supply the forces to shape the materials in the dies. There are dies for cutting, bending, squeezing, and drawing. The material may be subjected to a combination of several, or all, of these operations in a single die or in a succession of dies. Certain operations are named for the manner in which the force is applied.

By far the largest amount of cutting, bending, squeezing, and shallow drawing is done with the drop hammer. In this machine, a falling or powered weight produces the force required to perform the necessary function.

In hydroforming, hydraulic pressure is furnished by a fluid to one side of a rubber diaphragm. This diaphragm presses against the metal blank and forms it around a punch that moves up into the forming mold cavity. The pressure causes the metal to flow evenly around the punch with little thinning or slipping. Hydroforming usually can achieve in one or two operations the same results that would require four or five draws in a normal press method. It usually is employed with higher strength alloys.

In stretch-forming, the sheet metal is pulled over a form block. The material is stretched beyond its elastic limit, causing it to take a permanent set in the desired contour. There is no spring-back, but allowance must be made for dimensional changes in the metal. Large parts with compound curvatures — such as the external surfaces of fuselages, cowlings, and fairings — may be made in this manner.

Spinning is used when cold forming circular, symmetrical sheet metal parts such as pans, covers, shields, and bullet nose shapes. In spinning, a flat circular blank is clamped on a die or chuck in a lathe type machine. The blank is revolved or spun and the metal is formed over the chuck, using hand held forming tools. Parts with return flanges may be formed by using hand held forming tools, or by using collapsible or take-apart spinning chucks. Parts can be trimmed to size on the machine, using conventional lathe cut-off tools. Wooden chucks can be made economically for prototype and short run parts whereas aluminum and steel chucks are suitable for high production.

In explosive, or high-energy, forming, shock waves are generated by explosives such as dynamite, exploding gases or an electrical discharge. The shock waves are transmitted through a liquid medium, usually water, to the work piece, fanning out in all directions and forcing the metal into a preformed die cavity. Materials of very high strength can be formed in this manner. The same materials formed by other methods would have excessive spring-back, but, due to the high energy rate and the uniformity of distribution, very little spring-back occurs after explosive forming.

17-2.5.2 Shop Fabrication

Although the machine-forming processes previously discussed are used in the fabrication of a large number of parts, the majority of the sheet-metal work involved in the manufacture of helicopters is performed in a sheet-metal shop. Here, sheet metal is cut into various flatwork patterns; punched, drilled, folded, seamed, crimped, beaded, grooved, turned, rolled, and burred; and then joined by clinching, soldering, brazing, welding, riveting, or adhesive bonding.

Many design and fabrication techniques are available to add strength and rigidity to simple sheet-metal parts. For example, strength can be incorporated into the structure by means of flanges, ribs, corrugations, beads, etc. These and similar metal-working operations can be performed either hot or cold. Hot-working may involve little or no strain hardening, whereas in cold-working, considerable dislocation and strain hardening can occur. Work hardening can produce beneficial effects, such as increases in tensile and yield strength, but accompanying decreases in ductility and toughness also may result.

Regardless of whether the sheet metal is machine-formed or worked in the shop with press brakes or by hand, the metal will be subjected to bending in many ways. The ease with which a metal can be bent is dependent on many factors. The most important are the nature of the metal itself, its hardness, temper, and prior working, and the orientation of the bend relative to the direction in which the sheet was rolled. In Table 17-1 data are presented to illustrate the wide variety of characteristics available from working sheet metals. Minimum radii are listed for representative metals. In Fig. 17-1 common flange and curling operations, and their recommended bend radii are illustrated.

17-3 MACHINING

17-3.1 GENERAL

This paragraph discusses machining practices and the relationship between the designer and the machine shop in the construction of helicopter parts. More comprehensive discussion, as well as detail design data, will be found in Chapter 10, AMCP 706-100, and MIL-HDBK-5, -693, -694, -697, -698, and -723.

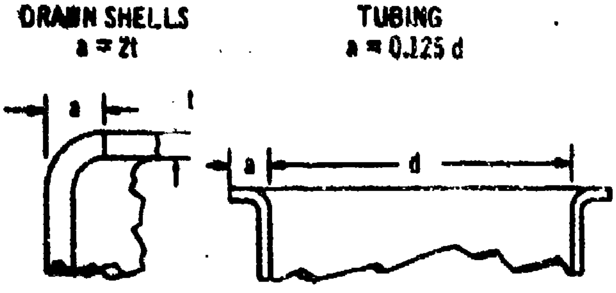
As distinguished from the forming operations discussed in the preceding paragraph, machining involves the removal of material from the work piece. Thus, proper design, dimensioning, and sequencing of operations are important because an error, once made, is not easily corrected. Proper design of the part is essential for a quality machining operation, and the design must be complete in every detail before any material enters the machine shop.

TABLE 17-1. BEND CHARACTERISTICS OF SELECTED METALS

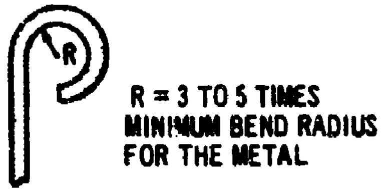
STEEL:

TEMPER AND CONDITION	BEND CHARACTERISTICS	
No. 1–HARD–R_B 84 Min	BEND ONLY ON LARGE RADII	
No. 2–1/2 HARD–R_B 70 to 85	r = t PERPENDICULAR TO ROLL	
No. 3–1/4 HARD–R_B	90 deg PARALLEL TO ROLL 180 deg PERPENDICULAR TO ROLL	
No. 4– R_B 65 No. 5– R_B 55	180 deg FLAT ON ITSELF IN ANY DIRECTION	
CARBON % 0.015 OR LESS	180 deg FLAT ANY DIRECTION	
CARBON % 0.150 TO 0.25	180 deg r = t	
THICKNESS t, in.	MINIMUM BEND RADIUS r FOR STEEL OF MINIMUM YIELD STRENGTH psi	
	45,000	50,000
TO 1/16	t/2	1t
1/16 to 1/4	1t	2t
1/4 to 1/2	2t	3t

ALUMINUM ALLOY	TEMPER	MINIMUM BEND RADIUS IN 1/32 in. FOR THICKNESS t, in. 0.016	0.025	0.032	0.040	0.050	0.063	0.090	0.125	0.250
3004, 5151	0	0	0	0	0	0	0	0	2	8
ALCLAD 3004	H32	0	0	0	1	1	2	3	4	18
5254 AND	H34	1	1	1	2	2	3	5	6	24
5254 AT TEMP.	H36	1	1	1	2	3	4	6	9	24
	H38	1	1	2	3	4	6	9	16	40



(A) FLANGES



(B) CURLING

Figure 17-1. Standard Bend Radii Practice—Minimum Bend Radii

17-3.2 MACHINING OPERATIONS

The machines referred to in this discussion are cutting tool types. Other types of machines, which are used to a lesser degree, include electrical discharge, electrochemical, chemical, ultrasonic, abrasive jet, and plasma arc. All can be programmed from numerical control by tape or by computer so that only nominal supervision is required. This results in substantial cost reductions for quantity and quality work.

Among the operations performed with cutting tool machines are turning, trepanning, planing, shaping, broaching, drilling and thread milling, countersinking, counterboring, and die threading. A tool that cuts with one point or edge is referred to as a single-point tool. Drills, reamers, and milling cutters have more than one cutting edge.

During machining operations, metal removal takes place through three distinct types of cutting action, depending upon the type of material being cut. Medium-hard materials, with a low coefficient of friction, generate a continuous chip that tends to foul the work piece, while soft and ductile materials with a high coefficient of friction give a continuous chip with a built-up edge. This action contributes to short tool life, requiring optimum tool geometry, proper cutting speed, and proper cutting fluid to remove heat and reduce friction. Brittle materials are removed by a combination of shear and fracture, causing chips to come off in segments. This makes the attainment of a fine finish difficult.

Important factors in the overall cost of machining are the tool material, the tool geometry, and the cutting fluid employed. A secondary factor is the machinability of the metal being machined. Methods of measuring the machinability of a given material are based upon cutting ratio, shear angle, horsepower, surface finish, tool temperature, and tool life. A major parameter is horsepower. Table 17-2 lists the relative values of horsepower required for turning, drilling, and milling representative metals. Horsepower, surface finish, and tool life may be used as a measure of machinability by comparing the results obtained for each with the results obtained under the same conditions with SAE B1112 steel, rated 100% machinable.

The three major machine shop operations are turning, milling, and drilling. All other operations are derived from these three. In turning operations, the work piece is turned against a cutting tool. In milling operations, the work piece remains at a fixed height and is moved back and forth under an arbor or spindle holding the multibladed cutting tool. The tool rotates on an axis parallel to the plane of the work table, and feed to the work piece is made by vertical adjustment of the tool. Drilling operations are conducted by means of a drill being turned into the work piece.

17-3.3 ELEMENTS OF MACHINING DESIGN

The smoothness of the surface produced by machine-cutting or metal-finishing operations is expressed in RMS microinches. Measurement standards for these surface textures are given by ANSI B46.1-1962. Representative values for several cutting operations are given in Table 17-3.

The tolerances and limits applied to a dimension on the drawing of a part will determine the kind of machining process that should be used. Thus, for a dimension requiring a tolerance of 0.002 in., it is appropriate to use a grinder rather than a lathe or milling machine since the time spent trying to hold this tolerance on a lathe would prove expensive. On the other hand, a part that carries a tolerance of 0.015 in. probably is more appropriately machined on a lathe or milling machine rather than precision ground.

The established classes of tolerance include running or sliding clearance (RC), location clearance (LC), location transition (LT), location interference (LN), and force or shrink (FN). In RC, one part moves inside the other. LC, LT, and LN are used mainly for the assembly of stationary parts. FN yields constant bore pressure. The relationship

TABLE 17-2. UNIT HORSEPOWER VALUES FOR REPRESENTATIVE METALS

MATERIAL	HARDNESS	UNIT HORSEPOWER/IN.^{3}/MIN FOR:		
		TURNING	DRILLING	MILLING
MAGNESIUM ALLOYS	40–90 B_r	0.2	0.2	0.2
ALUMINUM ALLOYS	30–150 B_r	0.3	0.2	0.4
COPPER ALLOYS	20–80 R_B 80–100 R_B	0.8 1.2	0.6 1.0	0.8 1.2
TITANIUM	250–375 B_r	1.0	1.0	1.2
PH STEELS	170–450 B_r	1.5	1.4	1.7
CARBON STEELS	35–40 R_c	1.6	1.4	1.5
ALLOY STEELS	40–50 R_c	2.0	1.7	2.0
TOOL STEELS	50–55 R_c	2.2	2.0	2.2
TUNGSTEN	321 B_r	3.5	3.3	3.6

TABLE 17-3. REPRESENTATIVE SURFACE FINISHES OBTAINED IN MACHINING OPERATIONS

OPERATION	RMS FINISH, micro in.
BURNISHING	2–4
LAPPING	2–8
HONING	2–10
POLISHING	2–10
REAMING	8–50
GRINDING	5–150
BROACHING	15–60
DRILLING	75–200
MILLING	20–300
TURNING	20–300
SHAPING	20–300
SAWING	250–1000

between the limits and allowances for holes and shafts for a Class 2 fit are illustrated in Table 17-4.

Among the more frequently encountered elements of design are cams and gears, keyways, splines, and serrations. Each gear type performs a specific role in power transmission. Gears may be cut on a milling machine or by hobbing, but more frequently are produced by a shaper-cutter working on a preformed, forged gear blank. The types of gears employed are discussed in detail in Chapter 4.

Splines, serrations, and keys are devices for attaching items such as gears, cams, pulleys, and torque bars to the power shaft in such a manner that there will be no interfacial slip caused by the torque imposed. Splines and serrations often are cut with a spline roller, keyways with a mill or shaper. There are many kinds of keys, such as feathered, gibhead, plain flat, plain square, round, saddle, tangential, and Woodruff.

17-4 JOINING

17-4.1 GENERAL

Joining operations include welding; brazing; soldering; mechanical fastening, including rivets, bolts, nuts, washers and screws; adhesive bonding; swaging; and cable splicing.

TABLE 17-4. VALUES TO BE ADDED TO OR SUBTRACTED FROM BASE DIMENSION FOR HOLES AND SHAFTS TO CALCULATE TOLERANCE

	CLASS AND FIT (In thousandths of an inch)				
	RC	LC	LT	LN	FN
HOLE	+0.9 -0.0	+1.4 -0.0	+2.2 -0.0	+1.4 -0.0	+1.4 -0.0
SHAFT	-0.5 -1.1	0.0 -0.9	+0.8 -0.6	+2.5 +1.6	+3.9 +3.0
ALLOWANCE	+0.5 +2.0	+0.0 +2.3	-0.8 +2.8	-0.2 -2.5	-1.6 -3.9

17-4.2 WELDING, BRAZING, AND SOLDERING

17-4.2.1 Welding

Welding is the process in which pieces are joined or fused together, with or without a filler material, so that a cohesive bond is formed. There are two primary types of welding: fusion welding, in which molten metal is formed between the pieces to be joined; and forge welding, in which pressure is applied to cause the plasticized surfaces to diffuse.

A more recently developed fusion welding technique is electron beam (EB) welding. In this process, a stream of electrons emitted by a hot cathode is focused to a fine beam by an electrostatic or magnetic lens. Extremely high densities are possible, and welds of high depth-width ratio, with consequent lack of distortion, can be obtained. The operation must be conducted in a vacuum, and consequently is expensive and time-consuming. The process is used mainly where high heat input, precise placing of the beam, and cleanliness of vacuum welding can be exploited. The process is useful on reactive, vacuum-melted materials such as titanium, zirconium, hafnium, and beryllium. It gives the same control of impurities as in the original material. This technique has been successful in welding titanium forgings for the main rotor hub.

