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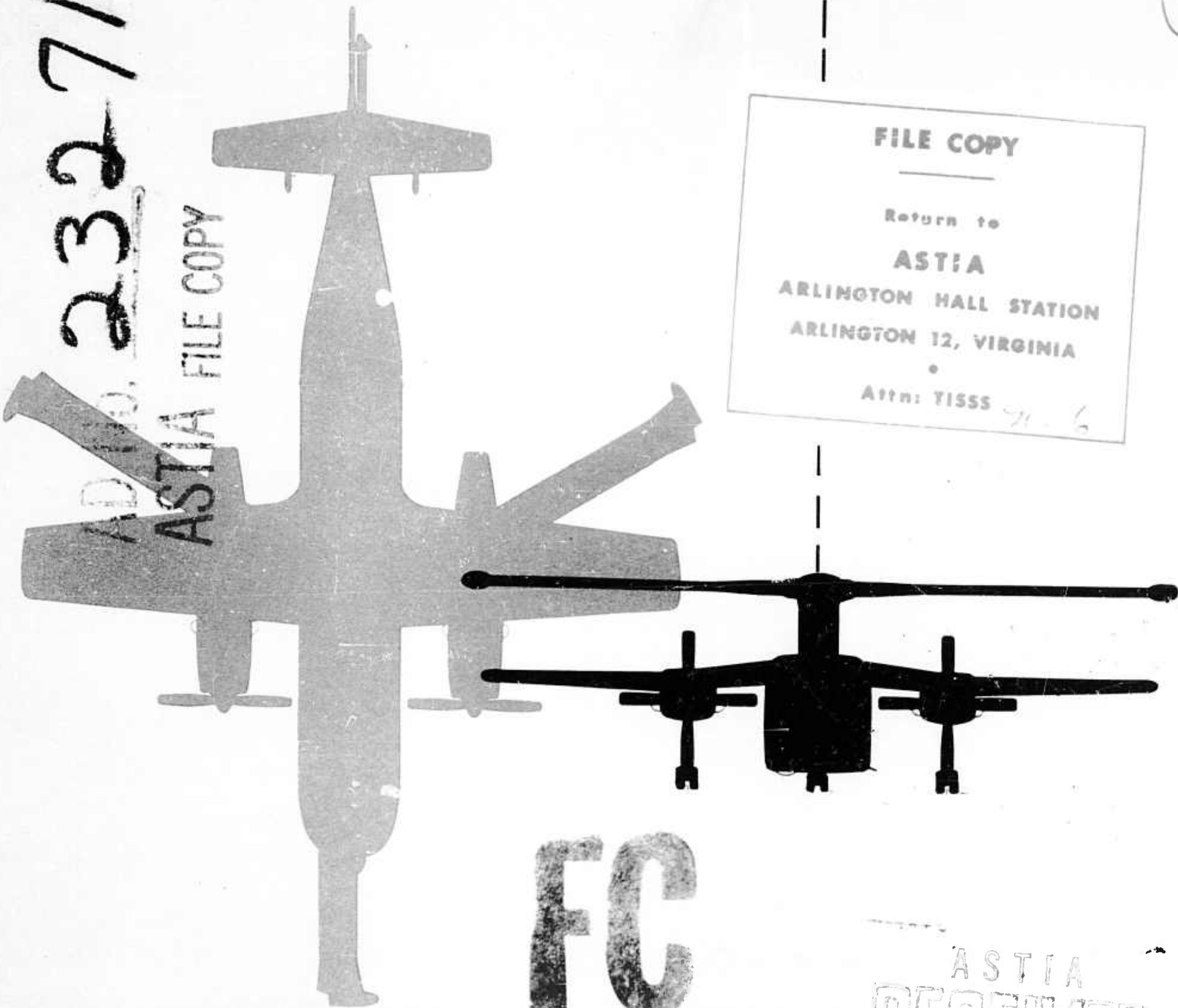

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REPORT NO. 7064 FEBRUARY 1, 1960

UNLOADED ROTOR COMPOUND HELICOPTER

LIGHT TRANSPORT AIRCRAFT STUDY

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DATE 1 FEBRUARY 1960

REVISED

UNLOADED ROTOR COMPOUND HELICOPTER

LIGHT VTOL TRANSPORT AIRCRAFT STUDY (U)

REPORT 7064 SERIAL NO.

McDONNELL Aircraft Corporation

LAMBERT-ST. LOUIS MUNICIPAL AIRPORT,, ST. LOUIS 3, MISSOURI

SUBMITTED UNDER

PROJECT NUMBER 9-38-04-000 TASK 510AV

TRECOM CONTRACT NO. DA 44-177-TC-588

U. S. ARMY TRANSPORTATION RESEARCH COMMAND, FORT EUSTIS, VIRGINIA

CRD 260

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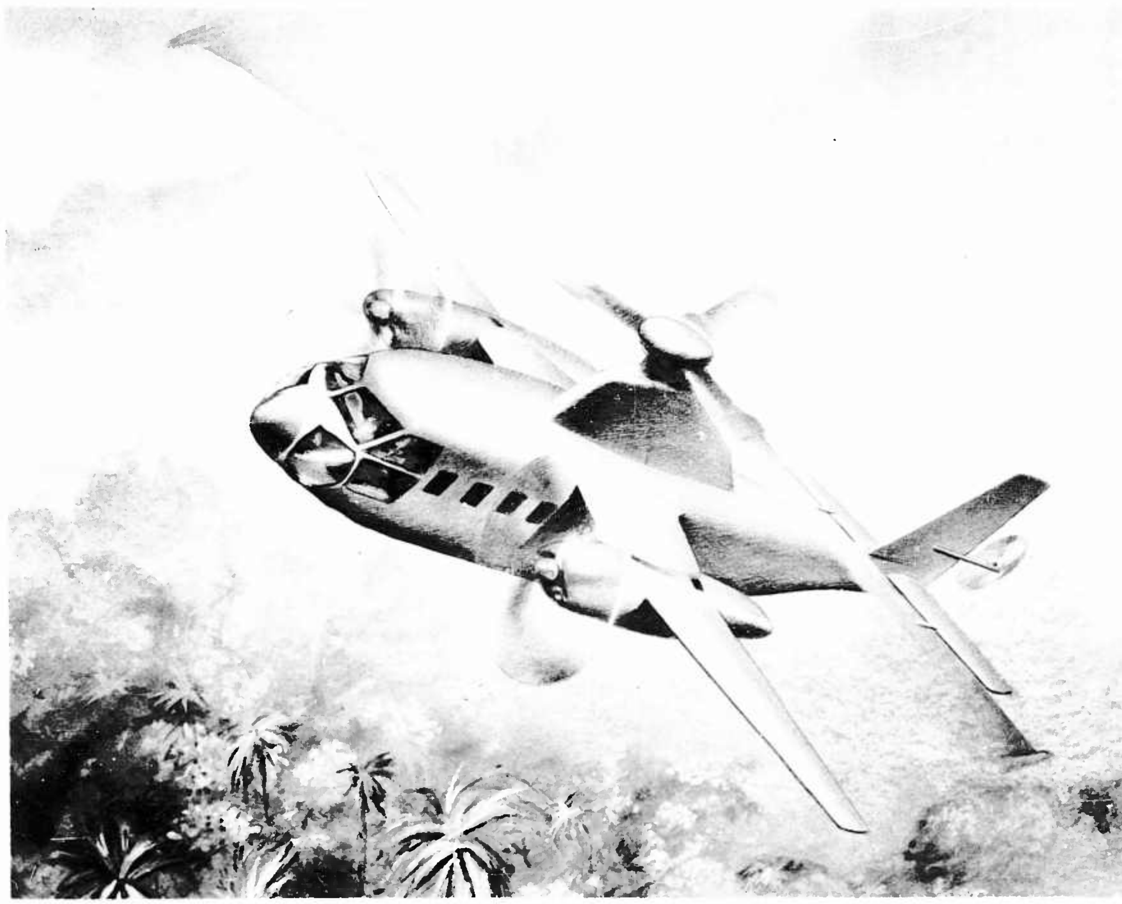


UNLOADED ROTOR COMPOUND HELICOPTER (XV-1 PRINCIPLE)

ARMY LIGHT VTOL TRANSPORT AIRCRAFT REQUIREMENTS

VTOL PAYLOAD :	4000 LB. OR MORE
HOVERING CEILING, OGE:	6,000 FT. , 95° F.
RADIUS OF ACTION :	200-500 NAUTICAL MILES
POWER PLANT :	TURBINES AVAILABLE 1960-1963
CONFIGURATION :	RAMP LOADING
DESIGN&CONSTRUCTION :	CURRENT DESIGN PRACTICE

XV-1

UNLOADED ROTOR COMPOUND HELICOPTER

IN

AIRPLANE FLIGHT (1956)

SECTION I

1. SUMMARY

A summary report of the McDonnell design study of the Army light VTOL transport aircraft incorporating the XV-1 principle is presented in fulfillment of Reference 12.1 contract.

A comparison is made of various VTOL concepts applicable to the Army light VTOL transport based on the design mission definitions of Reference 12.1 and the three parameters which establish usefulness and economy; the aerodynamic efficiency expressed as the lift to drag ratio, the structural efficiency expressed as the ratio of empty weight to design gross weight, and the propulsive efficiency expressed as the specific fuel consumption per thrust horsepower. This comparison indicates a large margin of superiority of the XV-1 type over either the pure helicopter or the turboprop VTOL.

The justification of component selection is given for the suggested VTOL aircraft which retains the basic unloaded rotor principle of the XV-1 with configuration improvements resulting from flight experience, development and test programs, and Model 113 design studies. The difference in mission requirements is also reflected in the proposed aircraft. The basic differences from the XV-1 configuration are the replacement of the reciprocating-pusher propeller system by a multiple gas turbine-tractor propeller system, replacement of the skid gear by a retractable tricycle gear, and replacement of the twin boom empennage by a conventional aft fuselage and empennage.

The recommended aircraft is of the fully unloaded rotor type with tip jet drive, combining good helicopter type handling characteristics in low speed flight with good airplane type handling characteristics in cruising flight where surface controls are used rather than rotor controls. The McDonnell rotor system incorporated in this aircraft provides for inherent dynamic stability about all axes in helicopter flight without the use of stabilizing bars or other devices to artificially introduce damping. These characteristics have been displayed in the XV-1 convertiplane and the McDonnell Model 120 helicopter.

The use of this rotor system with high solidity and small diameter permits the establishment of dynamic characteristics which eliminate ground resonance and mechanical instability. This feature and the absence of rotating propellers during low speed flight permits safe operation completely independent of the take-off and landing terrain.

In common with other helicopter type VTOL aircraft the selected version possesses safety characteristics of good autorotational capabilities, lift augmentation in ground proximity, and a source of rotational kinetic energy which can be used for partial power or power-off flares. In addition, one engine out performance data show a high level of safety and emergency mission completion, especially for the three- and four-engine aircraft; this is characteristic of the unloaded rotor helicopter.

A spectrum analysis of VTOL aircraft utilizing the XV-1 principle is presented and covers design gross weight variations from 15,000 to 60,000 pounds to show the size of the aircraft required to meet any specific payload-radius combination within the bracket defined in Reference 12.1 contract.

A 30,000-pound aircraft is recommended for the Army light transport. Three different engine installations are treated: four T58-GE-8, three T55-L-7, and two

T64-GE-2. With two T64 engines a 2-ton payload can be carried over a 340-nautical mile radius of action with a take-off gross weight at which hovering out of ground effect at 6000 feet 95°F is possible. If the hovering requirement is reduced to standard sea level conditions, the same aircraft can transport a payload of 4000 pounds over a radius of action of 925 nautical miles or a payload of 14,800 pounds over a radius of action of 250 nautical miles.

The unloaded rotor compound helicopter lends itself well to STOL operation. Theoretical analyses of STOL overload operations for the Model 113 show that the take-off weight for a running take-off to clear a 50-foot obstacle within 500 feet is about 5 percent greater than the maximum vertical take-off weight. These analyses agree with flight test data on overload capabilities of pure helicopters. It is also shown that when plotted against take-off altitude at a 250-nautical mile radius the payload increases by about 1500 pounds on a 95°F day for STOL operation and that the standard day capability is increased by about 1800 pounds. For sea level conditions the 95°F day STOL payload is 20 percent greater and the standard day payload is 15 percent greater than the VTOL payload. If runways are available for take-off without the 500-foot restriction, these payload increases can be approximately doubled.

The high cargo flow and resulting high productivity for the pressure jet driven unloaded compound helicopter stem from the combination of moderately high lift to drag ratio (L/D = 10) and the low ratio of empty weight to gross weight. This low ratio is primarily the result of the power plant weight saving achieved through the application of the pressure jet system which possesses an inherently low ratio of installed power to rotor power.

When the McDonnell rotor system is combined with tip jet drive, the maintenance is unusually low compared with conventional rotor systems. This is a result of low rotor speed operation in cruising flight; elimination of all bearings under centrifugal load; elimination of dampers, stabilizers, or other devices on the blades; and use of lubrication-free Teflon bearing surfaces in all oscillating bearings.

Total military direct operating cost including maintenance, fuel and oil, and crew costs but excluding depreciation is estimated at 181 dollars per flight hour. Development schedules and costs for five and ten prototypes and production costs are estimated for production rates of 25 and 100 aircraft per year.

SECTION II

2. INTRODUCTION

This final report presents the results of preliminary design studies of light VTOL Army transport aircraft performed under contract to TRECOM dated 30 June 1959 (Reference 12.1). The statement of work in the contract calls for studies of VTOL transport aircraft with a VTOL payload of approximately but not less than 2 tons at 6000 feet 95°F out of ground effect hovering and with a radius of action from 200 to 500 nautical miles. The aircraft are to incorporate, where applicable, the features and principles of the McDonnell XV-1 research aircraft. The power plants are to be selected from those expected to be available in the 1960-1963 time period. Two alternate cargo compartment sizes are to be studied: height, width, and length, respectively, of 78, 72, and 288 inches and of 78, 96, and 360 inches. The aircraft performance is to be determined for various mission profiles and also for conditions less stringent than the Army hot day.

In addition to VTOL performance studies, the statement of work calls for a discussion of STOL and ferrying capabilities; stability and control characteristics including transition techniques; and power system characteristics with respect to mechanical complexity, power losses, cooling requirements, vibrations, service life, system weight, and safety. Finally the contract calls for recommended military and technical specifications for this class of aircraft; a discussion of sacrifices associated with the selected criteria; and estimated development, production, maintenance, and operating costs. The contract emphasizes the inclusion in the final report of substantiating data, methods, and assumptions used in the preliminary design in such a way that a valid comparison with other VTOL designs can be derived.

In general, the methods used to establish aerodynamic and performance data and stability and control characteristics follow the material developed in Reference 12.2. Conventional helicopter and airplane methods of performance analysis are followed. Where applicable, all procedures and methods are modified to incorporate current practices and latest available information gained from wind tunnel data, NACA reports, etc. Stability and control characteristics are obtained primarily from wind tunnel test data for similar configurations and from full scale flight tests of the XV-1 convertiplane.

In the unloaded rotor principle explored by the XV-1, the lifting rotor, when relieved of its three functions to provide lift, propulsive force, and control, is capable of autorotating at advance ratios several times higher than those to which a pure helicopter is limited. An aircraft designed to this principle is capable of three distinct flight regimes. The first is helicopter flight utilizing accepted helicopter principles of lift generation and control. The second is autogyro flight with propellers supplying forward thrust and the rotor remaining at a relatively high rpm carrying about one-half of the lift with the wing carrying the remaining half. The third is airplane flight, with typical airplane flight controls and also with propeller drive, where the rotor autorotates at less than half its helicopter rpm contributing only a relatively small portion to the total aircraft lift and drag. A rotor speed sensing governor actuates the longitudinal cyclic pitch mechanism to maintain constant rpm; thus for airplane flight speeds the rotor advance ratio varies from approximately .5 to 1.2. The rotor has no primary control function in the airplane flight regime.

The results of extended flight testing of the McDonnell XV-1 research aircraft by company and by Air Force pilots during 1955-57 (Reference 12.3 summarizes the

Air Force evaluation of this aircraft) prompted the company to start the preliminary design phase of a VTOL transport designated Model 113 which incorporates all the features and principles of the XV-1 research aircraft. This aircraft meets the TRECOM criterion for a light VTOL Army transport with a payload of approximately but not less than 2 tons at 6000 feet $95^{\circ}F$ and with a radius of action of more than 200 nautical miles. A large number of design variations of this VTOL transport have been studied since 1956 (Reference 12.4). Extensive wind tunnel tests with a scale model of one configuration, including the rotor, have been conducted.

The light unloaded rotor transport configuration with four T58-GE-8 free turbine engines is designated Model 113P. Alternate versions of the Model 113 have three T55-L-7 or two T64-GE-2 turbine engines. The Model 113P version is a 30,000-pound VTOL transport designed for the Army criterion of hovering out of ground effect at 6000 feet $95^{\circ}F$ (11 percent power augmentation) and carrying, under these stringent conditions, a payload of 3 tons and fuel sufficient for a radius of action of 100 nautical miles. With a payload of only 2 tons, the radius of action is extended to 230 nautical miles. The cruising speed is approximately 200 knots.

The design of Model 113P is based not only on the flight experience with the XV-1 research aircraft but also on the experience with the Navy 75-foot, 50,000-pound thrust jet driven helicopter rotor which accumulated over 150 hours of whirlstand operation. During the extensive studies of the problems connected with the development of a 2- to 3-ton, 200-knot VTOL aircraft, optimum parameters of such a craft were established considering not only performance criteria but also dynamic, structural, weights, stability, control, and flight conversion criteria.

SECTION III

3. RECOMMENDED LIGHT VTOL TRANSPORT REQUIREMENTS

3.1 Purpose of Recommendations - The usefulness and economy of any aircraft depends to a large degree on the design requirements and design specifications. In the case of the VTOL aircraft it is especially important not to compromise the design by requirements which would impose deviations from the optimum combination of design parameters. For example, studies have shown that for a VTOL aircraft a cruising speed of about 200 knots results in higher transport efficiency than a cruising speed of 300 knots. If transport economy is desired it would, therefore, be detrimental to this purpose to specify a VTOL design requirement for a cruising speed of 300 knots.

The following recommended transport requirements have been selected with the purpose of avoiding compromise of the optimum design of an XV-1 type VTOL aircraft. However, when applied to other VTOL types, these requirements may result in severe penalties.

3.2 Performance and Mission Profiles

3.2.1 General Performance - Reference 12.5, approved by Departments of Army-Navy-Air Force, is recommended, with some exceptions, as the general performance specification for the Army light VTOL transport aircraft. The desired specific performance such as hovering criterion, radius of operation, payload, cruise speed, and cruise altitude, and deviations necessary because of the special characteristics of VTOL aircraft should be in the type specification. A combined requirement of hovering out of ground effect at 6000 feet 95°F with a 2-ton payload and for 500-nautical mile radius of operation is considered too stringent and causes a disproportionate increase in gross weight. The requirements recommended are a radius of 250 nautical miles for the 6000-foot 95°F hovering OGE criterion, and a radius of 400 nautical miles for a hovering OGE criterion of sea level 100°F ambient temperature.

The engine out performance requirement has a decisive influence on the over-all aircraft design and should be carefully considered. Two of the one engine out requirements of Reference 12.5 refer to the service ceiling; for normal gross weight the maximum power service ceiling shall not be less than sea level on a hot day, and for overload gross weight maximum power service ceiling shall not be less than sea level on a standard day. It is recommended that for such types of VTOL aircraft which are capable of one engine out emergency landing with a touchdown forward speed less than 30 knots, the engine out service ceilings of Reference 12.5 be replaced by sea level standard day for normal gross weight. If the low emergency landing speed is not attainable, the requirements of Reference 12.5 should apply. The subject study, as recommended by TRECOM, is based on sea level standard day for normal gross weight.

3.2.2 Cruising Speed - Some studies have indicated that cruise speeds of 150 to 200 knots are optimum from a vulnerability standpoint for "nap of the earth" type of operation. Lower speeds increase vulnerability from ground fire. Higher speeds increase hazards in low altitude flying. High altitude increases vulnerability through radar detection. A study of VTOL types indicates that optimum empty weight and productivity for the light VTOL transport aircraft occur at a cruise speed in the vicinity of 200 knots. Therefore, it is recommended that a cruising speed of not less than 180 knots be required.

3.2.3 Cruising Altitude - The best utility of the light VTOL transport aircraft appears to be in the 200- to 300-nautical mile radius regime. To cover the majority of such cases, a design cruise altitude of 10,000 feet or less is recommended. This avoids the necessity for oxygen and pressurization which are additional hazards in military operations.

3.2.4 Ferry Range - For complete global mobility, it is recommended that a ferry range of not less than 2000 nautical miles be required. Running take-off under standard day conditions at sea level should be permitted to obtain this range.

3.3 Stability and Control - Basically the helicopter flying qualities specification, Reference 12.6, is recommended for hovering and low speed stability and control requirements; the piloted airplane flying qualities specification, Reference 12.7, is recommended for cruise and high speed stability and control requirements. By following these specifications, the Army light VTOL transport aircraft will have the desirable low speed handling characteristics of the helicopter while retaining the cruise flying qualities of the conventional fixed wing transport.

It is recommended that the Army VTOL aircraft possess inherent levels of static and dynamic stability sufficient for emergency operation in any possible flight regime. Use of automatic stabilization equipment to augment stability during normal operation is considered acceptable. Aircraft types incapable of attaining inherent stability levels sufficient to meet the emergency requirement should be required to have completely separate, dual stabilization systems.

Direct aerodynamic control, i.e., cyclic and collective pitch and/or surface area control, of the VTOL aircraft in all flight regimes is recommended. Such systems rely only on structural integrity and generally provide control for a minimum loss of power and/or weight penalty. Control powers should be sufficient to meet the response requirements of the flying qualities specifications recommended, even when artificial stability is obtained by primary control deflection. In addition, control power should be adequate to compensate any moment unbalance resulting from an emergency such as power loss, whether partial or complete.

3.4 Basic Structural Criteria

3.4.1 Flight Criteria - The structural criteria should conform to the applicable portions of existing Military Specifications such as References 12.8 and 12.9. The maneuvering loads should be determined in accordance with the wing and aircraft stall characteristics in conjunction with the control characteristics which are consistent with the recommended flying qualities outlined in Section 3.3. Based upon the degree of maneuverability required, the recommended symmetrical limit load factors at the aircraft center of gravity are 3.0 positive and -1.0 negative. Gust effects should be determined in accordance with Reference 12.9 where applicable.

The flight V-n envelope should cover the entire range of flight speeds with particular attention to the principal flight regimes such as hovering and slow speed, and the airplane flight regime in which a large portion of the aircraft weight is supported by a wing. Transition regimes which involve short periods of time with the absence of maneuvering are likely to be noncritical with respect to the structural weight. A representative flight V-n envelope obtained by the above methods is shown in Figure 7.22.

3.4.2 Landing Criteria - It is recommended that the landing, taxiing, and ground handling criteria be determined in accordance with Reference 12.8 for the helicopter class of VTOL capable of performing a helicopter type autorotational landing; and in accordance with Reference 12.9 for airplane class of VTOL.

3.4.3 Fatigue Criteria - Fatigue criteria similar to those of Reference 12.8 should apply at all flight speeds.

3.5 Center of Gravity Travel - The allowable center of gravity travel should be at least equal to that of comparable fixed wing aircraft. The c.g. travels of the following aircraft were investigated: C-47B, Convair 340, Convair 440, C-123B, C-119B, DC-6, C-69, C-121A, C-130, C-97, C-124A, DC-8, and XC-99. It was found that the average available travel is approximately 15 percent of the wing mean aerodynamic chord. Therefore, it is recommended that the allowable c.g. travel of the light VTOL transport aircraft be equivalent to 15 percent of the M.A.C. Such a center of gravity travel should be available in hovering and low speed flight as well as in normal cruise flight.

3.6 Cargo Compartment Configuration - The cabin size should be based upon adequate accommodation of the desired payload in troops as well as adequate space for vehicles within the allowable payload. The cargo compartment cross section should be rectangular in shape and free from obstructions. The recommended compartment size for the payload capability of this aircraft is 6.5 feet high x 8 feet wide x 30 feet long. An aft loading ramp is recommended to facilitate loading and unloading from ground level, from truck bed height, and by fork lifts. Provisions should be made for rapidly loading and unloading large prepackaged cargo.

3.7 Safety Features and Procedures - The Army requirement of operation in "the nap of the earth" accentuates the need for consideration of safety features and procedures in the design of a VTOL aircraft. By following the recommended performance and flying qualities specifications, the maneuverability and emergency performance of the VTOL aircraft will be acceptable from the safety viewpoint. Contributions to VTOL aircraft safety are associated with other features of which the following are recommended:

a. Means to instantaneously balance lift and propulsive force asymmetry caused by partial power failure.

b. Landing gear with minimum of 35 degrees side turnover angle.

c. No rotating elements less than 8 feet above the ground.

d. Safe conversion capability at low altitude with one engine inoperative.

SECTION IV

4. DISCUSSION OF VTOL CONCEPTS

4.1 VTOL Aircraft Classes - The VTOL aircraft types may be divided into two classes: helicopter type VTOL and airplane type VTOL. In vertical and slow speed flight, the first type uses accepted helicopter principles of lift generation and of control. Lifting rotors with hinged blades, with collective and cyclic blade pitch control, and with moderate rotor disc loadings up to about 10 pounds per square foot give the helicopter type VTOL slow speed performance and handling characteristics very similar to those of the pure helicopter. Roll and pitching velocities are well damped because of the flapping effects of hinged rotor blades. Autorotation in case of power system failure is possible with moderate sinking speeds and glide angles; collective and cyclic pitch flares from autorotational or partial power descents allow vertical touchdown by utilizing the large rotational energy stored in the lifting rotors and by utilizing the substantial ground effect which increases the rotor lift in the order of 20 percent close to the ground without requiring an increase in rotor power. In cruising flight of the helicopter type VTOL aircraft where part or all of the aircraft weight is carried by a fixed wing, propulsion is obtained either by forward inclination of the lifting rotors up to 90° or by separate means of propulsion such as additional propellers or jet propulsion.

While the helicopter type VTOL craft is basically a helicopter to which features of the propeller or jet airplane are added, the airplane VTOL craft is basically a propeller or jet airplane to which means of lift generation in vertical flight are added. For vertical flight the propellers, fans, or jets are rotated to the vertical position - either with or without simultaneously rotating the wing - or the slipstream is deflected in such a way that its direction is essentially vertical. The vertical lift devices used in the various types of airplane VTOL craft are characterized by:

a. A much higher disc loading than used for helicopters.

b. A much higher power consumption per pound of vertical lift generation.

c. The absence of roll or pitch damping in vertical flight.

d. The absence of autorotational capability.

e. The absence of lift augmentation in ground proximity.

f. The absence of a large source of rotational kinetic energy which could be used for partial power or power-off flares.

In cruising flight the airplane VTOL types are mismatched in power available versus power required which results in very high optimum cruise altitudes or in very low cruising efficiencies because of the high SFC of the turbine engines associated with operation at a low percentage of normal power unless the procedure of shutting down and restarting engines in flight is used.

4.2 Aircraft Efficiency Parameters - The three parameters which establish the economy and usefulness of an aircraft are its aerodynamic efficiency expressed as the lift to drag ratio L/D, its structural efficiency expressed as the empty weight to design gross weight ratio K_E, and its propulsive efficiency expressed as the specific fuel consumption per thrust horsepower SFC/η where η considers the sum of all power transmission losses, such as from gearing, from torque compensation, and from propulsive devices. Assuming the use of modern turbine engines and of well-designed power transmission systems, optimum values of SFC/η in the order of .75 can be expected for all VTOL types. The difference in the various types is then limited to differences in the two parameters L/D and K_E. The structural efficiency K_E depends very much on the definition of what establishes the design gross weight. For the purpose of this study the definition is given by the TRECOM requirement of hovering capability at design gross weight out of ground effect at 6000 feet 95^{o}F. Retaining this requirement and the assumption of SFC/η = .75 constant for all types of VTOL aircraft, there is a definite trend of reduced structural efficiency K_E with improved aerodynamic efficiency L/D and vice versa. The pure helicopter at one end of the spectrum has a good structural efficiency in the order of K_E = .60 but a poor aerodynamic efficiency in the order of L/D = 5. Trying to improve the aerodynamic efficiency of the helicopter, for example, by adding fixed wings and propellers results in penalties in structural efficiency. Typical airplane VTOL craft at the other end of the spectrum have a relatively good aerodynamic efficiency in the order of L/D = 15 (the L/D ratio of the VTOL airplane suffers as compared to the pure airplane from the necessity of a small wing aspect ratio), but they have a poor structural efficiency in the order of .80. The obvious reason is that attempts to provide hover capability in a pure airplane require a large increment in installed power, large propellers or turbo-fans, and additional means for hovering control. This can only be achieved with a severe penalty in structural efficiency, that is, with a high K_E number.

4.3 Optimum Aircraft Type - In selecting an aircraft type most suited to the Army Light VTOL Transport requirement, the criterion of best economy of transport performance is important. An aircraft will operate most economically if it can perform a certain transport mission with the lowest empty weight and with the highest cargo flow per unit empty weight. In this study the payload is given as 4000 pounds. The transport efficiency or productivity is defined by $4000/W_E \times V_B$ where W_E is the empty weight in pounds and V_B the block speed assuming a 15-minute turn around time. The dimension of the "productivity" or transport efficiency is usually given as ton knots of cargo flow per ton of empty weight. It was found that the fully unloaded rotor helicopter with rotor jet drive is by far the most economic VTOL transport for the mission specified by TRECOM (Reference 12.10). For the same mission, the VTOL aircraft at either end of the spectrum, the pure helicopter and the VTOL airplane, require a much higher empty weight and they both produce a much lower cargo flow per unit empty weight. The next best VTOL type is the partially unloaded rotor helicopter with rotor jet drive represented by the Fairey Rotodyne. The empty weight of this type for the TRECOM mission, assuming 250 nautical miles radius of action, would be about the same as that of the fully unloaded rotor helicopter; however, the cargo flow would be less because of the lower cruising speed.

Another VTOL type studied is the fully unloaded rotor helicopter with shaft drive. Although the aerodynamic efficiency of this type is about the same as for jet drive, the lower structural efficiency - higher K_E number - results in a considerably increased empty weight for the assumed mission and in a considerably reduced cargo flow per unit empty weight. The main reason for the superiority of the unloaded rotor helicopter with rotor jet drive over the other VTOL types is that the additional weight from fixed wing and propellers is partly compensated by weight savings when substituting jet drive for shaft drive. This special advantage together with the optimum location in the speed spectrum for any VTOL type is the reason for the economic superiority of the XV-1 type over other VTOL types.

SECTION V

5. JUSTIFICATION OF COMPONENT SELECTION

5.1 Over-all Considerations - As discussed in the introduction, the McDonnell Model 113 meets the design criteria laid down by TRECOM in Reference 12.1 contract. The selection of the design parameters for Model 113 is based on several years of preliminary design work and makes use of the flight test experience with the Army XV-1 research aircraft, of the ground test experience with the 50,000-pound thrust Navy rotor, and of extensive wind tunnel tests with a complete scale model of Model 113. For this reason the process of optimum design parameter selection was not repeated for the TRECOM light VTOL aircraft study. Instead, a justification of the component selection for Model 113 is given in this section.

The over-all configuration of the Model 113, while retaining the basic unloaded rotor principle of the XV-1, reflects differences in mission requirements. It also incorporates improvements resulting from experience gained during the XV-1 test programs and subsequent pressure jet development programs (Navy 75-foot rotor and McDonnell Model 120 helicopter programs). The XV-1 reciprocating-pusher propeller system is replaced in the Model 113 by multiple gas turbine-tractor propeller systems; the skid gear is replaced by a retractable tricycle gear; the twin boom empennage support with abrupt afterbody contraction is replaced by a conventional fuselage and empennage. These modifications have resulted in major gains in structural and aerodynamic efficiency. To illustrate, the maximum L/D for the XV-1 configuration was approximately 6.6 while scale model wind tunnel test of the Model 113 configuration demonstrated 9.0 for the maximum L/D value (see Reference 12.11). This gain is obtained by reducing the parasite drag area and the local interference drag in the pylon-fuselage-wing junctions through configuration selection without compromise of flying qualities.

In addition to performance and flying qualities aspects the Model 113 aircraft configuration selected to fulfill the Army light VTOL transport requirements reflects consideration of minimum silhouette, "nap of the earth" operation, and tripartite Service application. The aircraft is capable of complete operation from CVS and LPH aircraft carriers when power folding of rotor blades and wing outer panels is incorporated (see Figure 5.2).

In the following paragraphs, the Model 113 components are discussed and the bases for selection presented. References 12.4 and 12.12 provide information in support of the selections made as well as detailed descriptions of the components. Figure 5.1 presents the general arrangement of Model 113P.

5.2 Rotor

5.2.1 Rotor System - Almost all of the significant characteristics of the unloaded rotor compound helicopter depend upon the rotor system. It must provide the lift, propulsive force, and basic flying qualities in helicopter flight; yet it must not adversely affect the flying qualities in airplane flight.

Through the combined support of the Army, Navy, Air Force, and McDonnell Aircraft Corporation, it has been possible to maintain a high level of theoretical and experimental effort for developing a rotor system to fulfill these requirements. Flight tests of the XV-1 and whirl tests of the Navy 75-foot rotor during this time provided full scale experimental verification of the attainment of the desired rotor characteristics. The resulting McDonnell rotor system incorporated in the Model 113 permits flight at speeds nearly twice those possible with conventional rotor systems on current operational rotary wing aircraft, and at the same time shows outstanding attributes with respect to vibration, flying qualities, and maintenance.

The McDonnell rotor incorporates three blades of high inplane stiffness attached to the hub by two bundles of thin retention straps, as shown schematically in Figure 5.3. Spherical coning hinges permit the blades to flap and pitch with respect to the hub. The hub is gimbal mounted to the rotor support cone. Cyclic and collective control are obtained through a control stem as shown schematically in Figure 5.4. The pitch control link is oriented such that flapping motion of a blade resulting from motion of the hub about its gimbal produces little pitch change. Flapping of the blade with respect to the hub is accompanied by a large pitch change because of the large pitch-cone ratio, tan δ_3. In helicopter flight the hub is free to float about the gimbal axes, and motion of the blade about the offset hinge is primarily a coning motion due to lift. In airplane flight the collective pitch is reduced to the position where the hub is locked to the control stem (see Figure 5.4). Both flapping and coning motions are required from rotation about the offset hinge in this condition. For further details, see Figure 7.5.

The improved flying qualities of the McDonnell rotor system are reflected mainly in increased lateral and longitudinal stability and reduced gust response. Figure 5.5 shows that the damping obtained experimentally on a model and the XV-1 rotor is two to three times that of a conventional rotor, theoretically equal to $16/\gamma\Omega$. In addition, the rotor exhibits a stable aircraft pitching moment with changes in angle of attack. With the McDonnell rotor system the XV-1 and the McDonnell Model 120 helicopter are both inherently stable about all axes in helicopter flight without the use of stabilizing bars or other devices to artificially introduce damping. The reason for the reduced response to gusts is the reduced lift-curve slope resulting from the pitch-cone feature of the rotor as shown in Figure 5.6. Accompanying this lift reduction is a reduction in flapping response in the locked hub configuration which eliminates dangerously high flapping angles due to gusts in high speed airplane flight.

There are several dynamics improvements accrued in this rotor system. The combination of high blade inplane stiffness and the axially stiff retention straps allows the inplane frequency to be kept sufficiently high to preclude mechanical instability, thus eliminating the need for lag dampers. The high inplane frequency coupled with the hub locking feature and pitch-cone ratio permits the rotor to be started and stopped in high winds without excessive loads or flapping angles. Also attributed to the stiff inplane blades and pitch-cone coupling are low vibration level in the aircraft in helicopter flight and the elimination of frequent retracking and rebalancing of the rotor, a characteristic proven in the XV-1 and Model 120 operations. The high speed capability of the rotor is permissible dynamically because of the high torsional stiffness embodied in the rotor design.

Maintenance of this rotor system is substantially reduced compared to that of conventional rotor systems. In cruising flight the rotor rotates at half of the hovering rpm reducing the effects of wear and fatigue on oscillating and rotating parts. The strap retention system avoids all bearings under centrifugal load. There are no dampers, stabilizing bars, or other devices on the blade. All oscillating bearings are manufactured with Teflon bearing surfaces for long life and require no lubrication. Operation of two different full scale rotors and life cycle laboratory tests have justified the selection of these bearings.

The detailed geometry of the rotor is determined from a combination of aerodynamic, propulsion, dynamic, structural, and weight considerations. The choice of rotor solidity, tip speed, airfoil section, and blade twist is discussed in the next two paragraphs. Other aspects of the design are discussed in Section 7.

5.2.2 Solidity - The effect of solidity, tip speed, and rotor diameter on the hovering performance of the aircraft configuration selected was investigated in Reference 12.4. For a constant aerodynamic blade loading and tip jet thrust, the useful load and useful load to design gross weight ratio were determined as a function of rotor diameter for values of solidity between .08 and .10 and for tip speeds between 650 and 750 feet per second. The results of this study indicated the following trends:

a. The useful load has a relatively flat optimum as a function of rotor radius.

b. The greater the solidity, the lower the optimum rotor diameter.

c. The lower the diameter, the higher the tip speed.

d. The lower the rotor diameter, the higher the useful load ratio and disc loading.

Inasmuch as no great performance advantage was shown for any combination of the main rotor variables, the selection of solidity, tip speed, and rotor diameter was justified on the basis of other criteria.

Two factors which are influenced by solidity, or more particularly by solidity per blade, are the propulsive efficiency and rotor dynamic characteristics. The pressure loss from flow through the blade will reduce with increased solidity per blade. For similar cross-sectional geometry, the nondimensional rotor blade vibration frequencies will increase almost linearly with solidity per blade, σ_b, while the torsional divergence speed which controls the forward speed flutter limit increases at least with the square root of σ_b. Although there are other means to adjust the vibration frequencies and torsional stiffness in a detailed design, the initial choice of blade solidity is most important. From experience on the XV-1, the Navy 75-foot rotor, and Model 113P blade designs, the desired dynamic characteristics for the unloaded rotor compound helicopter can be obtained most easily using a σ_b of .03 (total solidity of .09) and a 15 percent thick airfoil section. The use of a substantially lower solidity leads to difficulty in maintaining the inplane frequency high enough to prevent mechanical instability; the use of higher solidity leads to difficulty in maintaining sufficient control system stiffness.

5.2.3 Aerodynamic Blade Loading, Tip Speed, Blade Section and Blade Twist - In order to select the other blade factors, it is first necessary to establish the rotor aerodynamic blade loading. The blade loading, C_T/σ, is by definition:

$$C_T/\sigma = \frac{K_1 W}{\rho \pi R^2 (\Omega R)^2 \sigma}$$

where

W = Gross Weight
K_1 = Hovering Downwash Factor (1.06 for Model 113P)

ρ = Density
πR^2 = Rotor Disc Area
ΩR = Rotor Tip Speed
σ = Rotor Solidity
C_T = Rotor Thrust Coefficient

A maximum permissible rotor aerodynamic blade loading must be established; first, to assure acceptable hovering (VTOL) control characteristics and second, to avoid premature blade stall in forward flight. For pure helicopters the C_T/σ values dictated by these considerations are approximately equal. For the unloaded rotor compound helicopter, forward flight velocity alleviates the rotor thrust load through reduction or elimination of the hovering downwash load and through development of fixed wing lift. Thus, the hovering control problem determines the maximum allowable blade loading for VTOL operation. Under conditions of marginal power and too high a blade loading, hovering control difficulties are created by rotor blade stall induced by cyclic pitch variation. Blade angles of attack higher than the steady state values can be obtained during rapid control displacements where the rotor induced velocity lags the change in blade pitch. Helicopters with low rotor damping require continuous cyclic control during hovering flight thereby accentuating the hovering stall problem. For these aircraft a lower level of blade loading must be maintained than that permitted for rotors with increased damping characteristics as exhibited by the stabilizer bar rotor and by the McDonnell pitch-cone rotor. Based on full scale tests and analyses, a limiting aerodynamic blade loading of .11 is selected for the Model 113 compound helicopter. This limit combined with the selection of rotor geometry, rotor tip speed, and density establishes the maximum aerodynamic VTOL capability of the aircraft.

The rotor blade tip speed should be as high as compressibility considerations permit. Figure 5.7 summarizes NASA whirlstand experimental data showing the effect of airfoil section and blade twist on the ratio of test to incompressible values of profile torques. Minimum profile torque ratios at the design aerodynamic blade loading can be obtained at the highest tip speeds (700-750 fps) from the NASA 6 series airfoils. The effect of negative twist is seen to be favorable from the comparison at zero and minus 8 degrees. Additional gains in hovering tip speed could be derived by the use of higher twist; however, in autorotation in airplane flight the effect of twist is reversed. As a compromise the blade twist of minus 8 degrees has been chosen for the Model 113. A $63A_2(1.5)15$ blade section has been selected which permits, according to Figure 5.6, the use of tip speeds up to 750 fps at $C_T/\sigma = .11$ without noticeable compressibility losses. In order to account for a possible increase in local Mach number from the tip burner influence, 735 fps has been chosen for the Model 113 design.

5.3 Propulsion

5.3.1 Rotor Drive System - Of the many methods of rotor drive adaptable to the compound helicopter, the pressure jet system and the gear driven system by current design practices appear to be the most competitive. The pressure jet system has the advantages of reduced weight and maintenance; the gear driven system the advantages of reduced noise and specific fuel consumption. For the short periods of rotor powered flight envisioned for transport aircraft, the benefits of the reduced empty weight-design gross weight ratio (K_E) far exceed the penalty of increased fuel consumption.

An increased K_E factor for a gear driven system arises from the shafting, gear boxes, and increased tail rotor and installed power requirements. Considering installed power requirements, pressure jet system analyses of Model 113 show the ratio of maximum tip jet thrust to installed power to be .85. Assuming a rotor tip speed of 735 feet per second,

$$\frac{\text{Maximum Rotor Shaft Power}}{\text{Military Power Plant Rating}} = \frac{.85 \times 735}{550} = 1.13$$

On the other hand, the gear driven rotor shaft power is less than the installed power by the sum of the gear, cooling, inlet, and torque compensation losses:

$$\frac{\text{Maximum Rotor Shaft Power}}{\text{Military Power Plant Rating}} \approx .80 \quad \text{(hovering flight)}$$

To obtain equal hovering rotor shaft horsepower, the gear driven-pressure jet installed power ratio is 1.13/.80 or 1.41. Weight analyses show a 7.5 percent gross weight incremental difference in the K_E factor for the two systems.

The pressure jet system is recommended for the rotor drive system of the unloaded rotor compound helicopter capable of fulfilling the Army light VTOL transport requirements because:

a. The aircraft empty weight-design gross weight ratio is reduced which permits a smaller aircraft size and weight to achieve a given payload-radius capability; thus, reduced developmental and maintenance costs are incurred.

b. The growth potential of such systems appears greater than that of existing competitive systems. Disadvantages of the pressure jet system (noise, halo) are subject to improvement through further, active development.

5.3.2 Primary Power Plant - Of the gas turbine power plants available in the 1960-63 time period, only the free turbine, turboshaft versions are considered appropriate to the integrated design of the pressure jet unloaded rotor helicopter. The free turbine feature provides superior characteristics with regard to pressure jet system matching and with regard to utilization of a simple, fixed pitch propeller for cruise flight. Included in the free turbine engine spectrum are the T58-GE-8, T64-GE-2, T53-L-3, and T55-L-7 power plants.

Insofar as is possible, power plant installations are restricted to wing locations. Engine nacelles on wings are advantageous because:

a. Power plants are removed from critical fuselage areas.

b. Such locations provide direct propeller drive.

c. Nacelles may be combined with housing for retractable landing gears.

d. Maintenance and inspection of power plants are facilitated.

Three power plant installations are recommended in this Army light VTOL transport study: (4) T58-GE-8, (3) T55-L-7, and (2) T64-GE-2 (see Sections 7 and 8). Generally, performance capabilities and maintenance reliability are increased with the improved specific fuel consumption and reduced engine number of the T55 and T64 power plants but safety reliability is reduced by use of fewer engines. The preference of installation depends upon the relative degree of importance attached to these characteristics. The presentation of data for all three permits the selection of the one most comparable to the characteristics of other competitive VTOL types.

5.3.3 Propeller Type - Propeller type, i.e., fixed pitch or variable pitch, for use with free turbine engine installations is selected on the basis of complexity, maintenance, and performance characteristics. Since directional control for hovering and low speed flight is obtained by a relatively small tail rotor (see Paragraph 5.5.1), no requirement exists for propeller operation during helicopter flight; thus take-off and low speed operating conditions are not propeller design requirements and the design may be based on climb, cruise, and maximum speed.

Fixed pitch propellers offer reduced weight, maintenance, and complexity, and thus reduced operational costs. Combined with free turbine engine installations, this type provides a range of rpm that is compatible with the flight velocity envelope of the unloaded rotor helicopter. For cruise and high speed flight, either a fixed pitch propeller designed for high speed or a variable pitch propeller gives comparable propulsive efficiencies; little aerodynamic advantage of one installation over the other is obtained. For airplane climb flight, the variable pitch propeller installation offers about a 6 percent thrust horsepower advantage through the combined effect of increased engine efficiency at the relatively high and constant speed, and a small increase in propeller efficiency. In autogyro flight at low flight velocities, the variable pitch propeller shows greater benefits in thrust horsepower. Comparisons of fixed pitch and variable pitch propeller characteristics during climb and cruise flight are discussed in Reference 12.2.