Ref. 2 provides a more detailed discussion of welding practices. In addition, MIL-STD-22 discusses welded joint designs for manual and semiautomatic arc and gas welding processes, although it does not apply as a standard to aeronautical equipment. Additional information is found in AMCP 706-100, MIL-HDBK-5, -693, -694, -697, -723, MIL-STD-20, and ANSI Y 32.3-1969.

Wherever possible, welded joints should be made with smooth-flowing lines that blend gradually with the parent metal. Any abrupt change in the surface contour causes a stress concentration at the point of change. Many parts fail in service because of stress concentrations, which tend to cause failure by fatigue — even when the regions of stress concentration are small — and are almost always traceable to improper design or fabrication. The effect of joint design on stress concentration is illustrated in Fig. 17-2.

Joint design must take into consideration the prior history of the parts being joined and the particular characteristics of the metal or metals. Thus highly cold-worked materials have locked-up stresses that are relieved by the heat of welding, and this may result in distortion or a decrease in strength. The rapid heating and cooling of the weld can produce thermal stresses from expansion and contraction that are quite large. Heat cracks are more apt to occur in weld metal than in the base metal because the weld metal cools last and essentially is cast metal with columnar grains. Corrosion properties also may be affected. For example, although Type 304 stainless steel is corrosion-resistant, chromium carbides form in the grain boundaries in welded areas, leaving the rest of the grain unprotected. Compensation for such alloy changes may be provided by the filler material.

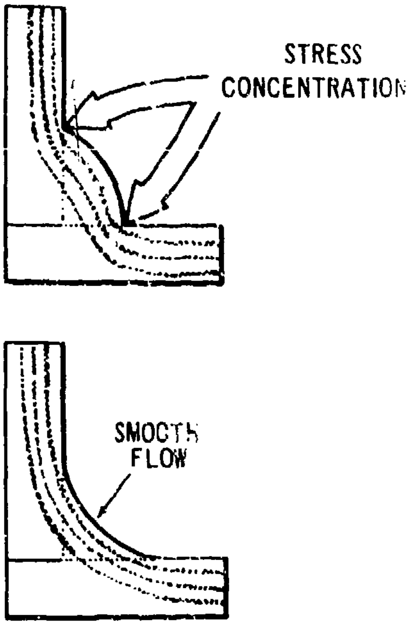
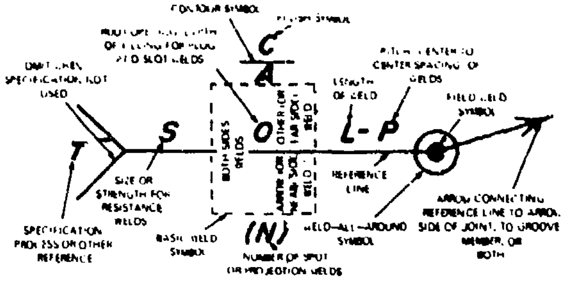


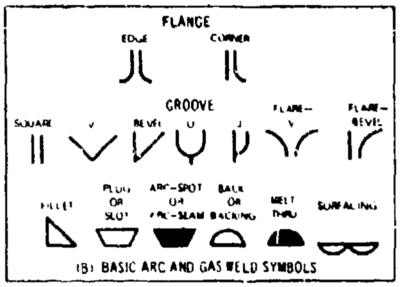
Figure 17-2. Weld Contour and Stress Concentration

Joints should be located so that the entire weld groove is visible to the welder and no obstructions impair the accessibility for welding. On all joints welded from both sides, the root of the first weld should be ground to sound metal before welding the second side. Commonly used welding symbols are given in Fig. 17-3. Representative butt joints are shown in Fig. 17-4, representative corner joints in Fig. 17-5, and representative tee joints in Fig. 17-6.

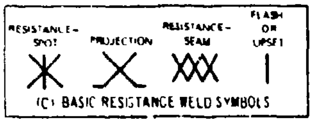
The welding procedure to be followed in the fabrication of a part will be determined by the design and will be defined by the contractor as a process specification. The cost and effectiveness of a welded construction will be determined by the prescribed procedure. Prior to welding, the welding procedure and the welding operator must be qualified in accordance with MIL-STD-248. Qualification assurance requirements for welds are discussed in Chapter 6, AMCP 706-203.


CONTOUR SYMBOL
FINISH SYMBOL
ROOT OPENING; DEPTH OF FILLING FOR PLUG AND SLOT WELDS
OMIT WHEN SPECIFICATION NOT USED
PITCH (CENTER TO CENTER SPACING) OF WELDS
LENGTH OF WELD
FIELD WELD SYMBOL
C
A
T
S
O
L-P
BOTH SIDES WELDS
OTHER (OR FAR SIDE) WELD
ARROW (OR NEAR SIDE) WELD
SIZE OR STRENGTH FOR RESISTANCE WELDS
REFERENCE LINE
ARROW CONNECTING REFERENCE LINE TO ARROW SIDE OF JOINT, TO GROOVE MEMBER, OR BOTH
SPECIFICATION PROCESS OR OTHER REFERENCE
BASIC WELD SYMBOL
(N)
WELD-ALL-AROUND SYMBOL
NUMBER OF SPOT OR PROJECTION WELDS

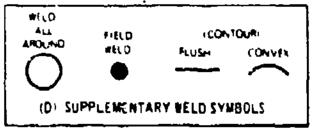
(A) ELEMENTS OF A WELDING SYMBOL


FLANGE
EDGE
CORNER
GROOVE
SQUARE
V
BEVEL
U
J
FLARE-V
FLARE-BEVEL
FILLET
PLUG OR SLOT
ARC-SPOT OR ARC-SEAM
BACK OR BACKING
MELT THRU
SURFACING

(B) BASIC ARC AND GAS WELD SYMBOLS


RESISTANCE-SPOT
PROJECTION
RESISTANCE-SEAM
FLASH OR UPSET

(C) BASIC RESISTANCE WELD SYMBOLS

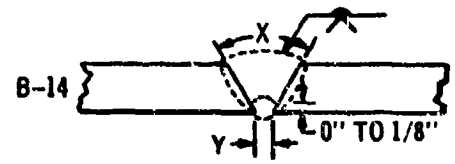

WELD ALL AROUND
FIELD WELD
(CONTOUR)
FLUSH
CONVEX

(D) SUPPLEMENTARY WELD SYMBOLS

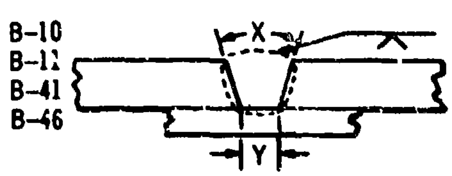
Figure 17-3. Welding Symbols

17-4.2.2 Brazing

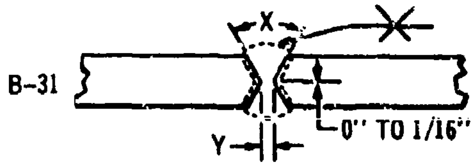
The joining of two parts by brazing requires the use of nonferrous fill rod, film, or powder. The pieces to be joined and the filler material are brought to a temperature that is below the melting point of the materials to be joined but above the melting point of the brazing material. The brazing material wets the surfaces of the pieces to be joined, and through capillary action then draws the melt into the space between the parts. The optimum strength occurs when adhesion is present between the molecules of the brazing materials and the molecules of the base materials. Under these conditions, some alloying takes place.


X
B-14
Y
0" TO 1/8"

(A) SINGLE-V BUTT JOINT, WELDED BOTH SIDES


B-10
B-11
B-41
B-46
X
Y

(B) SINGLE-V BUTT JOINT, WELDED ON BACKING


X
B-31
Y
0" TO 1/16"

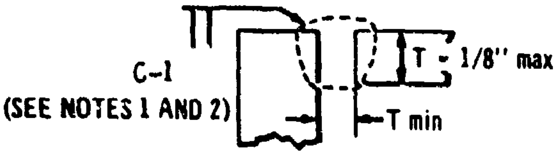
(C) DOUBLE-V BUTT JOINT, WELDED BOTH SIDES

JOINT NUMBER	ANGLE X min, deg	Y ROOT OPENING (min), in.	WELDING POSITIONS
B-14	60	1/8	ALL
B-10[1]	60	1/8	ALL
B-11	45	1/4	ALL
B-41	20	1/2	FLAT, VERT., OVER
B-46	12	1/2	FLAT
B-31	60	1/8	ALL

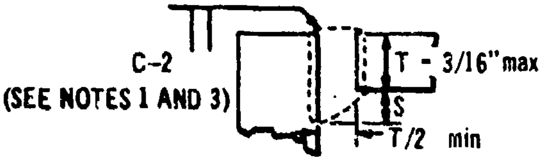
[1] B-10 MAXIMUM PLATE THICKNESS - 1/8 in.

NOTE: REINFORCEMENT OF GROOVE WELDS SHALL BE 1/32 TO 1/8 in. AS WELDED.

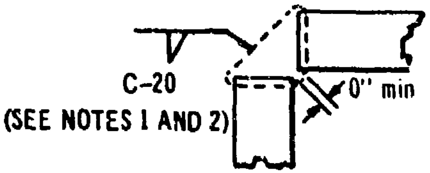
Figure 17-4. Representative Butt Joints


C-1
(SEE NOTES 1 AND 2)
T - 1/8" max
T min

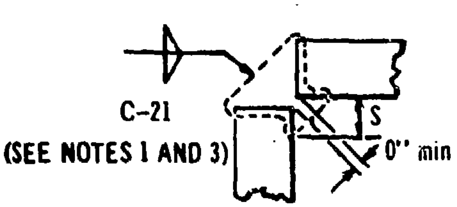
(A) OPEN SQUARE CORNER JOINT, WELDED ONE SIDE


C-2
(SEE NOTES 1 AND 3)
T - 3/16" max
S
T/2 min

(B) OPEN SQUARE CORNER JOINT, WELDED BOTH SIDES


C-20
(SEE NOTES 1 AND 2)
0" min

(C) OUTSIDE SINGLE-FILLET-WELDED CORNER JOINT

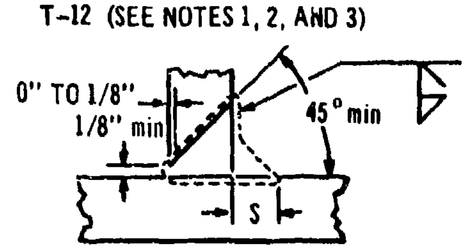

C-21
(SEE NOTES 1 AND 3)
S
0" min

(D) DOUBLE-FILLET-WELDED CORNER JOINT

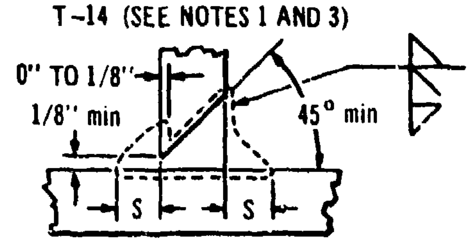
NOTES:

1. REINFORCEMENT OF WELD SHALL BE 1/32 TO 1/8 in. AS WELDED.
2. JOINT SHALL NOT BE USED WHEN ROOT OF WELD IS SUBJECT TO BENDING TENSION.
3. SIZE OF FILLET "S" SHALL BE AS GOVERNED BY DESIGN REQUIREMENTS.

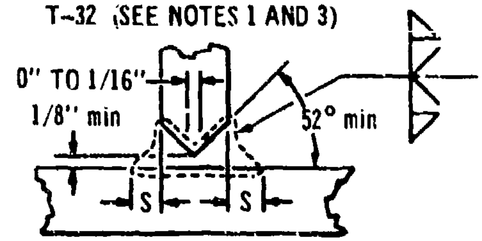
Figure 17-5. Representative Corner Joints


T-12 (SEE NOTES 1, 2, AND 3)
0" TO 1/8"
1/8" min
45° min
S

(A) SINGLE-BEVELED TEE JOINT, WELDED, ONE SIDE, FILLET REINFORCED


T-14 (SEE NOTES 1 AND 3)
0" TO 1/8"
1/8" min
45° min
S
S

(B) SINGLE-BEVELED TEE JOINT, WELDED BOTH SIDES, FILLET REINFORCED


T-32 (SEE NOTES 1 AND 3)
0" TO 1/16"
1/8" min
52° min
S
S

(C) DOUBLE-BEVELED TEE JOINT, FILET REINFORCED

NOTES:

1. REINFORCEMENT OF GROOVE WELDS SHALL BE 1/32 TO 1/8 in. AS WELDED.
2. JOINT SHALL NOT BE USED WHEN ROOT OF WELD IS SUBJECT TO BENDING TENSION.
3. SIZE OF FILLET "S" SHALL BE AS GOVERNED BY DESIGN REQUIREMENTS.

Figure 17-6. Representative Tee Joints

Brazing has the advantage that it can be used to join dissimilar metals. The melting point of brazing materials is above 800°F. Brazing materials include silver, copper and aluminum alloys; nickel-chrome; and silver-manganese. Brazing techniques include torch, resistance, induction, furnace, and dip brazing. A variety of fluxes is used.

Fits and tolerances are of particular importance in brazing. Lap joints are necessary whenever strength is a consideration. The only allowable loading of brazed joints is in shear. An overlap of three times the thickness of the thinnest member gives the greatest efficiency. Butt joints provide a smooth joint of minimum thickness, but are more difficult to fit. Scarf joints maintain the smooth contour of the butt joint, and at the same time provide the large area of the lap joint.