The fixed pitch propeller is selected for use on the unloaded rotor compound helicopter because:

a. It is 40 percent lighter in weight.

b. It is simple and almost maintenance free.

c. Take-off and low speed propeller flight is not required.

d. There is little loss in climb performance.

e. Efficiency is equal to that of the variable pitch propeller at cruise and higher speeds.

5.4 Empennage - The empennage size and configuration of the compound helicopter are established primarily by the selection of location, whether under or aft of the rotor disc. For the "under rotor" location, the rotor and ground clearance requirements reduce the area moment arm and fin aspect ratio of the empennage. This leads to increased area requirements generally obtained through surface folding and multi-fin arrangements that add to the empennage vibration and dynamics problem. The "aft of rotor" location more than doubles the moment arm and eliminates clearance restrictions on empennage size, permitting the use of conventional inverted tee-tail arrangements. The advantages of the "aft of rotor" empennage location are:

a. Area requirements to obtain inherent stability are reduced.

b. Empennage contributions to aircraft angular velocity damping are increased.

c. Longitudinal and directional surface control powers are increased.

d. Empennage contributions to hovering download and undesirable disturbances from the rotor inflow velocity are reduced.

e. Aft fuselage may be faired to reduce drag in forward flight.

The "aft of rotor" empennage structural support may consist of a twin boom arrangement as used on the XV-1 or a single boom extension of the fuselage. The fuselage extension approach is recommended in that this arrangement removes any abrupt contraction of the fuselage from the critical pylon-wing intersection, reducing the interference drag, and relieves a difficult dynamic and vibration problem of a twin boom.

In forward flight the rotor downwash flow at the empennage location creates special problems in horizontal stabilizer design. If a fixed area is used, an unstable speed stability contribution results. To avoid this instability, a free floating stabilizer similar to that used successfully on the XV-1 convertiplane is provided. This vee-tab controlled stabilizer proved to be very satisfactory in all phases of flight with the exception of low speed helicopter transition, zero to 40 knots. The stick reversal which existed in this flight regime can be alleviated by further development of the tail system.

Since the advantages of the "aft of rotor" empennage location far outweigh the advantages of the short coupled, under rotor location, the aft location was chosen for the Model 113. Section 7.3.3 presents detailed descriptions of the complete empennage system, including control systems. Reference 12.11 shows the inherent static stability about all axes.

5.5 <u>Control</u> - The unloaded rotor compound helicopter permits the use of developed and accepted means of control for both the helicopter and the airplane flight regimes without excessive penalty in weight or complexity. Primarily, control may be accomplished by helicopter type systems in low speed flight and airplane systems in cruise and high speed flight without complicated mixing mechanisms. A discussion and justification of Model 113 control systems appear in the following paragraphs.

5.5.1 <u>Directional Control</u> - The pressure jet rotor drive eliminates the requirement for anti-torque compensation typical of gear driven helicopters. However, the requirements for directional control in vertical as well as in forward flight remain. Hovering and low speed flight directional control may be provided by differential propeller pitch or by use of a small tail rotor aft of the empennage. In addition to the arguments cited in Paragraph 5.3.3 against using variable pitch propellers, the following reasons against providing directional control by differential propeller pitch are presented:

a. While propellers not used for directional control can be stopped in low speed flight and during landing and take-off, the use of differential propeller pitch for control purposes prevents the stopping of the propellers and thus presents hazards to embarking and disembarking personnel and to ground personnel.

b. Turning propellers while landing in unprepared terrain are liable to be damaged or destroyed from contact with underbrush, etc.

c. The power losses involved in directional control by differential pitch are very high; 6 to 8 percent of total engine power is consumed in flat pitch without control operation, and about 14 percent of total engine power is consumed when applying directional control.

d. The propeller control system for differential pitch control is more complicated than that for conventional variable pitch propellers.

Because of the great disadvantages of differential propeller pitch control, the selected solution for low speed directional control of Model 113 is a small tail rotor aft of the empennage and located out of reach of ground personnel. The tail rotor solution permits the use of fixed pitch propellers which can be stopped in helicopter flight and which do not present the hazards previously mentioned. The power loss from directional tail rotor control is only 2 percent of engine power (no anti-torque requirements) for fully deflected directional control and much less for neutral control position. As flight velocity increases, the tail rotor output is augmented by a rudder system which takes over complete directional control in cruising flight.

5.5.2 Lateral and Longitudinal Control - In slow speed helicopter type flight lateral and longitudinal control of Model 113 is obtained through cyclic pitch variation. As discussed in Paragraph 5.2, the McDonnell rotor system provides two to three times as much damping in roll and pitch as conventional rotor types. Together with the reduction in rotor lift slope with angle of attack this feature explains the good hovering and slow speed flight stability of the McDonnell XV-1 and the Model 120 and the unusual insensitivity to gust disturbances.

As forward flight velocity increases, the damping contributions of wing and empennage become greater while rotor lift and, therefore, rotor control power, is reduced through unloading of the rotor. Roll and pitch control in Model 113 is, therefore, augmented by surface controls which take over completely in the final cruising flight condition when the rotor is fully unloaded. In cruising flight the contribution of the rotor to the flight characteristics is very small and does not adversely affect the handling qualities (see Reference 12.3). A description of the Model 113 control system is presented in Section 7.

5.6 Fuselage - As required in the statement of work, two specific cabin sizes, 6.5 feet x 6 feet x 24 feet and 6.5 feet x 8 feet x 30 feet, are investigated. For the 30,000-pound aircraft class, weight analysis shows that the smaller fuselage size saves approximately 300 pounds, 1 percent of design gross weight. Factors other than weight are affected by fuselage size; one is the aircraft lift-drag ratio, another is payload restrictions imposed by cargo space limitations. Figure 5.8 presents the percent change in aircraft L/D versus aircraft design gross weight for assumptions of constant volume and constant payload density, the larger cabin size noted above being established as the base at 30,000 pounds gross weight. Since radius of operation is a linear function of aircraft L/D, the effect of fuselage size on radius is obvious. Table 5.1 presents the effect of cabin size on transport capability - cargo density, troop and litter capacity, and unit floor loading - for assumed payload capabilities of a 30,000-pound class unloaded rotor compound helicopter. Minimum cargo density for the smaller cabin is 60 percent higher than for the larger cabin, indicating a possibility of space restriction on payload. The larger cabin is shown to provide greater utilization of the aircraft payload potential for personnel and ambulance missions. Furthermore, military vehicles in the weight class dictated by the payload capability are predominantly of a size that requires the larger cabin size.

In addition to fuselage cabin size studies, fuselage loading aspects were investigated including both front and rear ramp loading. For forward ramp loading, an unobstructed entrance would require the cockpit to be raised which in turn would raise the rotor and increase the over-all height of the aircraft, sacrificing silhouette and possible Navy application. For rear ramp loading, the rotor height also limits the loading clearance height but the effect is less severe. By alighting gear extension or retraction, adequate clearance between the ground and aft fuselage is provided. Either truck or forklift loading or unloading is possible, as well as prepackaged cargo (see Figure 5.9).

As a result of these studies, the larger of the fuselage cabin sizes (6.5 x 8 x 30) with a rear ramp loading arrangement is recommended for the Model 113 compound helicopter.

5.7 Fixed Wing - In the unloaded rotor concept the aircraft total lift is divided between the rotor and the fixed wing; the rotor supplying the total lift in hovering, the fixed wing supplying the major portion of lift in cruise and high speed flight with the rotor autorotating at approximately half speed. The main effects of transferring the lift from the rotor to the wing are:

a. Removal of blade stall and vibration limitations of forward speed.

b. Improvement of aircraft aerodynamic efficiency (L/D).

c. Attainment of airplane flying qualities in cruising flight.

Wing aspect ratio and area have an effect on hovering download, the aircraft lift-drag ratio, cruise lift coefficient, wing stall characteristics, and component weight. Previous studies (References 12.4 and 12.12) and XV-1 flight experience have shown that the optimum compromise is attained by selecting wing areas of approximately 13.5 percent of rotor disc area and wing aspect ratio approaching 7.0 with wing tip fold provisions to reduce hovering download. Wind tunnel tests of the Model 113 (Reference 12.11) demonstrated that wing airfoil camber and wing aerodynamic center locations aft of the aircraft center of gravity are beneficial with respect to maximum lift coefficient, wing stall characteristics, and static stability. The wing geometry is shown in Figure 5.1.

5.8 Landing Gear - A conventional, retractable tricycle landing gear is selected for the basic configuration. Dual wheels and tires providing a unit construction index (UCI) of less than 20 enables the aircraft to take off and land vertically from the majority of unprepared terrain. Floats or skis for specific operations from water or soft snow can readily be provided; the drag and weight penalties do not justify permanent installation of such special gear.

TABLE 5.1

EFFECT OF CABIN SIZE ON TRANSPORT CAPABILITY

Cabin Size, H x W x L		(ft)	6.5 x 6 x 24	6.5 x 8 x 30
Cabin Volume		(ft^3)	975	1560
Assumed Average Payload Capability (lbs)	6000' 95°F Hovering	100 n.mi. radius	6800	6800
		250 n.mi. radius	4500	4500
	S.L. Std. Hovering	250 n.mi. radius	14,000	14,000
Cargo Density (100% volume) (lbs/ft^3)	6000' 95°F	100 n.mi. radius	7.0	4.3
		250 n.mi. radius	4.6	2.9
	S.L. Std.	250 n.mi. radius	14.3	9.0
Troop Capacity	Number of Troops Spacewise		24	32
	Weight of Troops (260 lbs/troop)	(lbs)	6240	8320
Litter Capacity	Number of Litters Spacewise		16	24
	Number of Attendants		2	2
	Litter plus Attendant Weights @ 250 lbs/litter and 200 lbs/attendant	(lbs)	4400	6400
Unit Loading	Cabin Floor Loading (100% area used)	(lbs/ft^2)	31 to 97	19 to 58

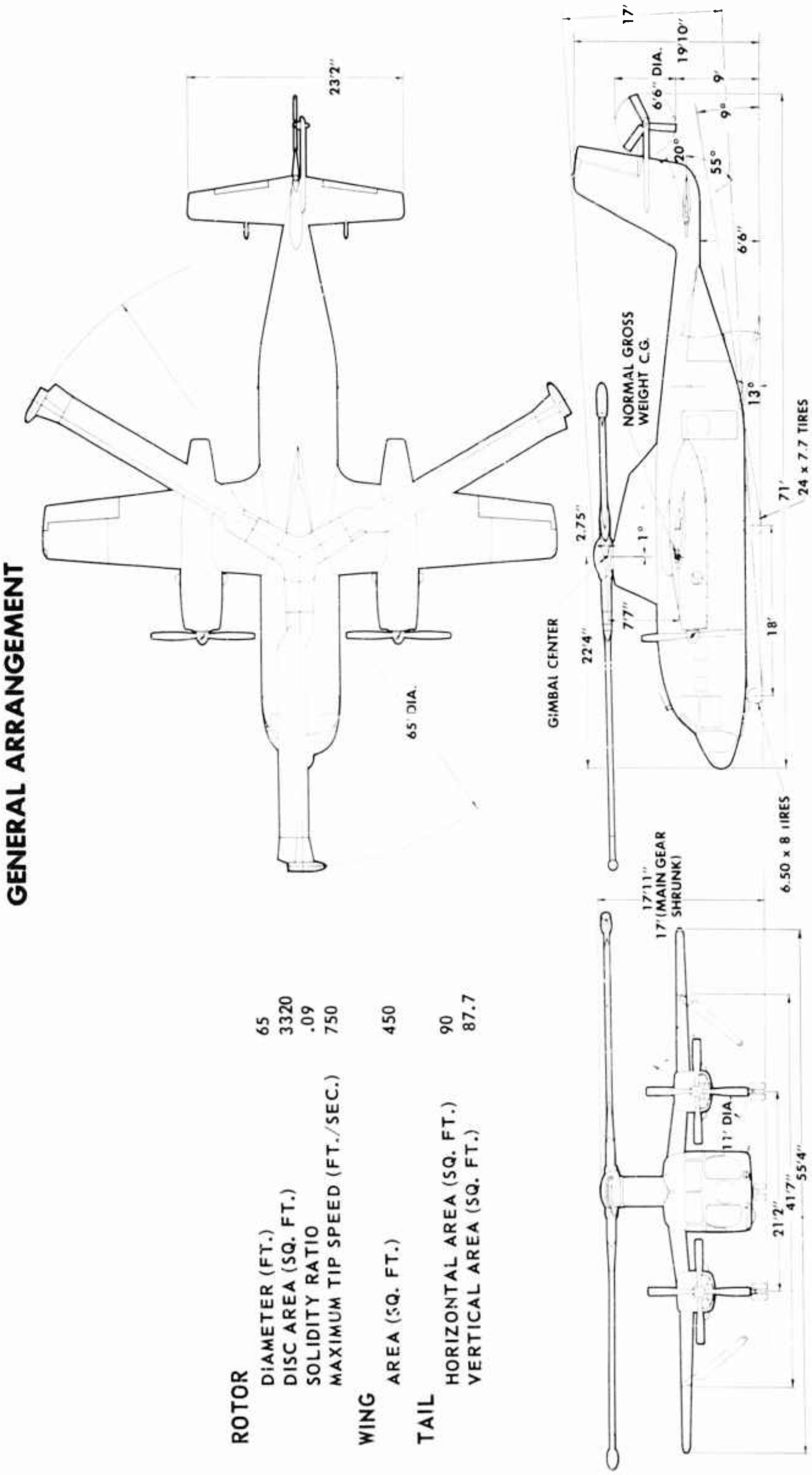

GENERAL ARRANGEMENT
ROTOR
DIAMETER (FT.) 65
DISC AREA (SQ. FT.) 3320
SOLIDITY RATIO .09
MAXIMUM TIP SPEED (FT./SEC.) 750
WING
AREA (SQ. FT.) 450
TAIL
HORIZONTAL AREA (SQ. FT.) 90
VERTICAL AREA (SQ. FT.) 87.7
23'2"
65' DIA.
17'
19'10"
6'6" DIA.
9'
9°
20°
55°
6'6"
13°
NORMAL GROSS WEIGHT C.G.
71'
24 x 7.7 TIRES
2.75"
1°
7'7"
22'4"
GIMBAL CENTER
18'
6.50 x 8 TIRES
17'11"
17'(MAIN GEAR SHRUNK)
11' DIA.
21'2"
41'7"
55'4"

FIGURE 5.1

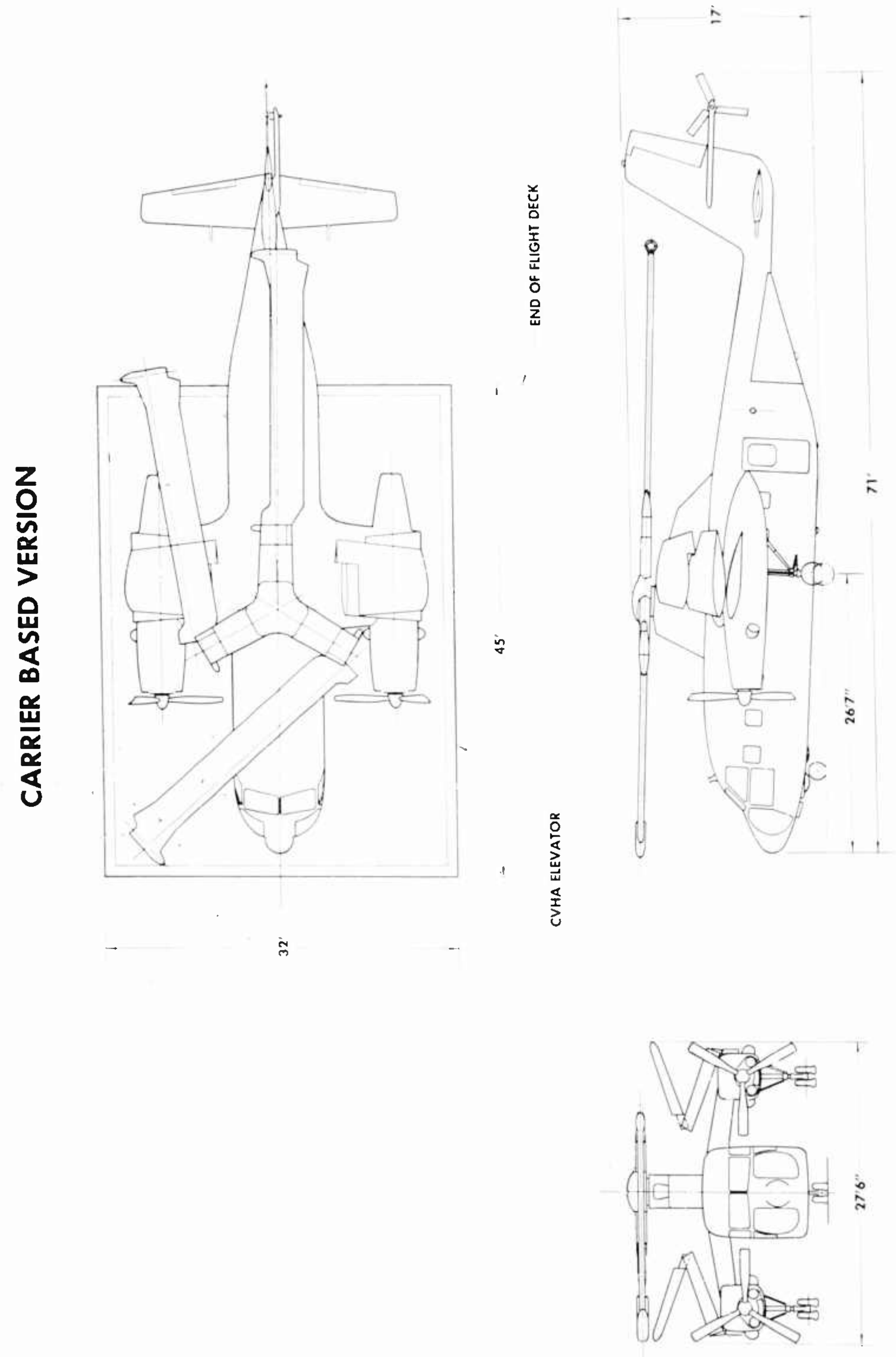

CARRIER BASED VERSION
END OF FLIGHT DECK
CVHA ELEVATOR
45'
32'
17'
71'
26'7"
27'6"

FIGURE 5.2

SCHEMATIC PLAN AND SIDE VIEWS OF
ROTOR AND ROTOR SUPPORT

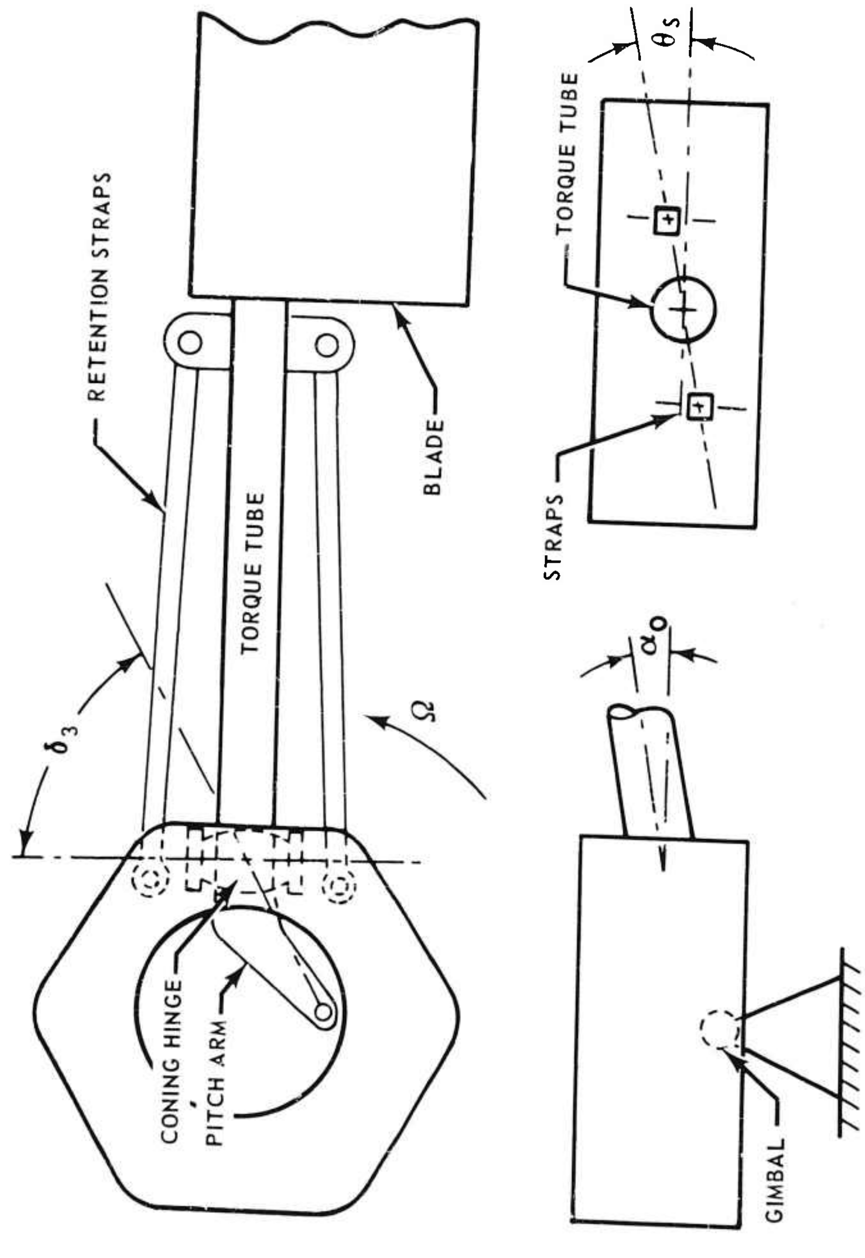


FIGURE 5.3

SCHEMATIC OF ROTOR HUB AND CONTROLS

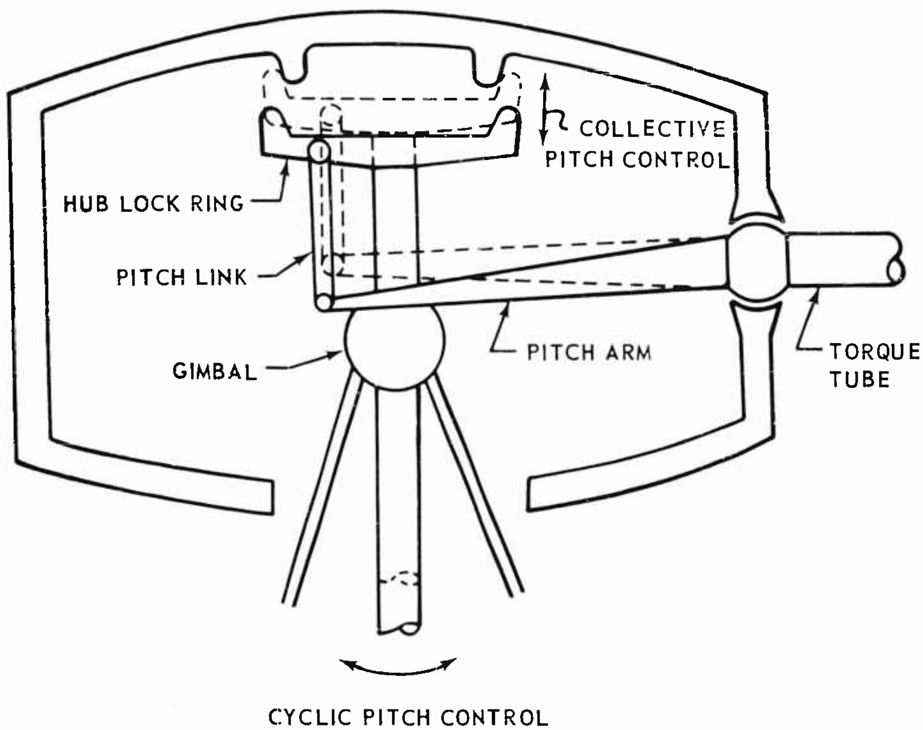


FIGURE 5.4

MODEL 113 VTOL TRANSPORT
ROTOR DAMPING IN ROLL AND PITCH

MODEL AND FULL SCALE ROTOR TESTS

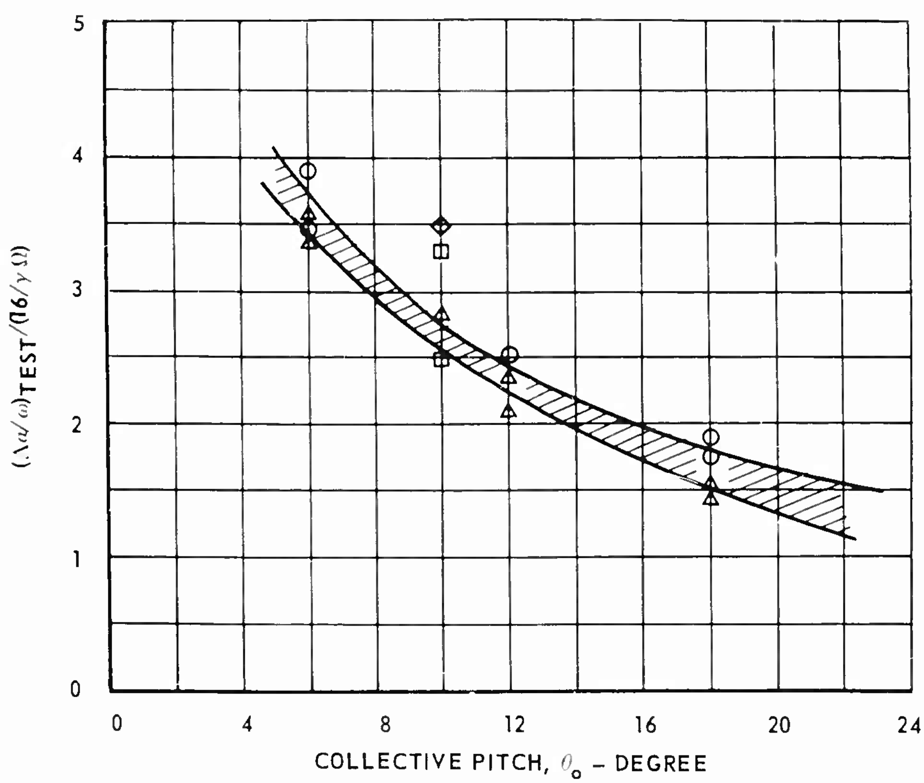


FIGURE 5.5

ROTOR LIFT CURVE SLOPE VS. ADVANCE RATIO

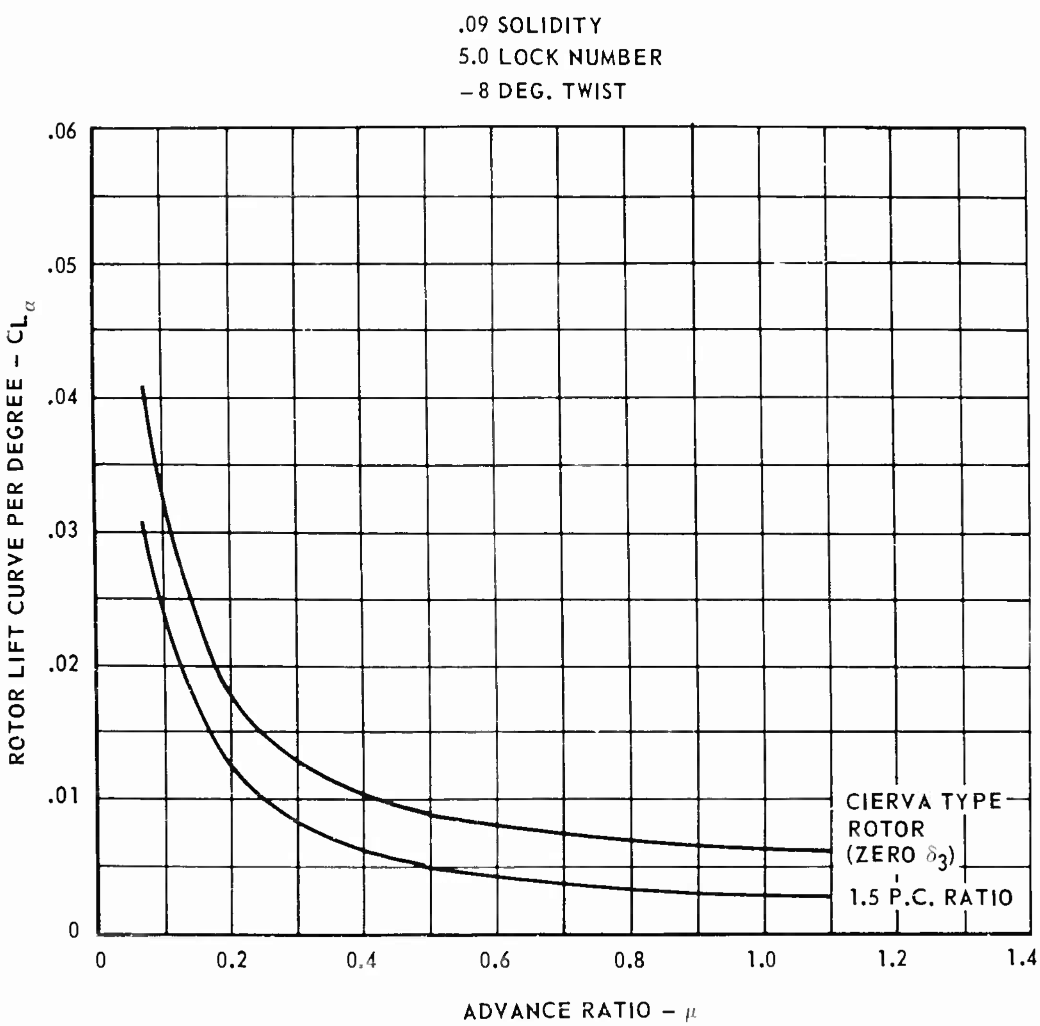


FIGURE 5.6

EFFECT OF BLADE TWIST AND AIRFOIL SECTION ON ROTOR COMPRESSIBILITY AND STALL LOSSES

	SOURCE	AIRFOIL	TWIST (DEG.)
long dash	NACA TR 1078	NACA 23015	ZERO & −8
solid	NACA TN 3850	NACA 63_2-015	−6.5
dash-dot	NACA RM L57F26	NACA 0018 (ROOT) NACA 0012 (TIP)	−7.5

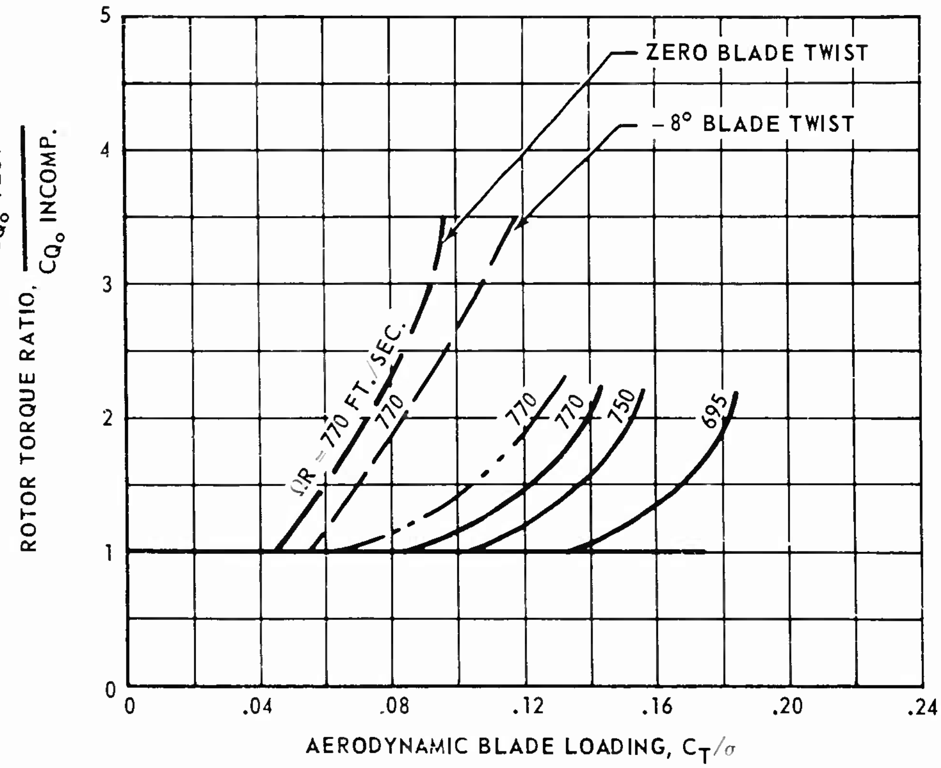


FIGURE 5.7

UNLOADED ROTOR COMPOUND HELICOPTER AIRCRAFT
EFFECT OF FUSELAGE SIZE ON LIFT/DRAG RATIO

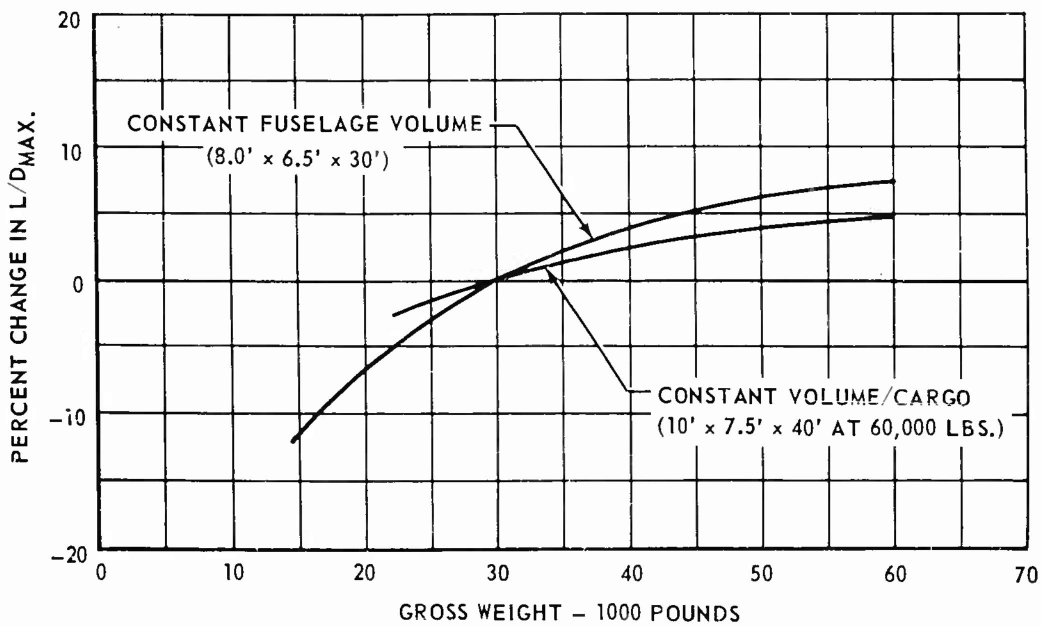


FIGURE 5.8

CARGO LOADING

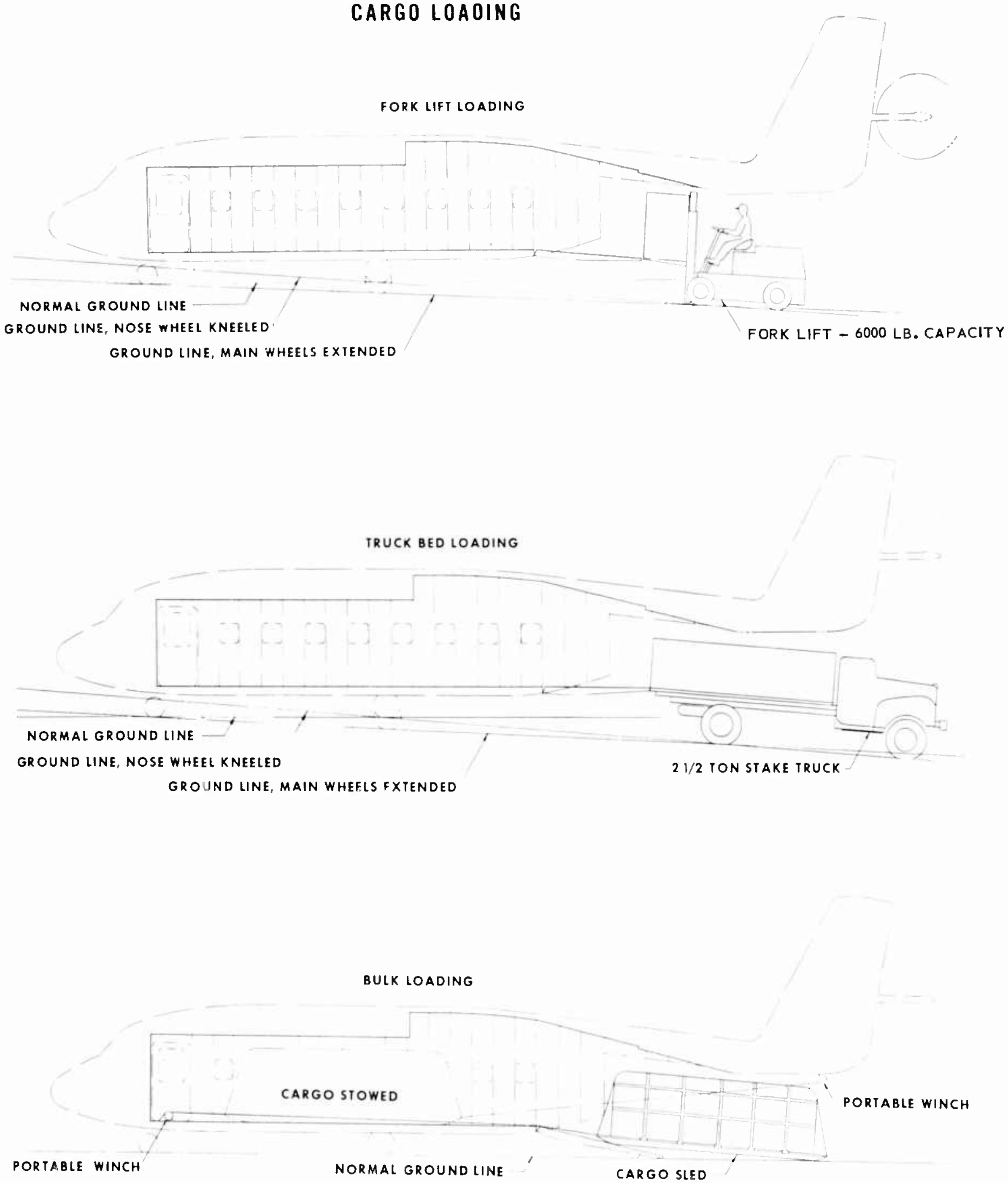


FIGURE 5.9

SECTION VI

6. AIRCRAFT SPECTRUM ANALYSIS

6.1 General Approach - A payload-radius spectrum analysis of the jet driven unloaded rotor compound helicopter is presented which permits preliminary selection of aircraft size or gross weight for specific payload-radius requirements. The spectrum presented is determined by use of a Breguet cruise approach, an estimated useful load ratio as a function of aircraft gross weight, and generalized T58-GE-8 gas turbine characteristics as discussed in succeeding sections. The Breguet cruise approach permits the estimation of the zero payload-radius of operation, while the useful load ratio gives the zero radius-payload capability. Fuel system weight for radii of 250 nautical miles or less is assumed to be constant. For the basic spectrum analysis, no variation in aircraft aerodynamic efficiency (L/D) or propulsion system efficiency (η/SFC or SFC/η) with aircraft size is considered. Small increases in L/D ratio with aircraft size occur through reduced fuselage frontal area-wing area ratios, but these are secondary effects. No variation in propulsion efficiency exists since generalized engines of specified characteristics are assumed. Alternate generalized power plant characteristics corresponding to the T55-L-7 and T64-GE-2 engines are included in the study and show increases in radius capability of all aircraft. Charts are provided to show the effect of deviations from the aerodynamic, propulsive, and structural efficiencies assumed for the basic spectrum analysis.

All aircraft are geometrically scaled to maintain constant disc loading and power loading, the Model 113P serving as a base for scaling purposes. The generalized T58-GE-8 engines require 11 percent power augmentation to meet the Army 6000 feet 95°F normal gross weight hovering criterion; the generalized T55-L-7 engines require no augmentation; and the generalized T64-GE-2 engines require 5 percent power augmentation. The aircraft performance levels including VTOL take-off capability, maximum speed, engine out capability, etc., are approximately maintained independent of aircraft size.

In all cases the aircraft normal gross weight, assumed variation 15 to 60 thousand pounds, is determined by the Army requirement of hover out of ground effect at 6000 feet 95°F. The VTOL overload weight to normal gross weight ratios at various altitude-temperature combinations for the Model 113P with T58-GE-8 engines are as follows:

Hovering or VTOL Atmosphere	Gross Weight Ratio	Power Augmentation
6000 feet, 95°F	1.00	11 percent
6000 feet, Standard	1.12	none
3000 feet, 100°F	1.00	none
Sea Level, 103°F	1.20	11 percent
Sea Level, 103°F	1.10	none
Sea Level, 59°F	1.33	none

The respective overload ratios for the two alternate power plants are given in Paragraph 6.4.2. As indicated by this table, the requirement of hover capability at 3000 feet 100°F in lieu of 6000 feet 95°F eliminates the need for power augmentation, a result of the combined improvement in gas turbine power available and in rotor thrust-tip jet thrust ratio through decreased aerodynamic blade loading.

6.2 Useful Load Ratio - The unloaded rotor compound helicopter useful load ratio as a function of aircraft size or gross weight and fuselage cargo volume has been determined based upon Model 113 design experience. Aircraft disc loading, power loading, wing-disc area ratio, and empennage-wing area ratios are held constant with increase in aircraft size. Power plants are generalized gas turbines based upon the T58-GE-8 characteristics. Two fuselage cargo volume assumptions are considered: first, a constant volume and cross section (8 feet by 6.5 feet by 30 feet); and second, a varying volume to maintain a constant cargo density of approximately 4 pounds per cubic foot for normal gross weight payload.

Figure 6.1 presents the results of the detail weights analysis for a normal gross weight variation of 15 to 60 thousand pounds. The useful load ratio is shown to be relatively insensitive to aircraft size for the weight range considered. At the 60,000-pound normal gross weight level, the constant fuselage volume assumption shows a 3.5 percent advantage in useful load ratio as compared to the constant cargo density assumption. This advantage represents approximately a 10 percent gain in useful load, illustrating that the aircraft fuselage, independent of any aerodynamic consideration, should be as small as is practical. Bulky items, such as trucks, missiles, etc., and the overload capability of the aircraft must be considered in the determination of a practical fuselage cabin volume.

Curve 3 of Figure 6.1 is selected as the curve of the useful load ratio as a function of normal gross weight to be used in the aircraft spectrum analysis.

6.3 Breguet Cruise Approach - Aircraft range characteristics are often determined for the assumption of cruise at constant angle of attack; i.e., by the Breguet equation:

$$\text{Range} = 325 \; (L/D) \; (\eta/\text{SFC}) \; \log_e W_o/W_1$$

where

L/D = Aircraft Lift-Drag Ratio
η = Over-all Propulsive Efficiency
SFC = Power Plant Specific Fuel Consumption
W_o = Aircraft Weight at Start of Cruise
W_1 = Aircraft Weight Less Cruise Fuel

For maximum range, an aircraft design must achieve maximum lift-drag ratio, maximum propulsive efficiency, and maximum useful load-gross weight ratio at minimum power plant specific fuel consumption. The attainment of maximum aircraft lift-drag ratio involves the use of minimum parasite area, thus retractable landing gear and minimum cross-sectional areas compatible with payload requirements, and minimum induced losses obtained by high aspect ratio and elliptical inflow distributions. For unloaded rotor compound helicopter flight velocities, maximum propulsive efficiencies are characteristic of propeller rather than turbojet systems. Lightweight turboprop engine development has provided a means for drastically increasing the useful load-gross weight ratio of an aircraft at a small penalty in specific fuel consumption. The degree of improvement plus the potential gas turbine development is such that reciprocating engines are practically noncompetitive except for an extreme range application. Only turboprop power plants are considered in the ensuing analysis.