Joint clearance is the distance between the surfaces of the joint into which the brazing material must flow. For any given combination of base and filler metals, there is a best joint clearance. Values below the minimum clearance are weak because the alloy does not flow into the joint. Clearances beyond the maximum also result in joints with lower strength. The normal range of clearance is 0.002 to 0.010 in.

Brazing materials and processes, with approved specifications, are listed in MIL-B-7883.

17-4.2.3 Soldering

The principles and techniques for soldering are much the same as for brazing, except that soldering alloys are primarily tin-lead base alloys that melt below 800°F. Soldering is restricted primarily to joining sheet-metal surfaces and wiring for electrical connections. Copper and steel surfaces, and surfaces coated with copper, tin, zinc, or other compatible materials, may be soldered. Aluminum and stainless steel are examples of metals which are difficult to solder. Cleanliness and proper selection of the solder and the flux in relation to the surfaces to be joined are of utmost importance. No porosity can be tolerated in soldered joints. Electrical continuity and complete sealing against fluids are normal requirements.

17-4.3 MECHANICAL FASTENING

Because of the convenience of mechanical fastening, parts that need never be disassembled often are joined mechanically, with resultant poor distribution of stresses and increased weight. Such parts well may be joined more effectively by welding, brazing, or adhesive bonding.

There also is the question of which of the hundreds of types of mechanical fasteners to use. The result often is overdesign, with poor strength balance between the fasteners and the parts joined. Moreover, there is little in the literature to guide the designer in making broad choices in mechanical fastening. However, there are more than 20 Military Specifications on specific fasteners, and MIL-F-19700 and MS-17855 contain general specifications and standards for screw threaded fasteners. Following is a discussion of various methods of mechanical fastening.

17-4.3.1 Rivets

For all aircraft applications, particularly where dissimilar metals are involved, rivets *shall* be set with primer. Often an adhesive is placed between the parts being riveted in order to dampen vibrations and to minimize failure due to fatigue.

Important items of rivet joint design are pitch, the spacing between rivet centers; back or transverse pitch, the spacing between centerlines of rows of rivets; diagonal pitch, the spacing between the nearest rivet centers of adjacent rows; and margin, the spacing between the edge of a part and the centerline of the nearest row of rivets.

For structural and machine member joints, the pitch should be such that the tensile strength of the plate in the distance between rivets in the outer row is equal to the shear strength of the repeating sections of rivets. The rivet diameter d is selected so that $d = 1.1\sqrt{t}$ to $d = 1.4\sqrt{t}$ where t is the thickness of the plate. Good design practice calls for placing the center of the first row of rivets a minimum of one and one-half rivet diameters away from the edge of the plate. For multiple rows, the transverse pitch is $1.75d$ (see Fig. 17-7). It is also required that:

1. Bearing stress of rivet and plate be uniformly distributed over the projected area of the rivet
2. Tensile stress be uniform in the metal between rivets
3. Shearing stress be uniform across the rivet
4. Accumulation of stresses on rivets be minimized.

Rivet holes usually are made several thousandths of an inch larger than the nominal diameter of the rivet. On compression, the rivet expands to fill the hole while also forming the driven head. Punching operations may cause degradation in the strength of the plate surrounding the hole. An annealing operation may be desirable to restore strength in these areas. Structural and machine member rivets usually are made of wrought iron or soft steel; but copper, aluminum alloy, Monel, and Inconel rivets may be

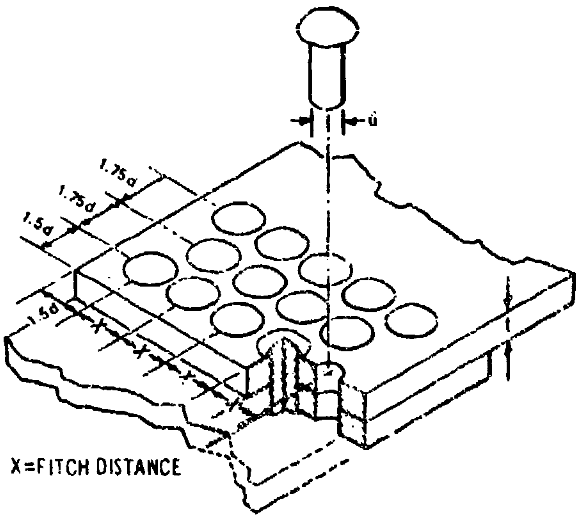


Figure 17-7. Rivet Spacing

required where weight or resistance to corrosion are important.

Although there are no general Military Specifications on riveting, there are more than 50 Military Specifications covering particular types of rivets. A general treatment for riveted joints in steel, aluminum, magnesium, and titanium is given in Chapter 8, MIL-HDBK-5.

17-4.3.2 Bolts, Nuts, and Washers

Bolts and nuts often are used when the joint will not be permanent and disassembly can be anticipated. Maximum strength of bolted joints can be attained only when the grip length of the bolt is at least equal to the thickness of the parts being joined. No threads are to be in bearing in the holes through the parts. Washers are added as required to permit tightening the bolt sufficiently to develop the load-carrying capability of the joint.

All bolted joints must be locked or safetied, and bolts in critical locations, such as in control linkages, *shall* have two separate locking provisions. Self-locking provisions include nylon inserts in the bolt or nut. Locking and safetying are discussed in Chapter 16.

If failure of a threaded assembly should occur, it is preferable for the bolt to break rather than for either the external or the internal thread to strip. Thus, the length of the mating threads should be sufficient to carry the full load necessary to break the bolt without stripping. The critical areas of mating threads are:

1. Effective cross-sectional or tensile stress area of the external thread
2. Shear area of the external thread (dependent upon the minor diameter of the tapped hole)
3. Shear area of the internal thread (dependent upon the major diameter of the external thread).

When bolts are used to join dissimilar metals, or to join materials dissimilar from the bolt material, differences in the coefficients of thermal expansion and in the temperature extremes specified for the helicopter must be considered in calculating the maximum stresses. Provisions also must be made for corrosion protection in such installations.

Box wrench clearances are given in Ref. 3. For wrench access, bolt centers should be placed at a minimum distance from obstruction of two times the wrench clearance.

Plain washers are defined by ANSI B27.2-1965. These washers are available in narrow, regular, and wide series, with proportions designed to distribute loads over larger areas of lower-strength materials. Plain washers are made of ferrous, nonferrous, plastic, or other materials. ANSI B27.1-1965 defines helical-spring-lock and tooth-lock washers. Helical-spring-lock washers have a dual function. First, they compensate for looseness that may be developed during the use of a bolt or screw fastener, preventing loss of tension between component parts of the assembly. Secondly, they act as hardened thrust bearings to facilitate assembly and disassembly of bolted fasteners by decreasing the frictional resistance between the bolted surface and the bearing surface of the head or nut. Tooth-lock washers bite into the bearing surfaces and increase frictional resistance to motion. These washers are made of carbon steel, corrosion-resistant steel, aluminum-zinc alloy, phosphor bronze, and K Monel of various series.

17-4.3.3 Screws

Screws, particularly machine screws, may be confused with bolts. Actually, the equations for strength and the precautions on thread bearing are the same for both. ANSI B18.6.3-1962 covers both slotted and recessed-head machine screws. Threads on machine screws may be either unified coarse (UNC) or fine thread (UNF). Head height of countersunk screws is the distance, parallel to the axis, from the bearing surface at the diameter of the screw to the largest diameter of the bearing surface. For screws less than 2 in. long, the threads will run to within two turns of the head. For longer screws, the minimum thread length is 1-3/4 in. The body is the unthreaded cylinder portion of the shank. The designation of a screw thread consists of nominal size (inches or number), number of threads per inch, letters of the thread series, and class of tolerance. On drawings, the designation may be followed by the pitch diameter tolerance. A left hand thread is identified by the letters LH following the class designation. The formulas for tolerances and allowances for the several series and classes of threads are given by ANSI B1.1-1960.

The heads of machine screws may be recessed, hex, slotted, round, countersunk, pan, cheese, or mushroom. Tapping and metallic drive screw types include round head, flat head, flat and oval trim head, undercut, Fillister, Struess, pan, and hex head.

Sheet-metal screws are defined in ANSI B18.6.4-1966. Some of these, when turned into a hole of the proper size, form a thread by displacing the sheet metal, while others form a thread by cutting action. There are 12 types, each having preferred applications with sheet metal, plywood, nonferrous castings, plastics, etc.

Set screws are used for preventing a pulley, gear, or other part from turning relative to a shaft. Generally speaking, 1/4-in.-diameter screws will hold against a force of 100 lb; 3/8-in., 250 lb; 1/2-in., 500 lb; and 1-in., 2500 lb.

Self-tapping screw thread inserts are hard bushings with internal and external threads. They often are used in nonferrous castings. Helical, diamond-shaped coils of stainless steel with phosphor bronze inserts often are used to repair old, threaded holes.

There are many Military Specifications dealing with particular types of screws, including MIL-STD-9 and MIL-S-7742. Additional data will be found in Chapter 8, MIL-HDBK-5.

Pins, such as cotter pins are used to secure nuts upon bolts, or upon other pins and fasteners. Cotter pins may be obtained for hole sizes of 3/64 to 3/4 in. The extended-prong type is secured in place most easily. Clevis pins (ANSI B5.20-1958) frequently are used as lock bolts and may be secured with a cotter pin. Dowel pins (ANSI B5.20 1958) are used either to retain parts in a fixed position or to preserve alignment. They normally are subject to shearing strain only at the junction of the two parts being held, and two usually are sufficient. For parts that frequently are disassembled, the taper dowel is preferred. This type also is preferred for joints of close tolerance. Pins having diameters of 1/8 to 3/16 in. are satisfactory in most cases.

When soft parts are to be joined, the hole should be about 0.001 in. smaller than the pin. For a locking fit, a longitudinally grooved pin is preferred.

17-4.4 ADHESIVE BONDING — STRUCTURAL

In order to realize the maximum benefits of adhesive bonding, a structure must be designed initially with this method of joining in mind. The first requisite is an understanding of the basic loading conditions to which a bonded joint may be subjected. In the past structural applications of adhesives were based on tension, shear, cleavage, and peel (as illustrated in Fig. 17-8). These criteria resulted in the development of adhesives that met purely structural applications based on design loads. Today, adhesives not only must meet these requirements but also must include resistance to environmental conditions experienced in current adhesive specifications for resistance to moisture penetration under load (environmental cyclic creep) and under dynamic loads (fracture mechanics and fatigue). These include tension, shear, cleavage, and peel, as illustrated in Fig. 17-8.

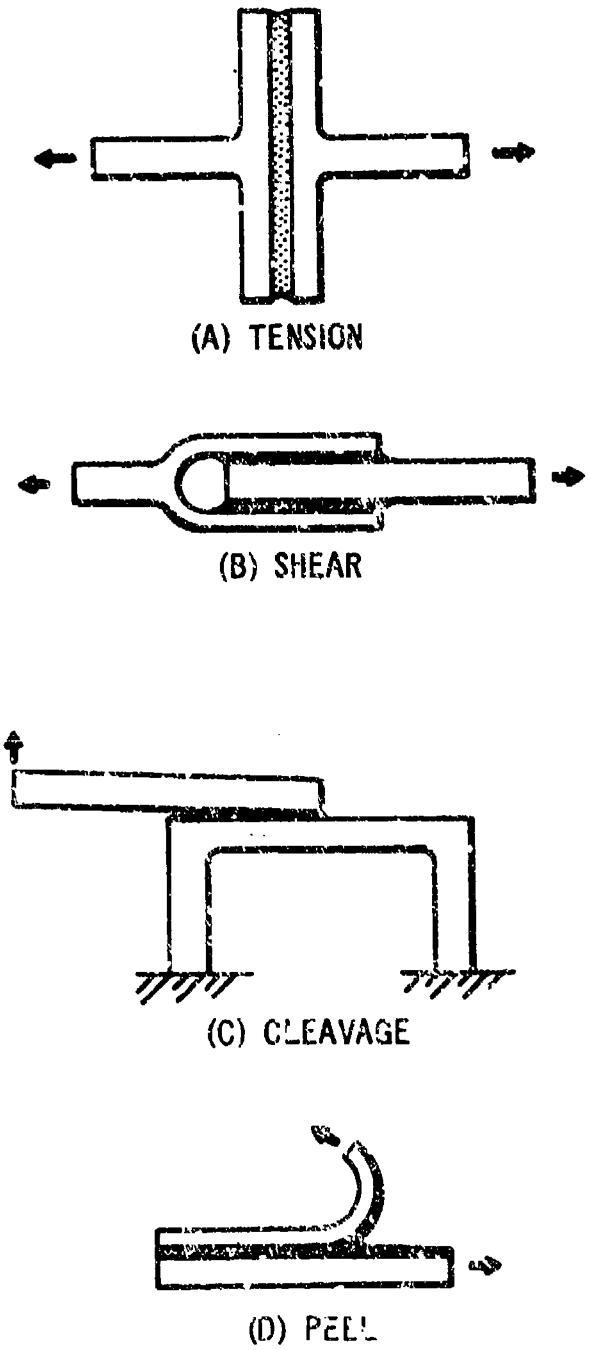


Figure 17-8. Types of Loading for Bonded Joints

An ideal structural joint is one in which the load is distributed as uniformly as is possible over the entire bonded area. This condition most nearly is approached when the basic stress is tension or shear (Figs. 17-8(A) and 17-8(B)). While the stresses that cause failure in the cleavage or peel situation (Figs. 17-8(C) and 17-8(D)) may be described as tensile stresses, they are concentrated heavily in local regions of the joint; thus, the load capability of the joint is unrelated to the total bonded area. Therefore, peel and cleavage situations should be avoided.

The designer must examine carefully even the tension and shear joints to minimize any eccentricities or deflections that would cause an unfavorable redistribution of stresses in the joint. For example, a simple lap shear joint under load, if not restrained, will undergo a deflection that causes high concentrations of stress at Points 1 and 2 as illustrated in Fig. 17-9.

The designer should keep in mind that joints with uniformly distributed shear or tension joints are a design goal, and then should apply common sense to achieve this goal. Such a common-sense approach to a joint design problem is illustrated in Fig. 17-10. The spar of the rotor blade might be considered to be a rigid member. Air loads acting upon the aft section create a bending moment that tends to pry, or peel, the lightweight skins away from the inside of the spar at Point A. Therefore, a "keeper" channel that greatly rigidizes this portion of the structure is incorporated, permitting the joint between the skin and spar to react the bending moment through a shear couple.

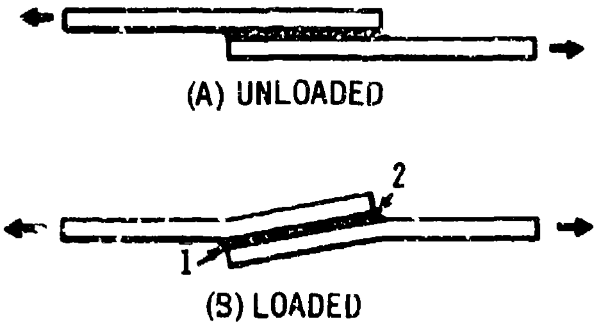


Figure 17-9. Lap Shear Joint Deflection Under Load

Another typical rotor blade design is shown in Fig. 17-11. Again, there is a tendency for the skin to peel from the spar. In this case, the skin deflections that would aggravate the peel situation are prevented by the incorporation of a simple angle (A).

The design of adhesive-bonded joints demands the proper mating of parts with regard to dimension and tolerance. It is imperative that adhesive-joined components fit each other without relying upon the adhesive to hold them in the proper shape. Where overall external dimensions are critical, it is necessary to account for the thickness of the cured bond lines in the final assembly. A reasonable thickness allowance per bond line is 0.005 in., although this value should be determined specifically in each case for the particular adhesive and method of manufacture.

In the case of honeycomb-sandwich structures, no allowance is made for adhesive between the honeycomb core and the face sheets. The cells of the honeycomb cut through the film of adhesive completely,

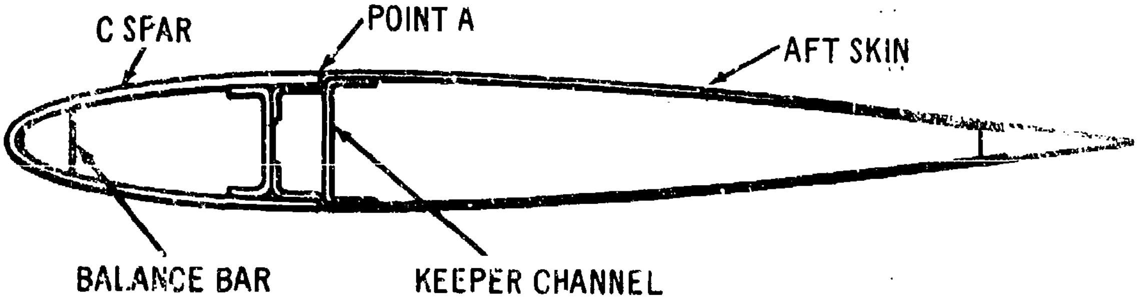


Figure 17-10. Typical Rotor Blade Design — Alternate 1

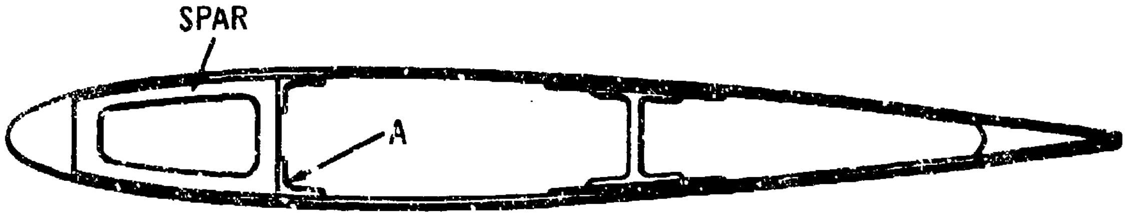


Figure 17-11. Typical Rotor Blade Design — Alternate 2

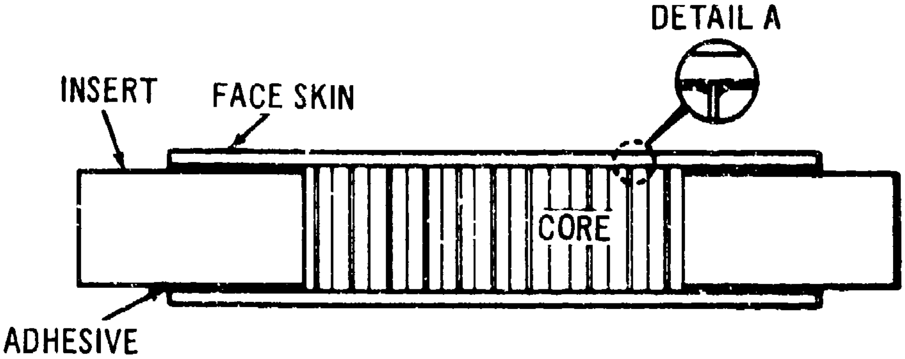


Figure 17-2. Honeycomb Sandwich Structure

and actually contact the face skin. (Fig. 17-12, Detail A). The bond of face-to-core is achieved by the fillet of adhesive between the face and the cell wall, which is another example of a shear joint. In Fig. 17-12, the insert might be an extrusion or any other form that results in a metal-to-metal joint, as opposed to a face-to-core joint. Thus, if the desired overall dimension is 1.00 in. and the face skins are each 0.020 in. thick, the inserts would be 0.95 in. thick, thereby allowing for two 0.020-in. skins and two 0.005-in. bond lines. The core, on the other hand, would have to be 0.96 in. thick, allowing only for the skins. An exception to this rule is the case of the adhesive that incorporates a carrier, or scrim, which is composed of woven or randomly oriented fibers. In such a case, the thickness of the carrier should be allowed for between core and faces.

Fig. 17-12 actually is oversimplified in that the abrupt change in section between the honeycomb sandwich and the rigid insert or adjoining member should, in most cases, be made more gradual either by shaping the solid member as shown in Fig. 17-13(A), or by the addition of doublers as in Fig. 17-13(B). In either case, the rules for adhesive thickness allowance still apply. If a construction like that shown in Fig. 17-13(A) is employed, a foaming type of adhesive may be used on the portion of the core that fits against surface A of the closure. Such an adhesive can be pressed into the core cells prior to assembly and thereby permit fitting of the parts.

Another example of a design technique that assures a fit of parts and, thus, a uniform, unstrained bond line is that employed for the balance bar shown in Fig. 17-10. The bar is bonded into the nose radius of the C spar. Because the spar is very stiff in this highly curved portion, while the balance bar is solid, the required bonding pressure is achieved best by applying a force against the back of the bar. It will be noted that the nose radius of the bar is larger than the inside nose radius of the C spar. This assures that the bar will never bottom out and, therefore, pressure always will be applied at the side of the bar. This technique is not restricted to rotor blades, but can be employed in any situation where the bonding pressure is provided by a component of force that is applied in a direction other than normal to the bond line.

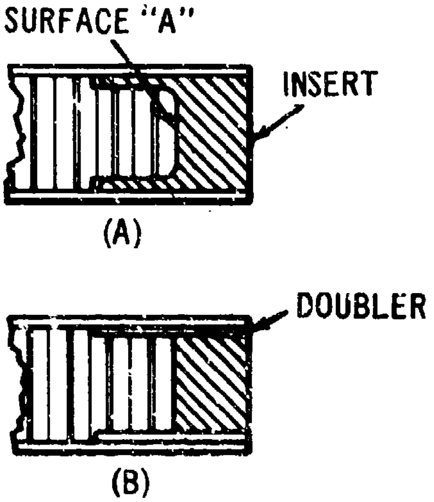


Figure 17-13. Addition of Doublers to Honeycomb Structure

While it was stated previously that parts usually should be formed to fit each other without the aid of the adhesive bond, the degree to which this rule should be imposed is dependent upon the stiffness of the members being joined and nature of the assembly. Generally, sheet-metal parts should not be so closely toleranced in their free state that there is no allowance for clearances when assembling them. There also must be space for 0.010 in. or more of uncured adhesive between the components during assembly. This can be achieved by an over-bend or under-bend allowance, but this allowance never should be so great as to prevent hand pressure from bringing the part into the final form that is desired in the finished assembly. The residual strains caused by such a condition are negligible.

Bonded structures usually are assembled in tooling. This precludes a visual examination of the relative locations of components during the curing stage. Because the adhesive becomes fluid at some point during the cure, and thermal expansions are taking place, some shifting of one part relative to another usually must be anticipated. Thus, adjoining parts must be dimensioned and toleranced to accommodate such movement. Fig. 17-14 illustrates such a case. The bottom of the figure shows a gap where two members butt together. This gap must be programmed in the design and usually should be from 0.020 to 0.060 in., depending upon the size of the structure. In the example shown at the top of Fig. 17-14, insufficient allowance was provided and the resulting joint either will have a void or, at best, will be subject to peel. Usually, the small gap in the properly designed joint will fill with adhesive squeeze-out or may be filled later with a fairing-sealing compound.

Currently, all structural bonds require pressure during their curing This pressure must be reacted in some way. Either one of the members being joined must be stiff enough so that it will not deflect significantly under the pressure required to bond the other members to it, or provisions must be made for tooling that will provide the reaction force. When such tooling is required, the detail design must provide space for the tooling and a means of removing it after the cure cycle is completed.

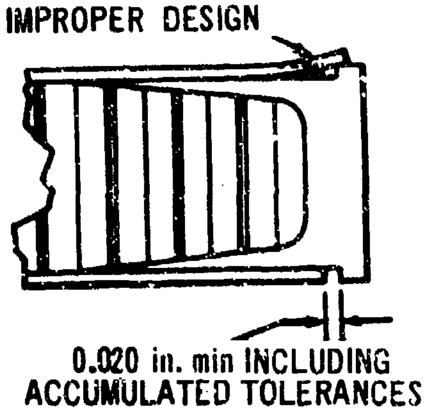


Figure 17-14. Balance Bar Design

When dissimilar metals are bonded, serious distortions can occur due to differential expansions, because the bonds are being cured at temperatures ranging from 225° to 350°F. Such distortions of the structure can be minimized if the member with the lowest coefficient of expansion is stretched or otherwise strained by the application of external force while undergoing cure. This often means that some extra length must be provided to permit gripping the member or pinning it to the bonding fixture. This extra material can be removed after the bonding is completed.

It often is impractical to complete an entire assembly in a single bonding operation. This creates the necessity for secondary bonding, in which a part of the assembly is reheated to bonding temperature while additional parts are bonded to it. The most commonly used adhesives are quite weak at the temperature at which they were cured and, thus, there is a risk of destroying the original bond during a secondary bonding operation if proper precautions are not taken. One such precaution is the use of a lower-temperature curing adhesive for the secondary bond. Another, more reliable technique is to reapply pressure to any of the primarily bonded joints that will be subjected to the heat of secondary bonding. Provisions must be made in the design to assure that reapplication of pressure is possible in such cases.

Finally, the designer always should consider providing, for physical testing, some kind of extension of the basic structure that can be removed after bonding. Nondestructive testing techniques are being improved constantly, but there is no substitute for destructive tests of joints that are built along with, and duplicate, the actual structure.

17-4.5 SWAGING AND CABLE SPLICING

Wire rope and cable may be used in helicopter control mechanisms, although push rods are preferred. Aircraft cable made of high-carbon steel wire, electrolytically galvanized and drawn to size, has the highest strength and greatest resistance to fatigue of any cable.

Representative wire rope fittings are illustrated in Fig. 17-15. For aircraft, the more commonly used types range in size from 1/16 to 5/8 in.

Swaged fittings on wire rope have a strength rating equivalent to the strength of the wire rope. These fittings are applied to the end or the body of wire rope by the application of high pressure, causing the steel to flow around the wires and strands of the wire rope to form a union as strong as the rope itself. The necessary high pressure and flow are accomplished by means of special dies. Machines for this purpose are described in MIL-S-6180 and MIL-S-80035.