For the spectrum analysis presented, an adaptation of the Breguet equation is used; the adaptation consisting of the changes necessary to obtain radius rather than range. This involves estimating fuel requirements for the take-off, climb, etc.; estimating the weight at the start and end of cruise both on the outbound and inbound leg of the radius mission; and then performing an iteration process to equate the outbound and inbound distances. Cruise flight is assumed to be established by operation at maximum aircraft L/D, thus constant C_L, and 90 percent normal rated power; thus two equations involving density and velocity may be written:

$$C_L = 295 \frac{L}{S \sigma V^2}$$

$$(THP)_{Req.} = \frac{V\, L}{325.5\ (L/D)_{Max}} \quad \text{and } (THP)_{Avail.} \approx f\ (\sigma, \eta, V) \text{ for } 90\% \text{ NRP}$$

where

- L = Aircraft Total Lift or Gross Weight, W
- S = Reference Area
- σ = Density Ratio, ρ/ρ_o
- THP = Thrust Horsepower at 90 percent NRP (SHP x η)
- V = Cruise Velocity (knots)

Knowing aircraft gross weight or wing loading, power plant altitude and velocity (ram effect) characteristics, and over-all propulsive efficiency, the unique altitude-velocity combination for 90 percent power and cruise at maximum lift-drag ratio is defined. Therefore, the $(\eta/SFC)\ (L/D)_{max}$ product for assumed wing loadings may be determined and the radius iteration of the Breguet equation completed.

$$(L/D)\ (\eta/SFC) = \frac{SHP\ \eta_p\ \eta_i}{SHP \times SFC} \times \frac{L\ V}{325.5\ THP} = \frac{W\ V}{325.5 \times \text{Fuel Flow/Hour}}$$

where

- η_p = Propeller Efficiency
- η_i = Installation Efficiency (includes inlet, gear box, etc., losses)
- SHP = Engine Shaft Horsepower
- SFC = Specific Fuel Consumption, pounds fuel/SHP-hour

The $(L/D)_{max}$ used in the spectrum analysis is 9.0; a value substantiated by wind tunnel test of the unloaded rotor compound helicopter configuration proposed. Figure 6.2 presents the power plant and (L/D) (η/SFC) characteristics as a function of altitude that were used in the basic aircraft spectrum analysis. The transport mission assumed for all spectrum aircraft is as follows:

Time	Mission Breakdown	Remarks
2 Minutes	Warm-up, Cockpit Check	Normal Rated Power
1 Minute	Take-off, Conversion	Maximum Rotor Power
	Climb to Initial Cruise Altitude	Military Rated Power
	Cruise Out at Constant C_L	90% NRP
	Descend, Land, Unload 1/2 Payload	No Distance Credit
	Repeat Sequence for Return Trip	
	Reserve Fuel	10% Initial T.O. Fuel
	Service Allowance (Ref. 12.5)	5% SFC Increase

All missions are performed for sea level take-off and a NASA standard atmosphere. A constant weight of 940 pounds is assumed for three crew members, oil, trapped fuel and oil, and cargo tie-down straps. Fuel system weight for fuel in excess of design capacity is estimated at .4 pound per gallon of added fuel.

Deviations in aerodynamic efficiency (L/D), propulsive efficiency (η/SFC), or useful load ratio from the values assumed in the basic spectrum analysis will alter the estimated payload-radius capabilities. Aircraft $(L/D)_{max}$ deviations may result from fuselage size assumption or, more important, from the basis of estimation, whether estimated from wind tunnel test or theoretical analysis. Propulsive efficiency deviations arise from use of power plant characteristics other than the T58-GE-8 characteristics assumed. The payload-range capability with alternate power plant characteristics is shown in Paragraph 6.4.2. Figure 6.3 is presented to show the radius variation for an assumed variation in useful load ratio. The zero radius payload may be estimated directly for any alternate useful load ratio. Thus, the revised payload-radius characteristics for an assumed deviation in useful load ratio can be estimated by use of Figure 6.3.

6.4 Results

6.4.1 Aircraft with T58-GE-8 Power Plant Characteristics - The results of the unloaded rotor compound helicopter spectrum analysis are presented as payload-radius charts for constant gross weight levels determined by VTOL capability at assumed altitude-temperature combinations, Figures 6.4 to 6.8 inclusive. Figure 6.4 presents data for the normal gross weight design condition; namely, for take-off weights determined by the ability to hover out of ground effect at 6000 feet 95°F. The combined requirements of hover OGE at 6000 feet 95°F, 4000-pound outbound payload, and a 500-nautical mile radius of operation cannot be met by an unloaded rotor compound helicopter powered by gas turbines with T58-GE-8 characteristics. The maximum radius of operation for the combined hover-payload requirement is approximately 375 nautical miles (normal gross weight of 60,000 pounds). Use of an alternate power plant with decreased specific fuel consumption and/or increased power permitting cruise at higher altitude would increase the maximum radius of operation (see discussion, Section 6.4.2). For a 2-ton payload requirement little radius benefit accrues by exceeding the 45,000-pound normal gross weight aircraft class. However, at reduced radius of operation, increases in payload are associated with the larger aircraft.

It is seen from Figure 6.4 that the zero payload radii for 30,000, 45,000, and 60,000 pounds gross weight are approximately the same. For constant useful load ratios and for constant crew and miscellaneous weight ratios the zero payload radius would be independent of aircraft size since lift-drag ratio and propulsive efficiencies have been assumed constant. The actual reduction in zero payload radius with reduction in gross weight is explained by the lower useful load ratios from constant fuselage size (see Figure 6.1) in combination with the assumption of a constant 940 pounds for crew and fixed miscellaneous weight. Figures 6.5 to 6.8 show the payload-radius capability of the unloaded rotor compound helicopter for altitude-temperature combinations other than the design combination of 6000 feet 95°F. The four curves on each chart represent the four normal gross weights of 15,000, 30,000, 45,000 and 60,000 pounds. Table 6.1 presents the approximate radius capabilities of a 2-ton payload unloaded rotor compound helicopter of approximately 45,000 pounds normal gross weight. Table 6.2 presents the approximate payload capability for a 250-nautical mile radius of operation as a function of VTOL take-off condition and reference aircraft normal gross weights determined by hover OGE at 6000 feet 95°F. Data for alternate power plant characteristics are shown in Table 6.2 to illustrate their effect on the payload radius capabilities as discussed in more detail in the following section.

6.4.2 Aircraft with Alternate Power Plant Characteristics - The free turbine turboshaft power plant considered for the 1960-63 time period are the T58-GE-8, the T55-L-7, and the T64-GE-2; the free turbine feature is necessary for the fixed pitch propeller configuration selected as most desirous for the unloaded rotor compound helicopter. As shown by Tables 6.1 and 6.2, increases in payload and/or radius of operation occur through the use of generalized T55-L-7 and T64-GE-2 power plant characteristics as compared to the T58-GE-8 characteristics. These benefits arise through changes in power plant specific fuel consumptions as well as through increased power ratings that permit higher altitude (lower SFC) cruise conditions. The specific weight and SFC characteristics are summarized by the following:

Power Plant	T58-GE-8	T55-L-7	T64-GE-2
Specific Weight, lbs/MIL SHP (without Gear Box)	.220	.297	.268
Relative SFC @ 90% NRP (at Constant Altitude)	1.0	.91	.82

The increased power of the T55-L-7 and T64-GE-2 power plant installations, besides improving cruise SFC through higher altitude operation, alters the degree of power augmentation required to meet the 6000-foot 95°F hover criterion; 5 percent required for the T64-GE-2 installation, no augmentation required for the T55-L-7 installation. As a result, the VTOL weight ratios for the other altitude-temperature combinations differ from those used in the T58-GE-8 spectrum analysis:

VTOL Atmosphere	Ratio of $\frac{\text{GW permitted by VTOL Atmosphere}}{\text{Normal Gross Weight}}$		
	T58-GE-8	T55-L-7	T64-GE-2
6000 feet, 95°F	1.0$^{(1)}$	1.0	1.0$^{(2)}$
6000 feet, Standard	1.12	1.12	1.12
3000 feet, 100°F	1.0$^{(3)}$	1.115	1.04$^{(3)}$
Sea Level, 103°F	1.20$^{(1)(3)}$ 1.10$^{(3)}$	1.24	1.13$^{(3)}$
Sea Level, Standard	1.33	1.33	1.33

(1) 11 percent power augmentation.
(2) 5 percent power augmentation.
(3) power limited.

The weight ratios given are defined either by power limitation or by VTOL aerodynamic blade loading limit ($C_T/\sigma = .11$).

The combined effect of these alternate power plant characteristics is shown by the T58-GE-8 versus T64-GE-2 comparison of the normal gross weight payload-radius capabilities, Figure 6.9 and by Tables 6.1 and 6.2. The influence of power plant selection on the recommended Army light VTOL transport aircraft is shown in Section 7.

Table 6.3 presents some specific unloaded rotor compound helicopter possible with the three gas turbines considered. The first three aircraft presented are the McDonnell Model 113 with alternate power plant installations. These aircraft fulfill the Army light VTOL transport requirements as described in detail in Section 7. The other two aircraft are four-engine transports of increased size. Figure 6.10 presents the normal and maximum VTOL overload gross weight estimated payload-radius characteristics of these larger four-engine transport aircraft.

TABLE 6.1

RADIUS OF OPERATION FOR 2-TON PAYLOAD, 45,000-POUND NGW AIRCRAFT

VTOL Atmosphere		Maximum Radius of Operation (Nautical Miles)		
Altitude (ft)	Temperature	Power Plant T58-GE-8 Characteristics	Power Plant T55-L-7 Characteristics	Power Plant T64-GE-2 Characteristics
6000	95°F	340 (1)	380	465 (2)
6000	Standard	520	595	680
3000	100°F	340	580	535
S.L.	103°F	620 (1)	775	690
S.L.	Standard	760	905	1010

(1) 11 percent T.O. power augmentation assumed.
(2) 5 percent T.O. power augmentation assumed.

TABLE 6.2

AIRCRAFT PAYLOAD CAPABILITY FOR 250-NAUTICAL MILE RADIUS OF OPERATION

Power Plant	Ref. Normal Gross Weight (lbs.)	Outbound Payload for VTOL Altitude and Temperature Conditions				
		6000 ft. 95°F	6000 ft. Standard	3000 ft. 100°F	S.L. 103°F	S.L. Standard
T58-GE-8 Characteristics	15,000	1100(1)	2700	1100	4000(1)	5700
	30,000	3900(1)	7100	3900	9600(1)	13,000
	45,000	6300(1)	11,200	6300	14,400(1)	20,000
	60,000	8000(1)	14,900	8000	19,400	26,650
T55-L-7 Characteristics	15,000	1200	3000	2850	4600	6000
	30,000	4300	7800	7600	9800	13,700
	45,000	6700	11,700	11,600	16,500	20,700
	60,000	8800	15,600	15,400	22,200	27,500
T64-GE-2 Characteristics	15,000	1900(2)	3600	2500	3700	6500
	30,000	5200(2)	8700	6300	9000	14,800
	45,000	8200(2)	13,300	9900	13,700	22,000
	60,000	10,900(2)	17,700	13,200	18,300	29,500

(1) 11 percent T.O. power augmentation assumed.
(2) 5 percent T.O. power augmentation assumed.

TABLE 6.3

SPECIFIC AIRCRAFT WITH AVAILABLE POWER PLANTS

Power Plant Number & Designation	Military Rated Power (shaft horsepower)	Aircraft Normal Gross Weight (Hover OGE 6000 ft. 95°F) (pounds)	Aircraft Rotor Diameter (feet)
(2) T64-GE-2	5300	29,650 (1)	65
(3) T55-L-7	5610	29,650	65
(4) T58-GE-8	5000	29,650 (2)	65
(4) T55-L-7	7480	39,900	75.5
(4) T64-GE-2	10,600	56,500	90

(1) 5 percent power augmentation assumed.
(2) 11 percent power augmentation assumed.

UNLOADED ROTOR COMPOUND HELICOPTER AIRCRAFT
USEFUL LOAD/GROSS WEIGHT RATIO VS. GROSS WEIGHT

113 P CONFIGURATION ASSUMED

1 CONSTANT FUSELAGE VOLUME
(8.0' x 6.5' x 30')

2 CONSTANT VOLUME/CARGO
(10' x 7.5' x 40' AT 60,000 LBS.)

3 CURVE USED FOR SPECTRUM ANALYSIS

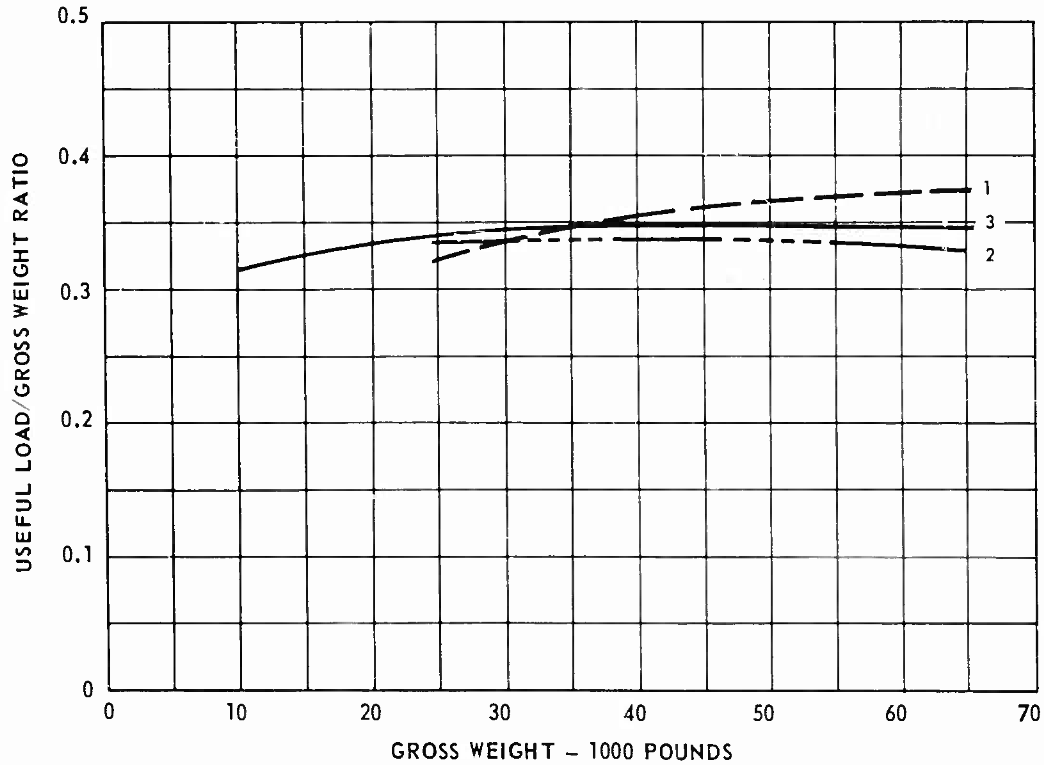


FIGURE 6.1

UNLOADED ROTOR COMPOUND HELICOPTER SPECTRUM ANALYSIS

ENGINE SFC, $\frac{\eta}{C}$ RATIO, $\left(\frac{\eta}{C}\right)\left(\frac{L}{D}\right)_{MAX}$ PRODUCT AND VELOCITY VS. ALTITUDE CHARACTERISTICS FOR T58-GE-8 ENGINES

$(L/D)_{MAX} = 9.0$
90% NRP

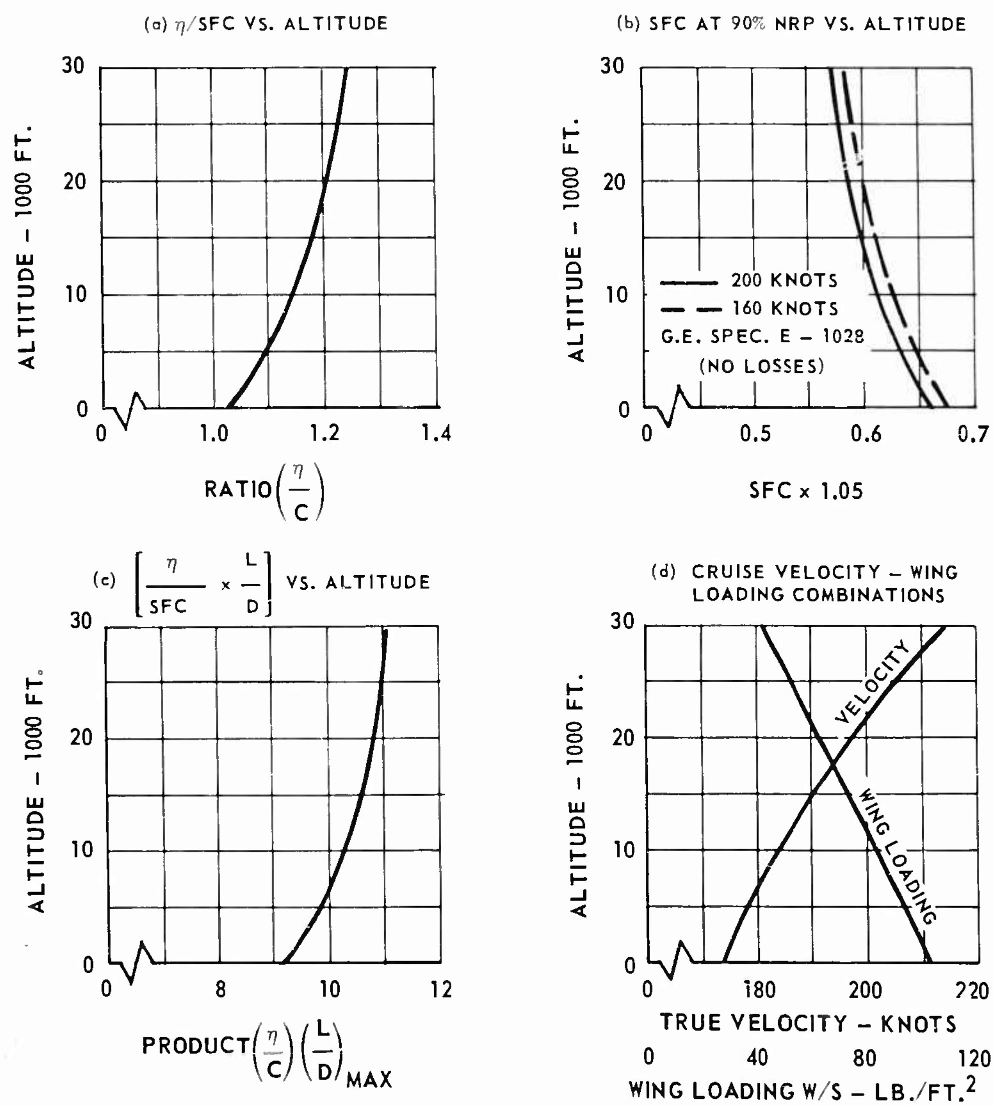


FIGURE 6.2

UNLOADED ROTOR COMPOUND HELICOPTER AIRCRAFT
EFFECT OF USEFUL LOAD RATIO DEVIATION
FROM
VALUES ASSUMED FOR SPECTRUM ANALYSIS

CREW OF THREE
NASA STD. ATMOSPHERE
OPTIMUM ALTITUDE CRUISE
PAYLOAD OUTBOUND=TWICE PAYLOAD INBOUND

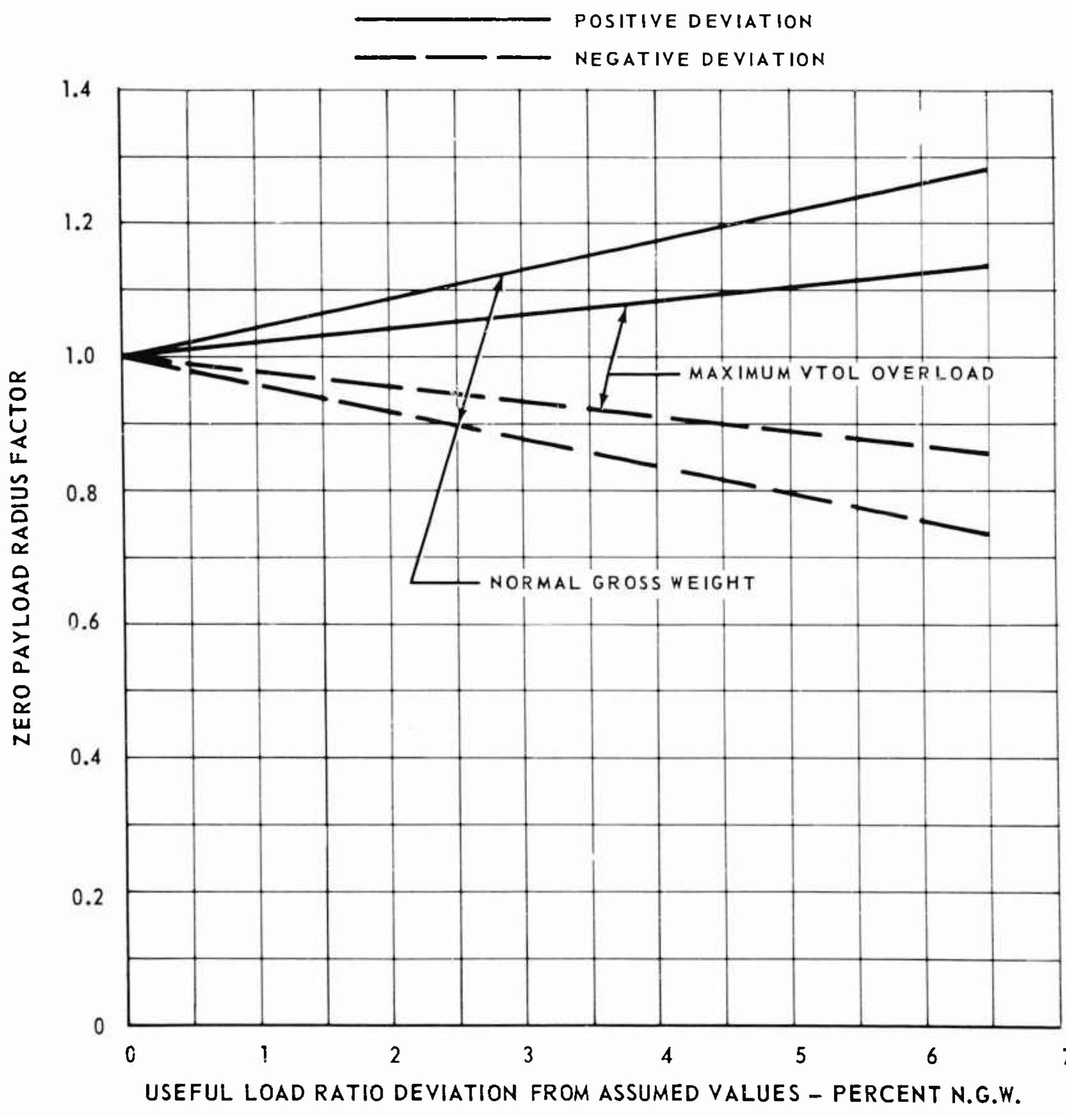


FIGURE 6.3

UNLOADED ROTOR COMPOUND HELICOPTER AIRCRAFT SPECTRUM

T58-GE-8 ENGINES
CREW OF THREE
NASA STD. ATMOSPHERE
OPTIMUM ALTITUDE CRUISE
PAYLOAD OUTBOUND=TWICE PAYLOAD INBOUND

NORMAL GROSS WEIGHT
TAKE-OFF WEIGHT – HOVER O.G.E. 6000 FT. 95°F (11% POWER AUGMENTATION)
TAKE-OFF WEIGHT – HOVER O.G.E. 3000 FT. 100°F

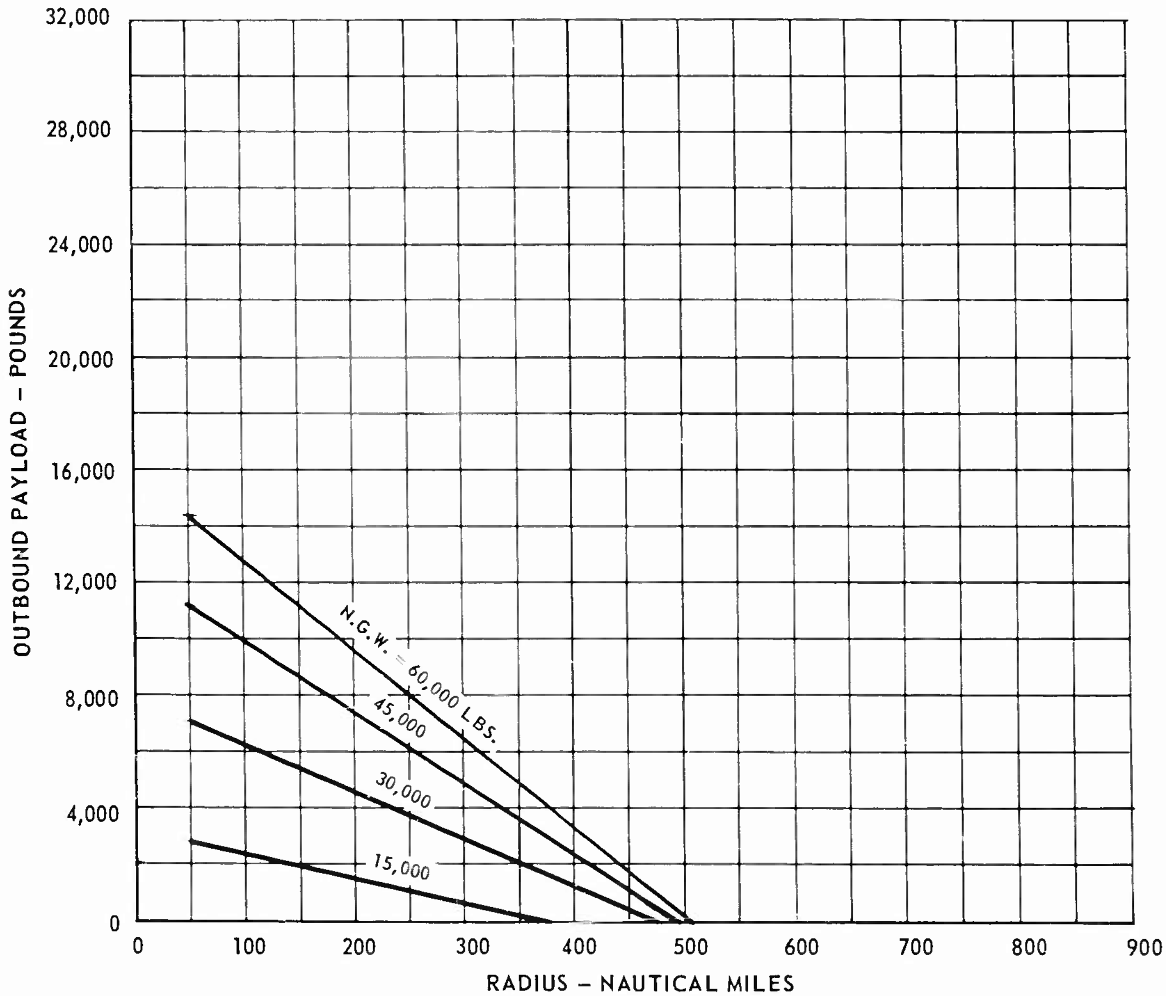


FIGURE 6.4

UNLOADED ROTOR COMPOUND HELICOPTER AIRCRAFT SPECTRUM

T58-GE-8 ENGINES
CREW OF THREE
NASA STD. ATMOSPHERE
OPTIMUM ALTITUDE CRUISE
PAYLOAD OUTBOUND=TWICE PAYLOAD INBOUND

TAKE-OFF WEIGHT – HOVER O.G.E. 6000 FT. STD. DAY $\left[\frac{\text{TAKE-OFF G.W.}}{\text{NORMAL G.W.}} = 1.12\right]$

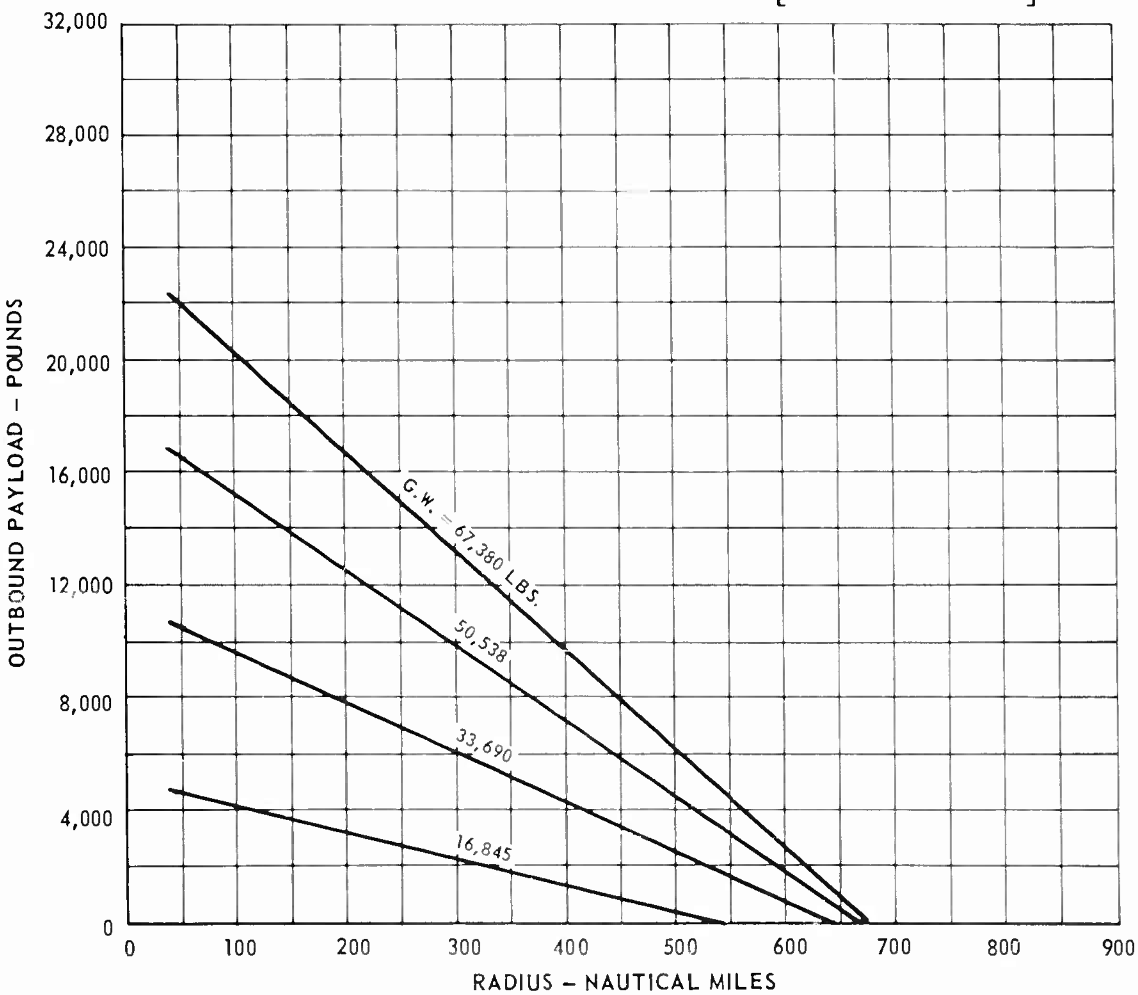


FIGURE 6.5

UNLOADED ROTOR COMPOUND HELICOPTER AIRCRAFT SPECTRUM

T58-GE-8 ENGINES
CREW OF THREE
NASA STD. ATMOSPHERE
OPTIMUM ALTITUDE CRUISE
PAYLOAD OUTBOUND=TWICE PAYLOAD INBOUND

HOT DAY GROSS WEIGHT

TAKE-OFF GROSS WEIGHT – HOVER AT SEA LEVEL 103°F $\left[\frac{\text{HOT DAY G.W.}}{\text{NORMAL G.W.}}\right] = 1.10$

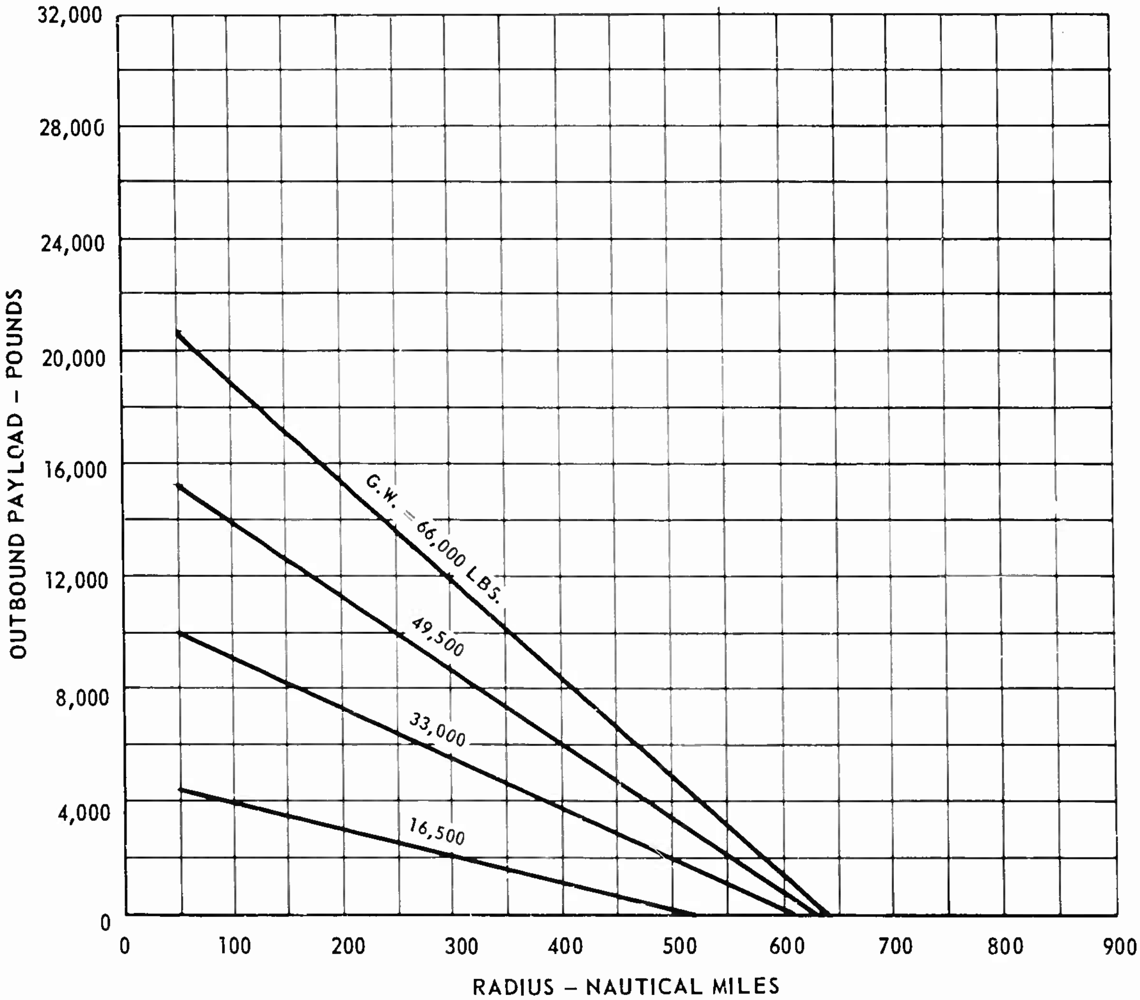


FIGURE 6.6

UNLOADED ROTOR COMPOUND HELICOPTER AIRCRAFT SPECTRUM

T58-GE-8 ENGINES
CREW OF THREE
NASA STD. ATMOSPHERE
OPTIMUM ALTITUDE CRUISE
PAYLOAD OUTBOUND=TWICE PAYLOAD INBOUND

HOT DAY GROSS WEIGHT

TAKE-OFF GROSS WEIGHT – HOVER AT SEA LEVEL 103°F (11% POWER AUGMENTATION)

$$\left[\frac{\text{HOT DAY GROSS WEIGHT}}{\text{NORMAL GROSS WEIGHT}}\right] = 1.2$$

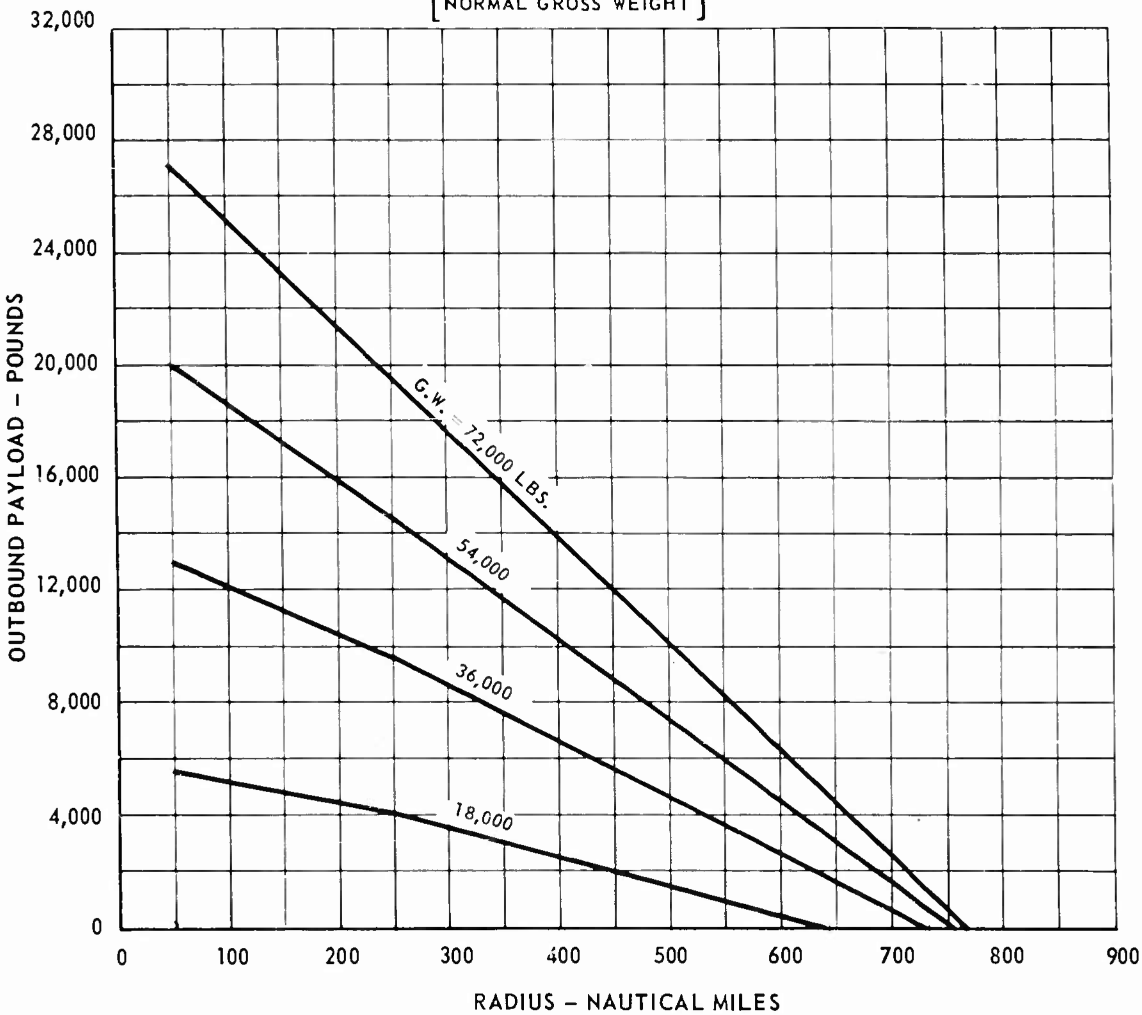


FIGURE 6.7

UNLOADED ROTOR COMPOUND HELICOPTER AIRCRAFT SPECTRUM

T58-GE-8 ENGINES
CREW OF THREE
NASA STD. ATMOSPHERE
OPTIMUM ALTITUDE CRUISE
PAYLOAD OUTBOUND=TWICE PAYLOAD INBOUND

VTOL OVERLOAD GROSS WEIGHT

TAKE-OFF WEIGHT – HOVER O.G.E. SEA LEVEL 60°F $\left[\dfrac{\text{OVERLOAD WT.}}{\text{NORMAL G. WT.}}\right] = 1.33$

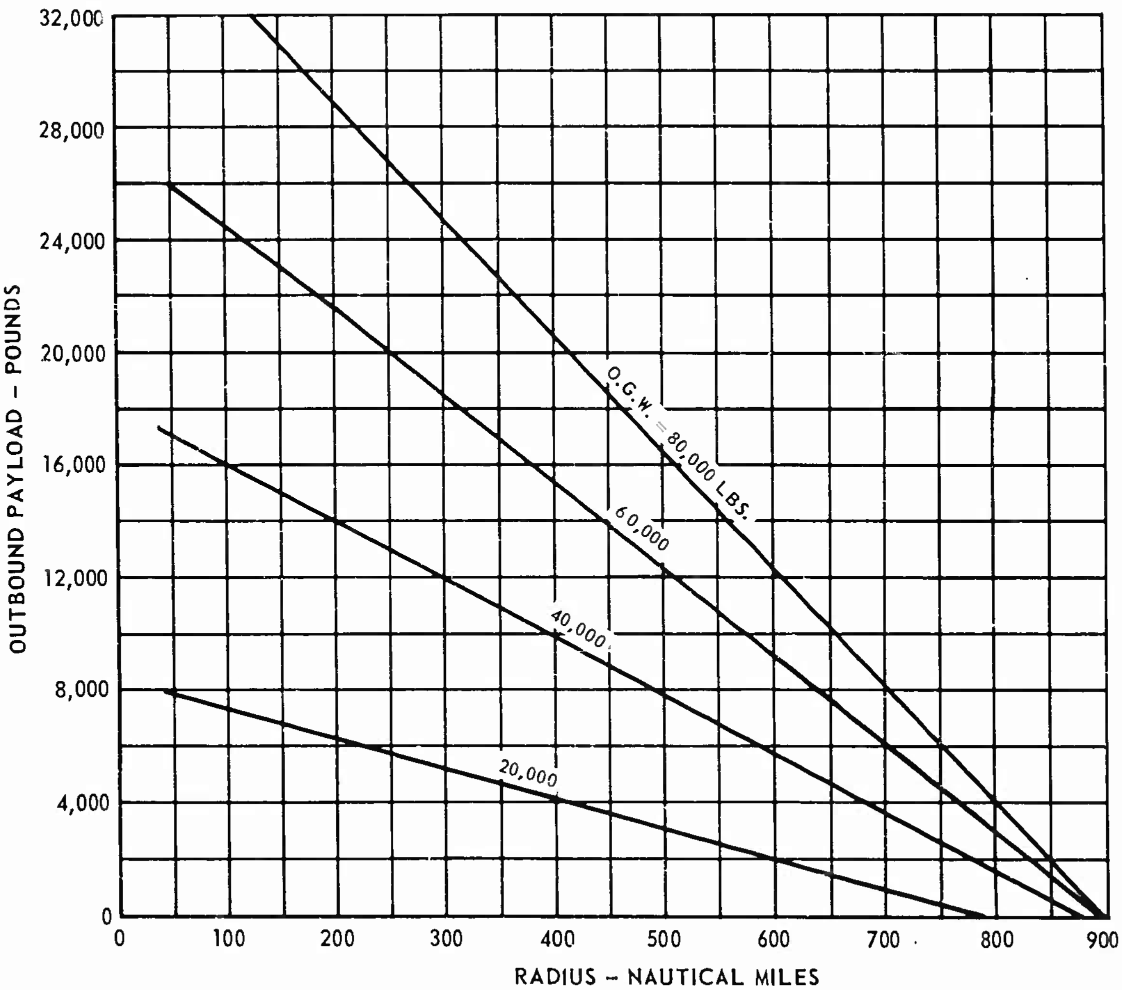


FIGURE 6.8

UNLOADED ROTOR COMPOUND HELICOPTER AIRCRAFT SPECTRUM

CREW OF THREE
NASA STD. ATMOSPHERE
OPTIMUM ALTITUDE CRUISE
PAYLOAD OUTBOUND=TWICE PAYLOAD INBOUND

NORMAL GROSS WEIGHT

TAKE-OFF WEIGHT – HOVER O.G.E. 6000 FT. 95°F (11% POWER AUGMENTATION FOR T58's) (5% POWER AUGMENTATION FOR T64's)