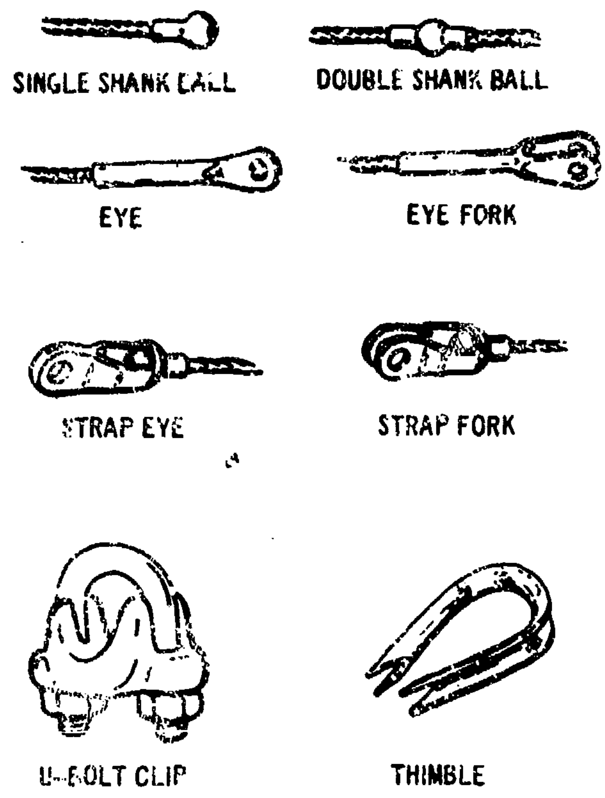


Figure 17-15. Aircraft Wing Rope Fittings

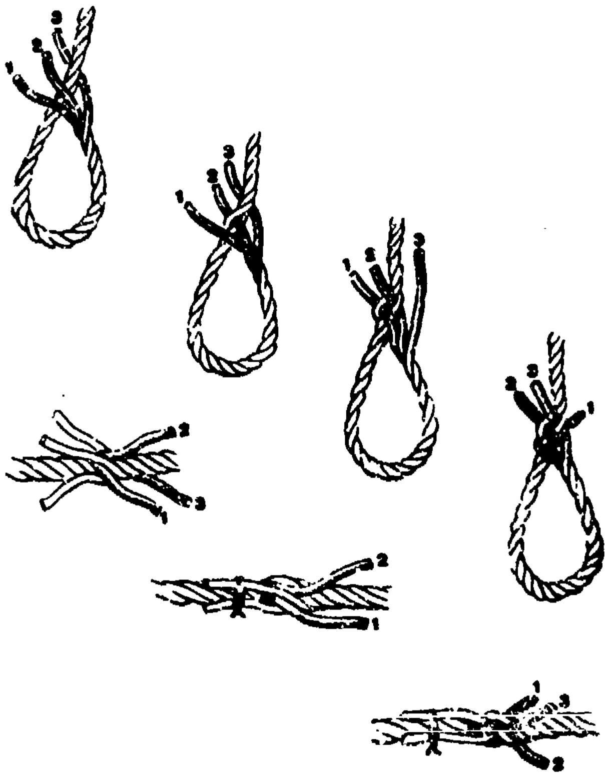


Figure 17-16. Cable Splicing

Another effective method of attaching fittings is by means of poured zinc. A special high grade of pure zinc is used to fill the socket. (Babbitt and other alloys will not hold properly.) When properly prepared, the joint strength approaches that of the wire rope.

For temporary connections, U-bolt clips may be employed. When forming a loop with a U-bolt clip, a wire rope thimble should be placed in the loop to prevent kinking. It is essential that the saddle or base of the clip bear against the longer or live end of the rope, while the U-bolt bears against the shorter or dead end. The end of the wire rope should be seized properly. The strength of a clip fastening is less than 80% of the strength of the cable.

A seldom-used means of joining cables and making endloops is cable splicing. The method may be used in the field for repair when equipment or fittings are unavailable. To join two cables, each of the free end strands is worked over and under a strand in the other cable, working against the lay. A total of four tucks for each strand is adequate for most purposes. An eye splice can be made as a loop of any size, or tight around a thimble. The ends are tucked under the main strands against the lay and around the cable. Normally, three tucks for each strand are all that can be managed. Splicing is illustrated in Fig. 17-16. Field-spliced cables never should be exposed to more than 50% of the normal breaking load for the cable.

Heavy abrasion, overloading, and bending around sheaves or drums that are too small in diameter are the principal reasons for deterioration of wire rope. A kinked, bent, or flattened area cannot be tolerated in a control cable. The rope or cable should be kept well lubricated, using the lubricant supplied by the manufacturer. Normally, the lubricant must be heated to 200°F in order to penetrate. Nylon or vinyl coatings are effective for sealing in lubricant and for protecting wire rope from dirt and corrosion.

17-5 HEAT TREATMENT

17-5.1 GENERAL

Heat treatment is defined as the application of time-temperature-cooling cycle relationships in order to cause atomic, molecular, or crystalline transformations in materials. These transformations are selected to impart desirable properties for particular end-uses. Although such transformations occur in plastics, ceramics, and other nonmetallic materials, the term heat treatment usually is applied exclusively to metals. The engineering metals include ferrous alloys, copper alloys, aluminum, titanium, and magnesium.

Heat treatment is applied to all forms of metals, including castings, forgings, machined parts, bars, plates, and sheets. The heating cycle may be conducted in various types of furnaces with a variety of atmospheres, in liquid media such as molten salt or molten lead, or by some other means. The holding and cooling cycles may involve vacuum, controlled inert atmospheres, oil bath quenching, air cooling, or water quenching.

The facilities, equipment, processes, controls, and acceptance standards for the heat treating of metal parts have been defined in numerous Military Specifications including MIL-H-6088, -6875, -7199, and -81200.

17-6.2 HEAT TREATMENT METALLURGY

There are many kinds of heat treatment, with time-temperature cycles designed to develop the crystalline and grain structure characteristics of the particular material. Those characteristics include strength, toughness, hardness, machinability, freedom from residual stress, or any other property necessary for the part to perform its design function. Each metal and metal alloy has its own heat treatment characteristics that are determined by its chemical constitution.

The more commonly employed heat treat operations are annealing, normalizing, stress relief, tempering, and aging.

17-6.2.1 Annealing

Most heat processes are variations of annealing. A "full" anneal consists of heating the metal to a temperature that allows the grains to recrystallize in a desired pattern. This process is described interchangeably as solution heat treatment, to indicate that the metal is heated to a temperature at which the alloying elements are dissolved and placed in solid solution. Then the metal is cooled in order to precipitate the desired structure.

17-6.2.2 Normalizing

This annealing step often is accomplished after forging or machining in order to restore uniform grain structure. The metal is heated to just above the solution temperature — in no case long enough or hot enough to cause the grains to grow to any appreciable extent — and then is cooled in still air to room temperature. The effect is to restore uniform cutting conditions in the previously distorted crystals.

17-6.2.3 Stress Relief

This process is conducted at considerably lower temperatures than those for the other heat treatment processes. Stresses imposed by forming or machining are relieved by allowing the diffusion of boundaries, or the diffusion of hydrogen from a part where hydrogen embrittlement is a hazard.

17-6.2.4 Tempering

This is the process of reheating a normalized or quench-hardened alloy to a temperature just below the transformation range and then cooling it at a suitable rate. Tempering is used to obtain desired properties of strength and toughness.

17-6.2.5 Aging

Aging is a tempering process in which certain alloys are held at a constant, relatively low temperature long enough to permit the precipitation of particles and grain structure transformations that develop the desired properties of strength, toughness, hardness, or formability. In some alloys, aging may be accomplished at room temperature.

The data in Table 17-5 are indicative of the temperatures at which heat treatment operations are conducted for each of the engineering metals.

17-6.3 FERROUS ALLOYS

The conditions for annealing, hardening, and tempering individual ferrous alloys are given in the applicable Military Specification or Aerospace Material Specification (AMS), or in the producer's literature. Air, combusted gases, protective atmospheres, inert atmospheres, vacuum, fused salts, and molten metal are acceptable heating media. Temperature control must allow the entire lot of heat-treated material to achieve its desired properties. Corrosion, contamination, and embrittlement of the metals being treated *shall not* be permitted.

Processes such as induction heating, flame hardening, carburizing, nitriding, austempering, and martempering are recognized processes but are beyond the scope of this discussion.

The equipment for heat treating must provide for adequate handling and uniform treatment of the charge, as well as for precise control of the time-temperature-cooling program, to achieve the desired results. The quenching processes involving oil, water, and air must be controlled to function with adequate speed.

TABLE 17-5. REPRESENTATIVE HEAT TREAT TEMPERATURES

METAL	TEMPERATURE IN °F FOR: SOLUTION H T ANNEALING FURNACE COOL	NORMALIZE AIR COOL	HARDEN QUENCH	TEMPER QUENCH	STRESS RELIEF AIR COOL
STEEL–4140	1525–1575	1600–1700	1525–1600	1075	1600–1700
TITANIUM 4 Al–3Mo–1V	1620–1700		900–975		900–1100
COPPER–Be ALLOY no 175	1700		900		
ALUMINUM 2024	900–93[illegible]		370–380		
MAGNESIUM AZ91C	775–790		400–425		

The following limitations and controls are imposed upon the treating processes:

1. Transformation-hardening steels *shall* be quenched to not less than 93% martensite or 93% lower bainite, as specified by design documentation. Mixed structures are not acceptable.

2. Cooling of stainless steels from the annealing range must not exceed 50 deg F/hr down to 1111°F.

3. Transformation-hardening steels *shall* be cooled to or below quenching-bath temperature before tempering.

4. The maximum permissible increase in depth of any zone of decarburizing *shall not* exceed 0.003 in. unless this zone will be removed by subsequent machining or grinding.

17-5.4 NONFERROUS ALLOYS

17-5.4.1 Aluminum Alloys

Air, fluidized beds, combusted gases, protective atmospheres, and molten salt baths are acceptable media for the heat treating of aluminum alloys, provided that no damage is done to the material. Items processed in air chamber furnaces must be shown by test to be free from high-temperature oxidation. Salt baths must be of the proper type and grade; nitrate baths will attack aluminum-magnesium alloys, for example. Time-temperature-cooling cycle controls must provide the desired properties. Parts must be substantially free from lubricants and other foreign matter, which could harm the material being heat treated. Quenching normally is conducted by total immersion in water of 100°F for wrought alloys and 150° to 212°F for castings and forgings, except that certain alloys may be oil-quenched or air-quenched.

17-5.4.2 Copper Alloys

Furnaces having vacuum or controlled atmosphere frequently are used for copper. Air atmosphere may be used when the loss of material due to oxidation and scaling is not detrimental to the finished part. Bright hardening requires a controlled, nonoxidizing and noncombustible gas atmosphere in a chamber furnace. Molten salt baths *shall not* be used for solution heat treatment because of the susceptibility of copper alloys to corrosive attack by molten salts at solution heat treatment temperatures. Time-temperature-cooling cycles must be adequately controlled and the parts adequately cleaned before charging. Cleaning may involve vapor degrease, acid pickling, or bright dipping. Neutral salt baths may be employed for age hardening, but must be removed carefully and neutralized. Quenching is in water. Copper-beryllium mill products and forgings normally are supplied in a condition suitable for precipitation heat treating, hence, solution heat treating is performed only when welding or cold working has required a softening treatment.

17-5.4.3 Titanium Alloys

Furnaces having a slightly oxidizing or inert atmosphere are employed for heat treating of titanium alloys. Reducing or endothermic atmospheres such as hydrogen or cracked ammonia *shall not* be used.

Hydrogen embrittlement presents a major problem because hydrogen is absorbed readily from baths and gases. Quenching is in water, and quench delay times must be minimal, except that the product *shall* be air cooled after stress-relieving operations. The treatment of titanium at temperatures above 1100°F under oxidizing conditions may result in severe scaling and oxygen diffusion to form a hard, brittle surface layer. Titanium alloys are susceptible to stress corrosion by halides at temperatures above 550°F.

17-5.5 DESIGN ASPECTS OF HEAT TREATING

The heat treating processes to which a particular part may be subjected in the course of its fabrication and assembly are an integral part of its design. The heat treat requirements must be defined clearly on the drawings and in the process specifications for the part. The heat treatment must be performed in the proper sequence to achieve the required end-item properties. Because the end-item properties serve to qualify the entire process, the design must be such that the properties realistically can be achieved. Assistance in the heat treating aspects of the design of metal parts can be found in a number of military documents, including MIL-HDBK-5, -693, -694, -697, -698, and -723. Additional information will be found in Ref. 4 and in Chapter 11, AMCP 706-100.

17-6 WORK HARDENING

17-6.1 GENERAL

When a stress is imposed upon a metal, some or all of the atoms in its grains are moved from their equilibrium positions, resulting in crystalline deformation. If enough stress is applied to deform the lattice permanently, the atoms will not return to their former positions; the metal has undergone plastic deformation. When this occurs below the recrystallization temperature, the metal is said to be "cold worked"; i.e., the grains have been distorted, elongated, and fragmented. As stress is increased, the dislocations in the crystals move to grain boundaries or other imperfections, where they are stalled and present increasing resistance to further plastic deformation. The metal then is work hardened.

Work hardening has some distinct disadvantages. It can cause cracking in sheet-metal forming, or it can require intermediate annealing steps during shaping operations. In machining, the metal chip may become severely deformed before breaking away from the work piece. Austenitic steels are difficult to machine because of their high rate of work hardening. And, as a consequence of their high ductility, much work hardening occurs before rupture of the chip. This results in increased power consumption and tool wear.

On the other hand, work hardening can be beneficial. Increases in tensile and yield strengths can be obtained, as can resistance to bending and buckling, along with increased fatigue resistance. Judiciously employed, the phenomenon can be used to provide lighter, stronger parts, thereby eliminating the need for subsequent, expensive heat treating equipment and processes. The designer must be careful to select material that will work harden to the proper degree during fabrication in order to produce a part having the required hardness or rigidity without excess weight. Knowledge of which gage and condition of material to select can produce significant savings. For instance, many materials can be purchased with varying degrees of cold working; aluminum, stainless steel, and brass may be purchased fully annealed, 1/4 hardened, 1/2 hardened, 3/4 hardened, and fully hardened by cold working.

17-6.2 FORMING

Forming operations represent the major area where savings can be obtained by judicious use of the process of work hardening. All operations that incorporate bending, stretching, or upset of metals, and result in plastic deformation below the crystallization temperature, involve work hardening. The effect may be advantageous or disadvantageous, depending upon the selection of the starting material and the rate and degree of deformation. For instance, stainless steel has high ductility but wrinkles easily with compression. A strong, light muffler header having increased rigidity and increased resistance to fatigue can be made by selecting the proper hardness of stainless steel and then designing the blanking and forming dies to provide the required degree of deformation in the proper places. In this case, an inner stretch and an outer compressive deformation would be required.