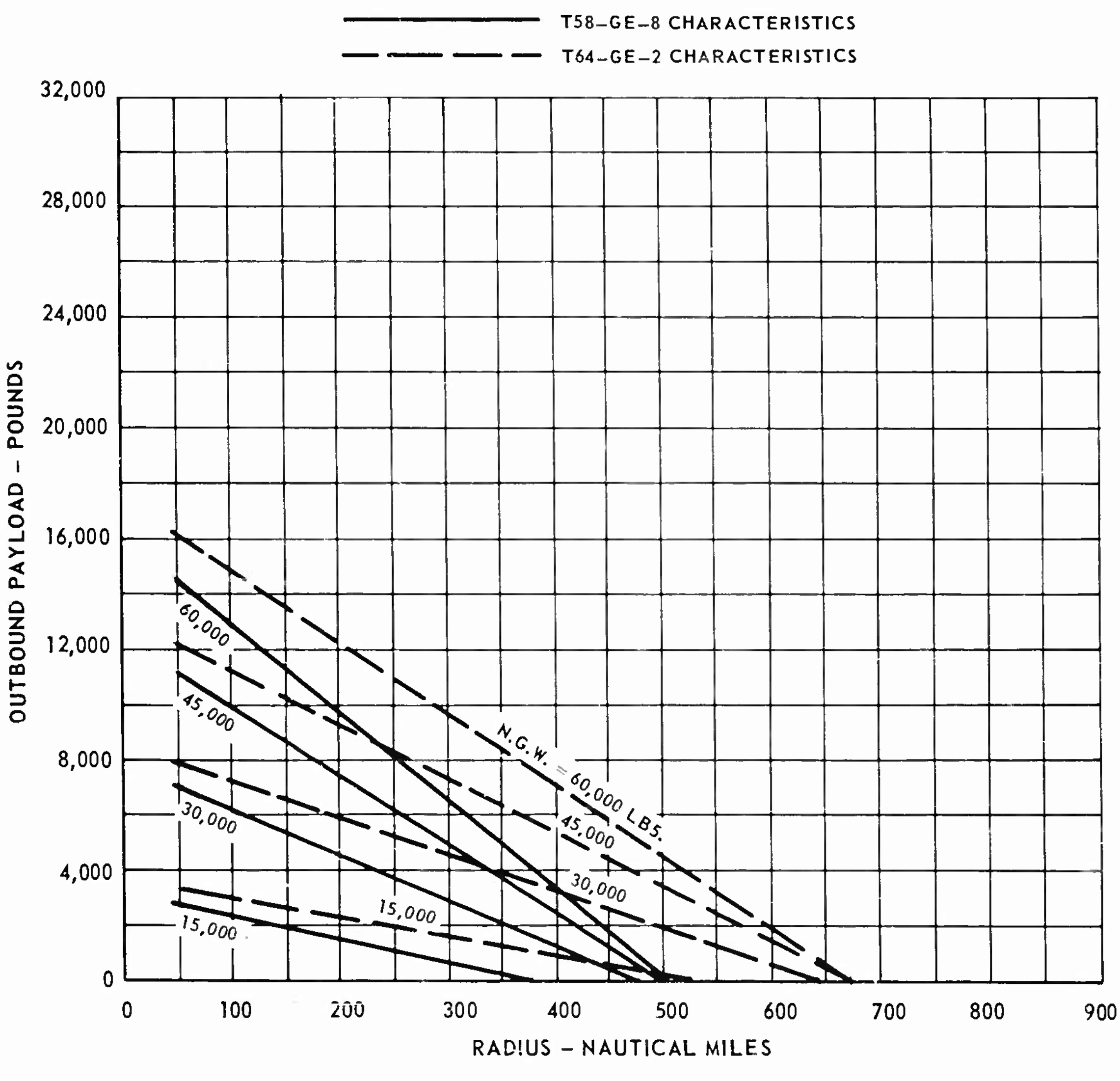


FIGURE 6.9

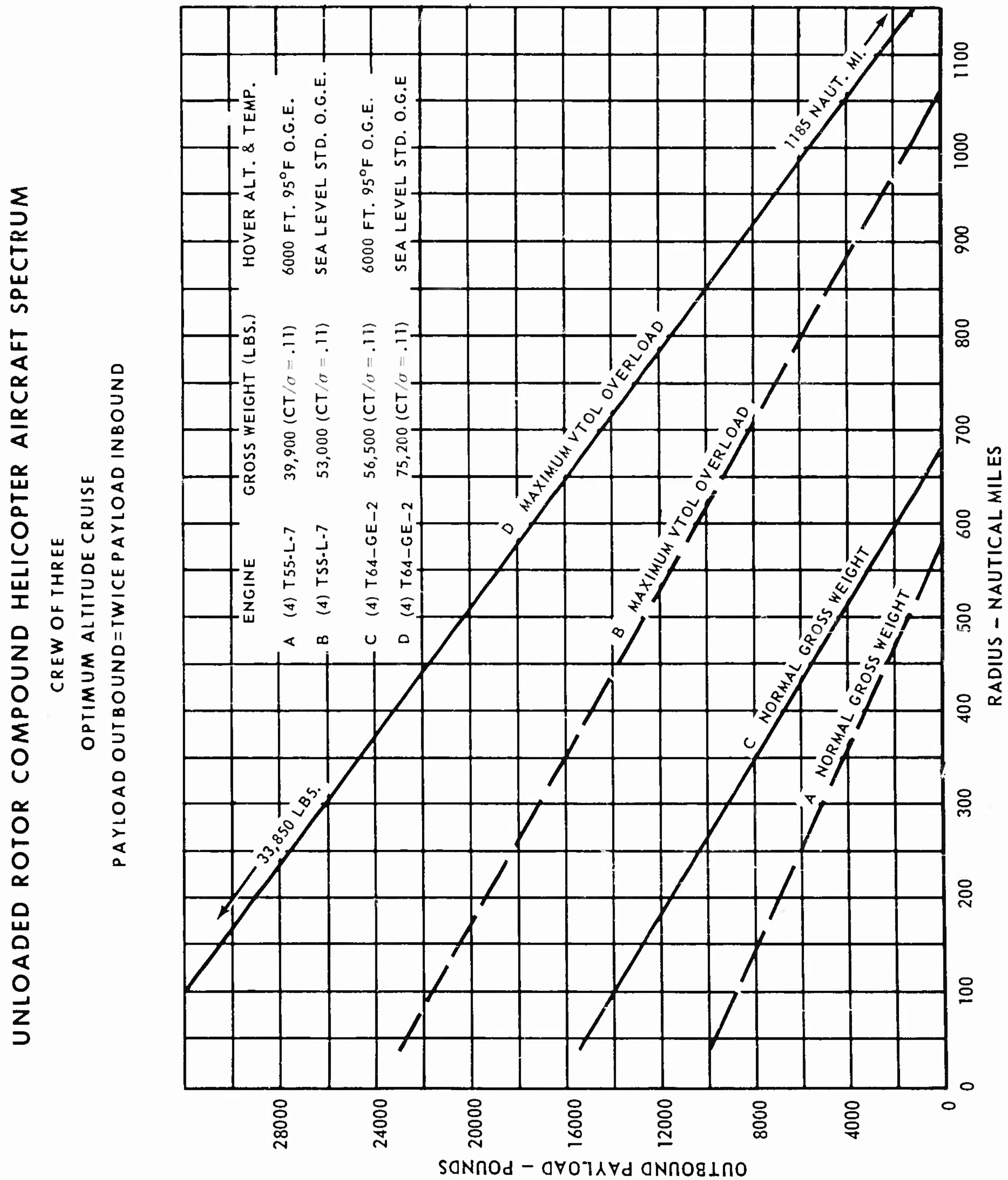

UNLOADED ROTOR COMPOUND HELICOPTER AIRCRAFT SPECTRUM
CREW OF THREE
OPTIMUM ALTITUDE CRUISE
PAYLOAD OUTBOUND=TWICE PAYLOAD INBOUND
ENGINE
GROSS WEIGHT (LBS.)
HOVER ALT. & TEMP.
A (4) T55-L-7 39,900 (CT/σ = .11) 6000 FT. 95°F O.G.E.
B (4) TSS-L-7 53,000 (CT/σ = .11) SEA LEVEL STD. O.G.E.
C (4) T64-GE-2 56,500 (CT/σ = .11) 6000 FT. 95°F O.G.E.
D (4) T64-GE-2 75,200 (CT/σ = .11) SEA LEVEL STD. O.G.E
33,850 LBS.
1185 NAUT. MI.
D MAXIMUM VTOL OVERLOAD
B MAXIMUM VTOL OVERLOAD
C NORMAL GROSS WEIGHT
A NORMAL GROSS WEIGHT
OUTBOUND PAYLOAD – POUNDS
RADIUS – NAUTICAL MILES
0
4000
8000
12000
16000
20000
24000
28000
100
200
300
400
500
600
700
800
900
1000
1100

FIGURE 6.10

SECTION VII

7. RECOMMENDED LIGHT VTOL TRANSPORT AIRCRAFT

7.1 Reasons for Recommendation - The results of the spectrum analysis in Figure 6.9 show that the 6000-foot 95°F VTOL gross weight for 4000 pounds of payload varies from about 25,000 pounds for 200 nautical miles radius of action to about 52,000 pounds for 500 nautical miles radius of action. Figure 6.9 also indicates that 500 miles radius with 4000 pounds payload can be obtained only with a power plant of T64-GE-2 characteristics and not with T58-GE-8 characteristics. In Section 3 it is recommended that a radius of action of 250 nautical miles with 4000 pounds payload be required for the Army light VTOL aircraft designed for out of ground hovering at 6000 feet 95°F. According to Figure 6.9 this recommended requirement gives a gross weight of about 30,000 pounds for T58-GE-8 engine characteristics and a gross weight of about 25,000 pounds for T64-GE-2 characteristics. Selecting 30,000 pounds normal gross weight and T58-GE-8 characteristics, Figure 6.6 gives a value of 400 nautical miles radius when taking off with an overload gross weight of 33,000 pounds at which hovering out of ground effect at sea level 103°F is possible. For the same selection Figure 6.8 shows, again for 4000 pounds payload, a radius of action of 690 nautical miles when taking off with an overload gross weight of 40,000 pounds at which the aircraft will hover out of ground effect at sea level standard temperature. It is believed that a 4000-pound payload aircraft for which the VTOL gross weight varies from 30,000 to 40,000 pounds between the 6000-foot 95°F condition and the sea level standard condition for which the respective radii of action vary between 250 and 690 nautical miles is best suited for the Army light VTOL transport missions.

Therefore, the 30,000-pound normal gross weight class, unloaded rotor compound helicopter is recommended. Further considerations leading to this recommendation are:

a. Compliance with Requirements - This class of aircraft is the smallest that satisfies the Army VTOL light transport aircraft design and performance requirements as outlined in Reference 12.1.

b. Aircraft Cost and Maintenance Levels - Minimum aircraft cost and maintenance levels are associated with the selection of the minimum gross weight aircraft capable of fulfilling the mission requirements inasmuch as these levels are established primarily by aircraft size once the aircraft type or concept is selected.

c. Power Plant Availability - Power requirements of this aircraft size are compatible with the availability of free shaft turbine power plants in the period 1960-63.

d. Tripartite Participation - Limitations on aircraft size are imposed by potential tripartite usage of the selected VTOL transport aircraft. The known transport mission requirements of the Army and Navy are sufficiently close to make Bi-Service participation feasible. Such participation, in addition to reducing individual Service developmental expenditures, would decrease unit production cost through increased number of units required.

The basic aircraft configuration is presented in Section 5. This selection plus the Army requirement of hover out of ground effect capability at 6000 feet 95°F establishes the aircraft power loading, thus the required installed power plant rating for the recommended 30,000-pound gross weight aircraft (approximately 5550 brake horsepower). A review of the 1960-63 free turbine turboshaft power plant spectrum shows that this power requirement is fulfilled by the following power plant installations:

a. Four T58-GE-8 gas turbines with 11 percent hovering power augmentation at 6000 feet 95°F.

b. Three T55-L-7 gas turbines without power augmentation.

c. Two T64-GE-2 gas turbines with 5 percent hovering power augmentation at 6000 feet 95°F.

The T58-GE-8 turboshaft installation in the 30,000-pound class, unloaded rotor compound helicopter is identified as the McDonnell Model 113P, an aircraft resulting from numerous engineering studies conducted during the past several years.

The two T64-GE-2 powered transport aircraft offer performance gains in payload-radius capability and maintenance gains for a compromise in the safety and operational reliability aspects. If greater payload-radius capabilities than those of the Army requirement were specified, a larger aircraft with four T64-GE-2 gas turbines would be preferred over an aircraft powered by an increased number of T58-GE-8 power plants. (See spectrum analysis, Section 6.) Dimensional data, performance, and mission data for the 30,000-pound normal gross weight aircraft are presented in the succeeding paragraphs.

7.2 Dimensional Data

7.2.1 Rotor Data

Airfoil section	$63_2A(1.5)15$
Rotor disc area, square feet	3320
Diameter, feet	65
Blade chord, inches	36.8
Number of blades	3
Theoretical blade twist, degrees	-8
Solidity ratio	.09

7.2.2 Wing Data

Airfoil section (Root)	64_3 -218
(Tip)	64_1 -212
Wing area, square feet	450
Span, feet	55.3
Aspect ratio	6.8
Mean aerodynamic chord, inches	101.1
Dihedral, degrees	-5.75° inboard .37 b/2 +1.2 ° outboard .37 b/2
Taper ratio	.50

7.2.3 Empennage Data

Horizontal Stabilizer	
Airfoil section (Root)	NACA 0015
(Tip)	NACA 0012
Area, square feet	90

Aspect ratio	6.0
Span, feet	23.2
Mean aerodynamic chord, inches	48.2

Vertical Surface	
Airfoil section (Root)	NACA 0015
(Tip)	NACA 0012
Area, square feet	87.6
Aspect ratio (geometric)	1.76
Mean aerodynamic chord, inches	87.9

Tail Rotor	
Diameter, feet	6.5
Number of blades	3
Solidity ratio	.24

7.2.4 Power Plant Data (113P)

Pressure jets (McDonnell Aircraft Corporation)
Engine (4) T58-GE-8

Propeller	
Gear ratio	13.44:1
Number	2
Diameter, feet	11
Activity factor	500
Blade angle at 3/4 radius, degrees	34.8

7.2.5 Weight Data (113P)

Weight empty, pounds	19,474
Design gross weight, pounds	29,650
Maximum take-off gross weight, pounds	40,000

7.3 Description of Systems

7.3.1 Propulsion System

7.3.1.1 General - The propulsion system for the unloaded rotor compound helicopter has two principle modes of operation. The gas turbine engines drive either the propellers during propeller powered autogyro and airplane flight or the compressor(s) which delivers air through the rotor hub and blades to the tip mounted burners equipped with noise and halo suppressing nozzles during rotor powered helicopter flight. In helicopter flight the rotor speed is controlled by a rotor fuel flow governor which governs the tip jet fuel flow to maintain the desired rotor speed. During conversion from rotor powered to propeller powered flight, the engine power is transferred from the compressor(s) to the propeller and vice versa by a twist grip on the collective pitch stick. Provisions are incorporated for partial power operation of both the rotor and propellers, if desired, during conversion. For propeller drive the compressor(s) is disengaged, the fuel flow to the tip jet units turned off, and the collective pitch reduced for rotor autorotation.

Only free turbine engines have been considered for the light VTOL transport because of their superior characteristics with regard to air compressor matching.

If engine output speed is held constant and the air compressor is sized to absorb full engine power on a standard day, insufficient power will be available to drive the compressor as ambient temperature increases above standard. Because of this, engines with limited speed variation capability must be matched with compressors sized to absorb full engine power on the extreme hot day; thus, full engine power cannot be absorbed at reduced ambient temperatures. Since the specific fuel consumption of shaft turbine engines increases as the power is decreased, a penalty in fuel consumption is imposed in addition to the loss in power. With a free turbine engine and its flat power versus output shaft speed characteristics, the compressor speed adjusts itself to absorb the full output of the engine as ambient temperature varies.

When reduced rotor power is required or when an engine is shut down, the pressure jet system air pressure drops below its maximum value. This in turn reduces the power required to drive the load compressors. If the load compressors are able to increase their speed under these conditions (as they are with free turbine engines), the full available engine power can still be absorbed by pumping an increased air flow. This increases the "engine out" rotor power available, and decreases the over-all specific fuel consumption during partial rotor power operation.

7.3.1.2 T58-GE-8 Configuration - This power plant package consists of two T58-GE-8 turboshaft engines, transmission system, single propeller shaft, and two air compressors consisting of the first seven stages of the T58-GE-8 engine compressor. This configuration is shown in Figures 7.1 and 7.2. As seen in Figure 7.1, the two engines are mounted side-by-side with a shaft and torque tube running forward from each main reduction gear to a combining and reduction gear box at the forward end of the power plant. A single propeller drive shaft comes out the front of this forward gear box. The aft power take-off on each engine drives into a speed increasing gear which, in turn, drives an axial flow air compressor. Identical hydraulically actuated clutches are used on both fore and aft power take-offs to permit the engaging or disengaging of each independently. The clutch horsepower and speed is compatible with the present state-of-the-art of clutch design. With free turbine engines, such as the T58-GE-8, the clutches can be engaged and disengaged at low shaft speeds and power setting, thus reducing the wear on the clutches. An accessory gear box is located at the aft end of the power plant and provides continuous power for driving aircraft accessories from either or both engines regardless of whether the compressors or propeller is being driven. The two engines in each nacelle are separated by a firewall with fire extinguishing protection, and are completely independent of each other in compressor drive during helicopter flight. In airplane and autogyro flight, either or both engines drive the fixed pitch propeller through a spur reduction gear box. Propeller spinner boundary layer bleed provides engine compartment cooling and uniform inlet pressure distribution in the short engine inlet ducts at the front of the nacelle. The air compressor air flow is supplied from a plenum chamber located over the wheel well with an inlet on the top side of the nacelle directly behind the wing carry-through structure. Separate power fan oil coolers are employed for cooling the rear gear boxes of each engine. A single ram air cooler is employed for the propeller gear box. Engine packages are interchangeable between left and right nacelles. Individual engines or complete packages are removable and replaceable in the field without special equipment through large quick-opening access doors in the bottom of each nacelle.

7.3.1.3 T55-L-7 Configuration - This configuration consists of three T55-L-7 turboshaft engines and one air compressor. Engines are located one in each nacelle and one in the rear of the rotor pylon. The single air compressor is located in the front of the rotor pylon. To transmit power to the compressor from all engines it is necessary to interconnect the three engines through a cross shaft. This insures one-engine-out capability and permits any or all of the engines to drive the propellers or air compressor. This configuration is shown in Figure 7.3. Engine input, propeller, and bevel cross-shaft drives are combined onto a single gear box attached as an integral part of the engine. A single gear box located in the pylon combines the engine drive cross shafts and, through a clutch, drives the compressor. One fan-driven oil cooler is provided for each gear box. Nacelle engine packages are made interchangeable from left to right by rotating the bevel drive housing. Engines alone are interchangeable in any of the three positions. Performance for this configuration has been based on the T55-L-7 engine which was on the approved list of engines for the Army Medium Transport Helicopter design competition in 1958. A similar version is currently under development for the Chinook helicopter, and a growth engine of higher performance may be available during the 1960-63 time period.

7.3.1.4 T64-GE-2 Configuration - This configuration consists of two T64-GE-2 turboshaft engines and one air compressor. The two engines are located one in each nacelle. The single air compressor is located in the rotor pylon. To transmit power to the compressor from the two engines, it is necessary to interconnect the two engines through a cross shaft. This insures one-engine-out capability and permits one or both of the engines to drive the propellers or air compressor. This configuration is shown in Figure 7.4. Engine input, propeller, and bevel cross-shaft drives are combined onto a single gear box attached as an integral part of the engine. A single gear box located in the pylon combines the cross-shaft drives and through a clutch drives the compressor. One fan-driven oil cooler is provided for each gear box. Ram air will aid in cooling the gear boxes during forward flight. An accessory gear box is located on the aft end of the main gear box to provide continuous power for driving aircraft accessories.

7.3.2 Rotor System - The rotor of an unloaded rotor compound helicopter provides the lift and propulsive force in helicopter flight. Almost all significant characteristics of the aircraft depend on the adequacy of the rotor system. Figure 7.5 shows a rotor hub cutaway while the system is discussed in the following paragraphs.

The McDonnell rotor incorporates three blades of high inplane stiffness. They are retained by retention straps to a gimbal-mounted hub. The retention straps carry centrifugal force and inplane bending loads. They consist of a bundle of thin straps, axially stiff, which permit blade flapping and pitching. The retention system eliminates all bearings, except for the lightly loaded spherical coning hinge bearing. The hub housing carries the loads from the blade torque tubes and retention straps. It is free to float in helicopter and autogyro flight by virtue of its gimbal ring attachment to the rotor support cone. Together with the lower air housing, it serves as a plenum chamber for distributing the ducted air to the torque tubes and out through the blades. For collective pitch values of less than 6 degrees, the hub housing becomes locked to the control stem through the hub lock rings and thus effectively becomes a fixed hub since the control system stem is restrained against tilting motion by irreversible cyclic controls. The hub is fully locked during airplane flight for a reference collective pitch (θ_0) of zero degrees. With the hub free there is a reduction in blade collective pitch when coning increases while flapping has no effect on pitch. With hub locked, blade pitch is effected also by flapping. A pitch link connects each pitch arm directly to the spider which in turn is mounted on the top of a tilting control stem.

Cyclic and collective pitch are accomplished through the spider which is an integral part of the control stem. The control stem provides cyclic control by tilting longitudinally and laterally with respect to the rotor support cone to which it is attached by means of the stem gimbal ring. Collective pitch is accomplished by raising and lowering the spider with respect to the outer stem, by means of the dual collective pitch cyclinders which are contained within the stem.

The rotor support cone supports the hub housing and the control stem and transfers the loads from the rotating system to the fixed rotor support base through the main rotor bearing. It carries the rotating portion of the rotor brake system and the ring gear of the auxiliary drive system. In addition to supporting the rotor cone and transferring its loads to the airframe, the rotor support base carries the air housing which is the fixed portion of the plenum chamber.

Flight action and operation of the McDonnell pitch-cone rotor system, as discussed in References 12.3 and 12.12, show the rotor system to exhibit outstanding stability and control characteristics.

7.3.3 Control System

7.3.3.1 General - The control systems are designed to give positive stick position and stick force stability and control about all three stability axes for all flight regimes. Surface and rotor controls, used individually or in combination, are utilized for control of the unloaded rotor compound helicopter. The control system combines push-pull controls and cables. Dual tandem hydraulic power cylinders are used in the rotor cyclic and collective controls. A single irreversible power cylinder is used for aileron control to avoid aileron feedback during helicopter flight. The rudder, stabilizer tab, and tail rotor controls are manually operated. A flight control schematic, showing fixed wing control surfaces, rotor controls, systems, etc., is presented in Figure 7.6.

7.3.3.2 Longitudinal Control - The combination of helicopter and airplane flight characteristics requires use of a suitable control system for both regimes of flight. Rotor downwash which reduces from large downwash angles in slow speed helicopter flight to small downwash angles in airplane cruise flight creates special problems in horizontal stabilizer design. A free floating stabilizer similar to the XV-1 convertiplane configuration is provided. Stability considerations with this free floating stabilizer system are discussed in Section 6.2; flight operation of this floating stabilizer is discussed in Reference 12.3. In essence the free floating horizontal stabilizer is hinged approximately at the aerodynamic center and is Vee-tab controlled by an anti-balance spring tab, deflected trailing edge up, and a servo anti-balance tab, deflected trailing edge down. The spring in the trim spring tab control tends to deflect the tab trailing edge up; at low airspeeds this positions the stabilizer nose up into the rotor downwash. As the airspeed is increased the tab deflection is reduced due to the increased air load, causing the stabilizer to move to lower incidence values. This is the desired movement for decreasing rotor downwash angles with increased speeds. The cyclic control is connected at all times to the servo tab of the horizontal stabilizer.

In helicopter flight longitudinal control is provided by a combination of rotor cyclic pitch control and the floating stabilizer control. Provisions are incorporated in the system to keep the stabilizer at the desired high incidence setting during hovering and very slow speed flight. The rotor system provides the propulsive force and major portion of the required lift.

In autogyro flight the longitudinal controls are identical to those in helicopter flight. The collective pitch lever is placed on a down-stop of a control shift mechanism which corresponds to autorotative collective pitch settings used during autogyro flight. The rotor rpm becomes a function of the longitudinal stick and tail trim position which control rotor incidence.

Longitudinal control in airplane flight is maintained essentially by the surface controls. The longitudinal rotor cyclic control is divorced from the pilot stick and its function is transferred to a rotor speed governor which senses rotor rpm and adjusts the rotor incidence to maintain a given autorotational speed. The control shift mechanism is moved to the bottom position corresponding to airplane flight. This action gives automatic rotor rpm control, centers and locks the hub with respect to the rotor shaft, and disconnects the cyclic stick from longitudinal control of the rotor.

7.3.3.3 Directional Control - In hovering and slow speed helicopter flight, directional control is obtained by use of a relatively small tail rotor; when sufficient forward speed is attained, additional directional control is provided by the rudder. For autogyro and airplane flight, directional control is obtained by rudder deflection.

7.3.3.4 Lateral Control - Lateral control is obtained by a combination of rotor lateral tilt and aileron deflection. The rotor lateral control is connected to the lateral cyclic stick for all flight regimes. The rotor and ailerons are connected to the cyclic stick through irreversible power cylinders.

7.4 Performance

7.4.1 Bases of Analysis

7.4.1.1 General - In general, the assumptions and methods of analysis used in the aircraft spectrum analysis are used to estimate the payload-radius characteristics of the recommended 30,000-pound normal gross weight aircraft. All aircraft payload-radius characteristics are determined for the Model 113 configuration using estimated weight statements (see Section 7.8), aerodynamic characteristics based on wind tunnel or whirlstand test data, and propulsion characteristics as obtained from the manufacturer's engine specification and/or McDonnell pressure jet system analysis. All aircraft are designed to meet the Army 6000-foot 95°F hover criterion. Power plant specific fuel consumptions are increased 5 percent for service tolerance, and a 10 percent initial take-off fuel is assumed as reserve for all missions.

Payload-radius charts are determined for three power plant installations - T58-GE-8, T55-L-7, T64-GE-2 - and three cruise altitudes - sea level, 10,000 feet, and optimum. For constant altitude cruise (sea level and 10,000 feet), the radii are determined from the nautical mile per pound of fuel approach; the cruise condition is established either by 99 percent maximum nautical miles/pound fuel or by a maximum power setting of 90 percent NRP. For optimum altitude cruise, the radii are determined by cruise at 90 percent power and constant lift coefficient or maximum aircraft lift-drag ratio (Breguet radius approach); thus the cruise altitude is variable. Mission profile charts are presented for these cruise conditions.

7.4.1.2 Hovering and Vertical Flight - The primary design objective for a hovering pressure jet rotor design is attainment of maximum rotor thrust per pound of net pressure jet thrust. This rotor thrust-tip jet thrust ratio (C_T/C_Q or T/F_J) is generally presented as a function of the hovering aerodynamic blade loading

(C_T/σ). The relationship between tip jet thrust and torque coefficient is

$$C_Q = \frac{F_J\ R}{\rho \pi R^2\ (\Omega R)^2\ R}$$

and the rotor horsepower tip jet relationship is

$$\text{Rotor Horsepower} = \frac{F_J\ (\Omega R)}{550}$$

Rotor thrust-tip jet thrust ratios versus aerodynamic blade loading for a pressure jet driven rotor with solidity of .09 and -8 degrees blade twist are detailed in Reference 12.23. These data are based upon whirlstand substantiated data from results of a Wright Field whirlstand calibration of the XV-1 rotor system. Included in the test data are allowances for pressure jet external drag, tip end plate effect, practical airfoil section, surface conditions, and compressibility. The equivalent maximum rotor figure of merit based upon shaft horsepower rather than installed power is .71.

Rotor efficiency in vertical climb is assumed to be the same as in hovering. This assumption is conservative since the additional mass of air handled in vertical climb reduces the induced power losses.

7.4.1.3 Helicopter Forward Flight - Rotor characteristics in helicopter forward flight are based primarily on NACA theory. As the NACA charts are for conventional rotors, certain modifications are required to account for the effects of tip jet drag, retreating blade stall, and compressibility. Tip jet units are assumed to have negligible effect on the aerodynamic characteristics of the rotor, except as a source of power loss due to their external drag. This drag contribution is expressed as a drag-lift ratio $(D/L)_J$ of the tip jet and is made a function of rotor advance ratio and rotor lift coefficient. The influence of retreating blade tip stall is derived from NACA tests for a conventional rotor. Compressibility effects on the profile power losses of a rotor have been investigated by the NACA for a hovering rotor. Adaption of the hovering investigation results for use in forward flight involves a survey of suitable blade element drag rise characteristics which, when applied to the rotor, produce results corresponding to the experimental results of the hovering tests. In applying the blade element drag rise data to match the experimental hovering torque rise results and in further applying appropriate data to obtain the profile drag-lift ratio increments for forward flight, the following principal relationships were useful:

Hovering,

Blade element angle of attack:

$$\alpha_r = \theta_{root} + x\ \theta_1 + \frac{a\sigma}{16} \left[1 - \sqrt{1 + \frac{32\ x}{a} (\theta_{root} + x\ \theta_1)} \right]$$

Blade element Mach number:

$$M_r = x\ M_t$$

Profile torque compressibility increments:

$$C_{Q_{OM}} = \frac{\sigma}{2} \int_x^1 \Delta C_{d_o} x^3 \, dx \quad \text{for } \Delta C_{d_o} = 0$$

Forward Flight,

Blade element angle of attack:

$$\alpha_r = \theta_{root} + x_1 \theta_1 + \frac{\lambda t + \mu a_1}{u_t} - a_1 \sin \psi$$

Blade element Mach number:

$$M_r = u_t M_t = (x + \mu \sin \psi) M_t$$

Profile drag-lift compressibility increment:

$$(D/L)_{OM} = \frac{1}{2\mu} \frac{1}{C_T/\sigma} \frac{1}{2\pi} \int_o^{2\pi} \int_x^1 u_t^3 \Delta C_{d_o} \, dx \, d\psi \quad \text{for } \Delta C_{d_o} = 0$$

The $(D/L)_{OM}$ so determined for compressibility is incorporated as an additive increment in determining helicopter forward flight power required.

7.4.1.4 Autogyro and Airplane Flight - Autorotational rotor characteristics for autogyro and airplane flight are obtained from summary data for a number of model rotor and full scale rotor wind tunnel tests. The data are for the complete rotor and, therefore, include drag of the rotor hub and retention assembly and whatever local intereference drag may be present. No corrections for Reynolds number are applied to rotor characteristics obtained from wind tunnel test data.

7.4.1.5 Parasite Drag - An estimate of parasite drag is obtained from a drag breakdown analysis. This analysis entails summation of component drag values obtained from component areas times their respective proper drag coefficients. The resulting equivalent flat plate area is increased by 10 percent to allow for unknown intereference factors. The equivalent parasite area used in the performance for airplane flight is 17.0 square feet, and for helicopter flight is 28.3 square feet.

7.4.2 Missions

7.4.2.1 General - Mission analyses are performed according to radius or range requirements, flight altitudes, and schedules. The basic transport radius mission schedule is the same as that used in the spectrum analysis in Paragraph 6.3 and is repeated here for discussion:

	Time	Mission Breakdown	Remarks
(1)	2 Minutes	Warm-up, Cockpit Check	Normal Rated Power
(2)	1 Minute	Take-off, Conversion	Maximum Power
(3)		Climb to Initial Cruise Altitude	Military Rated Power
(4)		Cruise Out at Constant C_L	90% Normal Rated Power
(5)		Descend, Land, Unload 1/2 Payload	No Distance Credit
(6)		Repeat Sequence for Return Trip	
(7)		Reserve Fuel	10% Initial T.O. Fuel
(8)		Service Allowance (MIL-C-5011A)	5% SFC Increase

Variations from the basic transport radius mission are accomplished by modifying items (4), (5), (6), and inserting desired hovering times as required for the mission, specified cruise altitudes, etc. In all cases items (1), (2), (3), (7), (8) remain the same.

7.4.2.2 Missions at Constant Altitude - For missions which require cruise at specified altitudes, the cruise distance is obtained from use of fuel consumption curves (nautical miles per pound of fuel) for a series of gross weights and altitudes in airplane flight. These curves evolve from the calculation of level flight power required curves and the corresponding power available curves. For cruise at a constant altitude or power setting the radius or range mission is determined by an iteration process, assuming average gross weight conditions, and applying the fuel consumption curves. Unless a specific cruise velocity or power setting is required, cruise at constant altitude is determined for conditions of 99 percent maximum nautical miles per pound of fuel (on the high velocity side) and not exceeding 90 percent normal rated power operation.

7.4.2.3 Missions at Optimum Altitude - Missions at optimum altitude are determined for conditions of maximum L/D and 90 percent normal rated power operation. The radius or range cruise distance is based upon the well known Breguet range equation. In essence, the missions consist of climbing to the altitude at which cruise flight originates and then maintaining conditions of maximum (L/D) ratio and 90 percent normal rated power as flight continues. This occurs as the altitude and velocity vary as fuel is consumed.

The lift-drag ratio of the unloaded rotor compound helicopter, unlike conventional aircraft characteristics, is dependent not only upon the operational lift coefficient but also upon the advance ratio at which the rotor is operating. For each rotor advance ratio condition, a corresponding L/D variation with lift coefficient therefore exists. For rotor operation at high advance ratios (μ values above about .80), the (L/D) ratios become coincident values; consequently, high advance ratio conditions are represented by one characteristic curve. This occurs in the operational range for cruise flight of the Model 113 aircraft.

The general Breguet range equation is:

$$\text{Range (nautical miles)} = 325.5 \left(\frac{\eta}{c}\ \frac{L}{D}\right) \log_e \left(\frac{W_o}{W_1}\right)$$

The product $\frac{\eta}{c}\frac{L}{D}$ is obtained from

$$\left(\frac{L}{D}\right)\left(\frac{\eta}{c}\right) = \frac{\text{SHP}\ \eta_p\ \eta_I}{\text{SHP x c}} \times \frac{LV_{kn}}{325.5\ \text{THP}} = \frac{WV_{kn}}{325.5 \times \text{Fuel Flow/Hour}}$$

where the gross weight (W) and velocity (V_{kn}) conditions are established by the unique altitude velocity combination for 90 percent power and cruise at maximum (L/D) ratio. The unique solution is obtained either directly from intersections on the power available and power required curves for the specified conditions, if the curves are available, or from the following equations involving density and velocity:

$$C_L = 295 \frac{L}{S\sigma V_{kn}^2}$$

$$(THP)_{req} = \frac{V_{kn}\, L}{325.5 \left(\frac{L}{D}\right)_{max}} \text{ and } (THP)_{avail} \quad f\,(\sigma,\ \eta,\ V) \text{ for 90\% NRP}$$

Knowing the aircraft gross weight or wing loading, power plant altitude and velocity (ram effect) characteristics, and over-all propulsive efficiency, the unique altitude-velocity combination for 90 percent normal rated power and cruise at maximum (L/D) ratio is defined.

Previous calculations and studies show that maximum range is obtained if the initial cruise altitude is reached in a minimum of time, i.e., climb to cruise altitude with military power. Once the initial cruise altitude is determined, the remaining cruise flight distance can be determined directly from the Breguet equation by substituting known quantities.

The method as presented is used to determine the cruise distance. To obtain the total distance, the distance covered during climb is added to the cruise distance; no distance is credited to the descent from cruise altitude. This procedure is used for both cruise out and return, the only difference being the altitude of operation.

The methods previously discussed are for cruise operation at specific conditions of $(L/D)_{max}$ and 90 percent normal rated power operation. Since free turbine engine characteristics show increased power with speed at near constant fuel flow (for constant altitude), there exists a condition of cruise operation at a higher velocity and at a slightly reduced (L/D) ratio in order to gain the effects of reduced fuel consumption. Studies show that conditions for maximum product of $[(\eta/c)\ (L/D)]$ for 90 percent normal rated power occur at altitudes about 1500 to 2000 feet lower and velocities from 5-10 knots higher than for the specific condition of $(L/D)_{max}$ operation. The values of $[(\eta/c)]\ [(L/D)_{max}]$ and $[(\eta/c)\ (L/D)]$ maximum product plotted against gross weight are nearly coincident. Mission analysis using both cruise methods results in nearly identical payload radius or range characteristics. This shows that cruise conditions for 90 percent power operation are relatively insensitive to altitude and velocity variation.

7.4.3 <u>Results</u> - As discussed in Section 6, the power plant SFC characteristics at assumed power setting and altitude give approximately a 10 percent radii improvement for the T55-L-7, and approximately a 22 percent radii improvement for the T64-GE-2 installation, relative to the T58-GE-8 installation. Also discussed is the effect of power augmentation at the design condition on the VTOL gross weight ratio for less stringent altitude-temperature combinations. The increased power ratings, especially the T55-L-7 gas turbine flat power rating, permit optimum cruise at higher altitude thus lower specific fuel consumption. The possibility of more efficient cruise operation at constant altitude (sea level and 10,000 feet) through shutting off engines also exists. Although this method is quite controversial some data on cruise with one engine shut off are presented. For the T55-L-7 installation, the most efficient cruise mode of operation is assumed; thus three-engine cruise is used for optimum altitude, two-engine cruise is used for the constant altitude conditions of sea level and 10,000 feet. For the T58-GE-8 installation, all data presented are for four-engine cruise even though some 7 to 10 percent radii benefits are achieved through shutting down one engine for constant sea level altitude cruise.

Table 7.1 presents a general performance summary for the recommended light VTOL transport aircraft; background data and methods of analysis are given in detail in Reference 12.2. Performance estimates are presented for two gross weight levels as limited by aerodynamic blade loading ($C_T/\sigma = .11$); normal gross weight at 6000 feet 95°F, and maximum VTOL overload gross weight at sea level standard atmosphere.

Figures 7.7 to 7.17, inclusive, present the payload-radius capabilities of Model 113. Figures 7.7 to 7.9 give the 2-ton payload-radius of operation as a function of take-off weight and cruise altitude. Figures 7.10 to 7.17 present the outbound payload-radius characteristics for each power plant installation, for three cruise altitudes (sea level, 10,000 feet, optimum), and for selected altitude-temperature combinations at initial take-off. Figures 7.18 and 7.19 show the mission altitude profile for the optimum cruise condition.

These data demonstrate that the Model 113 is capable of fulfilling the Army light VTOL transport aircraft requirements. The presentation of data for three different power plant installations permits the selection of the installation most comparable with competitive aircraft and brings to focus the effect of differences in power plant characteristics that must be eliminated in a valid comparison between aircraft concepts.

7.5 Flying Qualities - The flying qualities of the unloaded rotor compound helicopter incorporating the XV-1 principle have been ascertained through many years of theoretical study, model and full scale wind tunnel test programs, and experimental flight test of both the XV-1 convertiplane and the McDonnell Model 120, a crane, pressure jet helicopter designed around the XV-1 rotor system. References 12.3 and 12.13, the Air Force and Navy XV-1 flight evaluation reports, respectively, present specific flying qualities of the compound helicopter that confirm the high speed and outstanding stability, control, and vibration characteristics of this VTOL type. Reference 12.14 presents the XV-1 full scale wind tunnel data, and Reference 12.11 presents the results of wind tunnel test of a geometrically similar, eighth-scale model of the Model 113 incorporating configuration improvements resulting from the XV-1 test programs. The elimination of the XV-1 skid gear, twin boom empennage support system, and the rapid diffusion in the pylon, wing, fuselage junctions gave marked improvement in aircraft L/D and wing stall characteristics. Reference 12.15 presents an interim flight test report pertaining to the experimental development of the Model 120 helicopter, especially the development of low speed flying qualities. Figures 7.14 and 7.15 present some typical control response characteristics obtained during flight test of the XV-1 and Model 120 aircraft. All of these data substantiate that the Model 113 unloaded rotor compound helicopter should significantly improve the flying qualities, both in comparison to current operational rotary wing aircraft and in comparison to other VTOL aircraft types.

In vertical and low speed flight, the Model 113 is basically a helicopter; it uses accepted helicopter principles of lift generation and of control, thereby retaining the slow speed performance and handling characteristics of the pure helicopter. Features such as collective and cyclic pitch control, high rotor damping of aircraft angular velocities, autorotation capability, and ground cushion effects characterize this VTOL aircraft type. Added to these are the desirable features of the conventional fixed wing aircraft, control surface areas and empennage that increase the levels of stability and control response.

Exceptional VTOL aircraft flying qualities comprising inherent positive static and dynamic stability and high control response levels about all axes are provided by the Model 113 configuration. Instrument and all-weather flight capabilities are obtained without dependence on automatic stabilization equipment as primary systems. These major contributions to VTOL aircraft flying qualities are attained by the selection of the configuration, discussed in Section 5. In helicopter or low speed flight, the major contributor to flying qualities is the rotor. The McDonnell rotor system in comparison to the conventional Cierva-type rotor contributes large improvements to the aircraft flying qualities which have also been discussed in Section 5.

Flying qualities in airplane flight, cruise conditions, are established primarily by the airframe components and secondarily by the action of the rotor as controlled by the rotor constant speed governor. The rotor input in airplane flight is relatively small and it is a stable contribution resulting in aircraft response to control input similar to that of a fixed wing aircraft. As an example, values denoting maneuvering capability are given in the following table:

	Turning Radius (ft)			
Gross Weight (lbs)	Speed at Sea Level (kts)		Speed at 10,000 ft. (kts)	
	150	200	150	200
Minimum 21,000	830	850	1200	1200
Normal 29,650	1250	1250	2100	1800
Maximum 40,000	2100	1800	8500	2700

Both theoretical analysis and wind tunnel test of a complete eighth-scale model of the Model 113 including rotor verify that positive static stability about all axes is provided by the proposed configuration. Theoretical analysis also predicts good dynamic characteristics. A discussion of the Model 113 flying qualities as predicted by a combined test-theory approach is given in Reference 12.2.

7.6 Basic Structural Approach

7.6.1 General Design Criteria

7.6.1.1 Discussion - The aircraft structure conforms wherever applicable to the criteria set both in existing military specifications.

The aircraft design and operational characteristics are such that two main separate flight regimes are considered; namely, a helicopter flight regime in the speed range from hovering to 145 knots, and a partly over-lapping airplane flight regime in the range from 100 knots to 240 knots. The basic loading criteria are obtained from Reference 12.8 for the helicopter flight regime and from Reference 12.9 for the airplane flight regime. During the transition from helicopter to airplane flight and vice versa the rotor passes through a partially unloaded, autorotational phase. This phase is in general noncritical for the aircraft, and need not be investigated for the purpose of determining structural weight except insofar as it may influence the rotor fatigue life.

7.6.1.2 Symmetrical Flight Criteria - There is a difference in rotor load between a temporary aircraft angle of attack change experienced in a gust and a sustained aircraft angle of attack change occurring during a maneuver. For the temporary gust angle of attack change the rotor rpm control is too sluggish to respond to an appreciable degree within the duration of the gust. This fact has been observed during flight testing of the XV-1 aircraft and is explained by the high rotor moment of inertia. The gust load distribution between wing and rotor is, therefore, based on the assumption that the rotor lift increases with angle of attack according to the constant speed rotor lift slope. For the sustained angle of attack change possible in a maneuver the rotor governor responds by keeping the rotor angle of attack constant and no increment in rotor lift occurs. The flight envelopes in Figure 7.22 are computed for the two cases of gust load (temporary) and maneuver load (sustained) and are in accordance with Reference 12.8 for the helicopter portion and with Reference 12.9 for the airplane portion.

The level flight high speed, V_{H_e}, of 220 knots corresponds to the aircraft power limitation with normal rated power in level flight. The maximum design speed, V_{max_e}, is limited to 240 knots to provide a margin of 15 percent below the rotor flutter limit of 275 knots at sea level and to maintain an acceptable level of rotor fatigue stress. Flutter limit speeds increase with altitude, varying inversely as the square root of density ratio.

7.6.1.3 Landing Criteria - The landing, taxiing, and ground handling provisions of Reference 12.8 apply. In connection with the above provisions, the following is derived:

a. The maximum limit landing load factor at the aircraft c.g. is 2.67.

b. The maximum forward speed in an autorotative landing is 35 knots.

c. Crash landing requirements do not apply to the landing gear.

d. The gross weight for taxiing and ground handling is the overload gross weight.

7.6.1.4 Rotor Criteria - The rotor structural criteria conform to Reference 12.8 in the helicopter flight regime. In the airplane flight regime the loads are determined in accordance with the rotor aerodynamic characteristics for the applicable flight conditions of Reference 12.9.

The principal structural requirements for the rotor are established by rotor fatigue criteria, as discussed in the following paragraphs.

7.6.2 Rotor Structural Design

7.6.2.1 Loads Analysis - Steady loads are determined using conventional analytical procedures. Alternating loads are studied using one or more of the following approaches, each of which is discussed more thoroughly in Reference 12.16.

7.6.2.1.1 Empirical Approach - Wind tunnel tests and full scale flight tests on dynamically similar rotors form the primary sources of loads information. The tests conducted as a part of the XV-1, the Model 120, and the Navy 75-foot rotor programs and additional wind tunnel tests of dynamic models of a 65-foot diameter rotor provide most of these data.

7.6.2.1.2 Analytical Approach - Analytical techniques are used primarily to select appropriate scale factors to be applied to the empirical data.

For design purposes, several fatigue conditions believed to be critical are selected. These are the equivalent of operation at 1.6 g's (blade stall) in helicopter flight and cruising speed in airplane flight. It is intended that a design based on these loads have a life expectancy exceeding the useful life of the aircraft.

Another approach, purely analytical in nature, has been used with success for the high speed airplane flight regime and may be extended to lower speeds. This approach consists of a simplified mode expansion method, based on the mathematical background of Reference 12.17 and using rigid rotor air load forcing functions. This method employs an integrated damping coefficient, but neglects cross-coupling between harmonics in the Fourier series employed as well as between different bending modes. It has been found to be applicable to McDonnell rotors and is employed to supplement other loads data and as a means of scale factor determination.

7.6.2.1.3 Statistical Approach - The basis of this approach is the observation that the variation of loads to which rotor components are subjected appears to be represented by the log-normal statistical distribution for each flight condition. Correlation has been substantiated by studies of data for the XV-1 for both helicopter and airplane flight regimes and for the MAC Model 120 helicopter. This approach, which is discussed in detail in Reference 12.18, accounts, in a rational way, for each cycle of loading imposed on the aircraft for each flight maneuver. When combined with a schedule of anticipated flight maneuvers for each intended mission, the complete load history can be predicted for the life of the aircraft. Efforts are continuing to verify the accuracy of this approach and to the extent which can be justified, this approach is used to supplement the empirical and analytical studies, especially for predicting fatigue life of major components. After initial flight tests, when measured data can be used to confirm or perhaps modify the predicted loads, this spectrum of loading may be applied to the fatigue test specimens and thus provide a more accurate means of life evaluation.

7.6.2.2 Structural Analysis - Analysis for limit and ultimate loads is made using conventional procedures based on the most critical combination of steady and alternating loads as determined by the methods of Paragraph 7.6.2.1. Analysis for fatigue loadings is based largely on conventional analysis techniques using allowable stresses determined primarily from extensive fatigue tests. Such tests, conducted as part of the XV-1 and XHCH development programs, are reported in References 12.19 through 12.22. The allowable stresses are selected to provide adequate allowance for scatter in test results. When analyzing for the loads described in Paragraph 7.6.2.1.2, it is intended that the fatigue life will exceed the useful life of the aircraft.

When the statistical approach to loads analysis is employed, as described in Paragraph 7.6.2.1.3, the life is tentatively evaluated based on Miners' theory.

Examples of this approach are presented in References 12.18 and 12.23. After preliminary flight tests have confirmed or modified the predicted loads, fatigue tests are made to the loads defined by the spectrum loading. This permits the determination of fatigue life without relying on Miners' theory. A further advantage of this approach is that only a time to failure need be established and not an S-N curve. Hence, for a given cost, the fatigue test program will result in greater accuracy.

A further discussion of these methods, including applications to the 65-foot diameter high speed rotor, are presented in Reference 12.18.

7.7 Dynamics

7.7.1 Background - The operational acceptability of any rotary wing aircraft is determined to a considerable extent by its dynamic characteristics. Coincident with the research and development on the McDonnell unloaded rotor compound helicopter a continuous program of tests and theoretical analyses directed toward understanding and solving the dynamics problems has been carried on since 1949. This program included 27 wind tunnel tests, over one and one-half years of whirlstand tests, many weeks of direct analog computer analysis, and innumerable hours of digital computer analyses. The results of this program are discussed briefly in the following paragraphs in terms of the means of control and method of prediction of the various dynamics problems. More detailed discussions of the results of the dynamics program are given in References 12.24 to 12.27.

7.7.2 Rotor Vibration and Mechanical Instability - The vibration characteristics of the rotor are important in controlling blade loads, aircraft vibration, and mechanical instability. The principal means of avoiding excessive blade loads and rotor induced aircraft vibration is to avoid resonances of the rotor blades with the harmonic aerodynamic excitation in the rotor operational speed range. Mechanical instability can be completely eliminated from the operational speed range by maintaining a high inplane frequency.

The frequency diagram to which McDonnell rotors are designed is presented in Figure 7.23 in nondimensional form where both the frequency and the rotor speed are referred to the operating speed Ω_0 in hovering flight. This figure shows only the frequencies of the two most important vibration modes for the considerations mentioned; namely, the blade first cantilever inplane bending mode and the second flapping mode (first elastic bending). The frequencies of these modes are essentially the same for both the free and locked hub conditions. It is seen that the inplane frequency is 35 percent above the rotor speed at $\Omega/\Omega_0 = \overline{\Omega} = 1.0$, which places the lowest possible region of mechanical instability at least 45 percent above the maximum operating speed and provides a comfortable margin with respect to the one per rev excitation. The absence of a one per rev resonance in the range from zero to maximum operating speed is very desirable from the loads point of view when starting and stopping the rotor in high winds. The rather high inplane frequency was selected to provide some margin with respect to resonance with the exciting frequency equal to the sum of the rotor speed and the pilot's natural stick cyclic frequency so as to minimize pilot-induced inplane bending moments.

It is also seen in Figure 7.23 that only two resonances occur in the entire operating speed range from the helicopter to the airplane flight regimes. Both the two per rev inplane and three per rev vertical resonances occur at $\overline{\Omega} = .65$ which is in the speed range traversed rapidly in conversion between the two main flight regimes.

Experience in the design of the Model XV-1 and the Navy 75-foot pressure jet rotors has shown that the above frequencies can be obtained most easily for blades with a solidity of .03 and a thickness ratio of .15. The use of either higher or lower solidity-per-blade tends to reduce the latitude one has in adjusting frequencies in the detail design. With lower solidity-per-blade, it is difficult to obtain a sufficiently high frequency, while for higher solidity the resonances occurring in conversion flight tend to be shifted upward where a somewhat longer period of operation is required during conversion.

The means available for adjusting the frequencies in the design stage consist primarily of varying the strap spacing for the inplane bending and shifting the spanwise location of the outboard retention fitting for vertical bending.

In general, there is good agreement between theoretical predictions and experimentally determined blade frequencies. Both digital and/or analog computer analyses are available for predicting the uncoupled blade frequencies and evaluating the modifications of these caused by collective pitch, free and locked hub, and coupling with the airframe.

7.7.3 Rotor Flutter

7.7.3.1 General - During the investigation of the blade flutter characteristics of the MAC rotor, five distinct types of flutter were observed, the boundaries of which are shown in the non-dimensional forward speed-rotor speed diagram of Figure 7.24 as $\mu\Omega$ versus Ω. At high rotor speeds and low advance ratios, μ, when the hub is freely floating there is a flutter boundary for flapwise bending-torsion flutter of the advancing blade. Also, at low advance ratios and in hovering at lower rotor speeds, there may be a region of chordwise bending flutter (dotted curve in Figure 7.24). Over the middle rotor speed range, when the hub is locked to the control stem, there is a nearly constant forward speed limit which is influenced by torsional divergence in reversed flow over the blade, and is characterized by a subharmonic flapping motion of the blades. At still lower Ω the rotor blades behave as though they were rigid and the stability limit follows a constant μ line. At very low rotor speed, the rotor blades rest on the droop stops and a fifth type of dynamic instability, which can be described as droop stop pounding, may occur. Because of the pitch-flap coupling of the MAC rotor starting and stopping of the rotor in winds up to 100 knots presents no dynamic problem.

A typical flight envelope for an unloaded rotor compound helicopter is superimposed on the stability diagram. The actual rotor dynamic design is such that the unstable region at low advance ratio is eliminated.

7.7.3.2 Low Advance Ratio Flutter Involving Inplane Blade Bending - This type of flutter is brought about by coupling of the inplane blade bending mode with blade pitching. The flutter frequency is approximately equal to the inplane bending frequency in the advancing sequence cyclic mode (frequency in the stationary system equals frequency in the rotating system plus the rotor speed). Chordwise blade flutter is influenced by advance ratio, by collective pitch setting (hub unlocked), and by inflow velocity. At high advance ratio, low collective pitch setting and low inflow velocity the rotor may be stable. As the collective pitch setting is increased or as advance ratio is decreased, the rotor speed range over which flutter occurs widens and the degree of divergence of the motion at the center of the unstable range increases.

The most effective means found by analysis and model tests for eliminating this instability is the introduction of positive strap-to-hub incidence, i.e., raising the front strap attachment point at the hub and lowering the rear one. A reduction in the detrimental effect of collective pitch setting can be obtained by the addition of tip weights.

Flutter involving inplane bending can be analyzed quite accurately using the direct analog computer. Results of the direct analog analysis of the Navy 75-foot rotor were in good agreement with the model test results. An analog analysis in the preliminary design stage will be used to prevent the occurrence of this phenomenon in any new rotor design.

7.7.3.3 <u>Low Advance Ratio Bending-Torsion Flutter</u> - This type of flutter is the one common to most helicopter rotors due to the inherently low torsional flexibility in the blades and control system. The stability limit (limiting rpm) can be raised effectively by a forward shift of the blade chordwise center of gravity location, particularly in the outer third of the blade.

Both digital and direct analog analyses will predict the flutter limits with sufficient accuracy to insure freedom from flutter in the operating range of a new rotor design.

7.7.3.4 <u>High Advance Ratio Flutter</u> - At high advance ratios it has been found that a rotor blade can become unstable in flapping motion with a frequency which is a subharmonic of the rotor speed. This type of flutter is of prime concern in the design of an unloaded rotor compound helicopter since it may limit the maximum forward flight speed. Because this is a new type of flutter not encountered in low speed helicopters, more than half of the wind tunnel tests and a large portion of the analog and digital computer analyses in the dynamics program have been directed to the investigation of this phenomenon.

Results of the analytic and experimental analyses of the high μ flutter have shown that in airplane flight regime the rotor will be operating in the rpm range where the flutter limit is influenced by blade flexibility (see Figure 7.24). Improvement in the limiting forward speed in this region has been found to result from a forward shift in the blade chordwise center of gravity, from added tip weight, and from increased torsional stiffness. The amount of improvement obtainable from shifting the center of gravity forward diminishes once the center of gravity is forward of the aerodynamic center in normal flow.

Examination of the theoretical and test data has demonstrated that there is a correlation of the flutter limits with the torsional divergence speeds in normal and reversed flow. This means that there is a requirement for high torsional stiffness, particularly in the control system and root end of the blade, to obtain high forward speed. As would be expected from the torsional divergence concept, the flutter speed is influenced by the blade mass factor, γ, the forward speed being proportional to a value between $1/(\gamma)1/2$ and $1/(\gamma)1/4$. This means that the flutter speed increases with increasing altitude.

Also, using the torsional divergence concept, the flutter limits are influenced by blade solidity through the variation of the torsional stiffness and aerodynamic pitching moment per unit pitch change. The flutter speed for a given type of blade construction and constant airfoil thickness ratio will be proportional to the solidity raised to the one-half power or somewhat greater. From experience on design of the XV-1 and the 75-foot rotor systems, it appears that blades with a solidity of .03 and a thickness ratio of .15, which give the desirable vibration characteristics, are also a favorable choice from flutter considerations. The use of higher solidity-per-blade, however, does lead to difficulty in obtaining sufficient control system stiffness to match the high blade stiffness.

The accuracy of the present available analog and digital computer analyses is sufficient for design purposes. The correlation with experimental results is best in the low Ω, high μ region where the highest forward speed is required. Experimental verification by wind tunnel model tests can be made in the final design phase.

7.7.4 Airframe Vibration - Since the vibration level of a rotary wing aircraft is dictated by the combination of rotor, airframe, and power plant dynamic system, careful determination of the dynamic characteristics during the design of the pylon, fuselage, wing, tails, and nacelles is of major importance. Preliminary estimates of the frequencies of the major structural components are obtained from statistical data to determine the most critical problems. These estimates are followed up by detail vibration analyses. An attempt is made to eliminate resonances from the helicopter and airplane flight rotor operating speed range by shifting the frequencies up or down depending upon the inherent frequency range in which the components fall. In the case of the pylon, however, the frequency is placed in the order of 20 percent above the three per rev exciting frequency in helicopter flight rpm so that it is not necessary to traverse this resonance in conversion flight. The power plant package is mounted in the nacelles with provisions for inclusion of vibration isolation mounts. The normal range of isolation mount stiffness variation is sufficient to allow tuning of the frequencies after the airframe is built.

Digital computer methods are available for analyzing airframe vibration. However, the vibration analysis of the complete airframe is of such complexity that there exists a margin of error between predicted and actual frequencies. For this reason, provisions for tuning of the critical aircraft components are incorporated wherever possible.

7.7.5 Airframe Flutter - The speed range of the unloaded rotor compound helicopter is in the range where wing-aileron and aft fuselage-stabilizer-control surface flutter must be investigated. Flutter involving conventional control surfaces can generally be suppressed by proper balance of the control surfaces. Experience with the Vee-tab controlled floating horizontal stabilizer on the XV-1 demonstrated that this control can be stabilized by suitable mass balancing. The balance requirements were found to be compatible with those for good aerodynamic operation.

Digital computer analysis will be used to investigate flutter of the airframe components. A dynamic model of the tail surfaces will be used to verify the stability limits before flight.

7.8 Weights

7.8.1 Weight Statements - Estimated Model 113 weight statements for three different engine arrangements are presented in Table 7.2. Column 1 shows estimated weights with four T58-GE-8 engines. Column 2 shows estimated weights with three T55-L-7 engines, and Column 3 is with two T64-GE-2 engines.

7.8.2 Cargo Center of Gravity Limits - The limits of the cargo placement in the cabin for varying payload are shown in Figure 7.25. With a 4000-pound payload, the cargo can be placed anywhere between stations 225 and 300, a distance of more than 6 feet. These allowable limits are compatible with the recommended allowable c.g. travel equal to 15 percent of the mean aerodynamic chord (Paragraph 4.4).

TABLE 7.1

RECOMMENDED LIGHT VTOL TRANSPORT PERFORMANCE SUMMARY

Power Plant Installation	(4) T58-GE-8		(3) T55-L-7		(2) T64-GE-2	
Performance	NGW	VTOL Overload	NGW	VTOL Overload	NGW	VTOL Overload (5)
Gross Weight (lbs)	29,650	40,000	29,650	40,000	29,650	40,000
Hover Ceiling, OGE (ft) Standard Atmosphere } Note (3) 95°F at Altitude	9700 6000 (1)	Sea Level --	9700 6000	Sea Level --	9700 6000 (2)	Sea Level --
Max. Rate of Climb' (ft/min) Airplane Flight, S.L. Standard	2000	1100	2400	1400	2300	1350
Max. Vertical Rate of Climb (ft/min) S.L. Standard	1500	0	2200	0	2000	0
Maximum Speed (knots)	236	228	257 (4)	250 (4)	238	228
Maximum Radius (naut. miles) 4000 lbs. Outbound Payload	235	695	285	825	350	925
Ferry Range (naut. miles)	1060	1870	1255	2200	1425	2450

(1) Requires 11 percent power augmentation.
(2) Requires 5 percent power augmentation.
(3) C_T/σ limit with ΩR = 735 ft/sec.
(4) Maximum free turbine RPM limit.
(5) VTOL overload of 40,000 pounds ignores engine out performance criteria.

TABLE 7.2

MODEL 113 - ESTIMATED WEIGHT STATEMENTS

	(4) T58-GE-8 Report # 6248	(3) T55-L-7	(2) T64-GE-2
	1	2	3
Gross Weight (Overload)	40000	40000	40000
Useful Load (O.L.)	20526	20401	20638
Crew (2)	430	430	430
Engine Oil	152	152	100
Unusable Fuel & Oil	42	42	42
Equipment	100	100	100
Water	112	--	55
Fuel Plus Payload	19690	19677	19911
Gross Weight (Design)	29650	29650	29650
Useful Load (Design)	10176	10051	10288
Crew (2)	430	430	430
Engine Oil	152	152	100
Unusable Fuel & Oil	42	42	42
Equipment	100	100	100
Water	112	--	55
Fuel Plus Payload	9340	9327	9561
Weight Empty	19474	19599	19362
Rotor Group	4954	4954	4954
Blades	1586	1586	1586
Retention	1681	1681	1681
Hub	1687	1687	1687
Wing Group	1820	1814	1817
Tail Group	525	525	525
Horizontal and Vertical Surface	360	360	360
Tail Rotor	165	165	165
Body Group	3087	3087	3087
Fuselage	2787	2787	2787
Pylon	300	300	300
Alighting Gear	870	870	870

TABLE 7.2 (continued)

	(4) T58-GE-8 Report # 6248	(3) T55-L-7	(2) T64-GE-2
	1	2	3
Flight Controls	380	380	380
Engine Section	500	482	466
Propulsion Group	5217	5376	5157
Engines	1068	1638	1420
Tip Burners	500	500	500
Load Compressors	240	374	346
Air Induction System	26	36	36
Exhaust System	62	94	62
Lubricating System	208	140	205
Fuel System (513 gals.)	480	480	480
Engine Controls	40	45	35
Starting System	304	301	292
Propellers	660	660	660
Engine Water Injection	41	--	26
Transmission System	1352	1058	1034
Prop. Gear Boxes & Clutches	658	712	504
Engine Gear Boxes	312	--	(1)
Nacelle Bevel Gear Boxes	--	--	218
Compressor Gear Boxes & Clutches	332	288	232
Engine Shafts	(1)	(1)	(2)
Compressor Shafts	--	8	--
Cross Shafting	--	50	80
Mounting Yoke	50	--	--
Rotor Air Ducts	236	50	61
Instruments & Navigation	125	115	110
Hydraulics	229	229	229
Electrical System	685	685	685
Electronics	602	602	602
Furnishings & Equipment	380	380	380
Heating & Vent. System	100	100	100

(1) Included with Propeller Gear Boxes.
(2) Included with Cross Shafting.