Similar considerations also may be appropriate for dished and flanged parts such as wheels, pulleys, and fairings; and in the shaping of bars and tubes, integral stiffeners, and large, stretch-formed shapes such as cowlings.

17-6.3 ROLLER BURNISHING

Roller burnishing is a method of improving finish and dimensional accuracy, and results in work hardening a surface without the removal of metal. The operation is employed primarily with internal bores. Bore diameter can be increased by 0.002 to

0.005 in., although this is not normally a primary objective. The operation frequently is designated for phosphor bronze and sintered bronze bushings. The depth of burnishing normally is limited to three times the diameter of the hole, but the insides of tubes 10 to 20 ft long have been roller burnished. Wall thickness is limited to no less than 1/16 in., unless the wall is supported properly by a backing material. Metals that work harden rapidly must be at a lower hardness.

Roller burnishing is a machine operation in which a set of steel rollers is caused by cam action to impact a surface at a rate of, perhaps, 2,000,000 blows per min. This produces a smooth surface, improves roundness or straightness, and increases surface hardness to a depth of 0.005 to 0.015 in. The surface may be finished to a tolerance of ±0.0001 in. The kneading action tends to reduce the stresses imparted by prior operations, such as welding and machining, and also introduces a compression stress to the surface. Greatly improved fatigue and impact resistance can be given to parts in this manner.

Roller burnishing has a limited amount of application to external surfaces. A special operation using a narrow roller of the required shape sometimes is employed for rolling fillets. In this case, the objective is an increased resistance to fatigue. Relatively small forces are employed — 100 psi or less. A plain roller of oil-hardened tool steel at Rockwell C-62 to C-65 may be used to burnish a fillet of 1/32-in. radius in about 10 passes. The rolling and pressing causes a combined rolling and sliding action on the metal in the fillet to relieve the stresses and to work harden the material locally to higher hardness and fatigue resistance.

17-6.4 SHOT-PEENING

Shot-peening is a process used on many helicopter components to increase fatigue strength. Compressive stresses are induced in the exposed surface layers of metallic objects by the impingement of a stream of shot, directed at the metal surface at high velocity and under controlled conditions. When the individual particles of shot contact the metal surface, they produce slight, rounded depressions in the surface, thus stretching it radially and causing plastic flow of surface material at the instant of contact. The layer of metal thus affected is 0 005 to 0.010 in. thick. The surface metal is in compression parallel to the surface, while the underlying metal is in tension. The compressive stress may be several times greater than the tensile stress, and therefore offsets an imposed tensile stress such as is encountered in bending. The fatigue life of the parts in service is improved markedly. The stress-concentration effects of notches, fillets, forging pits, surface defects, and decarburization are reduced greatly. Shot-peening can change to beneficial compressive stresses the residual tensile stresses that grinding usually imposes upon a metal surface.

A higher residual stress, approaching the full-yield strength, can be obtained by strain peening. This consists of peening the surface while it is being strained in tension. The surface tensile stresses that give rise to stress corrosion also can be overcome by the compression stresses induced by shot-peening. The brittle failure of a ductile material due to stress corrosion has been associated with brass, stainless steel, aluminum, zinc, magnesium, and titanium. The compressive stresses due to peening are stable in low-alloy steels to 550°F and in high-temperature steels to 800°F. The effectiveness of peening in improving fatigue resistance is illustrated in Table 17-6.

Shot size has been standardized by SAE J444, and the shot numbers range from S70 to S1320. The shot number is approximately the same as the diameter of the individual pellets expressed in ten thousandths of an inch. Cast steel shot is the most widely used peening medium. It has a useful life many times that of cast iron, and causes less wear and tear on the components of peening machines. Cast iron shot is used in peening operations requiring low initial cost. Where contamination with iron is not desirable — as in the peening of stainless steel, titanium, aluminum, and magnesium — glass beads are employed.

17-7 TOOLING

17-7.1 GENERAL

Tooling for helicopter manufacture is the responsibility of the manufacturer and, except as may be defined in the contract with regard to Government-furnished tools, is unique to the manufacturing facilities of the manufacturer. In any event, configuration control will be in accordance with MIL-STD-480.

Tooling is a significant element of helicopter manufacturing cost, and the more stringent the manufacturing tolerances, the more costly the tooling. Consequently, it is imperative that the tolerances specified by the designer be kept in perspective.

In modern production work, where mating parts are manufactured in different departments or by different contractors, some method is necessary for producing these parts so that they will fit correctly in the final assembly. Appropriate standards include MIL-STD-100, ANSI B4.1-1967, and ANSI Y14.5-1966.

As with airframe design, tooling design must be performed in accordance with standard practices and

TABLE 17-6. THE EFFECT OF SHOT PEENING ON THE FATIGUE PROPERTIES OF SELECTED SAMPLES

SAMPLE	STRENGTH GAIN BY PEENING, %
PLAIN ALUMINUM 2014 – T6 ROUND BAR	23
PLAIN ALUMINUM 2024 – T4 ROUND BAR	34
PLAIN ALUMINUM 7079 – T6 ROUND BAR	30
SPRING STEEL 5160 FLAT LEAF	51
PLAIN STEEL 1045 POLISHED	10
SINGLE GEAR TOOTH 4118 R_c60	29
S–11 STEEL – GROOVED	54
0.54 % C STEEL – V NOTCHED	73
4340 STEEL, POLISHED ROUND BAR	150

procedures. Tool design standards, prepared by the manufacturer, should include design drafting practices, design and shop techniques, standardized tool specifications, types of materials, standards of material strength and dimensions, and tool production and qualification processes.

For most projects, tooling will fall into two stages: prototype tooling and production tooling. When the design of a helicopter is released for manufacture, drawings and specifications for all of the parts and assemblies are used for tool planning and tool design. Tool engineers consider the number of units to be produced, the required rate of production, the equipment and resources of the plant, and contractual limitations, if any, on tool costs. When all factors have been weighed, the tooling plan will be defined. The production planners then can break down the manufacturing processes for each part into individual operations.

The helicopter will have been designed by engineers who are concerned primarily with the proper functioning of each part, although they will have kept in mind the factors of producibility and economy. The tool engineer, familiar with the manufacturing facilities and the tool stockpile, first will consider redesign of the part for easier or more economical manufacture. New material compositions or configurations, new manufacturing processes, and new tooling sequences are possible. For instance, a casting might be replaced by stamped sheet metal, requiring a forming die. Often, a suitable tool can effect economy in manufacture by reducing the amount of material scrapped.

When the machine sequence has been determined, the individual operations are listed. An operation consists of all of the work that can be done at one setup, or station. The operations are planned in an order that will reduce the number of special tools to a minimum. Thus, it is better to design dies for multiple operation on a single press than to require individual operations on a number of punch presses. The same multiple-use capability is desirable for jigs and fixtures.

Once an operation is listed, the tool to be designed is determined from the description of the given operation, the machine to be employed, and a set of detailed drawings of the part to be made. This tool — complete with assembly drawings, subassemblies, part details, and specifications — becomes an element of configuration control for the helicopter.

For both prototype construction and production, there are three broad categories of tooling: shop tooling, airframe tooling, and test tooling. The latter two categories may, in turn, require shop tooling.

17-7.2 SHOP TOOLING

Cutting operations in the shop are performed, for the most part, with standard tools or with expendable tools that do not become part of contract inventory. On occasion, however, special bar tools may be required for boring, reaming, recessing, grooving, undercutting, and similar operations. Most of the special tools for shop use are jigs and fixtures for holding the work piece and guiding the work performance.

17-7.3 AIRFRAME TOOLING

Fabrication of helicopters requires the transfer of numerous dimensions from an engineering drawing or prototype to the item being built. Where the number of items to be built is small, and the performance requirements not too great, the structure can be produced economically by extracting dimensions directly from a blueprint. However, for the production of large numbers of airframes, highly precise tools have been developed for transferring the configurations of nondimensional, lofted, full-size drawings to the actual structure, even though portions of that structure are manufactured by many different manufacturers. The five primary types of tools employed are:

1. Master tools, designed so that dimensions for the entire helicopter can be referenced to several master tools. Each tool is the three-dimensional representation of a key portion of the system.

2. Templates, a thin plate of metal or other suitable material that may be used as a guide or pattern. A template generally defines the profile, contour, or layout of holes; the bend lines of a part, or an assembly layout of several parts.

3. Optical tooling, a system of tools constructed with alignment telescopes at one end and targets at the other end. With the use of such line-of-sight tools — together with master gages, tooling bars, increment bars, optical micrometers, and a transit — the configuration of a system can be controlled from a single datum line in six degrees of freedom.

4. Jigs and fixtures, coordinated to the master tools, position and hold the detail parts in their relationship for drilling and fastening. Smaller or subassembly components are fabricated in fixtures that are coordinated to larger or main assembly fixtures. The large components then are loaded into joining or final assembly fixtures to complete the assembly of the airframe.

5. Plastic Tooling. The number of parts to be made and the material the parts are made from are determining factors in the type of tooling to be used. There are three common elements used in plastic tooling:

a. Mock-ups:
    (1) wooden
    (2) plaster filled grids
b. Molds:
    (1) high temperature epoxy
    (2) low temperature wood
    (3) matched
    (4) aluminum shell
c. Trim tools.

Special tools in these classifications may be designed for inspection, or for the application to testing of fabricated parts or systems using the test tools described subsequently.

Basic decisions as to tooling design must be made concurrently with design of the airframe. Tooling drawings will be initiated as soon as a flow chart and production breakdown are possible; the basic tool philosophy and procedures must be established at this time. The tooling system *shall* be designed to allow maximum latitude for airframe design changes while minimizing the need for redesign of tooling. Continuity of datum points and coordinate reference lines and planes must be retained at all times.

In most cases, the tooling processes employed will involve all five types of tooling discussed, but will rely more heavily upon one type for configuration control and quality assurance. The quality assurance program will be determined almost completely by the processes selected since the same systems are employed for inspection as for production.

Considerable variation can be introduced when numerical control lofting is employed. In this case, the engineering data are transmitted in mathematical form. Automatically programmed tools are possible, with a computer working from, or producing, numerical-control drawings.

Provision must be made for special situations. For example, because of the small cross sections and the extremely close tolerances required in the manufacture of rotor blade components, normal lofting practice may not be followed. These components may be fabricated in closely dimensioned detail. Rib and other blade components may be lofted actual size and provided with appropriately toleranced dimensions for inspection purposes.

When justified by the number of units and the number of subcontractors, the manufacturing and assembly processes may be built around the use of master tools. In this case, quality assurance can be improved and simplified because master tools eliminate individual interpretations of engineering values and permit duplication of close tolerances. Further, by having a control master as the dimensional authority, duplicate masters can be built for use by subcontractors. Protection also is provided against the loss of dimensional control should master tools be damaged.

17-7.4 TEST TOOLING

Most physical, mechanical, thermal, and electrical testing will be accomplished with standard input and readout equipment, and by use of expendable apparatus such as strain gages and resistance strips. However, some of the qualification requirements for a helicopter, and its assemblies or subassemblies, require testing in a configuration or in an environmental or fatigue condition that is not attainable with standard test equipment. Such tests, for example, may become necessary to design and fabricate test fixtures, test stands, load input equipment, and readout equipment peculiar to the helicopter system that is to be manufactured. Such special testing might involve the twisting loads on the helicopter fuselage, vibration inputs in specified attitudes and types of suspension, fatigue of assemblies, and determination of rotor stability characteristics.

Design and construction of special test tooling must be initiated as soon as the final configuration and the qualification specification have been defined. Verification of the test tools and test tool procedures must precede their application to the evaluation and qualification of the helicopter and its subassemblies and assemblies.

REFERENCES

1. *Metal Forming, ASM Handbook*. Vol. 5, American Society for Metals, Cleveland, OH.
2. *Welding Handbook*. American Welding Society, NY.
3. *SAE Drawing Standard Manual*. Society of Automotive Engineers, NY, 1970.
4. *Heat Treatment, ASM Handbook*. Vol. 2, American Society for Metals, Cleveland, OH.

APPENDIX A

EXAMPLE OF A PRELIMINARY HEATING, COOLING, AND VENTILATION ANALYSIS

A-1 HEATING AND VENTILATION ANALYSIS

The following is an example of a heating and ventilating system.

A-1.1 DESIGN REQUIREMENTS

1. The heating system *shall* be capable of maintaining a temperature of 60°F in occupied spaces where the outside air temperature is −65°F or above.
2. The ventilating system *shall* be capable of delivering no less than 2.25 lb of fresh air per min to each occupant.
3. The surface temperature of the ducts within reach of operating personnel *shall not* be in excess of 180°F.