T-58 ENGINE INSTALLATION

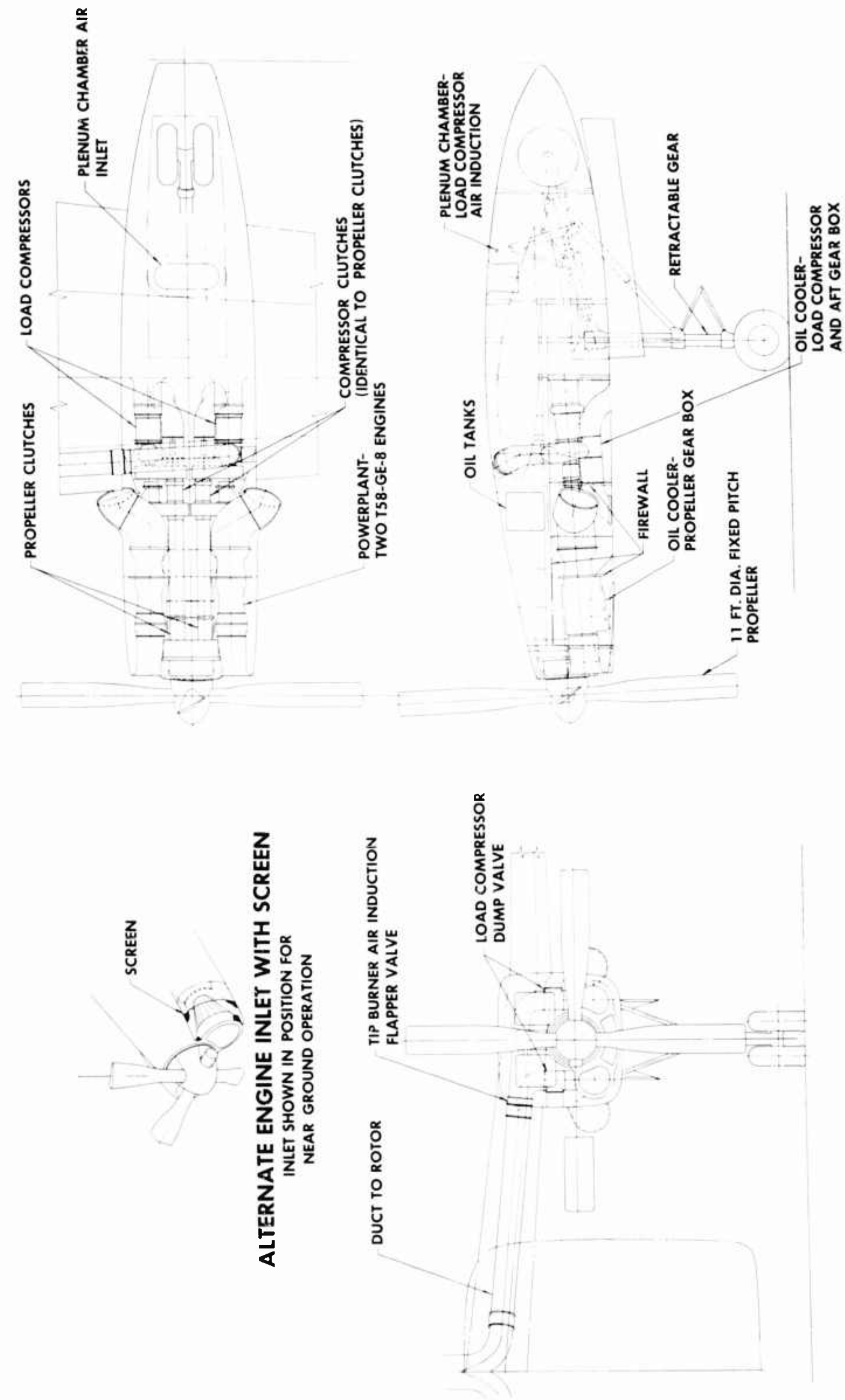


FIGURE 7.1

PROPULSION SYSTEM
T-58 ENGINE INSTALLATION

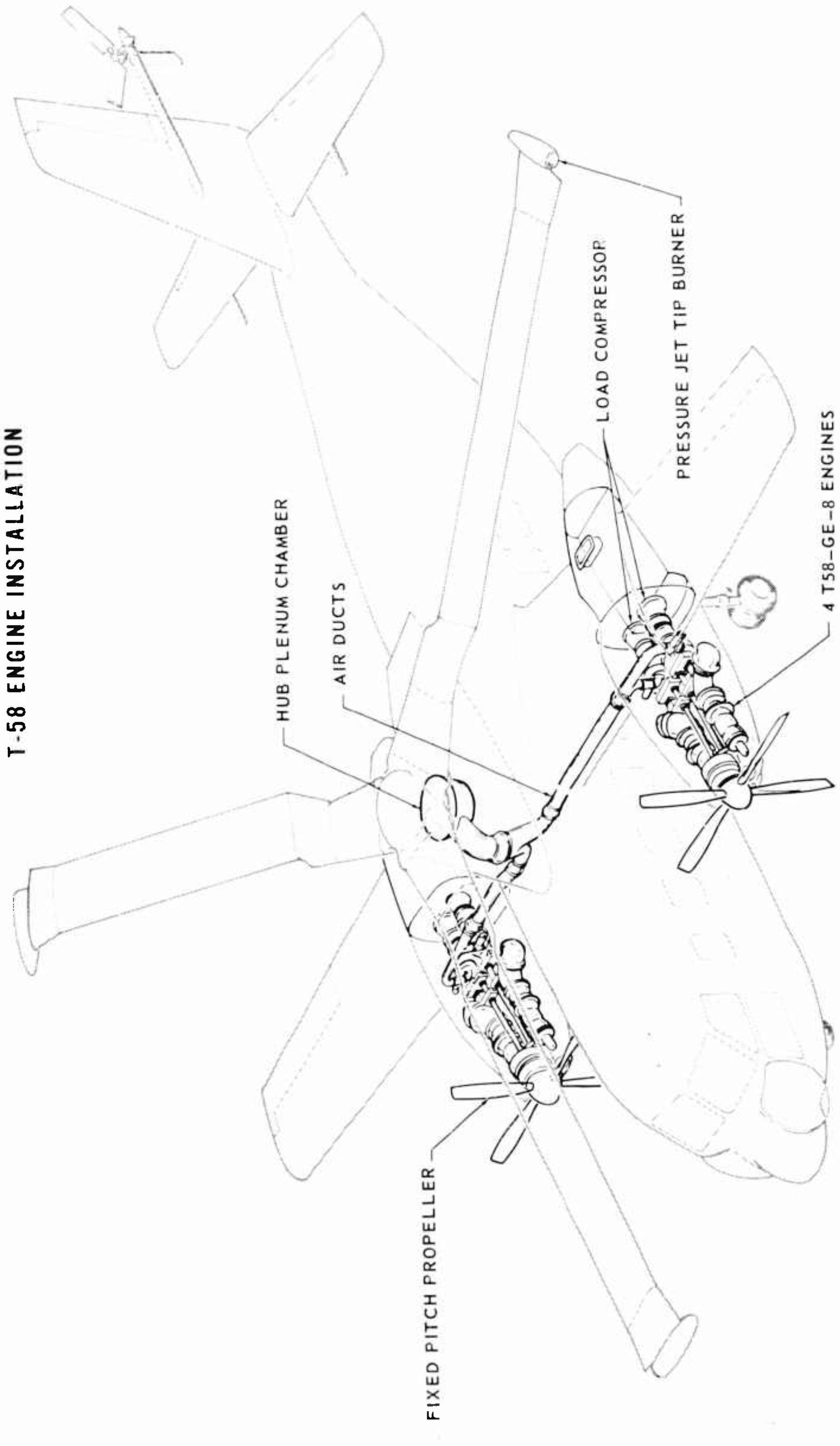


FIGURE 7.2

T-55 ENGINE INSTALLATION

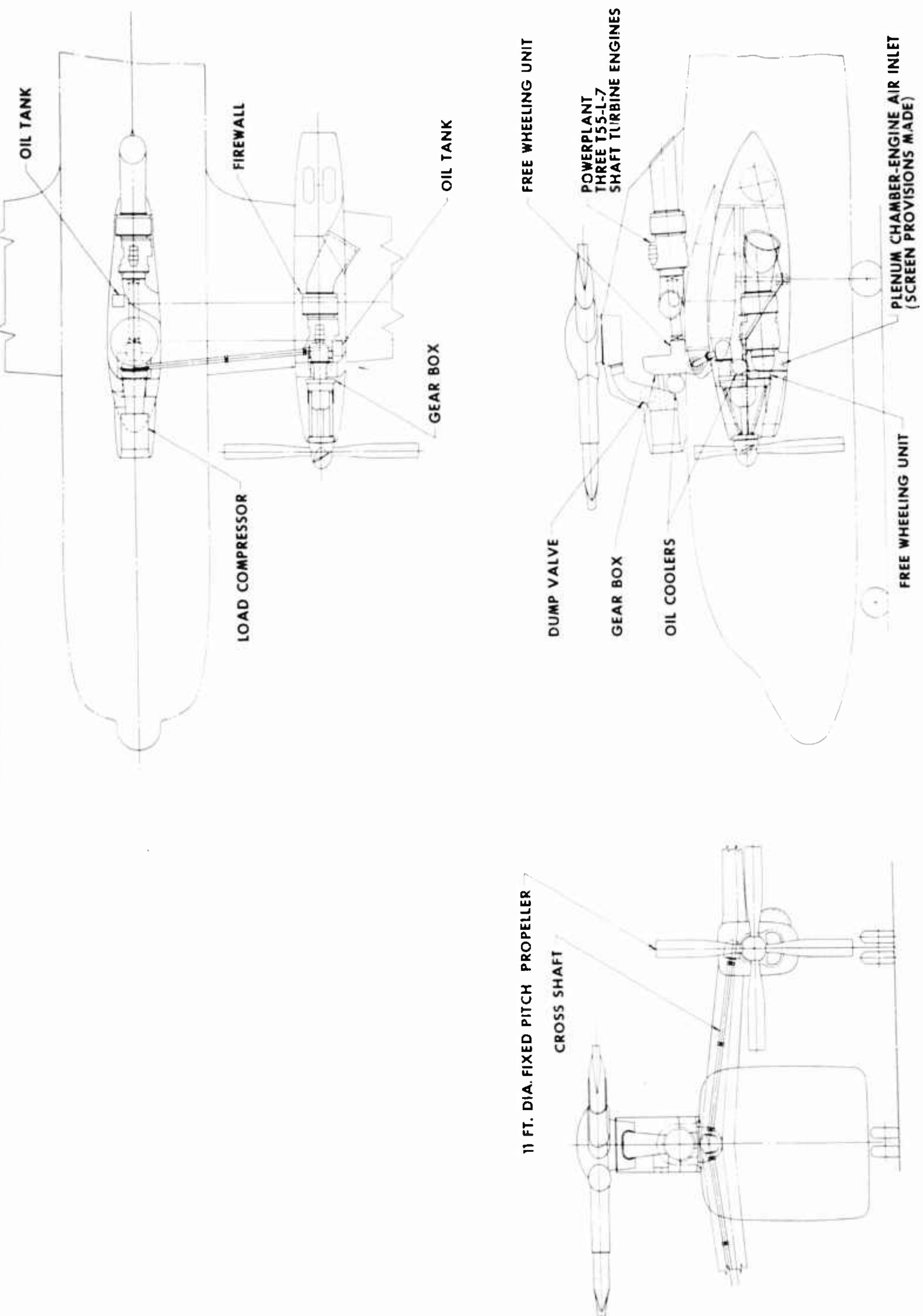


FIGURE 7.3

T-64 ENGINE INSTALLATION

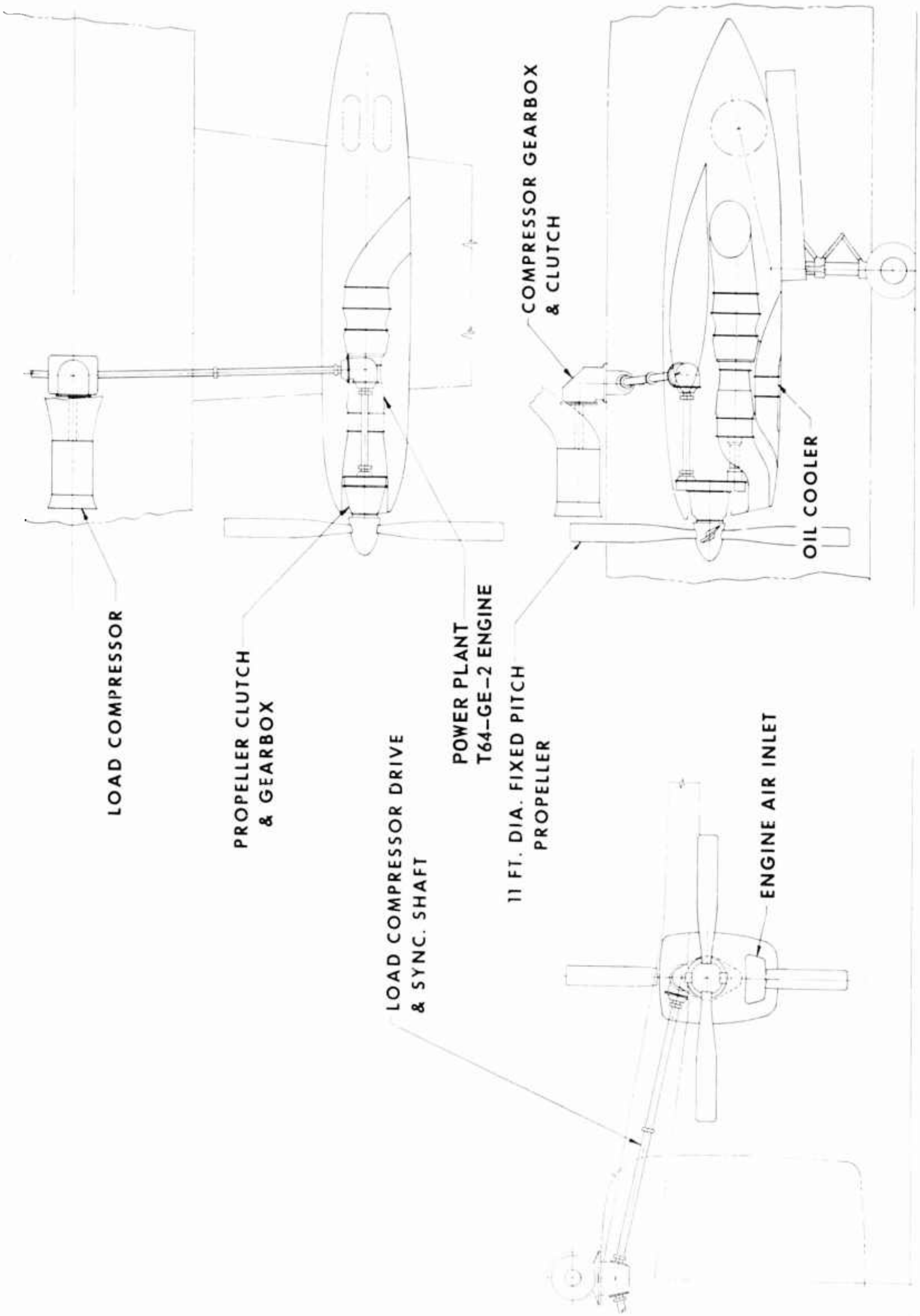


FIGURE 7.4

ROTOR HUB CUTAWAY

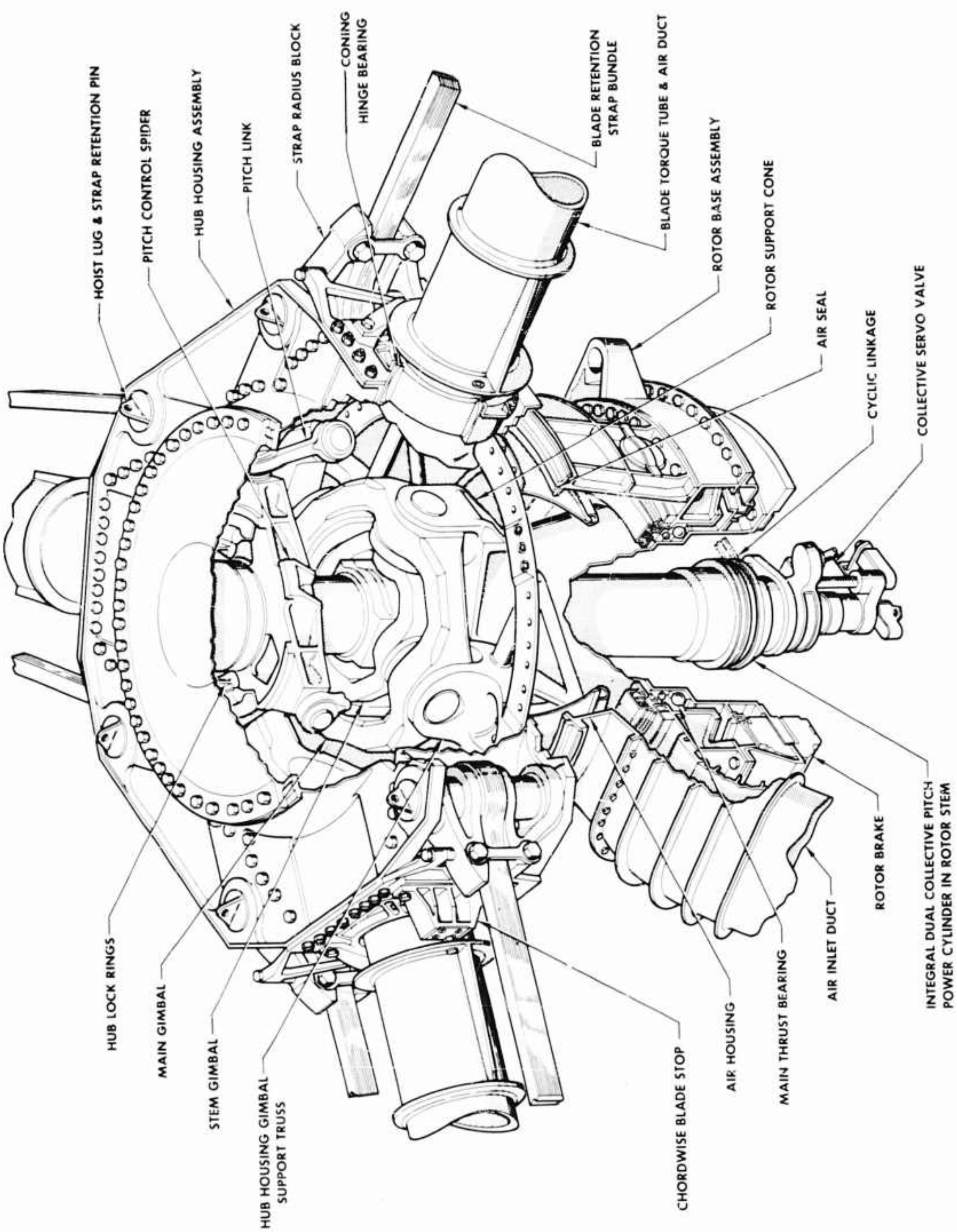


FIGURE 7.5

FLIGHT CONTROLS

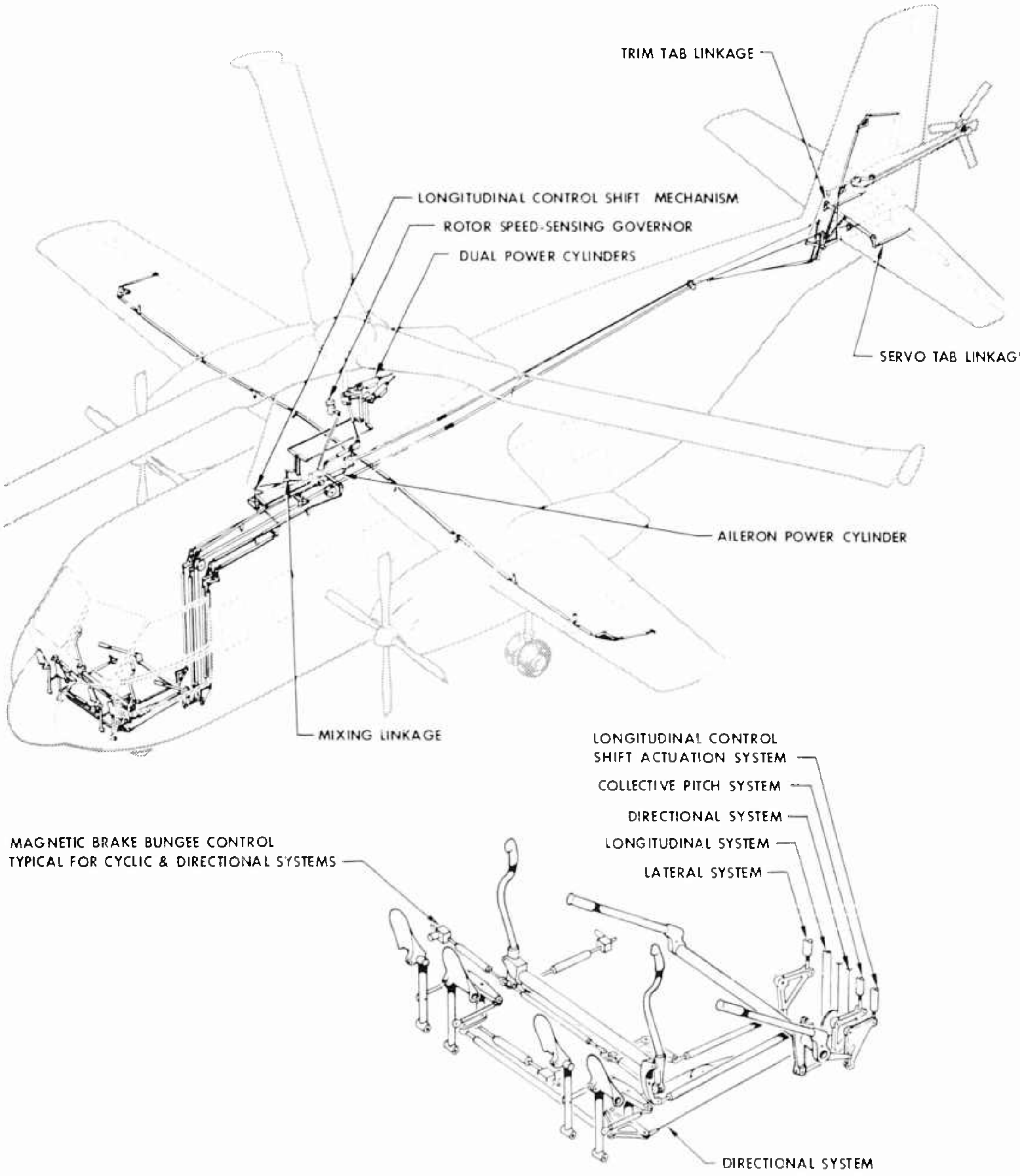


FIGURE 7.6

MODEL 113P VTOL TRANSPORT
RADIUS VS. TAKE-OFF GROSS WEIGHT

NASA STD. ATMOSPHERE
4 T58-GE-8 ENGINES
PAYLOAD = 4000 LB. OUTBOUND, 2000 LB. INBOUND

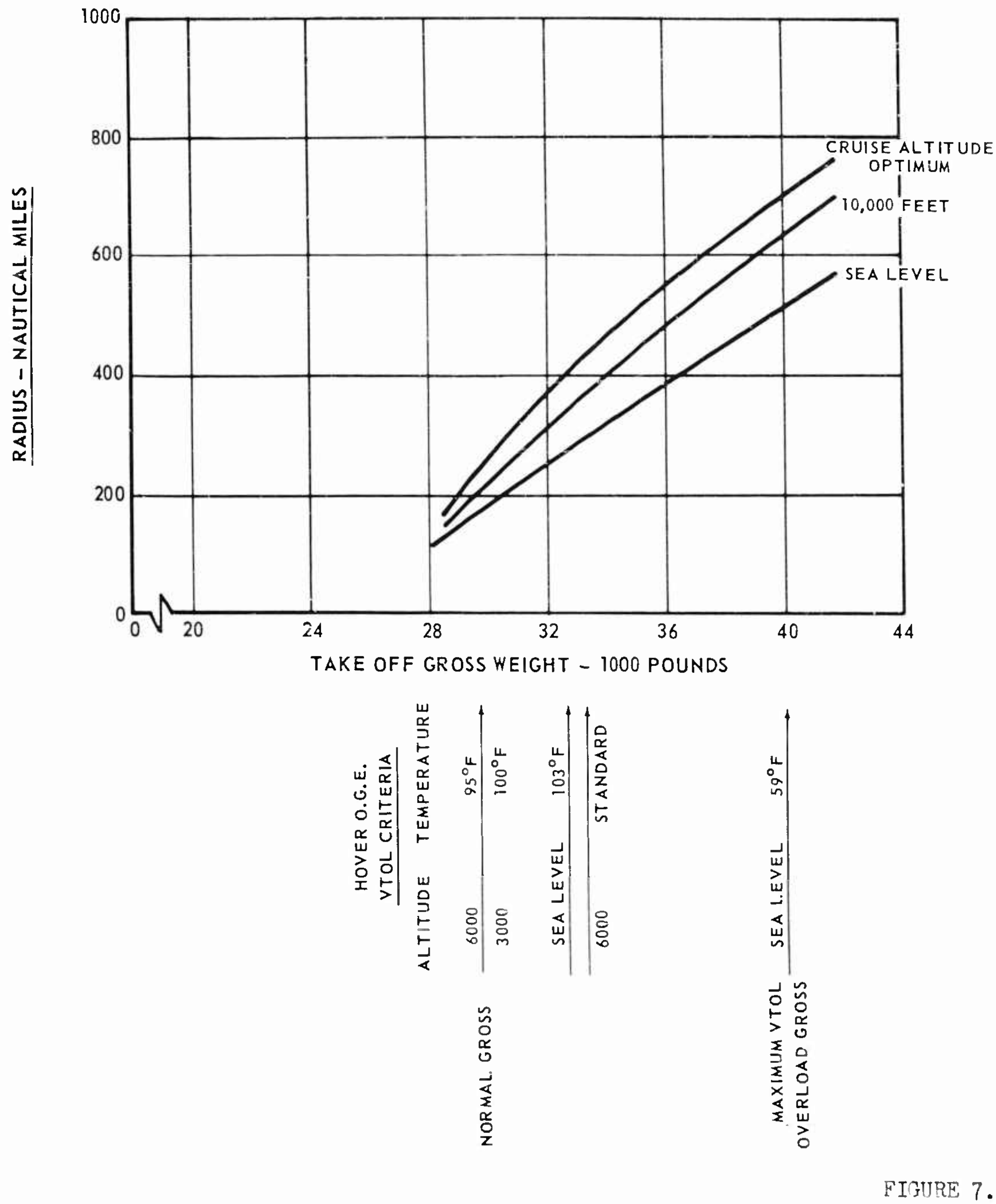


FIGURE 7.7

MODEL 113 VTOL TRANSPORT
RADIUS VS. TAKE-OFF GROSS WEIGHT

NASA STD. ATMOSPHERE
2 T64-GE-2 ENGINES
PAYLOAD = 4000 LB. OUTBOUND, 2000 LB. INBOUND

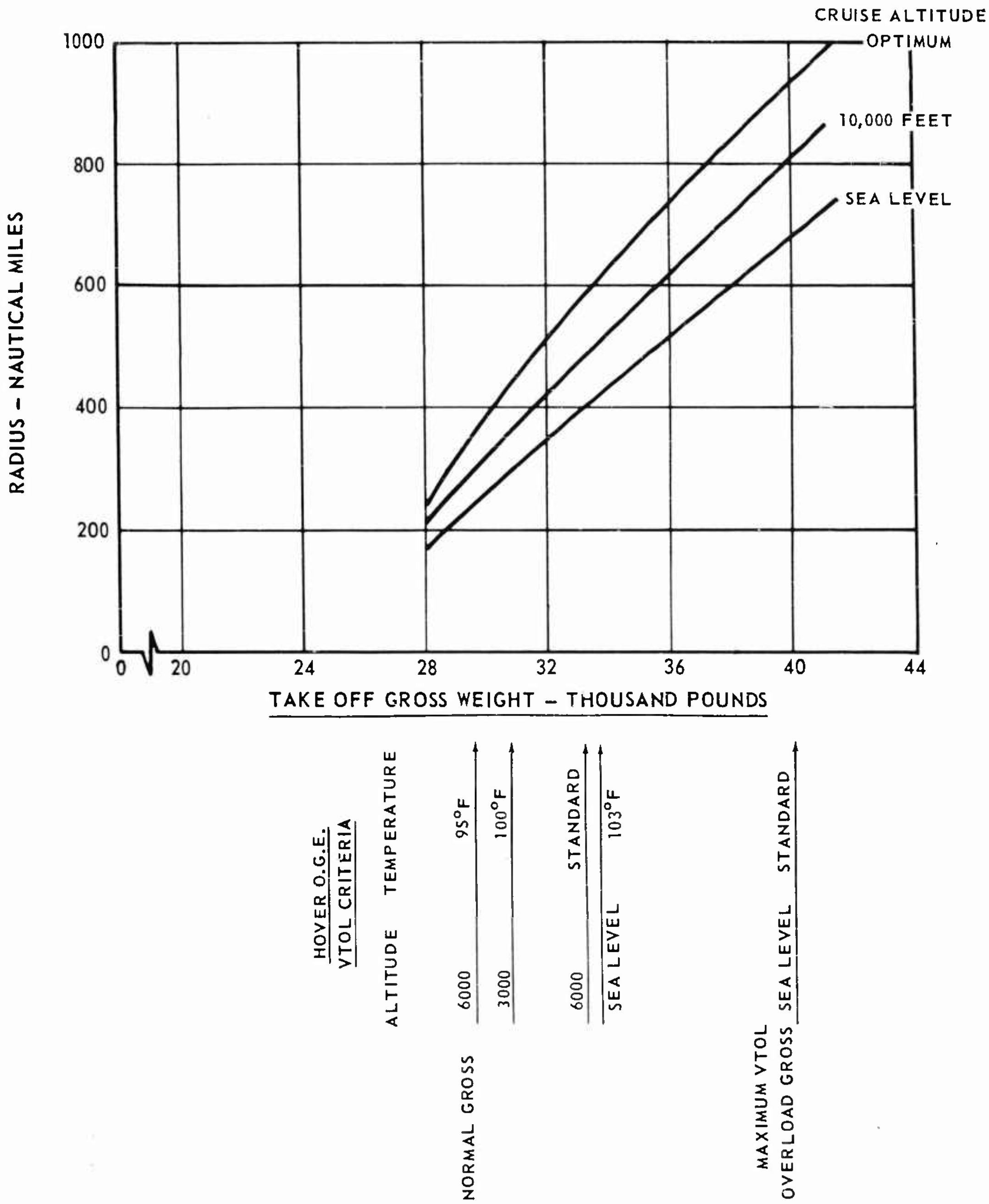


FIGURE 7.8

MODEL 113 VTOL TRANSPORT
RADIUS VS. TAKE-OFF GROSS WEIGHT

NASA STD. ATMOSPHERE
3 T55-L-7 ENGINES
PAYLOAD = 4000 LB. OUTBOUND, 2000 LB. INBOUND

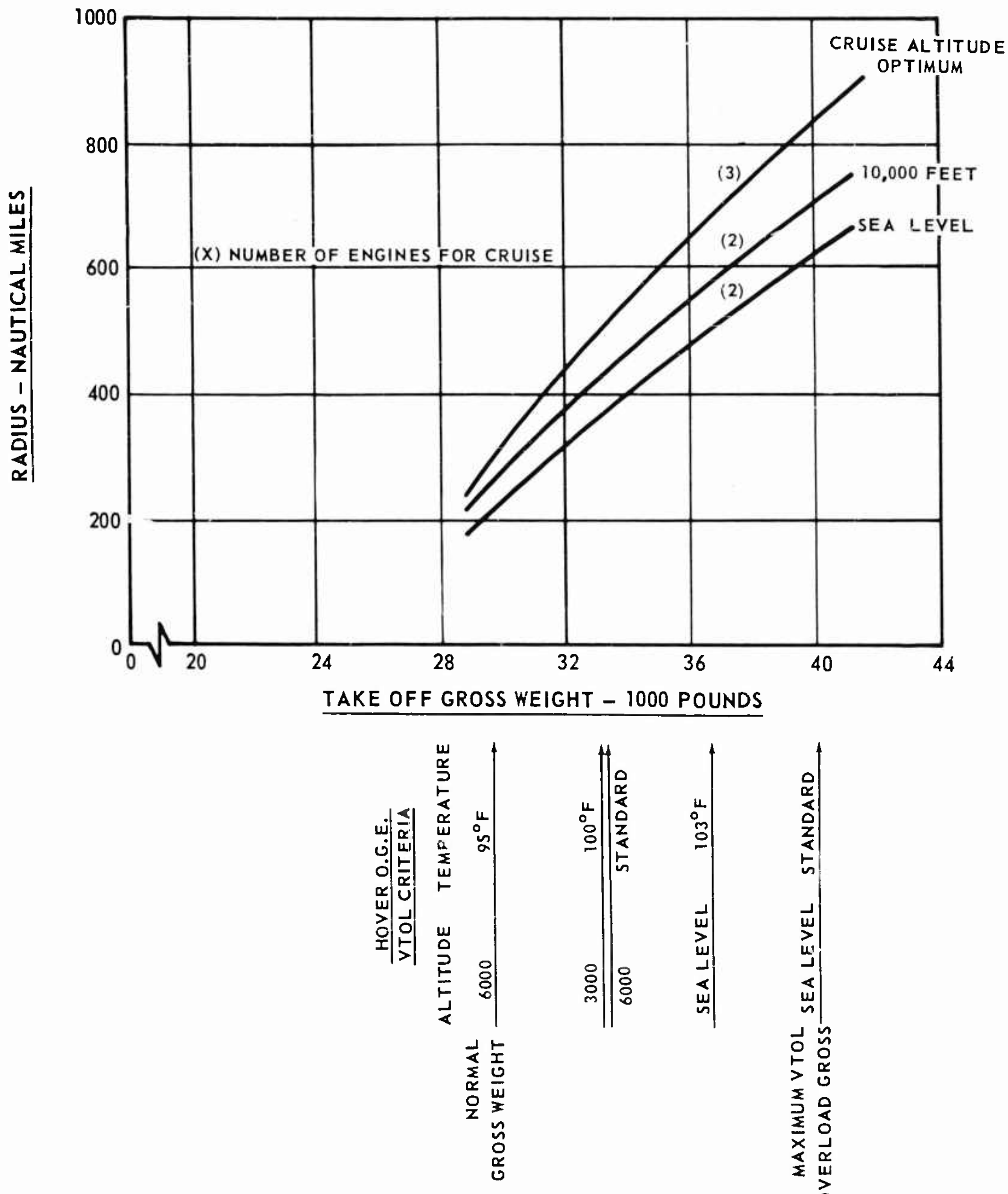


FIGURE 7.9

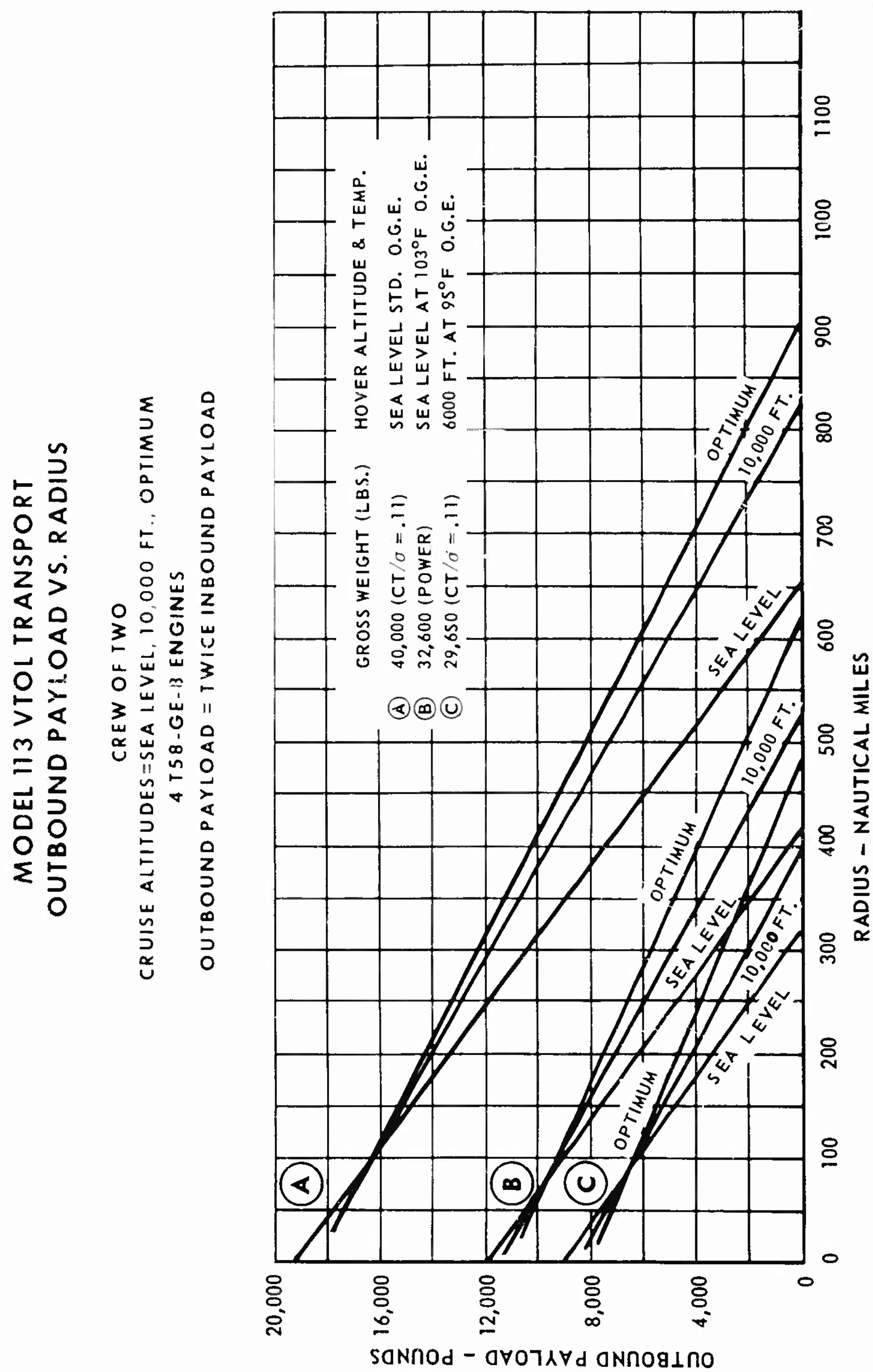

MODEL 113 VTOL TRANSPORT
OUTBOUND PAYLOAD VS. RADIUS
CREW OF TWO
CRUISE ALTITUDES=SEA LEVEL, 10,000 FT., OPTIMUM
4 T58-GE-B ENGINES
OUTBOUND PAYLOAD = TWICE INBOUND PAYLOAD
GROSS WEIGHT (LBS.)
HOVER ALTITUDE & TEMP.
Ⓐ 40,000 (CT/σ = .11)
SEA LEVEL STD. O.G.E.
Ⓑ 32,600 (POWER)
SEA LEVEL AT 103°F O.G.E.
Ⓒ 29,650 (CT/σ = .11)
6000 FT. AT 95°F O.G.E.
A
B
C
OPTIMUM
10,000 FT.
SEA LEVEL
OPTIMUM
10,000 FT.
SEA LEVEL
OPTIMUM
10,000 FT.
SEA LEVEL
OUTBOUND PAYLOAD – POUNDS
20,000
16,000
12,000
8,000
4,000
0
RADIUS – NAUTICAL MILES
0
100
200
300
400
500
600
700
800
900
1000
1100

FIGURE 7.10

MODEL 113 VTOL TRANSPORT
OUTBOUND PAYLOAD VS. RADIUS

CRUISE ALTITUDES=SEA LEVEL, 10,000 FT., OPTIMUM
CREW OF TWO
4 T58-GE-8 ENGINES
OUTBOUND PAYLOAD=TWICE INBOUND PAYLOAD

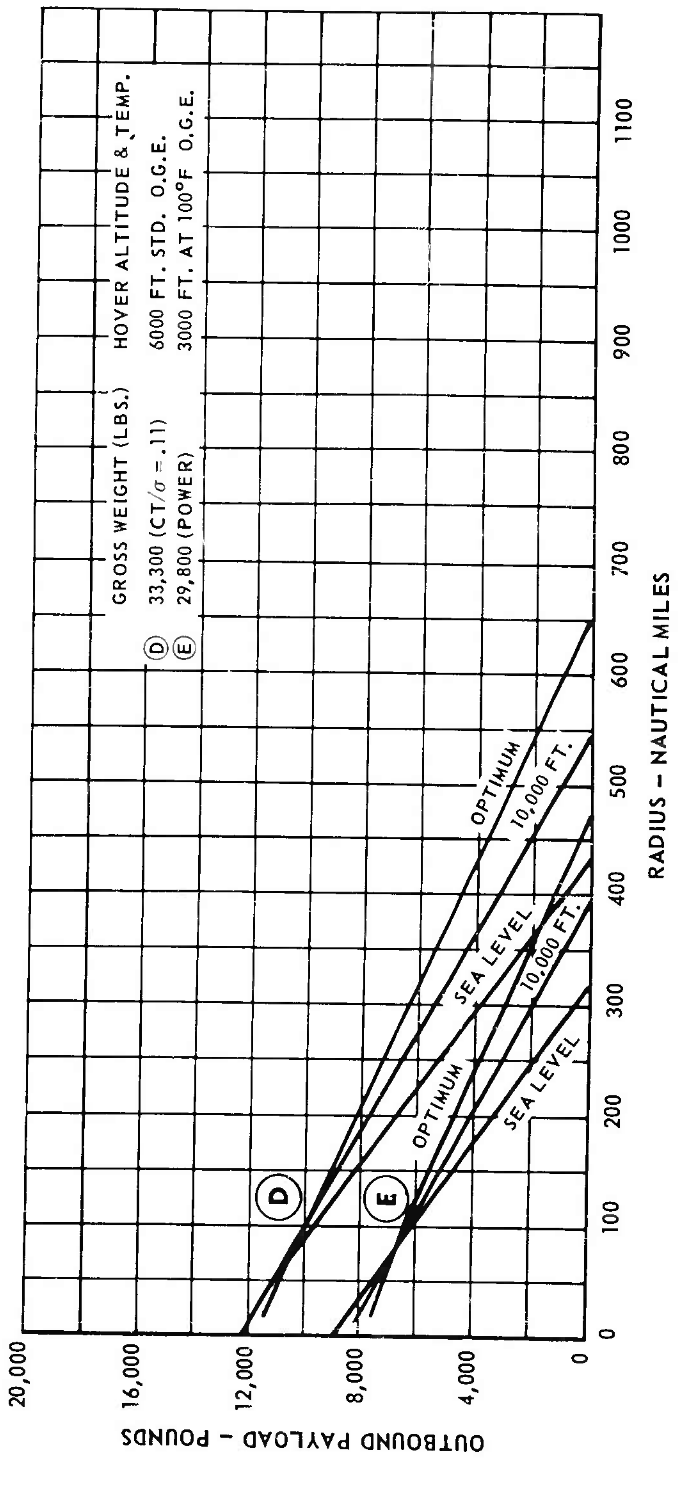


FIGURE 7.11

MODEL 113 VTOL TRANSPORT
OUTBOUND PAYLOAD VS. RADIUS

CRUISE ALTITUDES=SEA LEVEL, 10,000 FT., OPTIMUM
CREW OF TWO
2 T64-GE-2 ENGINES·
OUTBOUND PAYLOAD = TWICE INBOUND PAYLOAD

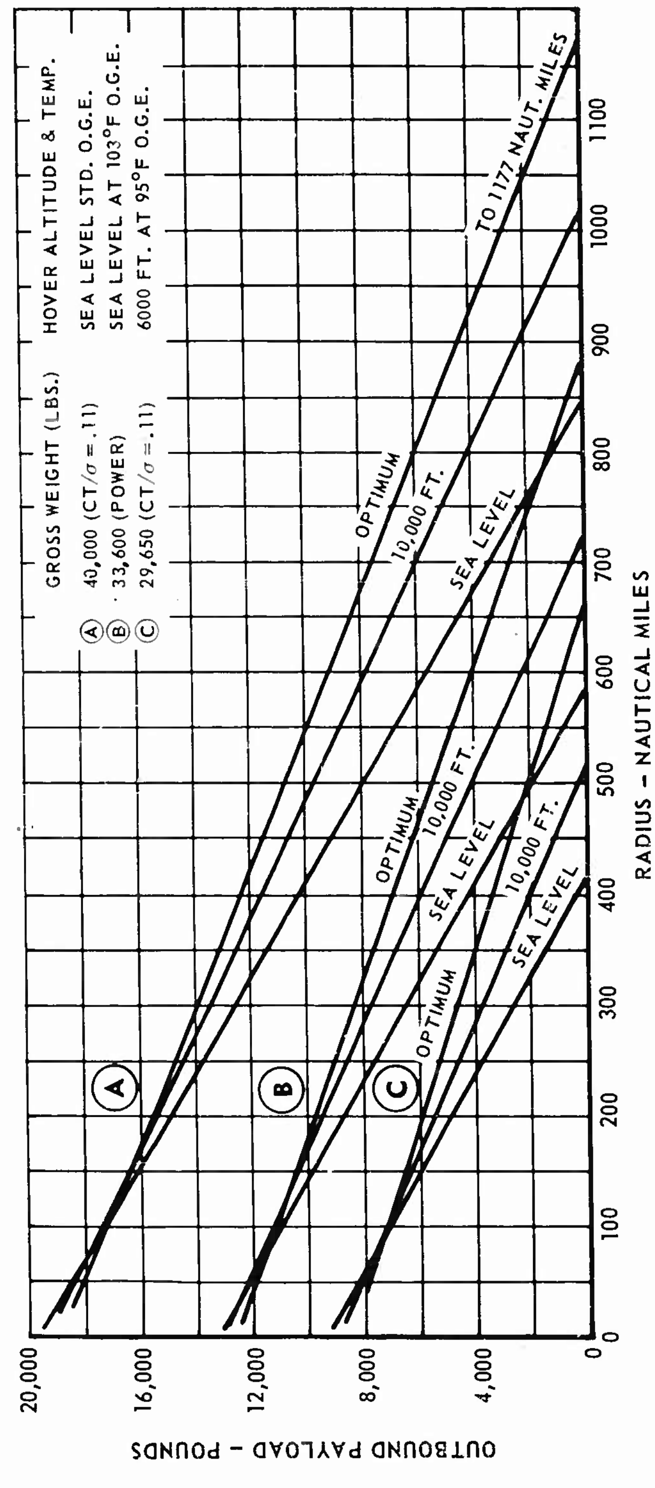


FIGURE 7.12

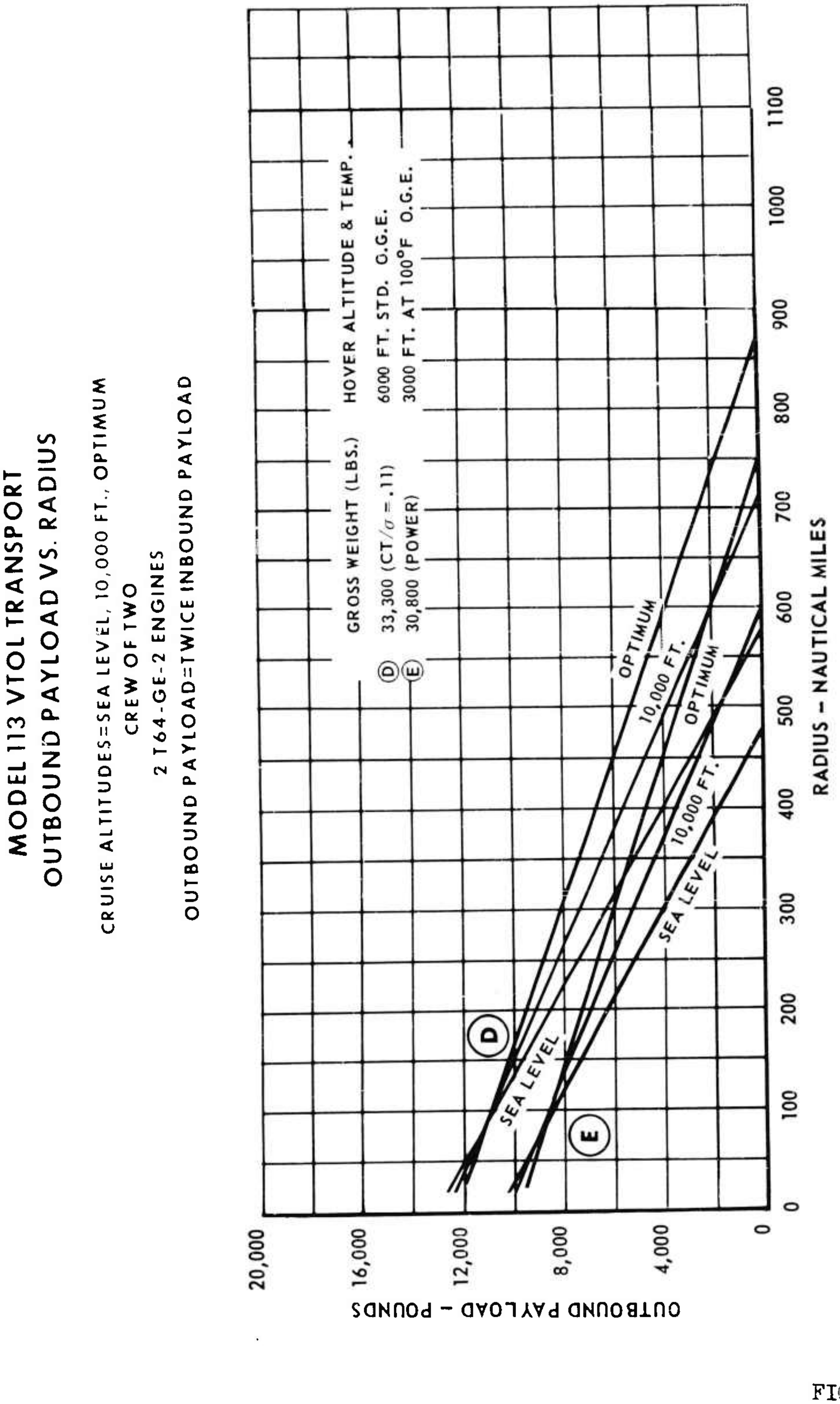

MODEL 113 VTOL TRANSPORT
OUTBOUND PAYLOAD VS. RADIUS
CRUISE ALTITUDES=SEA LEVEL, 10,000 FT., OPTIMUM
CREW OF TWO
2 T64-GE-2 ENGINES
OUTBOUND PAYLOAD=TWICE INBOUND PAYLOAD
GROSS WEIGHT (LBS.)
HOVER ALTITUDE & TEMP.
D 33,300 (CT/σ = .11) 6000 FT. STD. O.G.E.
E 30,800 (POWER) 3000 FT. AT 100°F O.G.E.
OPTIMUM
10,000 FT.
SEA LEVEL
D
E
OUTBOUND PAYLOAD – POUNDS
20,000
16,000
12,000
8,000
4,000
0
RADIUS – NAUTICAL MILES
0
100
200
300
400
500
600
700
800
900
1000
1100

FIGURE 7.13

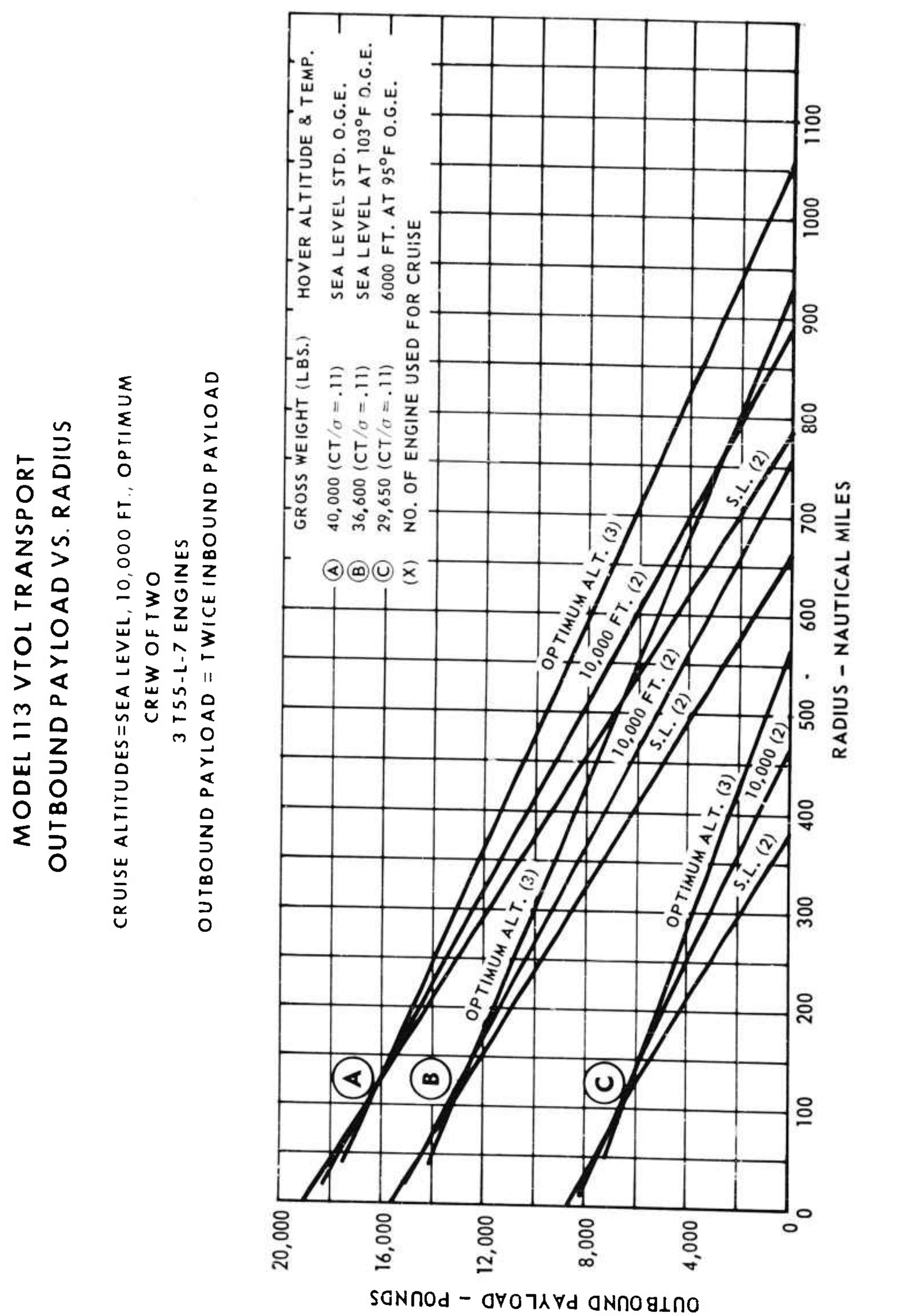

MODEL 113 VTOL TRANSPORT
OUTBOUND PAYLOAD VS. RADIUS
CRUISE ALTITUDES=SEA LEVEL, 10,000 FT., OPTIMUM
CREW OF TWO
3 T55-L-7 ENGINES
OUTBOUND PAYLOAD = TWICE INBOUND PAYLOAD
GROSS WEIGHT (LBS.)
HOVER ALTITUDE & TEMP.
(A) 40,000 (CT/σ = .11)
SEA LEVEL STD. O.G.E.
(B) 36,600 (CT/σ = .11)
SEA LEVEL AT 103°F O.G.E.
(C) 29,650 (CT/σ = .11)
6000 FT. AT 95°F O.G.E.
(X) NO. OF ENGINE USED FOR CRUISE
OPTIMUM ALT. (3)
10,000 FT. (2)
S.L. (2)
OPTIMUM ALT. (3)
10,000 FT. (2)
S.L. (2)
OPTIMUM ALT. (3)
10,000 (2)
S.L. (2)
A
B
C
OUTBOUND PAYLOAD – POUNDS
0
4,000
8,000
12,000
16,000
20,000
RADIUS – NAUTICAL MILES
0
100
200
300
400
500
600
700
800
900
1000
1100

FIGURE 7.14

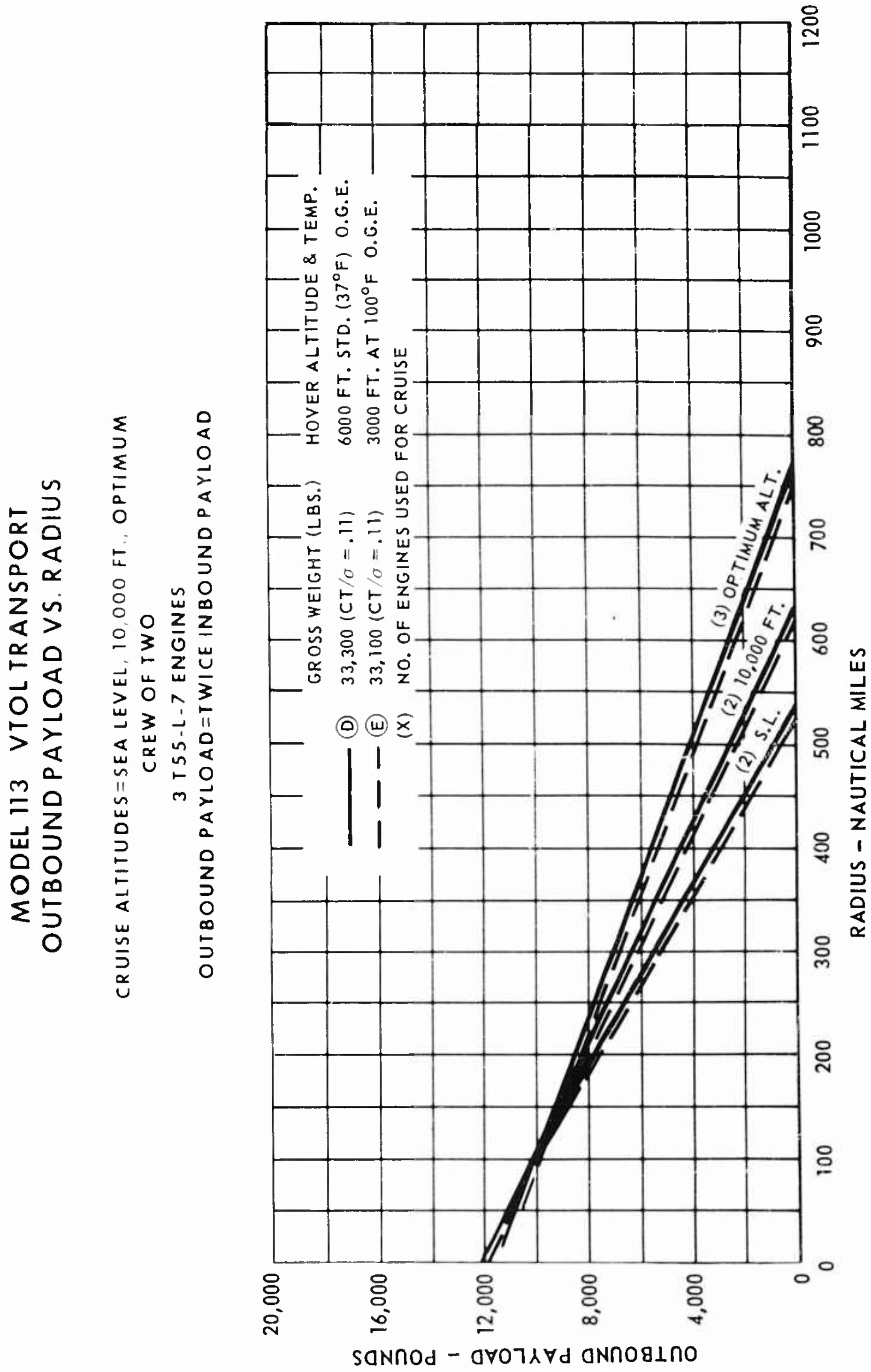

MODEL 113 VTOL TRANSPORT
OUTBOUND PAYLOAD VS. RADIUS
CRUISE ALTITUDES=SEA LEVEL, 10,000 FT., OPTIMUM
CREW OF TWO
3 T55-L-7 ENGINES
OUTBOUND PAYLOAD=TWICE INBOUND PAYLOAD
GROSS WEIGHT (LBS.)
HOVER ALTITUDE & TEMP.
(D) 33,300 (CT/σ = .11)
6000 FT. STD. (37°F) O.G.E.
(E) 33,100 (CT/σ = .11)
3000 FT. AT 100°F O.G.E.
(X) NO. OF ENGINES USED FOR CRUISE
(3) OPTIMUM ALT.
(2) 10,000 FT.
(2) S.L.
OUTBOUND PAYLOAD - POUNDS
20,000
16,000
12,000
8,000
4,000
0
RADIUS - NAUTICAL MILES
0
100
200
300
400
500
600
700
800
900
1000
1100
1200

FIGURE 7.15

MODEL 113 VTOL TRANSPORT
OUTBOUND PAYLOAD VS. RADIUS

CRUISE ALTITUDE = 10,000 FT.
CREW OF TWO
3 T55-L-7 ENGINES
OUTBOUND PAYLOAD = TWICE INBOUND PAYLOAD

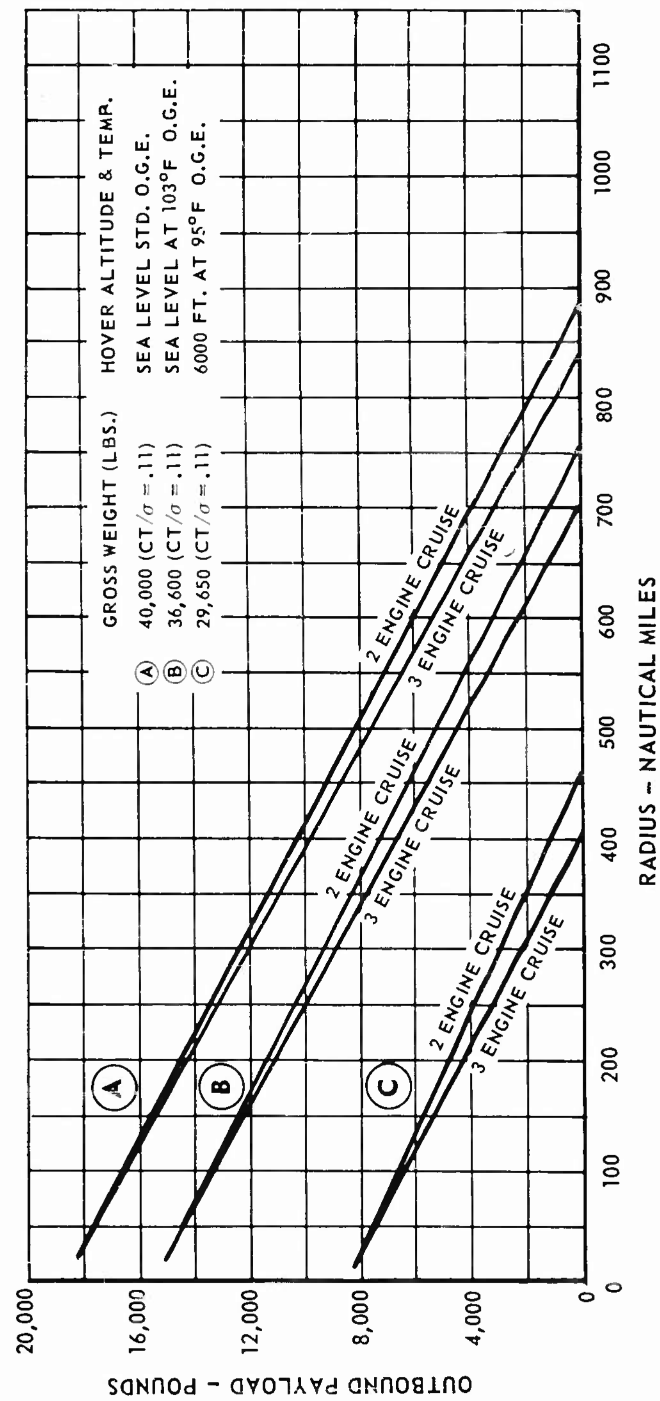


FIGURE 7.16

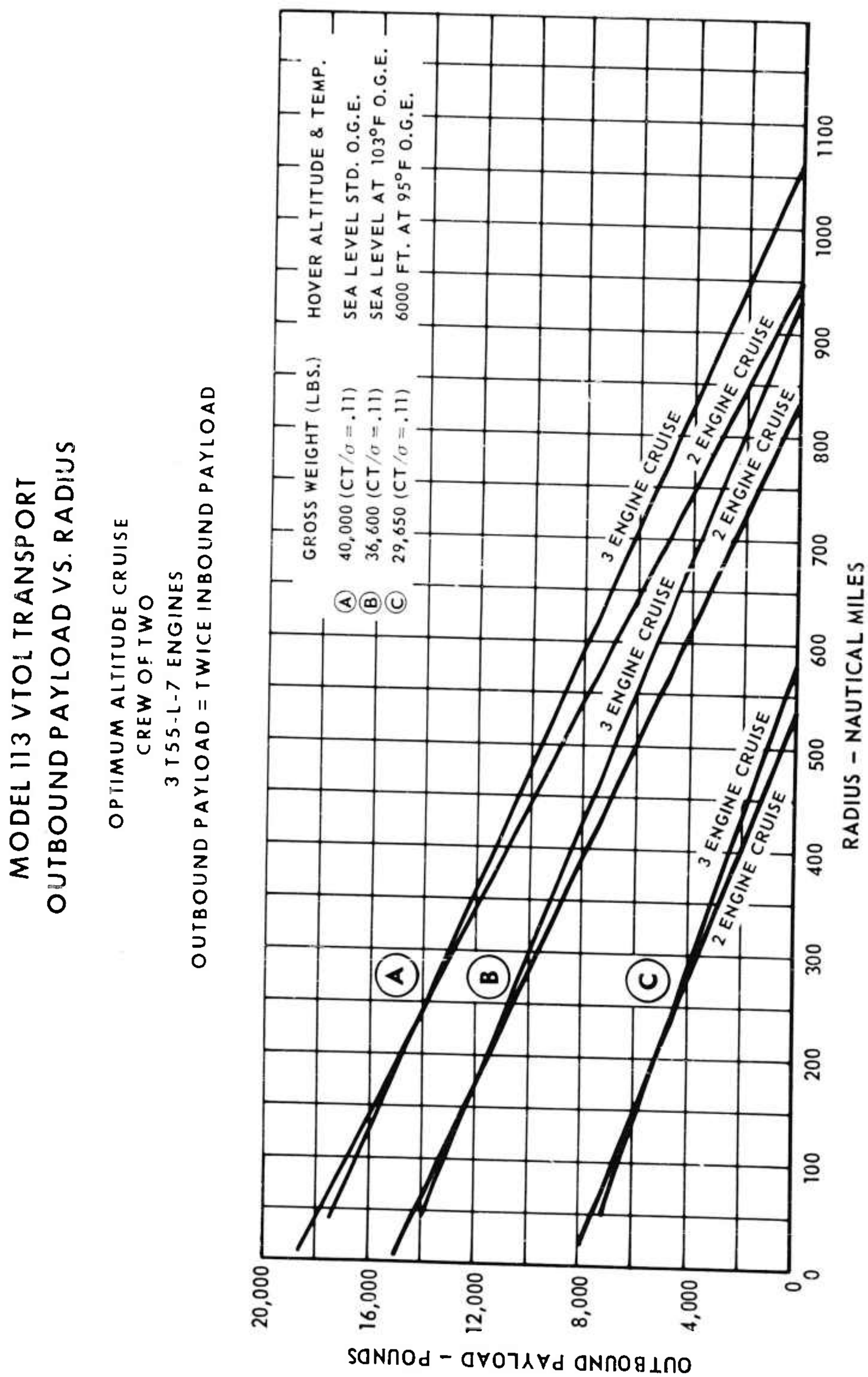

MODEL 113 VTOL TRANSPORT
OUTBOUND PAYLOAD VS. RADIUS
OPTIMUM ALTITUDE CRUISE
CREW OF TWO
3 T55-L-7 ENGINES
OUTBOUND PAYLOAD = TWICE INBOUND PAYLOAD
GROSS WEIGHT (LBS.)
HOVER ALTITUDE & TEMP.
A 40,000 (CT/σ = .11)
SEA LEVEL STD. O.G.E.
B 36,600 (CT/σ = .11)
SEA LEVEL AT 103°F O.G.E.
C 29,650 (CT/σ = .11)
6000 FT. AT 95°F O.G.E.
3 ENGINE CRUISE
2 ENGINE CRUISE
A
B
C
OUTBOUND PAYLOAD – POUNDS
RADIUS – NAUTICAL MILES
20,000
16,000
12,000
8,000
4,000
0
0
100
200
300
400
500
600
700
800
900
1000
1100

FIGURE 7.17

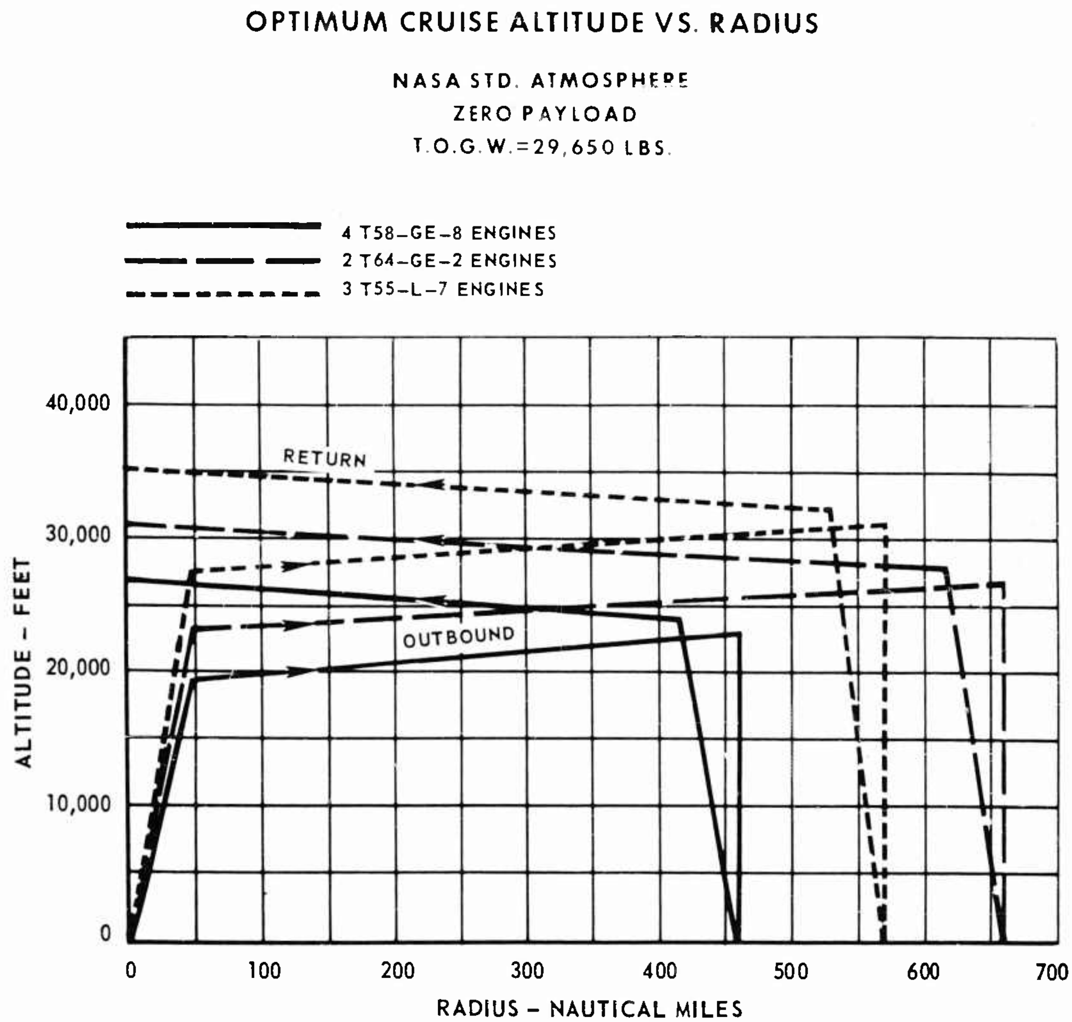
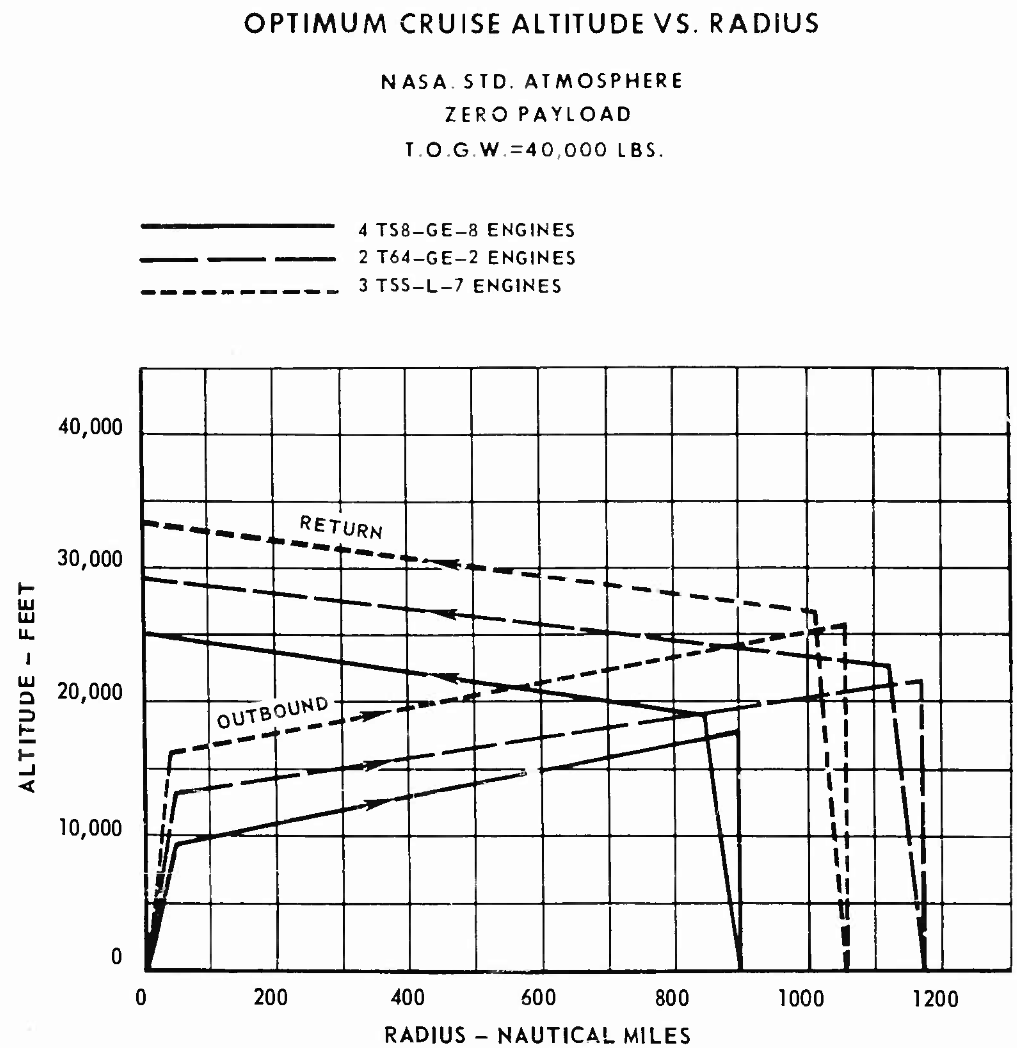