A-1.2 DESIGN ASSUMPTIONS

1. The number of occupants in the aircraft is 33: 3 in the cockpit and 30 in the cabin. Metabolic heat rate for a seated person (writing) is 400 Btu/hr per occupant.
2. Heat gain due to solar radiation at −65°F is negligible.
3. Outside air infiltration rates are: cockpit, 100 cfm; and cabin, 300 cfm.
4. Humidity effect at −65°F is negligible.
5. The heating system heat loss is equivalent to 20 deg F.
6. The blower volume flow rate is a constant.
7. Mechanical heat sources and fan work are negligible.
8. The cabin ceiling and upper 3.5 ft of the side walls are covered with 3 in. of insulation. The thermal coefficient U of the insulation is 0.07 Btu/hr-°F-ft^2.
9. Heat transfer coefficients:

Surface	U, Btu/hr-°F-ft^2
Transparent areas	1.69
Floor	0.7
Uninsulated wall	1.85
Insulated wall	0.07

10. Electrical equipment uses 0.225 kVA
11. Air: ρ = 0.10 lb/ft^3, c_p = 0.24 Btu/lb-°F
12. Heat transfer areas:

Surface	Area, ft^2
Cockpit windshield	50.0
Cockpit skin	50.0
Cockpit floor	50.0
Cabin rear ramp (uninsulated)	60.0
Cabin windows	26.7
Cabin floor	225.0
Cabin walls (uninsulated)	180.0
Cabin walls (insulated)	259.0
Cabin ceiling (uninsulated)	105.0
Cabin ceiling (insulated)	225.0

A-1.3 HEAT LOSSES

A-1.3.1 Cockpit

The heat losses result from convection and infiltration, i.e.,

$$Q_{cockpit_{total}} = Q_{convection} + Q_{infiltration} \tag{A-1}$$

A-1.3.1.1 Convection

The heat losses are through the transparencies, uninsulated skin, and floor, i.e.,

$$Q_{convection} = q_{transparency} + q_{skin\,(uninsulated)} + q_{floor} \tag{A-2}$$

where

$$\begin{aligned} q &= UA\Delta T \qquad \text{(A-3)} \\ &= (\text{Btu/hr-°F-ft}^2) \bullet (\text{ft}^2) \bullet (\text{°F}) = \text{Btu/hr} \end{aligned}$$

$$q_{transparency} = (1.69)(50)[60 - (-65)] = 10{,}560 \text{ Btu/hr}$$

$$q_{skin\,(uninsulated)} = (1.85)(50)[60 - (-65)] = 11{,}560 \text{ Btu/hr}$$

$$q_{floor} = (0.7)(50)[60 - (-65)] = 4{,}380 \text{ Btu/hr}$$

By Eq. A-2, the cockpit total convection heat loss is:

$$Q_{convection} = 10{,}560 + 11{,}560 + 4{,}380 = 26{,}500 \text{ Btu/hr}$$

A-1.3.1.2 Infiltration

The heat loss resulting from infiltration is:

$$Q_{infiltration} = c_p W \Delta T \quad \text{(A-4)}$$
$$= (\text{Btu/lb-°F}) \cdot (\text{lb/hr}) \cdot (\text{°F}) = \text{Btu/hr}$$

Based on the given infiltration rate of 100 cfm into the cockpit, the resulting pounds of air W are:

$$W = (\text{ft}^3/\text{min}) \cdot (\text{lb/ft}^3) \cdot (\text{min/hr}) = \text{lb/hr}$$
$$= (100)(0.1)(60) = 600 \text{ lb/hr}$$

Therefore, by Eq. A-4:

$$Q_{infiltration} = (0.24)(600)[60 - (-65)] = 18{,}000 \text{ Btu/hr}$$

A-1.3.1.3 Total Cockpit Heat Loss

By Eq. A-1, the total heat loss is

$$Q_{cockpit_{total}} = 26{,}500 + 18{,}000 = 44{,}500 \text{ Btu/hr}$$

A-1.3.2 Cabin

The heat losses result from convection and infiltration, i.e.,

$$Q_{cabin_{total}} = Q_{convection} + Q_{infiltration} \quad \text{(A-5)}$$

A-1.3.2.1 Convection

The heat losses are through the ramp, transparencies, floor, walls, and ceiling, i.e.,

$$\begin{aligned} Q_{convection} = {} & q_{rear\ ramp\ (uninsulated)} + q_{transparency} \\ & + q_{floor} + q_{uninsulated\ wall} \\ & + q_{insulated\ wall} \\ & + q_{uninsulated\ ceiling\ over\ ramp} \\ & + q_{insulated\ ceiling} \end{aligned} \quad \text{(A-6)}$$

Therefore, by Eq. A-3

$$\begin{aligned} q_{rear\ ramp\ (uninsulated)} &= (1.85)(60)[60 - (-65)] \\ &= 13{,}900 \text{ Btu/hr} \\ q_{transparency} &= (1.69)(26.7)[60 - (-65)] \\ &= 5{,}640 \text{ Btu/hr} \\ q_{floor} &= (0.7)(225)[60 - (-65)] \\ &= 19{,}690 \text{ Btu/hr} \\ q_{uninsulated\ wall} &= (1.85)(180)[60 - (-65)] \\ &= 41{,}630 \text{ Btu/hr} \\ q_{insulated\ wall} &= (0.07)(259)[60 - (-65)] \\ &= 2{,}270 \text{ Btu/hr} \\ q_{uninsulated\ ceiling\ over\ ramp} &= (1.85)(105)[60 - (-65)] \\ &= 24{,}280 \text{ Btu/hr} \\ q_{insulated\ ceiling} &= (0.07)(225)[60 - (-65)] \\ &= 1{,}970 \text{ Btu/hr} \end{aligned}$$

By Eq. A-6, the cabin total convection heat loss is

$$\begin{aligned} Q_{convection} &= 13{,}900 + 5{,}640 + 19{,}690 + 41{,}630 \\ &\quad + 2{,}270 + 24{,}280 + 1{,}970 \\ &= 109{,}380 \text{ Btu/hr} \end{aligned}$$

A-1.3.2.2 Infiltration

The heat loss resulting from infiltration is by Eq. A-4.

$$Q_{infiltration} = c_p W \Delta T$$

Since the infiltration rate is given as 300 cfm into the cabin, the resulting pounds of air W are:

$$W = (\text{ft}^3/\text{min}) \cdot (\text{lb/ft}^3) \cdot (\text{min/hr}) = \text{lb/hr}$$
$$= (300)(0.1)(60) = 1{,}800 \text{ lb/hr}$$

Therefore, by Eq. A-4

$$Q_{infiltration} = (0.24)(1{,}800)[60 - (-65)]$$
$$= 54{,}000 \text{ Btu/hr}$$

A-1.3.2.3 Total Cabin Heat Loss

By Eq. A-5, the total heat loss is

$$Q_{cabin_{total}} = 109{,}380 + 54{,}000 = 163{,}380 \text{ Btu/hr}$$

A-1.4 VENTILATING AIR REQUIRED

A-1.4.1 Based on Number of Occupants and Minimum Ventilating Rate

The ventilating air requirements are:

$$W_{a_{total}} = W_{a_{cockpit}} + W_{a_{cabin}} \quad \text{(A-7)}$$

The weight W_a of the air required is determined by:

$$W_a = (\text{lb/min-occupant}) \cdot (\text{occupant}) \quad \text{(A-8)}$$
$$= \text{lb/min}$$

Therefore, based on the given conditions of 3 cockpit and 30 cabin occupants each requiring 2.25 lb of fresh air per min:

$$W_{a_{cockpit}} = (2.25)(3) = 6.75 \text{ lb/min}$$
$$W_{a_{cabin}} = (2.25)(30) = 67.5 \text{ lb/min}$$

By Eq. A-7, the total minimum ventilation requirement is:

$$W_{a_{total}} = 6.75 + 67.5 = 74.25 \text{ lb/min}$$

A-1.4.2 Requirement Based on Maximum Allowable Temperature Difference

Since the surface temperature of the ducts cannot exceed 180°F, the maximum allowable temperature difference ΔT in occupied areas is:

$$\Delta T = 180 - 60 = 120 \text{ deg F}$$

Thus it is necessary to determine if this allowable ΔT is sufficient to satisfy the cockpit and cabin heat losses based on a circulation demanded by minimum ventilation requirements.

A-1.4.2.1 Cockpit Requirement

The required cockpit ΔT is

$$\Delta T_{req_{cockpit}} = Q_{cockpit_{total}}/(c_p W_{a_{cockpit}}) = \text{deg F} \tag{A-9}$$

Use the heat loss of 44,500 Btu/hr from par. A-1.3.1.3.

$$\Delta T_{req_{cockpit}} = 44{,}500/[(0.24)(6.75)(60)] = 458 \text{ deg F}$$

Since the 458-deg F temperature difference exceeds the allowable ΔT = 180 deg F, the airflow to the cockpit must exceed the minimum required for ventilation. The required amount of cockpit air based on the allowable ΔT = 120 deg F is

$$\begin{aligned} W_{a_{cockpit}} &= Q_{cockpit}/(c_p \Delta T) = \text{lb/hr} \qquad \text{(A-10)} \\ &= 44{,}500/[(0.24)(120)] \\ &= 1{,}545 \text{ lb/hr} = 25.8 \text{ lb/min} \end{aligned}$$

A-1.4.2.2 Cabin Requirement

Similarly, use the heat loss of 163,380 Btu/hr from par. A-1.3.2.3.

$$\Delta T_{req_{cabin}} = 163{,}380/[(0.24)(67.5)(60)] = 168 \text{ deg F}$$

Since the 168-deg F temperature difference also exceeds the allowable limit, the airflow to the cabin must exceed the minimum required for ventilation. The required amount of cabin air based on the allowable ΔT = 180 deg F is

$$\begin{aligned} W_{a_{cabin}} &= 163{,}380/[(0.24)(120)] \\ &= 5{,}670 \text{ Btu/hr} = 94.5 \text{ lb/min} \end{aligned}$$

A-1.4.2.3 Total Air Requirement

By Eq. A-7, the total ventilation requirement based on the allowable ΔT = 120 deg F is

$$W_{a_{total}} = 25.8 + 94.5 = 120.3 \text{ lb/min}$$

A-1.4.3 Total Heat Requirement

Since the system heating loss is given as 20 deg F, the total temperature difference between the outside air (−65°F) and the heating ducts (180°F) is:

$$\Delta T_{total} = 180 - (-65) + 20 = 265 \text{ deg F}$$

Accordingly, the total heat required is:

$$\begin{aligned} Q_{total} &= (0.24)[(120.3)(60)](265) \\ &= 459{,}100 \text{ Btu/hr} \end{aligned}$$

A-1.5 HEATER REQUIREMENTS

The net heat requirement to be supplied by the heater is the difference between that required and that gained from the occupants and electrical equipment, i.e.,

$$Q_{req} = Q_{total} - Q_{gain} \tag{A-11}$$

A-1.5.1 Heat Gained

The occupants and the electrical equipment are responsible for the heat gain, i.e.,

$$Q_{gain} = Q_{occupants} + Q_{electrical} \tag{A-12}$$

The heat generated by the 33 occupants based on the given metabolic heat rate of 400 Btu/hr is:

$$\begin{aligned} Q_{occupants} &= (\text{Btu/hr-occupant}) \bullet (\text{occupant}) \\ &= \text{Btu/hr} \\ &= (33)(400) = 13{,}200 \text{ Btu/hr} \end{aligned}$$

Since the electrical system power consumption is 0.225 kVA, the equivalent heat is:

$$\begin{aligned} Q_{electrical} &= (\text{kVA}) \bullet (\text{Btu/hr} - \text{kVA}) = \text{Btu/hr} \\ &= (0.225)(3{,}413) = 768 \text{ Btu/hr} \end{aligned}$$

By Eq. A-12, the total heat gained is:

$$Q_{gain} = 13{,}200 + 768 = 13{,}968 \text{ Btu/hr}$$

A-1.5.2 Net Heat Required

Since the total heat required by par. A-1.4.3 was 459,100 Btu/hr, by Eq. A-11 the net heat required is

$$Q_{req} = 459{,}100 - 13{,}968 = 445{,}132 \text{ Btu/hr}$$

A-1.5.3 Heater Size

The nearest available heater size to satisfy the heating requirement of 445,132 Btu/hr is a 600,000-Btu/hr heater.

A-1.6 BLOWER SIZE

The blower must provide air for both ventilation and combustion of fuel to heat the air, i.e.,

$$W_{a_{total}} = W_{a_{ventilation}} + W_{a_{combustion}} \qquad \text{(A-13)}$$

A-1.6.1 Volume of Air to be Delivered

The weight of ventilating air from par. A-1.4.2.3 is 120.2 lb/min.

To determine the combustion airflow, assume:

1. Heat transfer efficiency $\eta = 0.65$
2. Fuel with higher heating value $HHV = 18{,}400$ Btu/lb
3. Fuel-air ratio $W_f/W_a = 1/15$.

Then the airflow required for the 445,132 Btu/hr output is:

$$\begin{aligned} W_{a_{combustion}} &= Q_{req}/[(HHV)(W_f/W_a)\eta] \qquad \text{(A-14)} \\ &= (\text{Btu/hr})/[(\text{Btu/lb}) \cdot (W_f/W_a)\eta] = \text{lb/hr} \\ &= (445{,}132)/[(18{,}400)(1/15)(0.65)] \\ &= 558 \text{ lb/hr} = 9.3 \text{ lb/min} \end{aligned}$$

By Eq. A-13, the total airflow required is:

$$W_{a_{total}} = 120.3 + 9.3 = 129.6 \text{ lb/min}$$

or in terms of cfm

$$\begin{aligned} W_{a_{total}} &= 129.6/\rho = 129.6/0.1 \\ &= 1{,}296 \text{ cfm} \end{aligned}$$

A-1.6.2 Pressure Drop

Assume the following pressure losses:

1. Per foot of straight duct: 0.05-in. of water
2. Per 90-deg elbow: 1.0-in. of water
3. Fresh air inlet: 2.0-in. of water
4. Across the heater: 2.0-in. of water.

A blower with a pressure Δp rise of 13-in. of water is required, applying these unit losses.

A-2 COOLING AND VENTILATING ANALYSIS

The following is an example of a cooling and ventilation analysis.