OPTIMUM CRUISE ALTITUDE VS. RADIUS
NASA STD. ATMOSPHERE
ZERO PAYLOAD
T.O.G.W.=29,650 LBS.
4 T58-GE-8 ENGINES
2 T64-GE-2 ENGINES
3 T55-L-7 ENGINES
RETURN
OUTBOUND
ALTITUDE - FEET
40,000
30,000
20,000
10,000
0
0
100
200
300
400
500
600
700
RADIUS - NAUTICAL MILES

FIGURE 7.18



OPTIMUM CRUISE ALTITUDE VS. RADIUS
NASA. STD. ATMOSPHERE
ZERO PAYLOAD
T.O.G.W.=40,000 LBS.
4 TS8-GE-8 ENGINES
2 T64-GE-2 ENGINES
3 TSS-L-7 ENGINES
RETURN
OUTBOUND
ALTITUDE - FEET
40,000
30,000
20,000
10,000
0
0
200
400
600
800
1000
1200
RADIUS - NAUTICAL MILES

FIGURE 7.19

LONGITUDINAL AND LATERAL CONTROL RESPONSE IN HOVERING

MODEL 120 FLIGHT TEST, HOVER O.G.E.
GROSS WEIGHT=3825 LBS.
REVISED ROTOR HUB GEOMETRY
PITCH CONE RATIO=1.65, ADVANCE ANGLE=-14°56'

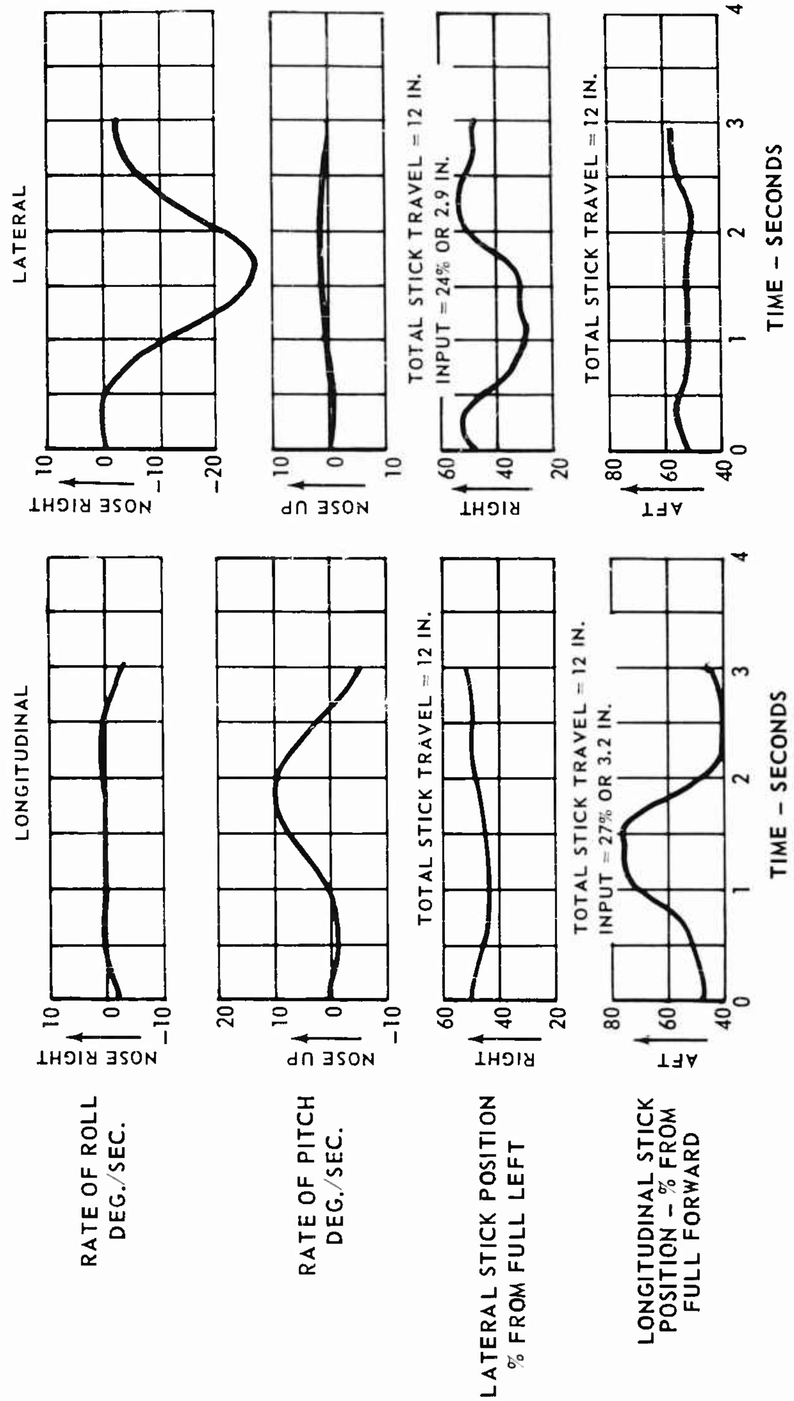


FIGURE 7.20

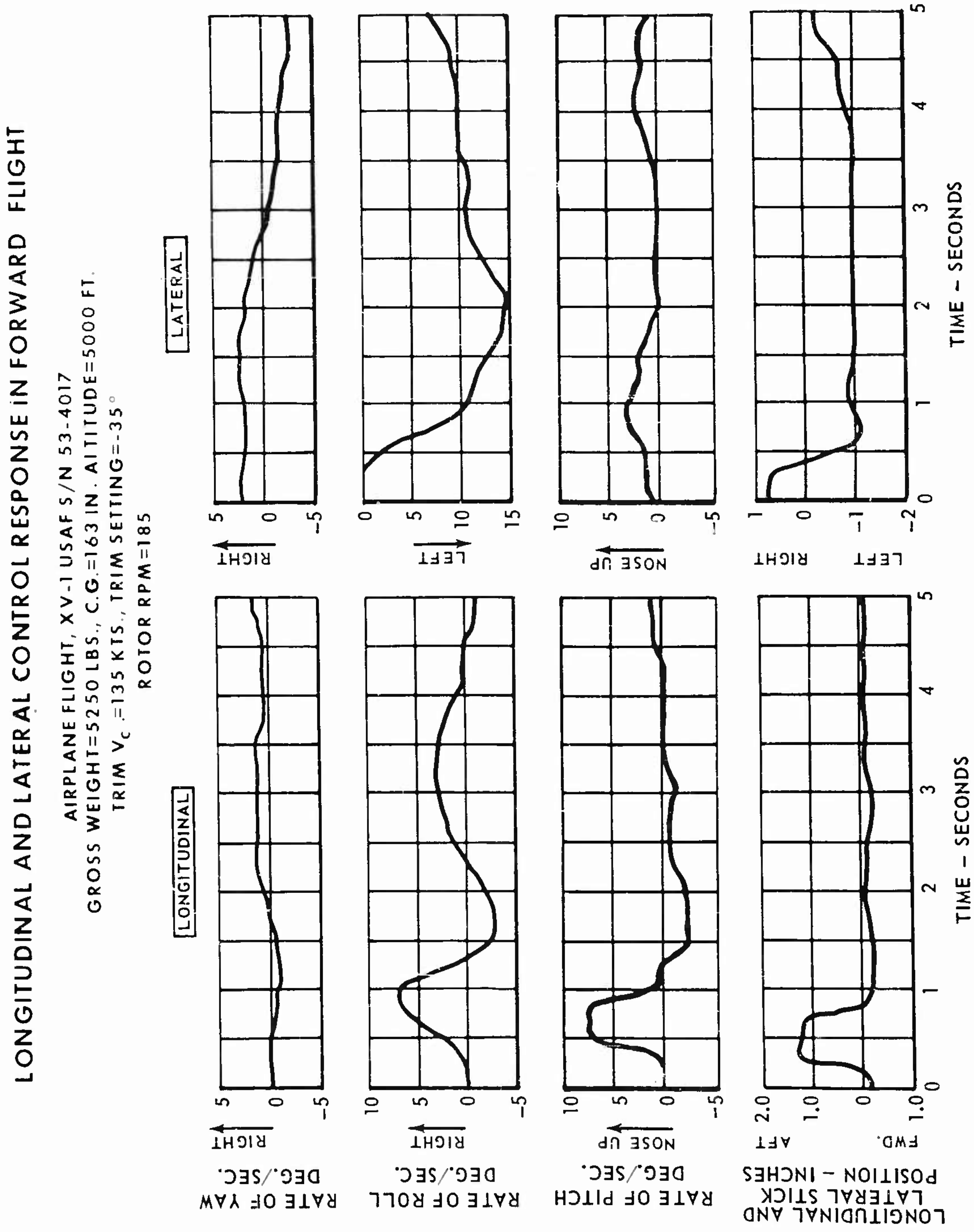

LONGITUDINAL AND LATERAL CONTROL RESPONSE IN FORWARD FLIGHT
AIRPLANE FLIGHT, XV-1 USAF S/N 53-4017
GROSS WEIGHT=5250 LBS., C.G.=163 IN. ALTITUDE=5000 FT.
TRIM V_c=135 KTS., TRIM SETTING=-35°
ROTOR RPM=185
LONGITUDINAL
LATERAL
RATE OF YAW DEG./SEC.
RIGHT
RATE OF ROLL DEG./SEC.
RIGHT
LEFT
RATE OF PITCH DEG./SEC.
NOSE UP
LONGITUDINAL AND LATERAL STICK POSITION – INCHES
AFT
FWD.
RIGHT
LEFT
TIME – SECONDS

FIGURE 7.21

FLIGHT V-n ENVELOPE AT DESIGN GROSS WEIGHT

SEA LEVEL STANDARD CONDITIONS

ROTOR GOVERNED IN AIRPLANE FLIGHT AT $\Omega R = 350$ F.P.S.

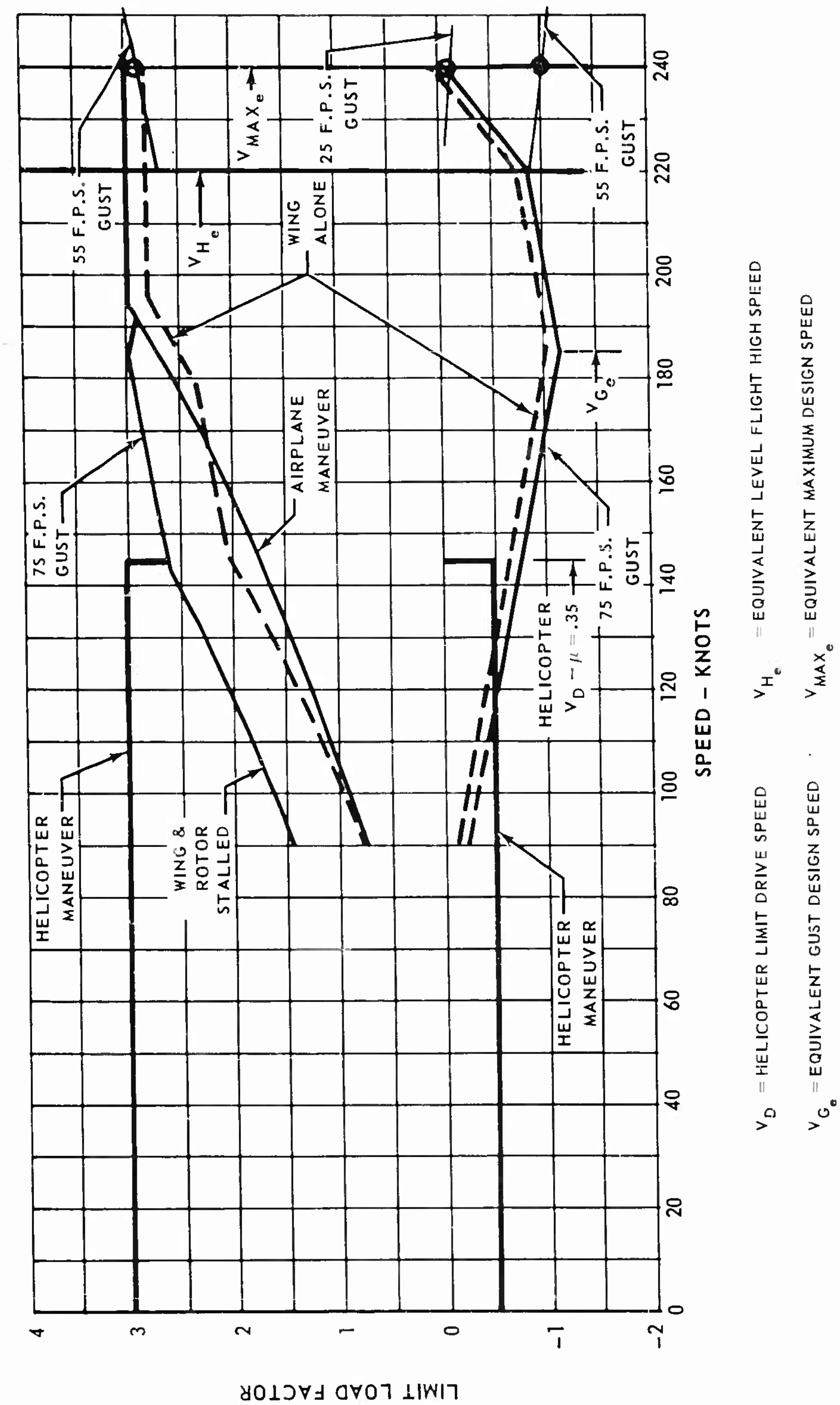


FIGURE 7.22

ESTIMATED ROTOR BLADE FREQUENCIES

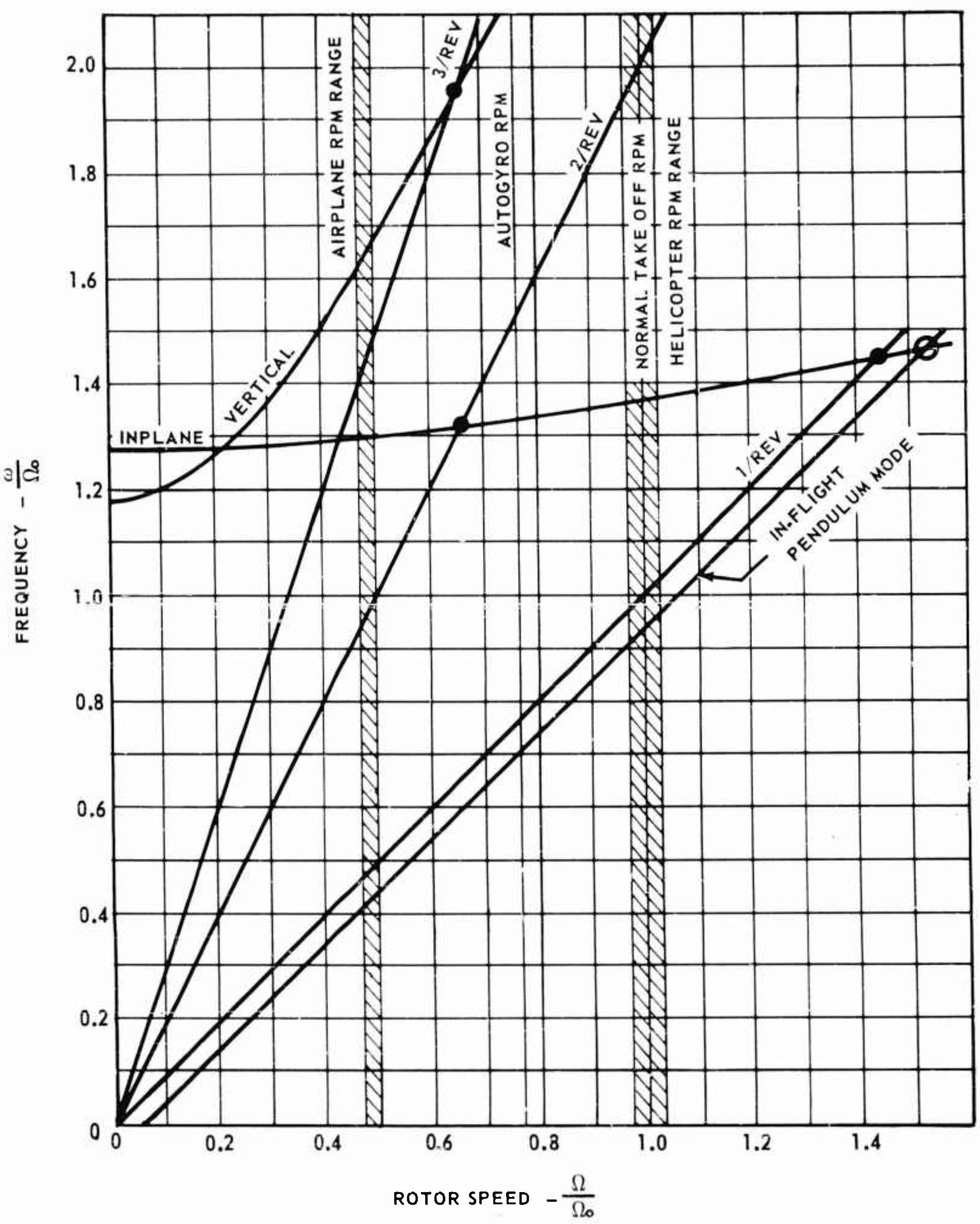


FIGURE 7.23

MAC ROTOR SYSTEM

GENERAL FLUTTER CHARACTERISTICS

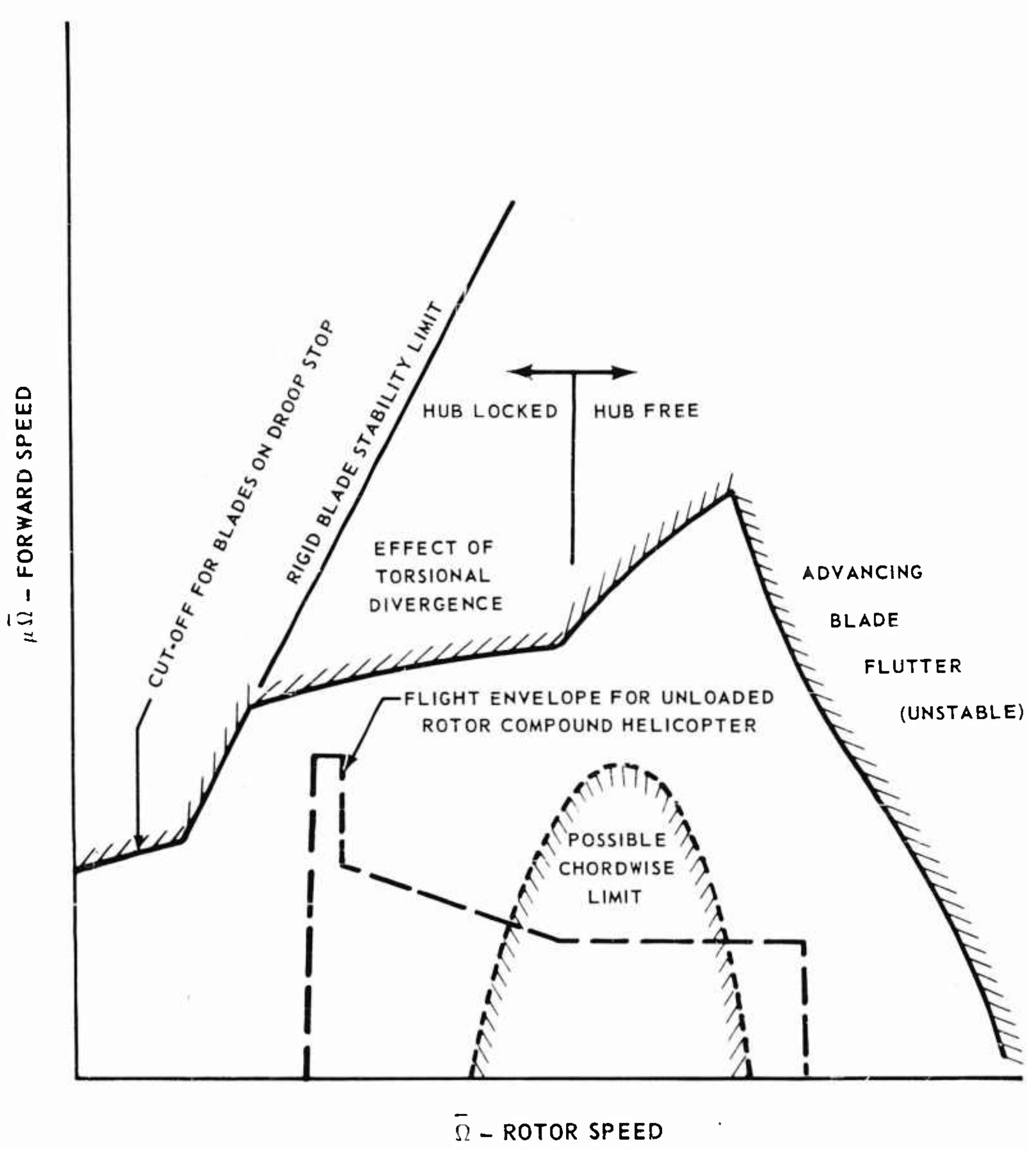


FIGURE 7.24

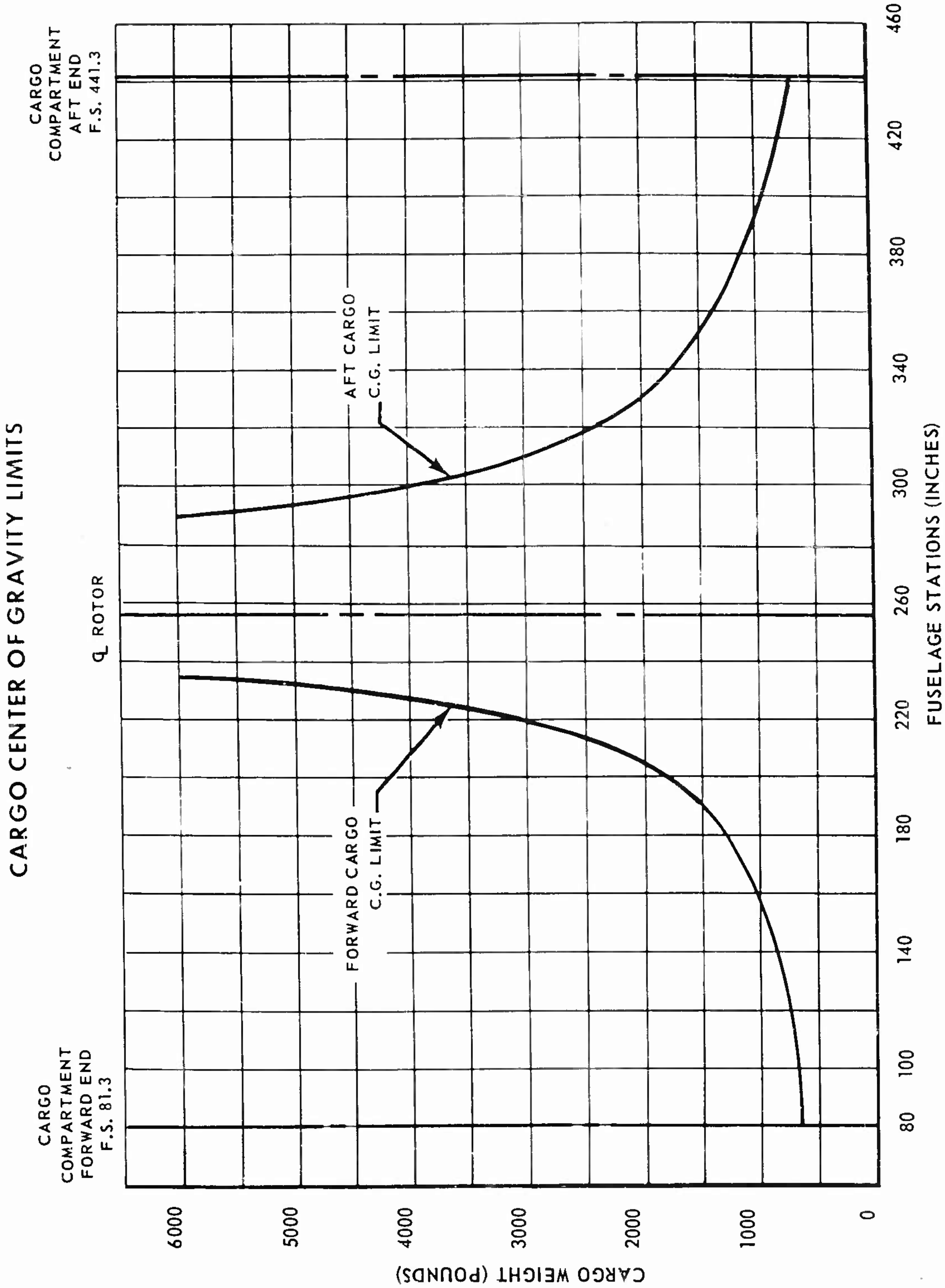

CARGO CENTER OF GRAVITY LIMITS
CARGO COMPARTMENT FORWARD END F.S. 81.3
CARGO COMPARTMENT AFT END F.S. 441.3
℄ ROTOR
FORWARD CARGO C.G. LIMIT
AFT CARGO C.G. LIMIT
80
100
140
180
220
260
300
340
380
420
460
FUSELAGE STATIONS (INCHES)
6000
5000
4000
3000
2000
1000
0
CARGO WEIGHT (POUNDS)

FIGURE 7.25

SECTION VIII

8. OPERATIONAL VARIABLES

8.1 VTOL and STOL Overload Limitations - The overload capability of an unloaded rotor compound helicopter is dictated by the establishment of design load factor, permissible rotor aerodynamic blade loading or mean lift coefficient levels, take-off altitude-temperature combination, and design power loading, or a combination of these factors. Each of these defining factors is considered individually in the following sections.

8.1.1 Overload Limitation by Minimum Load Factor - The design load factor for the normal gross weight established by the selected design altitude and temperature combination is specified as 3.0. By Reference 12.5, the minimum load factor defining the overload gross weight is 2.0; therefore,

$$\text{Maximum Permissible } \frac{\text{Overload Gross Weight}}{\text{Normal Gross Weight}} \text{ Ratio} = \frac{3.0}{2.0} \text{ or } 1.5$$

8.1.2 Overload Limitation by Maximum Rotor Aerodynamic Blade Loading - As discussed in paragraph 5.2.3 a limit in aerodynamic blade loading of C_T/σ = .11 has been imposed on VTOL operation of Model 113. For STOL operation the permissible C_T/σ may exceed the VTOL limit because of rotor thrust relief, landing gear load, removal of the hovering downwash load, and increased fixed wing lift with increased flight velocity. Thus the aerodynamic overload capability is extended through STOL operation; the extension is defined by take-off distance, forward flight blade stall and conversion characteristics, engine-out performance, or limit load factor. Further discussion of this operation is included in the subsequent paragraphs.

8.1.3 Effect of Take-off Altitude-Temperature Combination on VTOL Overload - The combination of maximum altitude and temperature specified for VTOL take-off defines a density which together with the VTOL blade loading limit, tip speeds, and given rotor geometry defines the design take-off weight or normal gross weight. To maintain a constant C_T/σ limit, the weight-density ratio must remain constant. Thus, as density increases with decrease in altitude or temperature, the weight increases proportionately, resulting in aerodynamic overload capability.

$$\frac{\text{Aerodynamic Overload Capability}}{\text{Normal Gross Weight}} \text{ Ratio} = \frac{1}{\rho/\rho_1}$$

where ρ = Density at Specified Altitude and Temperature
ρ_1 = Density at Alternate Altitude and Temperature

Thus, for the specified Army design condition (6000 feet 95°F), the sea level standard temperature, hovering take-off weight may be 1/.75 or 1.33 times the normal gross weight. The 6000-foot standard temperature VTOL weight may be 1/.90 = 1.11 times normal gross weight.

8.1.4 Power Limitation on VTOL Overload - The above take-off gross weights are established independent of power requirements. Whether or not VTOL capability exists under these overload conditions is a function of power plant temperature-altitude characteristics relative to the density ratio. In the case of the Model 113P with four T58-GE-8 engines, the power limitation for vertical take-off at standard sea level condition coincides with the aerodynamic loading limitation, so that the maximum VTOL overload weight is 1.33 times the design gross weight. For

Model 113 with three T55-L-7 engines there is a 9 percent power reserve, and for Model 113 with two T64-GE-2 engines there is a 7 percent power reserve available at the maximum aerodynamically limited vertical overload take-off weight of 1.33 times design gross weight.

The better matching of VTOL overload power and aerodynamic limitations at standard sea level conditions in case of the T58-GE-8 engines is a consequence of the 11 percent power augmentation at 6000 feet 95°F. Had the aircraft been designed without making use of power augmentation at the extreme hovering requirement an unusable VTOL power margin at sea level standard conditions would have been available of the same order of magnitude as for the other two power plant systems.

8.1.5 STOL Overload Limitations - Theoretical analyses of STOL overload operations for Model 113 show that the take-off weight for a running take-off to clear a 50-foot obstacle within 500 feet is 5 to 6 percent greater than the maximum vertical take-off weight. These analyses agree with flight test data on overload capabilities of pure helicopters. Further analyses indicated that a hovering take-off in ground effect in lieu of a ground roll if permitted by VTOL aerodynamic blade loading would result in similar take-off distances.

Figure 8.1 presents the VTOL and STOL (500 feet to clear 50 feet) gross weights as a function of take-off atmospheric density ratio for the three Model 113 power plant installations and two temperature conditions, standard and 95°F. The STOL gross weights are assumed 5.5 percent greater than the VTOL weights defined either by power or C_T/σ limitations. This assumption neglects differences in power margins between engines that would require changes in take-off gross weight increments to maintain equal climb out distances. A more detailed analysis of the STOL capabilities of Model 113 with T55-L-7 or T64-GE-2 engines may result in higher STOL weights for these two alternate power plants with their increased power margin. The maximum STOL overload is given by 45,000 pounds which is the limit with respect to overload load factor.

Figure 8.2 shows the Model 113P (T58-GE-8 power plant installation) VTOL and STOL outbound payload for 250-nautical mile radius of operation against take-off altitude for standard and 95°F conditions. The 95°F data are given for the without power augmentation case; the 11 percent power augmentation case is represented by a single VTOL point at 6000 feet. STOL operation increases the 95°F day payload capability by 1500 to 1600 pounds, the standard day capability by 1600 to 2000 pounds. For sea level conditions, the 95°F day STOL payload is 20 percent greater, and the standard day payload is 15 percent greater than the VTOL payload.

Since the fixed wing of the compound helicopter develops lift with forward velocity, greater take-off weights are permissible if greater take-off distances than 500 feet are specified. STOL operation of the compound helicopter may also be used to advantage for emergency take-off with one engine out or at altitude-temperature combinations which restrict VTOL take-off either through aerodynamic blade loading limits or through power available.

8.2 Mission Variation

8.2.1 Take-off Ambient Condition - The influence of take-off ambient condition on the VTOL-STOL capabilities of the unloaded rotor compound helicopter has been discussed in Sections 6 and 7 and Paragraph 8.1.3; the pertinent payload-radius charts are presented in Sections 6 and 7. Figure 8.3 presents the Model 113 transport aircraft payload for a 250-nautical mile radius of operation as a function of atmospheric density ratio. Each take-off ambient condition considered is identified as to atmospheric density ratio. Data for three power plant installations are shown.

The payload increase associated with decreased altitude and/or ambient temperature is shown by Figure 8.3. A relaxation of the take-off ambient condition from the design atmosphere to the sea level standard atmosphere increases the 250-nautical mile radius payload 3.5 times for the T58-GE-8 installation, 3.1 times for the T55-L-7 installation, and 2.8 times for the T64-GE-2 installation. The probability of occurrence of the 6000-foot 95°F take-off condition is roughly 2 percent of the total operating time of all transport aircraft (3 percent for the 3000-foot, 100°F condition). Therefore, a major portion of the service life of these transport aircraft will be devoted to overload payload operation; thus the productivity, payload ton miles per empty weight ton, during normal service usage is greater than the level associated with the design payload and cruise velocity.

8.2.2 Mission Profile - The effects of cruise altitude and number of engines for cruise on the payload-radius capabilities of the recommended light VTOL transport aircraft are presented and discussed in Section 7. Optimum altitude for cruise is not limited to 20,000 feet since interpolation between the 10,000 feet and optimum altitude curves is possible. Further analysis of mission profile variation appears unwarranted.

8.2.3 Hover Time - Figure 8.4 presents the effect of sea level hover time on the two-ton payload radius of operation for various take-off weights. Data are shown for the T58-GE-8 power plant installation and a constant cruise altitude of 10,000 feet. Whether the hover time is distributed over the total mission or concentrated at the mid-point makes little difference. Figure 8.5 presents the loss in payload at zero radius of operation for assumed sea level hover times. This chart plus Figure 8.4 permits the reconstruction of the T58-GE-8 payload-radius charts of Section 7 for an assumed mission hover period.

The data presented illustrate the importance of limiting the hovering or rotor powered flight time in a transport mission, especially for the extreme altitude-temperature take-off conditions. Although the penalty of hovering time is relatively large for the pressure jet unloaded rotor compound helicopter with respect to a gear driven helicopter, the VTOL concepts involving high disc loading will suffer similar radius penalty for constant payload as a result of this reduced design payload ratio and design power loading (see Section 3).

8.2.4 Cruise Velocity - The aircraft cruise velocities associated with maximum aerodynamic efficiency (L/D) and with optimum nautical mile per pound of fuel for constant cruise altitudes of sea level and 10,000 feet can be determined from Figures 8.6 and 8.7, respectively. The most efficient cruise conditions at altitudes between sea level and 10,000 feet correspond to approximately 90 percent throttle setting and 200 to 210 knots cruise velocity. However, the nautical mile per pound of fuel values are relatively insensitive to cruise velocity selection between 170 and 210 knots at sea level, 190 and 220 knots at 10,000 feet. Operation at velocities for maximum L/D at the less than optimum altitudes reduces the radius capability because of the low throttle settings required (low throttle setting - high SFC values). Maximum L/D cruise velocities and optimum altitudes for the Model 113 T58-GE-8 installation are presented in Section 6, Figure 6.2.

8.2.5 Ferry Capability - The ferry capability at optimum cruise altitude of the Model 113 unloaded rotor compound helicopter with three different power plant installations is shown by Figure 8.8. For VTOL operation at 6000 feet 95°F, the compound helicopter ferry range is approximately equal to the maximum ferry range of the present day, pure helicopter at maximum overload take-off condition (sea level standard). For VTOL operation at maximum overload, sea level standard atmosphere, the compound helicopter range is about twice that of the pure helicopter. These range capabilities provide transoceanic ferry possibilities that obviate the need for air transportability, thereby increasing Army mobility.

Ferry capability is further enhanced by the possibility of air-to-air refueling employing the probe and drogue technique and using existing fixed wing tanker airplanes.

8.2.6 Crane Capability - The compound helicopter may be used as a crane for carrying bulky cargoes externally. The method involves suspending the cargo from a single cable which is attached just below the aircraft center of gravity in the manner successfully demonstrated by the McDonnell Model 120 helicopter. External cargo may be carried in the helicopter flight regime at any airspeed up to that limited by power with no deleterious effects on stability or control; autogyro and airplane flight are not practical for the carrying of external payloads because of power limitations at the higher airspeeds of these regimes.

Figure 8.9 shows typical crane radius missions in which the aircraft carries external cargo on the outbound trip in the helicopter flight regime and flies back to the base without cargo in the airplane flight regime.

8.3 Conversion Characteristics - To avoid confusion of terms, transition and conversion are defined as follows:

a. Transition - The low speed, rotor powered flight regime between zero velocity (hover) and 30- to 40-knot flight velocity.

b. Conversion - The lift and/or propulsion transfer process between rotor powered helicopter flight and propeller powered airplane flight.

Conversion is accomplished by passing from a low speed, stable flight regime to a high speed, stable flight regime; the transitory region is also stable with the aircraft under complete control at all times. At any point in the conversion process, the pilot may proceed at will, reverse conversion, or dwell in intermediate conditions as circumstances dictate. In most cases the velocity overlap between helicopter and autogyro flight (a transitory phase) and the possibility of powering both the compressors and propellers permit the conversion to be completed without loss of altitude. Some altitude loss may occur at high altitude or high overload gross weight, but rotor powered service ceilings are sufficiently high to permit altitude loss in these cases.

Previous studies (References 12.3, 12.4, and 12.28) have shown that the conversion process is benefited to a greater extent through the use of increased autogyro blade angles than by increased wing area and/or flaps. XV-1 flight experience showed no unusual or dangerous flight conditions associated with the conversion procedures; in fact, flight test proved that a constant longitudinal trim setting could be used without objectionable control stick displacement. The flight experience gained during the XV-1 program is to be used to simplify the mechanical procedures for conversion, thereby relieving pilot input requirements.

8.4 Power Plant Failure

8.4.1 One Engine Out Performance - Estimated engine out performance for the two-, three- and four-engine transport aircraft of the XV-1 type is presented and discussed in this section.

Figure 8.10 shows the aircraft single engine out service ceiling as a function of gross weight for the three Model 113 power plant installations. Both the helicopter and airplane flight ceilings are given for the T58-GE-8 installation. Two charts are presented for the T64-GE-2 installation; one without cross shafting, the other with cross shafting. The TRECOM requirement of sea level, standard day, maximum power service ceiling is fulfilled by all aircraft.

The use of cross shafts in the T64-GE-2 installation improves the engine out performance by eliminating the drag of a fixed pitch windmilling propeller, reducing the drag from yaw correction and increasing propeller efficiency. However, the gain in propeller efficiency (a result of reduced power loading and rotational speed) is about compensated by the loss in free turbine power available resulting from the reduced turbine speed.

Figure 8.11 presents the airplane flight, one engine out, maximum rate of climb as a function of gross weight. The four- and three-engine Model 113 configurations show better climb potential for emergency operation than the two-engine version.

Figure 8.12 presents the hovering ceiling, in and out of ground effect, for the one engine out emergency. The order of merit is the three-engine single compressor arrangement, the four-engine four-compressor arrangement, and the two-engine single or dual compressor-cross shafted arrangement. The three- and four-engine aircraft with one engine out exhibit minimum hover ceilings in ground effect of at least sea level even at 95°F ambient temperature.

These one engine out performance data illustrate the high level of safety and emergency mission completion potential, especially for the three- and four-engine aircraft, that is characteristic of the unloaded rotor helicopter. In a great majority of emergency cases, these aircraft are capable of completing their assigned mission even when operating at maximum overload conditions. For the short radius-high take-off gross weight mission and engine failure immediately after take-off, a partial power descent to the take-off area is most logical even though cruise to the destination, partial power descent, and then return without payload is possible. The data presented should enable an evaluation of any emergency situation that may arise. The case of complete power loss is discussed in the following paragraph.

8.4.2 Total Power Loss - For unloaded rotor compound helicopter transport aircraft, a total power loss in any flight regime is a relatively safe emergency compared to other high speed VTOL aircraft types for the following reasons:

a. Autorotation with relatively low rate of descent is possible.

b. No abrupt loss of lift or unbalanced moment results from loss of power in any flight regime.

c. Pilot reaction time and procedure are less demanding.

d. Aerodynamic control and stability about all axes is inherent.

e. Ground cushion is available for deceleration.

f. High rotor inertia, thus rotation energy is inherent.

g. Low landing speeds are potentially possible.

Because of the possibility of autorotation at high pitch settings (pitch-cone rotor characteristic) and the elimination of any lifting surface incidence adjustment, the need for rapid pilot action and a large loss of altitude in establishing steady state emergency descent is removed. In the absence of power plant energy, the compound helicopter design offers the greatest ratio of expendable kinetic energy to the kinetic energy of descent of any of the VTOL aircraft with the possible exception of the pure helicopter. The energy available for arresting the descent is made up of flight velocity kinetic energy and a high level of rotor rotational energy created by the relatively high inertia pressure jet rotor system. The autorotation potential, the decelerating ground cushion, and the possibility of interchange of velocity energy for rotor energy permits a much greater reduction in landing speed than for those VTOL types wherein emergency lift is produced by dynamic pressure and fixed wing area.

Theoretical studies of in-flight total power loss emergencies of the compound helicopter (Reference 12.28) show that conversion from an initial high speed cruise condition at 200 feet altitude can be a straightforward procedure that results in a landing at 30 knots forward flight velocity. Thus, an aircraft of this type has exceptional potential for the "nap of the earth" operation envisioned by the Army.

8.4.3 Power Plant Reliability

8.4.3.1 General Concept - Reliability is a complex consideration which often involves compromises that can be evaluated more readily in terms of various classifications. The classifications of reliability normally considered are: Safety Reliability, Mission Reliability, Maintenance Reliability, and Overhaul Reliability. Past experience in producing aircraft and weapons systems is utilized in defining the problem area to obtain a high degree of reliability in the initial design. Each new design is given an appropriate analysis and evaluation by engineering reliability specialists to determine where emphasis is needed to provide a maximum of inherent reliability.

The engines under consideration in the Model 113 have insufficient operating statistics for analysis; hence all reliability calculations have been based on J47-GE-25 operating statistics. This approach is considered conservative since newer engines reflect in their design the experience gained from all previous engines.

The reliability of the gear boxes including clutches, based on best available data and experience, is approximately three times greater than that of the engines. The cross-shafting reliability is so high as to have a negligible effect on the calculations; and the same is true of the fixed pitch propeller.

8.4.3.2 Power Plant and Power Transmission Reliability - Figure 8.13 represents the probability of engine failure as a function of number of engines in the aircraft configuration. The twin engine aircraft with single engine capability is approximately twice as reliable as a single engine aircraft, while a four-engine aircraft with two-engine capability is five hundred to a thousand times more reliable than a two-engine aircraft.

The Safety Reliability curve presented in Figure 8.14 shows that in helicopter flight the four-engine aircraft with individual load compressors and two-engine capability is more reliable than the two- or three-engine aircraft cross shafted and with one compressor. Reliability of the three- and four-engine aircraft in airplane flight is greater than that of the two-engine aircraft. These relationships are based on reliability analyses taking into account engines, clutches, gear boxes, and shafting.

The Mission Reliability curve presented in Figure 8.15 shows the probability of mission success versus aircraft operating hours for the three configurations. This curve is based on the philosophy that any failure occurring in the power plant system would result in the mission being aborted. When Figures 8.14 and 8.15 are compared, it can be readily seen that a large increase in safety reliability can be acquired with a small decrease in theoretical mission reliability. However, the basic assumption that a single failure causes mission aborting may not apply under combat conditions. In actual practice, the aircraft having the greatest power reserve, one-engine inoperative, may show both higher mission reliability and higher safety reliability.