A-2.1 DESIGN REQUIREMENTS

The design condition is to maintain 90°F, 40% relative humidity (RH) maximum during a MIL-STD-210 hot day (103°F and 95% RH) in the cockpit only.

A-2.2 DESIGN ASSUMPTIONS

For solar radiation effects it is assumed that the helicopter is heading due south at 1400 hours in the afternoon of August 1. Other considerations remain the same as for the heating analysis. No heat effect due to mechanical sources will be considered.

A-2.3 DETERMINATION OF EFFECTIVE TEMPERATURE DIFFERENCES ASSOCIATED WITH VARIOUS SURFACES OF THE HELICOPTER

A-2.3.1 Effective Solar Temperatures

The outside surface temperature is dependent upon the outside heat transfer film coefficient f_o, the portion a of solar radiation I which is absorbed, and the outside ambient temperature T_o. The relationship for this effective surface temperature t_s (solar temperature) as given in Ref. A-1 is:

$$t_s = T_o + aI/f_o, \quad ^\circ\text{F} \qquad \text{(A-15)}$$

where

a = fraction of solar radiation absorbed, dimensionless

f_o = heat transfer coefficient (film), Btu/hr-ft²-°F

I = solar radiation, Btu/hr-ft²

For this case, $f_o = 13$, $T_o = 103$, and $a = 0.8$; therefore, from Eq. A-15:

$$t_s = 103 + 0.8I/13 = 103 + 0.061I, ^\circ\text{F} \qquad \text{(A-16)}$$

The temperature for each surface of the helicopter depends upon the sun angle, which affects I. The following values of I_i are taken from Ref. A-2 for the conditions of this example:

Surface	I_i, Btu/hr-ft²
$I_{East,\ Vertical}$	25
$I_{West,\ Vertical}$	160
$I_{North,\ Vertical}$	16
$I_{South,\ Vertical}$	105
$I_{Horizontal,\ Top}$	250
I_{Bottom}	16

From these I's, the following t_{si} are obtained from Eq. A-16:

t_{si}	Temperature, °F
t_{si_E}	104.5
t_{si_W}	113
t_{si_N}	104
t_{si_S}	109.5
t_{si_H}	118
$t_{si_{Bot}}$	104

A-2.3.2 Effective ΔT's

The ΔT's to be used for calculations are determined from Eq. A-17.

$$\Delta T = t_{si} - T_{cockpit}, \text{ deg F} \tag{A-17}$$

or since $T_{cockpit} = 90°\text{F}$,

$$\Delta T = t_{si} - 90, \text{ deg F}$$

The resulting ΔT_i follow:

ΔT_i	Temperature Difference, deg F
ΔT_E	14.5
ΔT_W	23
ΔT_N	14
ΔT_S	19.5
ΔT_{Hor}	28
ΔT_{Bot}	14

A-2.4 COCKPIT HEAT GAINS

The principal sources of heat gain considered are:

1. Convection, infiltration, and solar radiation
2. Occupants
3. Electrical system.

A-2.4.1 Convection, Infiltration, and Solar Radiation

The heat gain can be expressed as:

$$\begin{aligned} Q_{cockpit} &= Q_{convection} + Q_{infiltration} \\ &\quad + Q_{solar} \\ &= U A_i \Delta T_i + c_p W \Delta T_i + A I_r, \text{ Btu/hr} \end{aligned} \tag{A-18}$$

where A_i is the projected area facing each of the ΔT_i directions.

Assume uninsulated "wall" areas of the following dimensions:

A_i	Area, ft²
A_E	20
A_W	20
A_N	0
A_S	20
A_{Hor}	8
A_{Bot}	60

Assume transparent areas of the following dimensions:

A'_i	Area, ft²
A'_E	17.9
A'_W	17.9
A'_S	16.0
A'_{Hor}	12.8

Calculation of the heat gains based on the given data follow.

A-2.4.1.1 Convection Gains

The heat gains are through the walls (skin), floor, and transparent areas, i.e.,

$$\begin{aligned} Q_{convection} &= q_{walls} + q_{floor} \\ &\quad + q_{transparent\ areas} \end{aligned} \tag{A-19}$$

where

$$\begin{aligned} q_{walls} &= q_E + q_W + q_N + q_S + q_{Hor} \\ q_{floor} &= q_{Bot} \\ q_{transparent\ areas} &= q'_E + q'_W + q'_S + q'_{Hor} \end{aligned}$$

By Eq. A-3, where $U = 1.85$ for uninsulated walls,

$$\begin{aligned} q_E &= (1.85)(20)(14.5) = 537 \text{ Btu/hr} \\ q_W &= (1.85)(20)(23) = 851 \text{ Btu/hr} \\ q_N &= (1.85)(0)(14) = 0 \\ q_S &= (1.85)(20)(19.5) = 721 \text{ Btu/hr} \\ q_{Hor} &= (1.85)(8)(28) = 414 \text{ Btu/hr} \end{aligned}$$

and

$$\begin{aligned} q_{walls} &= 537 + 851 + 721 + 414 \\ &= 2{,}523 \text{ Btu/hr} \end{aligned}$$

Also by Eq. A-3, where $U = 0.7$ for floors,

$$q_{floor} = (0.7)(60)(14) = 588 \text{ Btu/hr}$$

Also by Eq. A-3, where $U = 1.69$ for transparent areas,

$$
\begin{aligned}
q'_E &= (1.69)(17.9)(14.5) = 439 \text{ Btu/hr}\\
q'_W &= (1.69)(17.9)(23) = 696 \text{ Btu/hr}\\
q'_S &= (1.69)(16)(19.5) = 527 \text{ Btu/hr}\\
q'_{Hat} &= (1.69)12.8)(28) = 606 \text{ Btu/hr}
\end{aligned}
$$

and

$$q_{transparent\ areas} = 439 + 696 + 527 + 606 = 2,268 \text{ Btu/hr}$$

Therefore total heat gained by convection (Eq. A-19) is

$$
\begin{aligned}
Q_{convection} &= 2,523 + 588 + 2,268\\
&= 5,379 \text{ Btu/hr}
\end{aligned}
$$

A-2.4.1.2 Infiltration Gain

By Eq. A-4, and the design assumptions and requirements of pars. A-1.2 and A-2.2,

$$
\begin{aligned}
Q_{infiltration} &= (0.24)(100)(0.1)(60)(103 - 90)\\
&= 1,872 \text{ Btu/hr}
\end{aligned}
$$

A-2.4.1.3 Solar Radiation Gains

The heat gain from solar radiation is determined by Eq. A-20.

$$Q_{solar} = A I I_t, \text{ Btu/hr} \tag{A-20}$$

The values for the transparent areas $A I$ are given in par. A-2.4.1; the values for solar radiation I_t in par. A-2.3.1.

$$Q_{solar_{total}} = q_{E_{solar}} + q_{W_{solar}} + q_{N_{solar}} + q_{S_{solar}} + q_{Hat_{solar}} \tag{A-21}$$

$$
\begin{aligned}
q_{E_{solar}} &= (17.9)(25) = 448 \text{ Btu/hr}\\
q_{W_{solar}} &= (17.9)(160) = 2,864 \text{ Btu/hr}\\
q_{N_{solar}} &= 0\\
q_{S_{solar}} &= (16)(105) = 1,680 \text{ Btu/hr}\\
q_{Hat_{solar}} &= (12.8)(250) = 3,200 \text{ Btu/hr}
\end{aligned}
$$

Therefore, by Eq. A-21 the total solar radiation is:

$$
\begin{aligned}
Q_{solar_{total}} &= 448 + 2,864 + 1,680 + 3,200\\
&= 8,192 \text{ Btu/hr}
\end{aligned}
$$

A-2.4.1.4 Total Heat Gain Due to Convection, Infiltration, and Solar Radiation

By Eq. A-18 the total heat gain is:

$$
\begin{aligned}
Q_{cockpit} &= 5,379 + 1,872 + 8,192\\
&= 15,443 \text{ Btu/hr}
\end{aligned}
$$

A-2.4.2 Occupants

The heat generated by the 3 cockpit occupants based on the given metabolic heat rate of 400 Btu/hr is:

$$Q_{occupant} = (3)(400) = 1,200 \text{ Btu/hr}$$

A-2.4.3 Electrical System

The $Q_{electric}$, from par. A-1.3.1, is 768 Btu/hr.

A-2.4.4 Total Cockpit Heat Gain

The total heat gain by Eq. A-18 is:

$$Q_{cockpit_{total}} = 15,443 + 1,200 + 768 = 17,411 \text{ Btu/hr}$$

A-2.5 AIR CONDITIONER SIZE

In this paragraph conditioning of the air for ventilation, fan heat and size, and pressure against which the fan must operate are considered in order to determine the size of the air conditioning equipment required.

A-2.5.1 Conditioning of Ventilation Air

In the example chosen fresh air is at 103°F, 95% RH. Since ventilation fans cool by using ambient air only, the design requirement of 90°F, 40% RH, cannot be met. Hence, air conditioning is necessary. Conditioning must provide for removal of the sensible heat of the air and the heat of condensation of the water.

The sensible heat removed from the air based on the 2.25 lb/min-occupant ventilation rate and 3 occupants follows:

$$
\begin{aligned}
W_{a_{ventilation}} &= (2.25)(3) = 6.75 \text{ lb/min}\\
Q_{vent\ air} &= c_p W \Delta T\\
&= (0.24)(6.75)(60)(103 - 90)\\
&= 1,264 \text{ Btu/hr}
\end{aligned}
$$

The amount of water removed from the air by reducing the RH from 95% to 40% may be determined from a psychrometric chart, i.e.,

water in 103°F, 95% RH air = 0.045 lb-water/lb-air
water in 90°F, 40% RH air = 0.012 lb-water/lb-air
water removed = 0.033 lb-water/lb-air

Since the heat of condensation of water is 1,037 Btu/lb-water, the total heat of condensation is:

$$Q_{condensation} = W_{a_{ventilation}}(\text{lb-water/lb-air}) \times (\textit{heat of condensation}) \tag{A-22}$$

$$
\begin{aligned}
&= [(6.75)(60)](0.033)(1,037)\\
&= 13,860 \text{ Btu/hr}
\end{aligned}
$$

The heat removed in cooling the water vapor or water from 103° to 90°F is very small and can be neglected.

A-2.5.2 Fan Size and Heat

Ventilation fans and any other fans moving air into the occupied space cool their own motors with the air they are moving. Many times, ventilation fans are installed in an exhaust mode. When this occurs, no heat is gained by the occupied space and fan heat is neglected. Air conditioners may be driven by compressor bleed air from either the engine or an APU and no direct heat is added to the occupied space. If the air conditioner is driven electrically, heat will in general have to be added to the system. For the example case, assume that:

1. Ventilation fans are on the input side, bringing fresh air into the cockpit at a rate of 6.75 lb/min.
2. Ventilation fans are moving the rest of the cooled air at a speed $v = 30$ ft/min.
3. Cross-sectional area A for the cockpit is 32 ft.²

The volume V_a of air to be moved is:

$$\begin{aligned} \bar{V}_a &= Av && \text{(A-23)} \\ &= (\text{ft}^2) \circ [(\text{ft/min}) \bullet (\text{min/hr})] = \text{ft}^3/\text{hr} \\ &= (32)[(30)(60)] = 57{,}600\ \text{ft}^3/\text{hr} \end{aligned}$$

In addition, the following amount of fresh air is introduced for ventilation (the density ρ of air at 103°F, 95% RH, is 0.064 lb/ft³):

$$\begin{aligned} V_a^{\circ} &= W_a/\rho && \text{(A-24)} \\ &= [(\text{lb/min}) \circ (\text{min/hr})] \bullet [1/(\text{lb/ft}^3)] \\ &= \text{ft}^3/\text{hr} \\ &= [(6.75)(60)](1/0.064) = 6{,}328\ \text{ft}^3/\text{hr} \end{aligned}$$

The total volume of air to be circulated by the fan is

$$V_{a_{total}} = 57{,}600 + 6{,}328 = 63{,}928\ \text{ft}^3/\text{hr}$$

Assume the following pressure losses:

1. Per foot of straight duct: 0.05-in. of water
2. Per 90-deg elbow: 1-in. of water
3. Across fresh air inlet: 2-in. of water
4. Across air conditioner: 2-in. of water.

Assume further that these unit losses result in a total loss in pressure $\Delta p = 13$-in. of water; also, the fan motor efficiency $\eta = 0.8$. The fan motor horsepower P is then

$$\begin{aligned} p &= V_{a_{total}}(\Delta P)/[(33{,}000)\eta] && \text{(A-25)} \\ &= (\text{ft}^3/\text{min}) \bullet (\text{lb/ft}^2)/(\text{ft-lb/hp-hr}) = \text{hp} \\ &= (63{,}928/60)[(13)(5.2)]/ \\ &\quad [(33{,}000)(0.8)] = 2.7\ \text{hp} \end{aligned}$$

since 1-in. of water = 5.2 lb/ft².

The equivalent heat of the fan motor is

$$Q_{motor} = (2.7)(2{,}545) = 6{,}872\ \text{Btu/hr}$$

since 1 hp = 2,545 Btu/hr.

A-2.5.3 Tons of Refrigeration Required

The total heat load to be removed from the cockpit to satisfy the design requirement — see pars. A-2.4.4, A-2.5.1, and A-2.5.2 — is:

$$\begin{aligned} Q_{refrig} &= 17{,}411 + 1{,}264 + 13{,}860 + 6{,}872 \\ &= 39{,}407\ \text{Btu/hr} \\ &= 39{,}407/12{,}000 = 3.3\ \text{tons refrigeration} \end{aligned}$$

Since 12,000 Btu/hr = 1 ton of refrigeration.

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