MODEL 113 VTOL TRANSPORT
TAKE OFF GROSS WEIGHTS FOR VTOL AND STOL OPERATION VS. ATMOSPHERIC DENSITY RATIO ρ/ρ_o

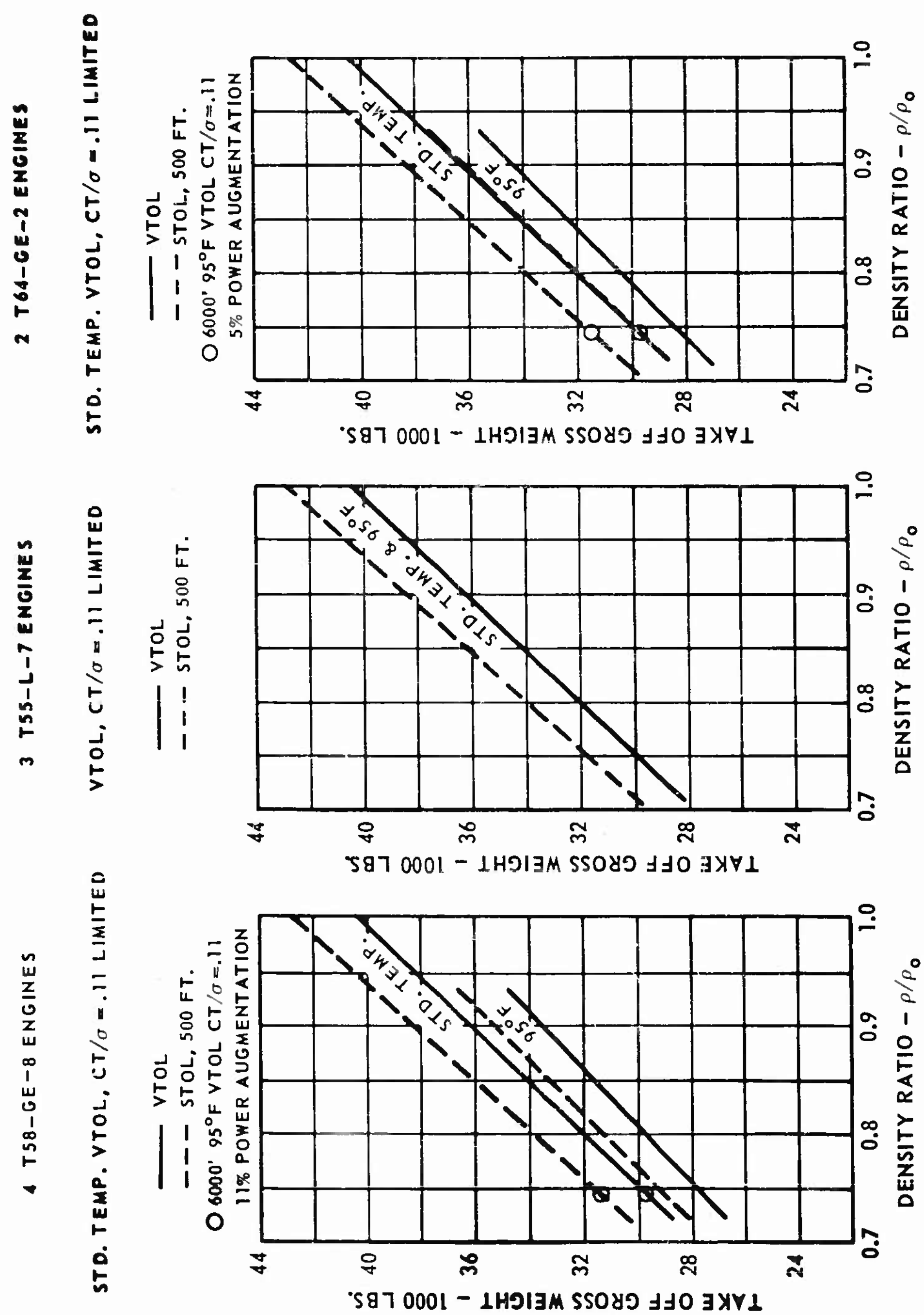


FIGURE 8.1

MODEL 113P VTOL TRANSPORT

VTOL AND STOL OUTBOUND PAYLOAD CAPABILITY FOR 250 NAUT. MILE RADIUS VS. ALTITUDE

CREW OF TWO

NASA STD. ATMOSPHERE

OPTIMUM ALTITUDE CRUISE

PAYLOAD OUTBOUND=TWICE PAYLOAD INBOUND

4 T58-GE-8 ENGINES

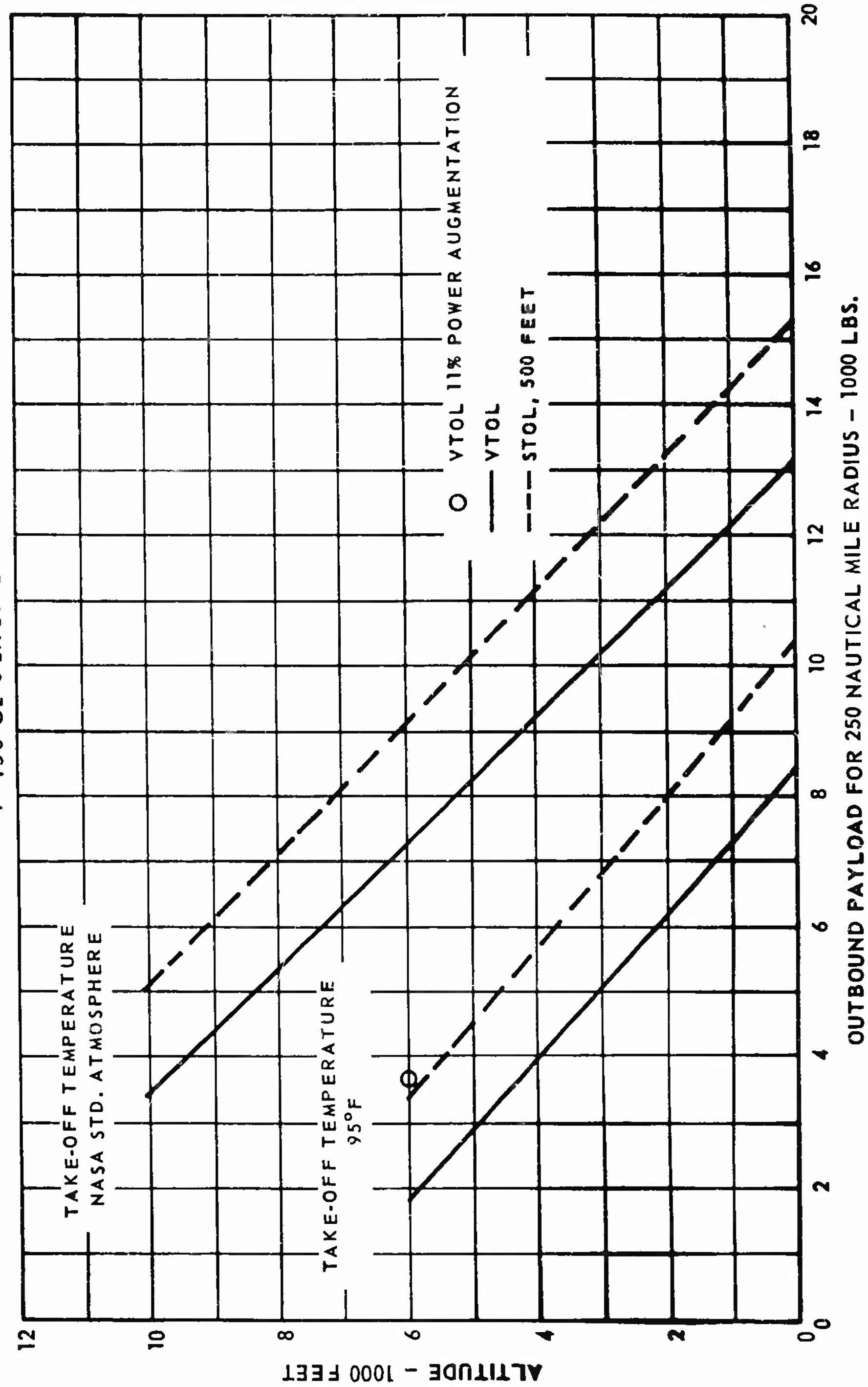


FIGURE 8.2

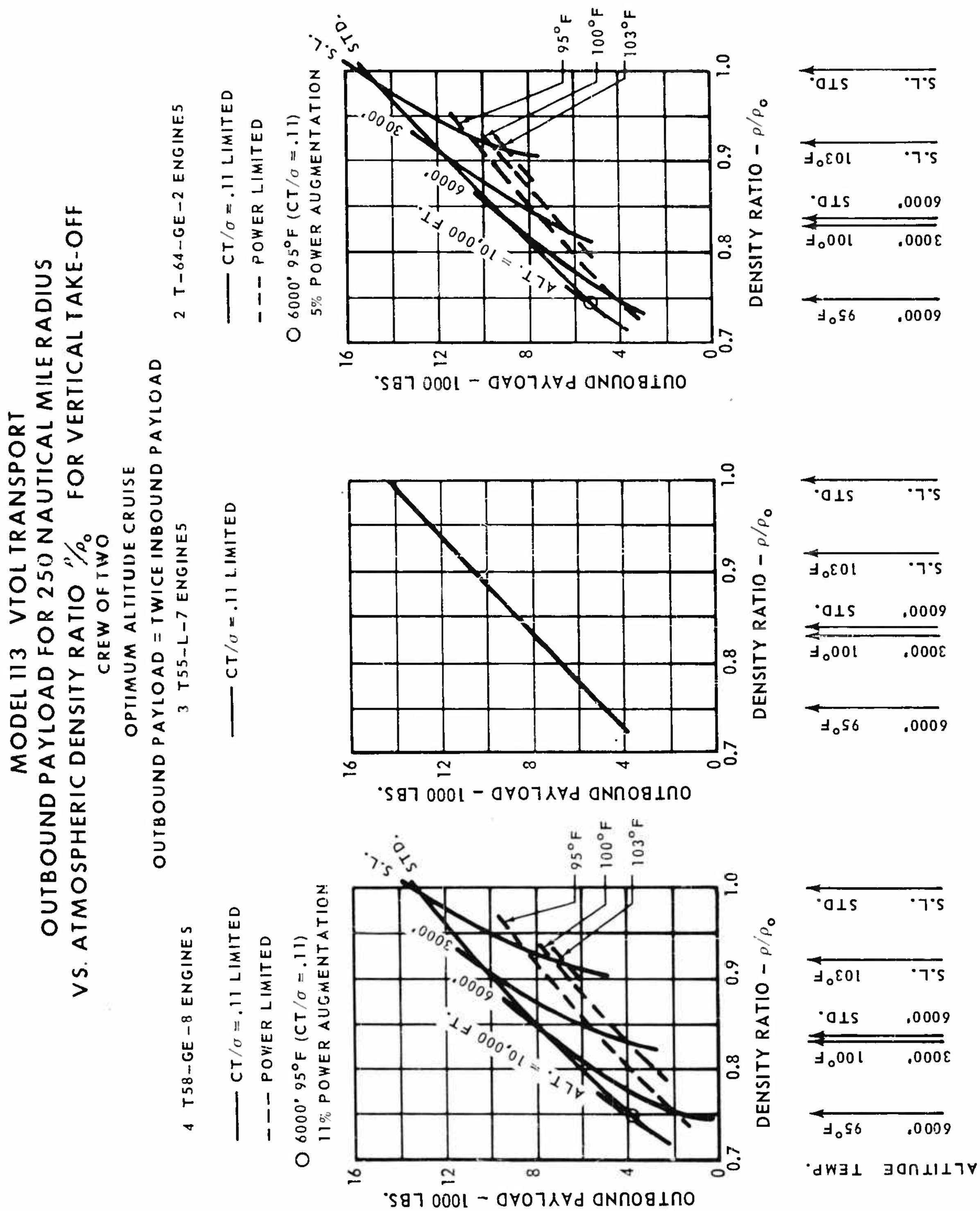

MODEL 113 VTOL TRANSPORT
OUTBOUND PAYLOAD FOR 250 NAUTICAL MILE RADIUS
VS. ATMOSPHERIC DENSITY RATIO ρ/ρo FOR VERTICAL TAKE-OFF
CREW OF TWO
OPTIMUM ALTITUDE CRUISE
OUTBOUND PAYLOAD = TWICE INBOUND PAYLOAD
2 T-64-GE-2 ENGINES
3 T55-L-7 ENGINES
4 T58-GE-8 ENGINES
CT/σ = .11 LIMITED
POWER LIMITED
6000' 95°F (CT/σ = .11) 5% POWER AUGMENTATION
6000' 95°F (CT/σ = .11) 11% POWER AUGMENTATION
S.L. STD.
95°F
100°F
103°F
3000'
6000'
ALT. = 10,000 FT.
OUTBOUND PAYLOAD – 1000 LBS.
DENSITY RATIO – ρ/ρo
ALTITUDE
TEMP.
S.L. STD.
S.L. 103°F
6000' STD.
3000' 100°F
6000' 95°F

FIGURE 8.3

MODEL 113 P VTOL TRANSPORT

EFFECT OF HOVER TIME ON MAXIMUM RADIUS

CRUISE ALTITUDE 10,000 FT.
CREW OF TWO
NASA STD. ATMOSPHERE
4 T58-GE-8 ENGINES
PAYLOAD = 4000 LB. OUTBOUND, 2000 LB. INBOUND

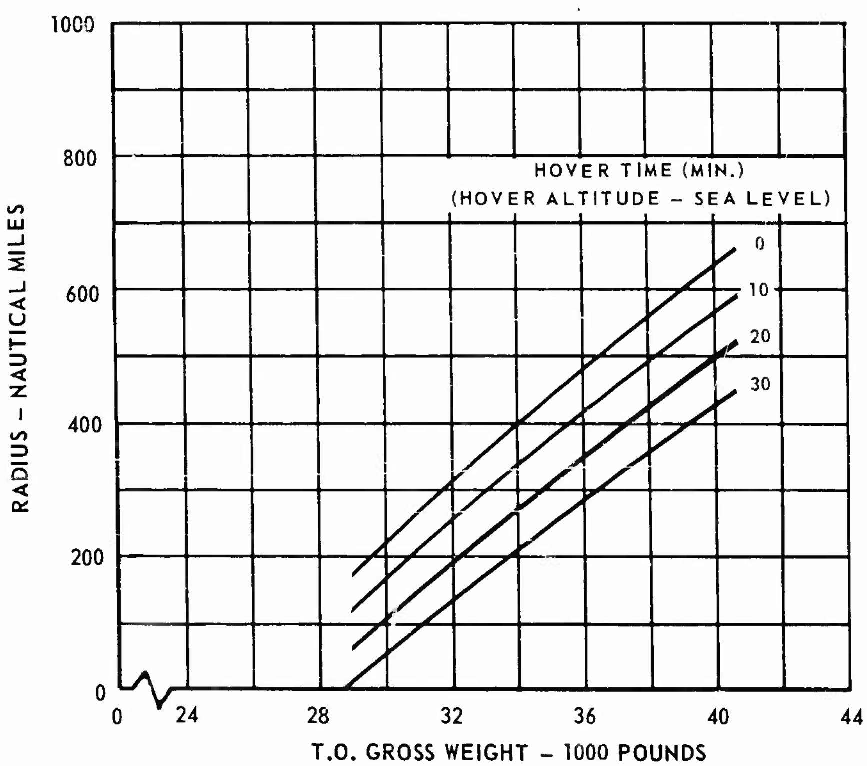


FIGURE 8.4

MODEL 113P VTOL TRANSPORT

EFFECT OF HOVER TIME ON PAYLOAD FOR ZERO RADIUS

4 T58-GE-8 ENGINES

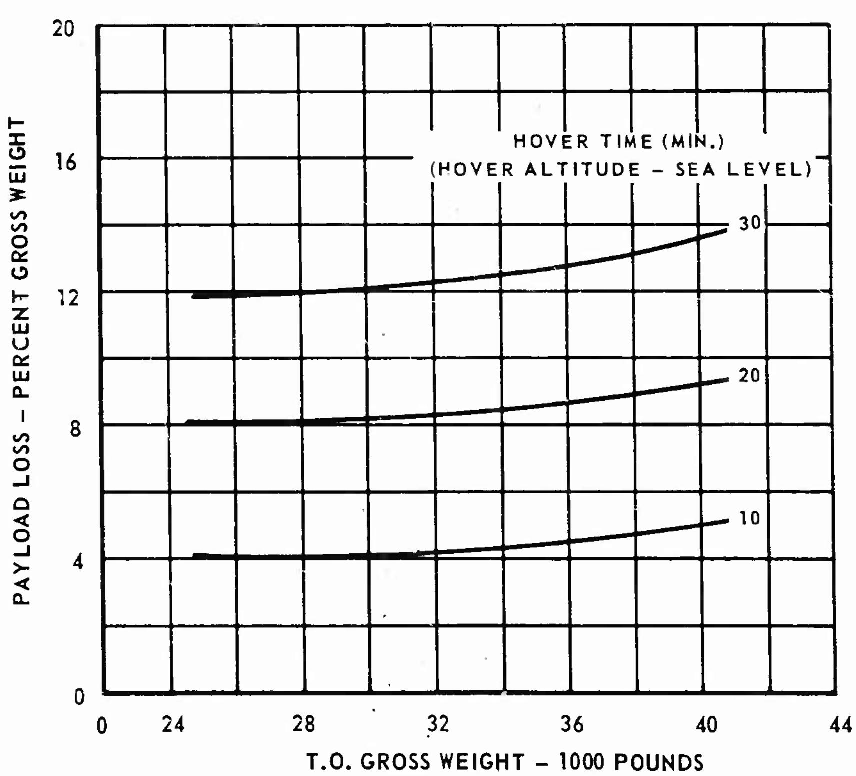


FIGURE 8.5

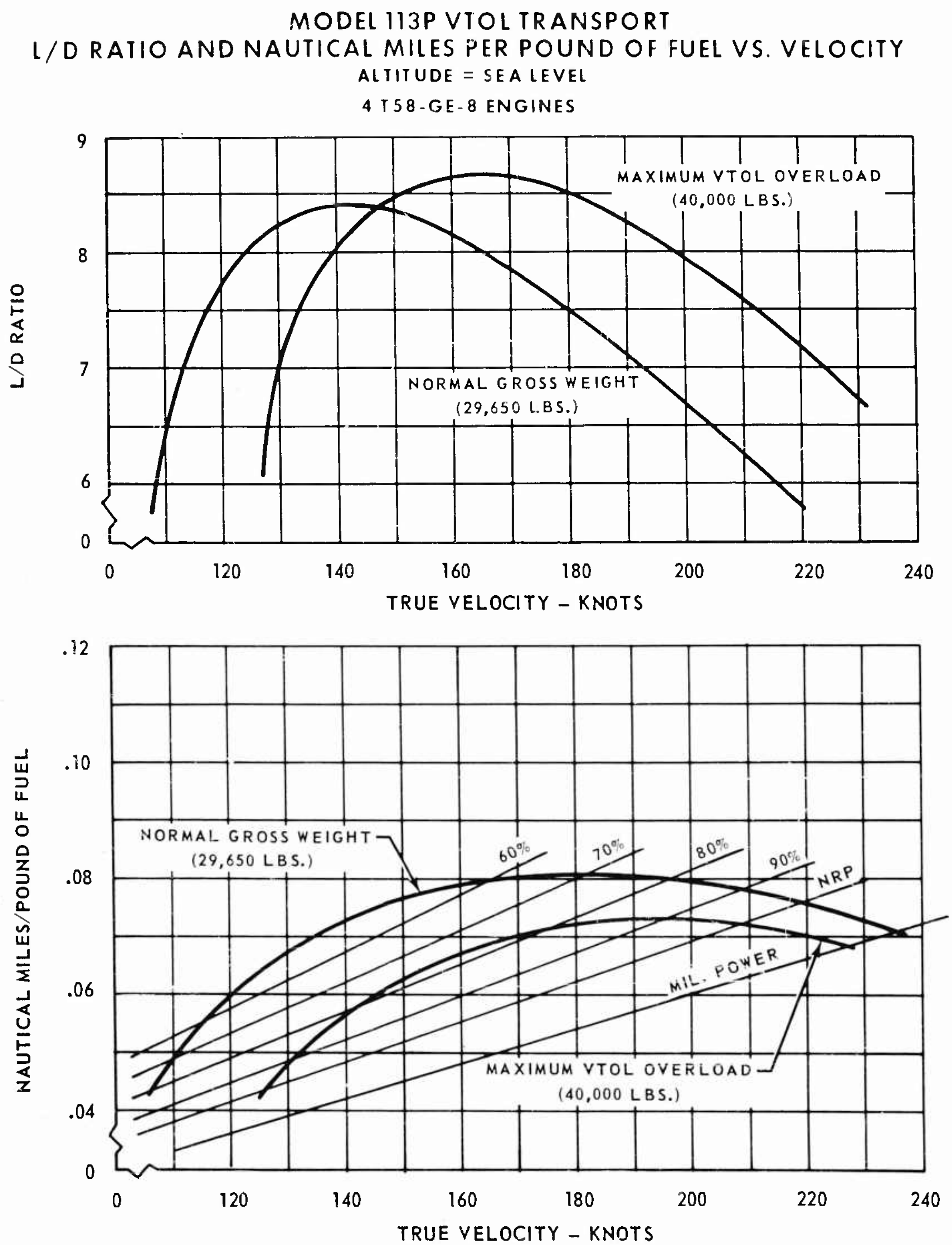
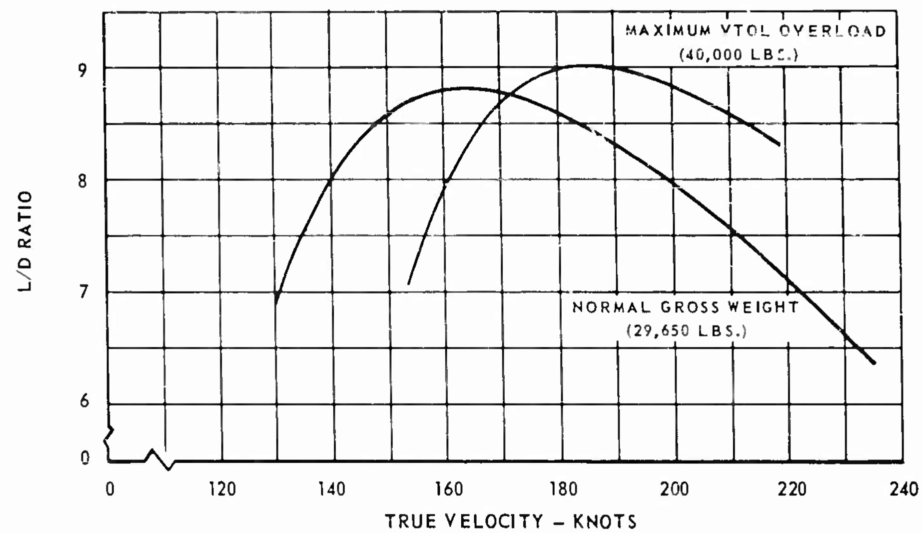

MODEL 113P VTOL TRANSPORT
L/D RATIO AND NAUTICAL MILES PER POUND OF FUEL VS. VELOCITY
ALTITUDE = SEA LEVEL
4 T58-GE-8 ENGINES
L/D RATIO
9
8
7
6
0
0
120
140
160
180
200
220
240
TRUE VELOCITY – KNOTS
MAXIMUM VTOL OVERLOAD
(40,000 LBS.)
NORMAL GROSS WEIGHT
(29,650 LBS.)
NAUTICAL MILES/POUND OF FUEL
.12
.10
.08
.06
.04
0
NORMAL GROSS WEIGHT
(29,650 LBS.)
60%
70%
80%
90%
NRP
MIL. POWER
MAXIMUM VTOL OVERLOAD
(40,000 LBS.)
TRUE VELOCITY – KNOTS

FIGURE 8.6

MODEL 113P VTOL TRANSPORT
L/D RATIO AND NAUTICAL MILES PER POUND OF FUEL VS. VELOCITY

ALTITUDE = 10,000 FT.
4 T58-GE-8 ENGINES



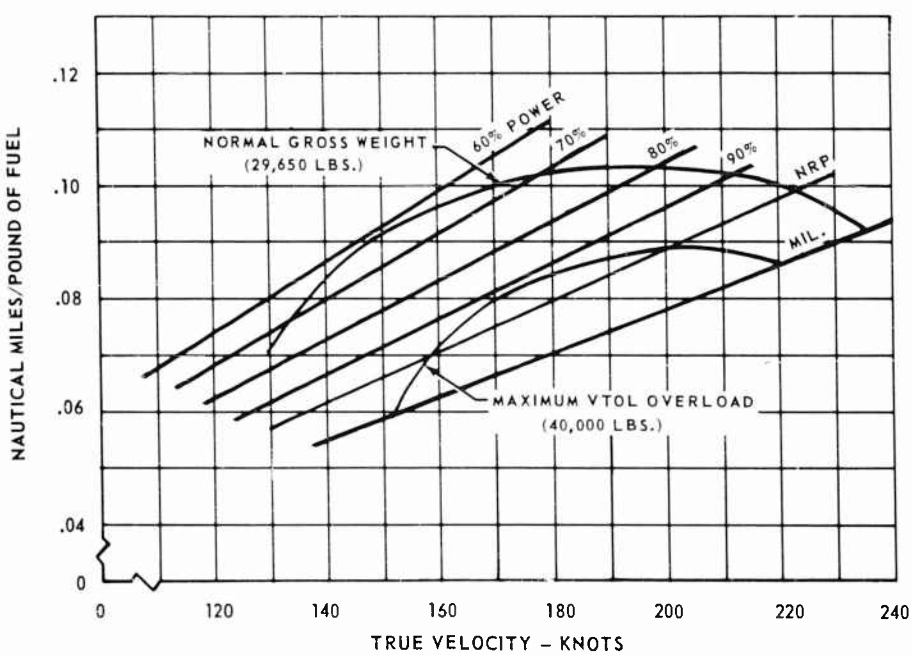


FIGURE 8.7

MODEL 113 VTOL TRANSPORT
FERRY RANGE VS. TAKE-OFF GROSS WEIGHT

OPTIMUM ALTITUDE CRUISE
CREW OF TWO

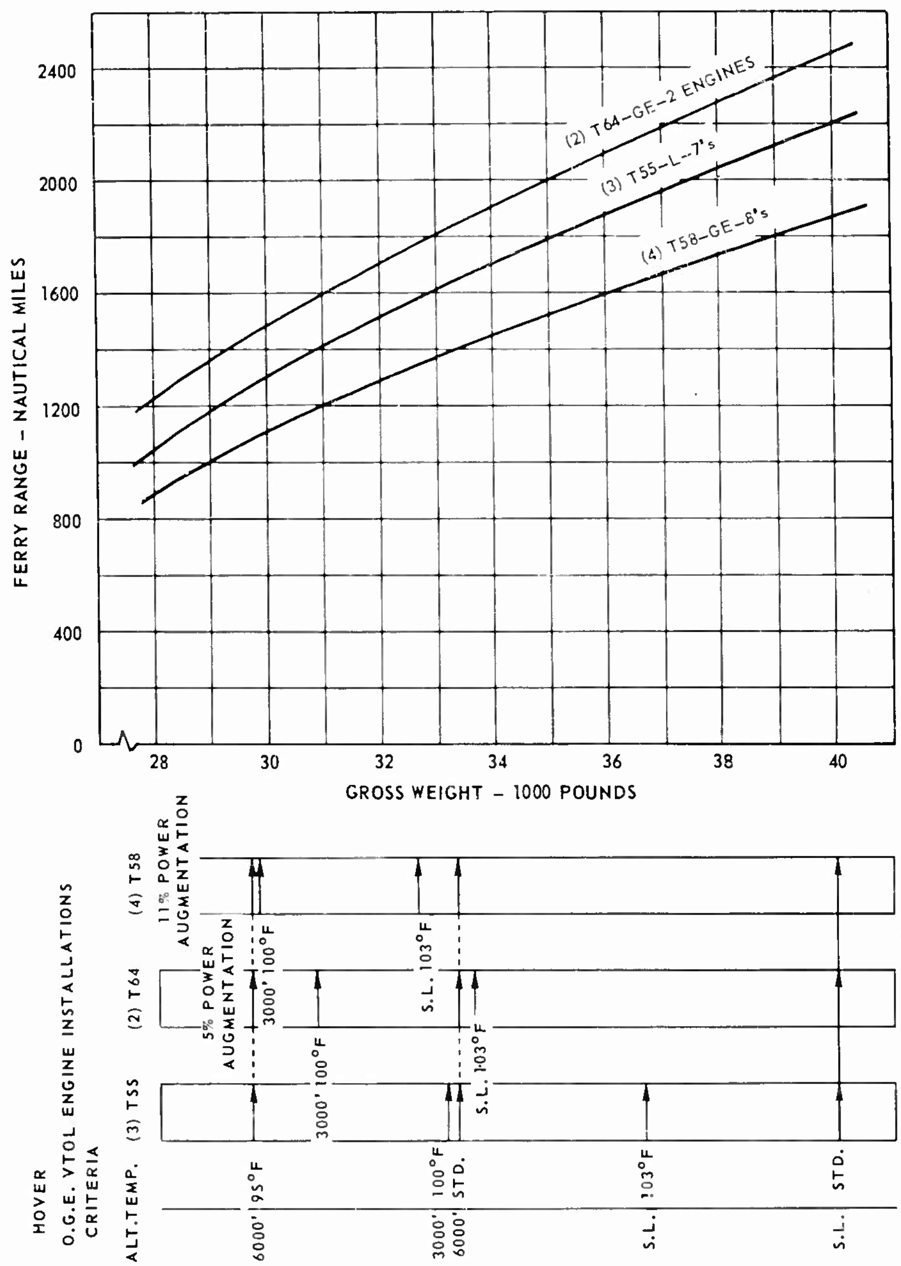
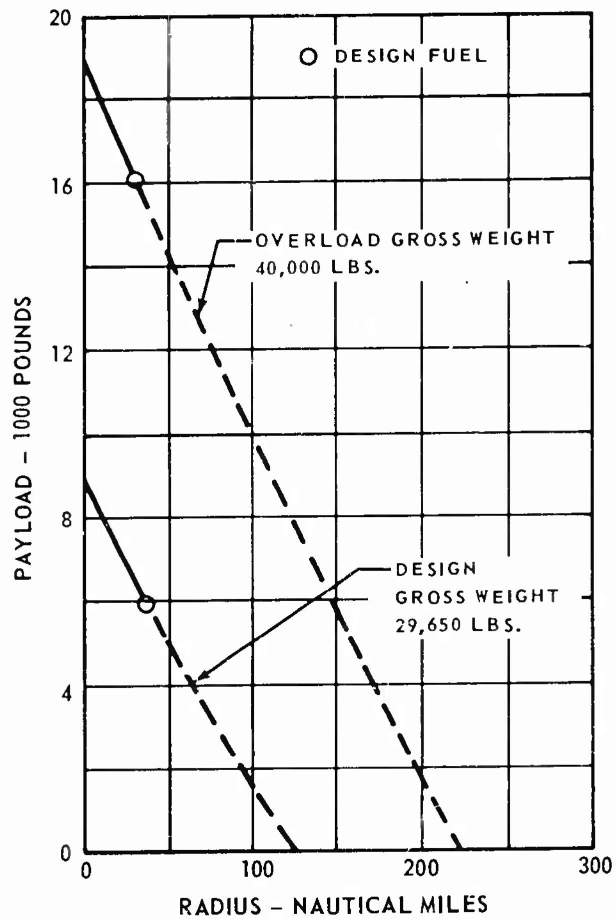


FIGURE 8.8

MODEL 113P VTOL TRANSPORT
CRANE PAYLOAD-RADIUS MISSION

SEA LEVEL CRUISE
EXTERNAL PAYLOAD
(f=50 sq. ft.)



FLIGHT PLAN

START & WARM-UP	2 MIN. NRP
T.O.	1 MIN. MAX.
PAYLOAD PICKUP	1 MIN. HOVER
CRUISE OUT (HELIC.)	SEA LEVEL
DROP PAYLOAD	1 MIN. HOVER
CRUISE BACK (AIRP.)	SEA LEVEL
LAND	
RESERVE FUEL	10%

EXTRA FUEL TANK WT. = 0.4 LB./GAL.

FIGURE 8.9

MODEL 113 VTOL TRANSPORT
ONE ENGINE INOPERATIVE SERVICE CEILING VS. GROSS WEIGHT
NASA STD. ATMOSPHERE
AIRPLANE FLIGHT

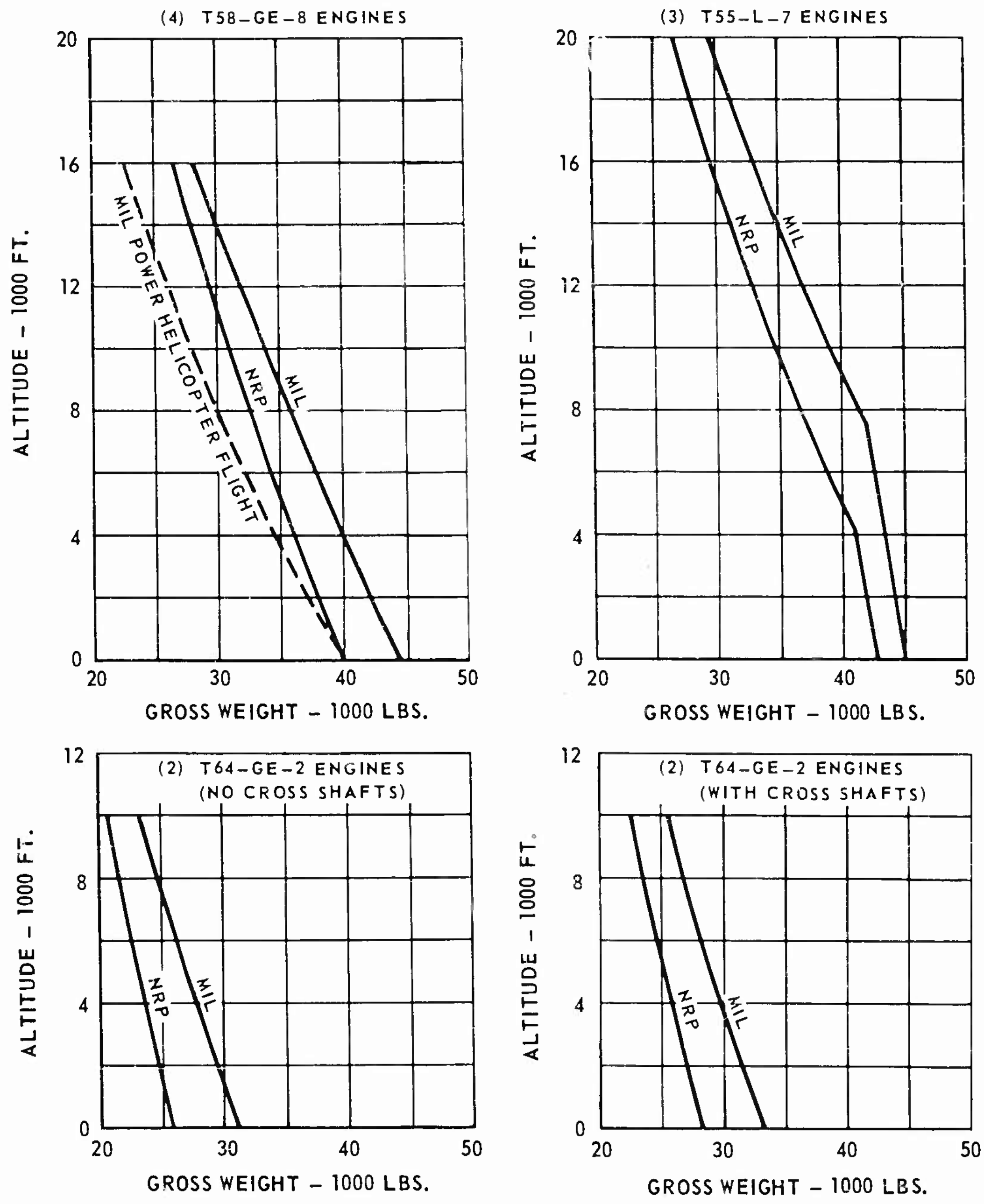


FIGURE 8.10

MODEL 113 VTOL TRANSPORT
ONE ENGINE INOPERATIVE MAXIMUM RATE OF CLIMB VS. GROSS WEIGHT

ALTITUDE = SEA LEVEL
AIRPLANE FLIGHT

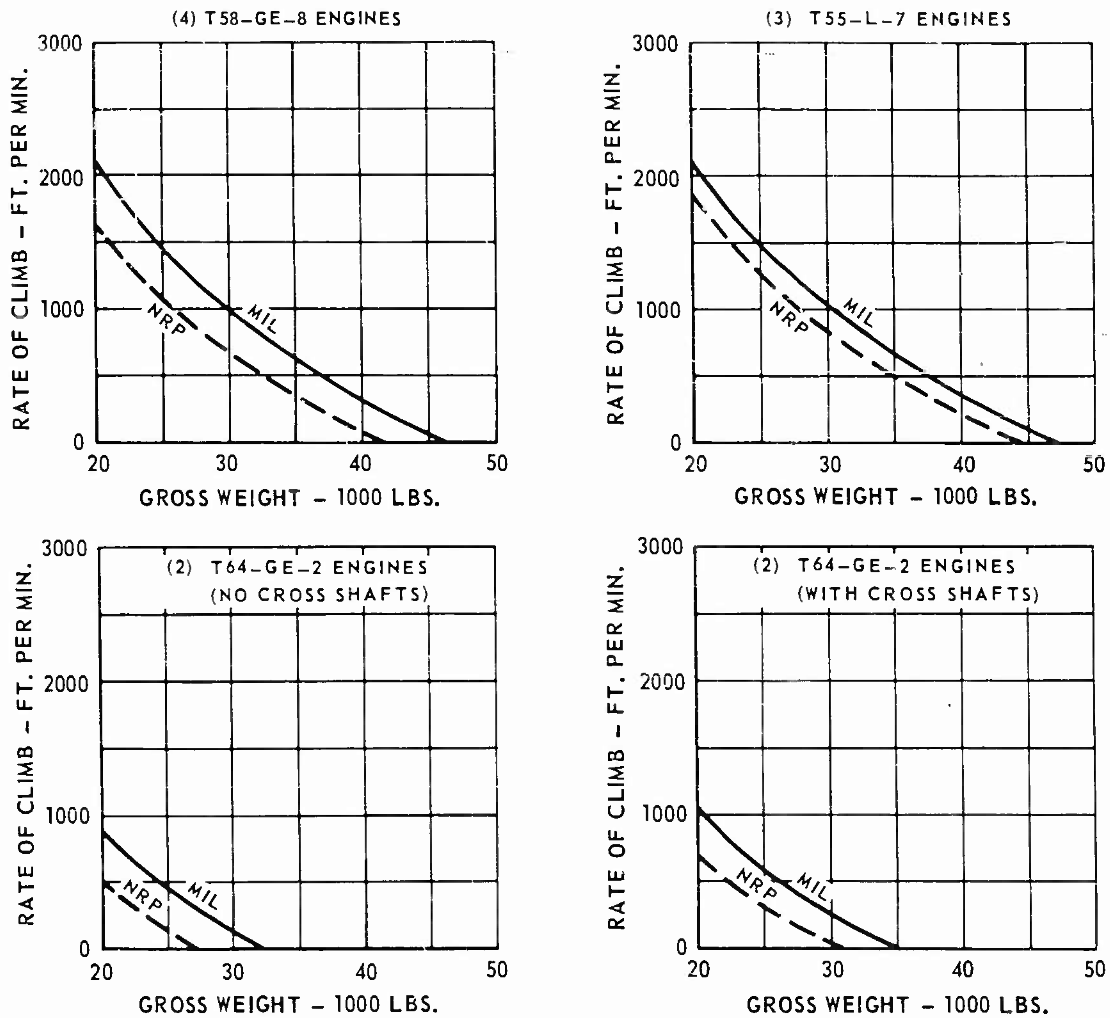


FIGURE 8.11

MODEL 113 VTOL TRANSPORT
ONE ENGINE INOPERATIVE HOVERING CEILING VS. GROSS WEIGHT

MAXIMUM ROTOR POWER
$\Omega R = 735$ FPS

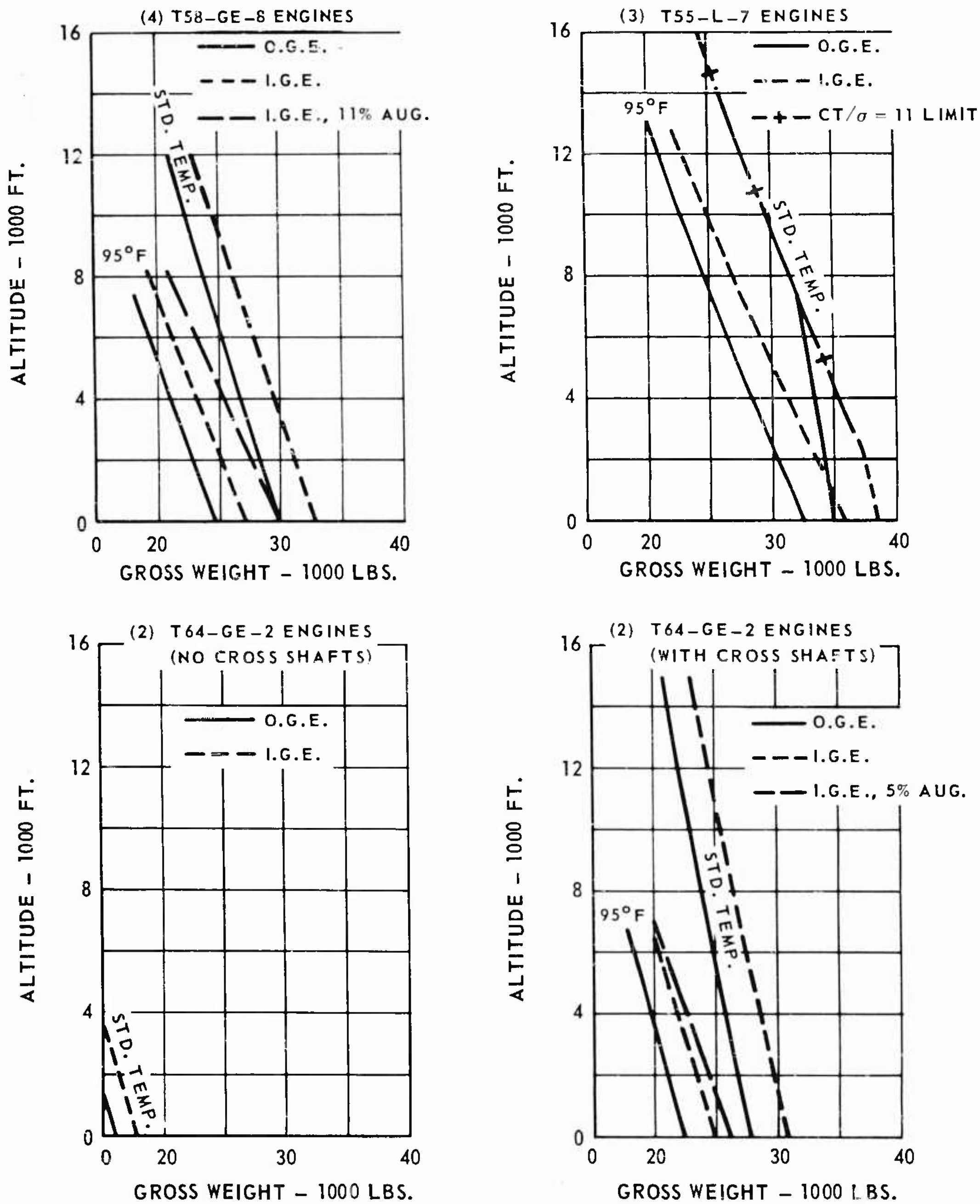


FIGURE 8.12

ENGINE RELIABILITY COMPARISON

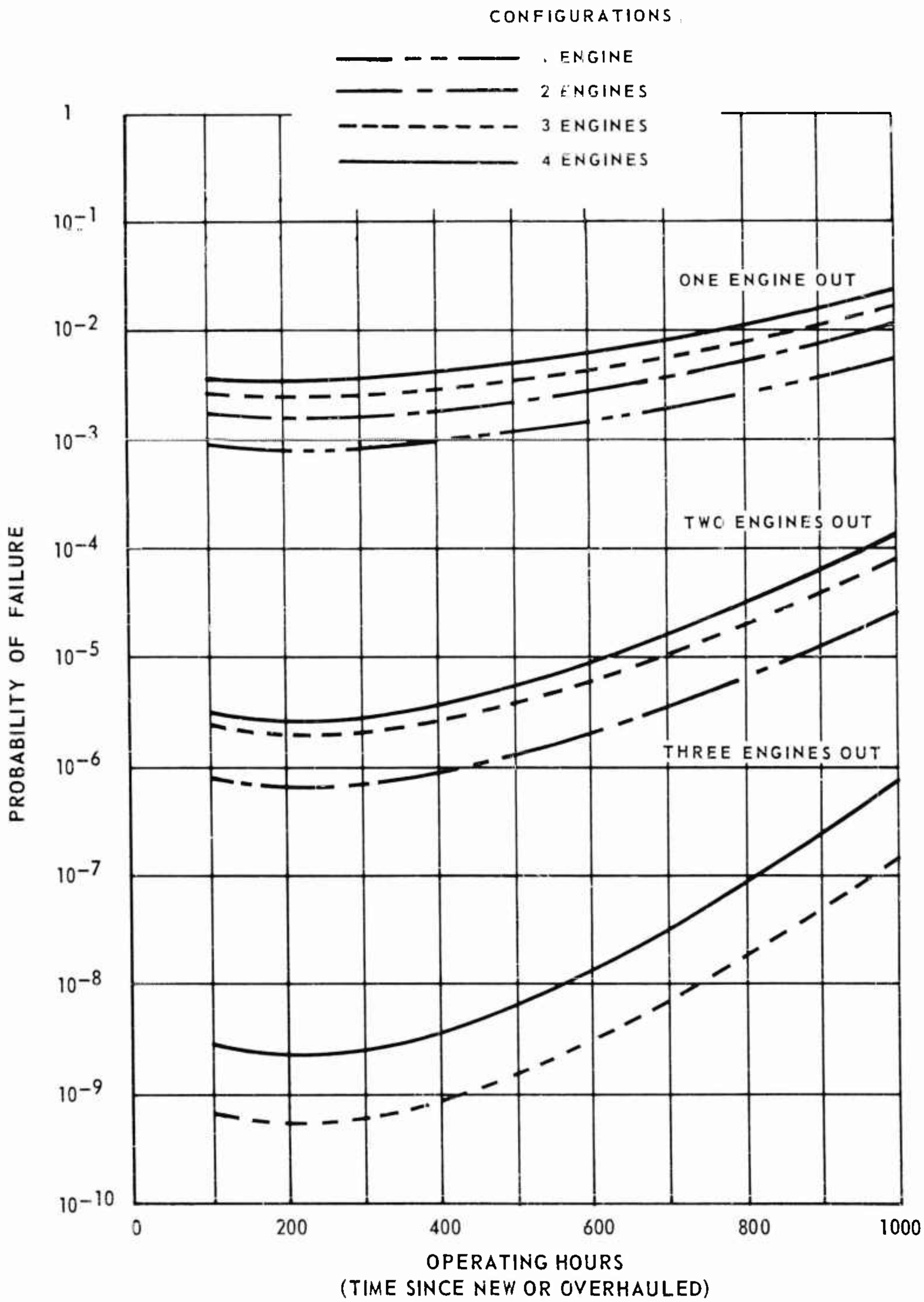


FIGURE 8.13

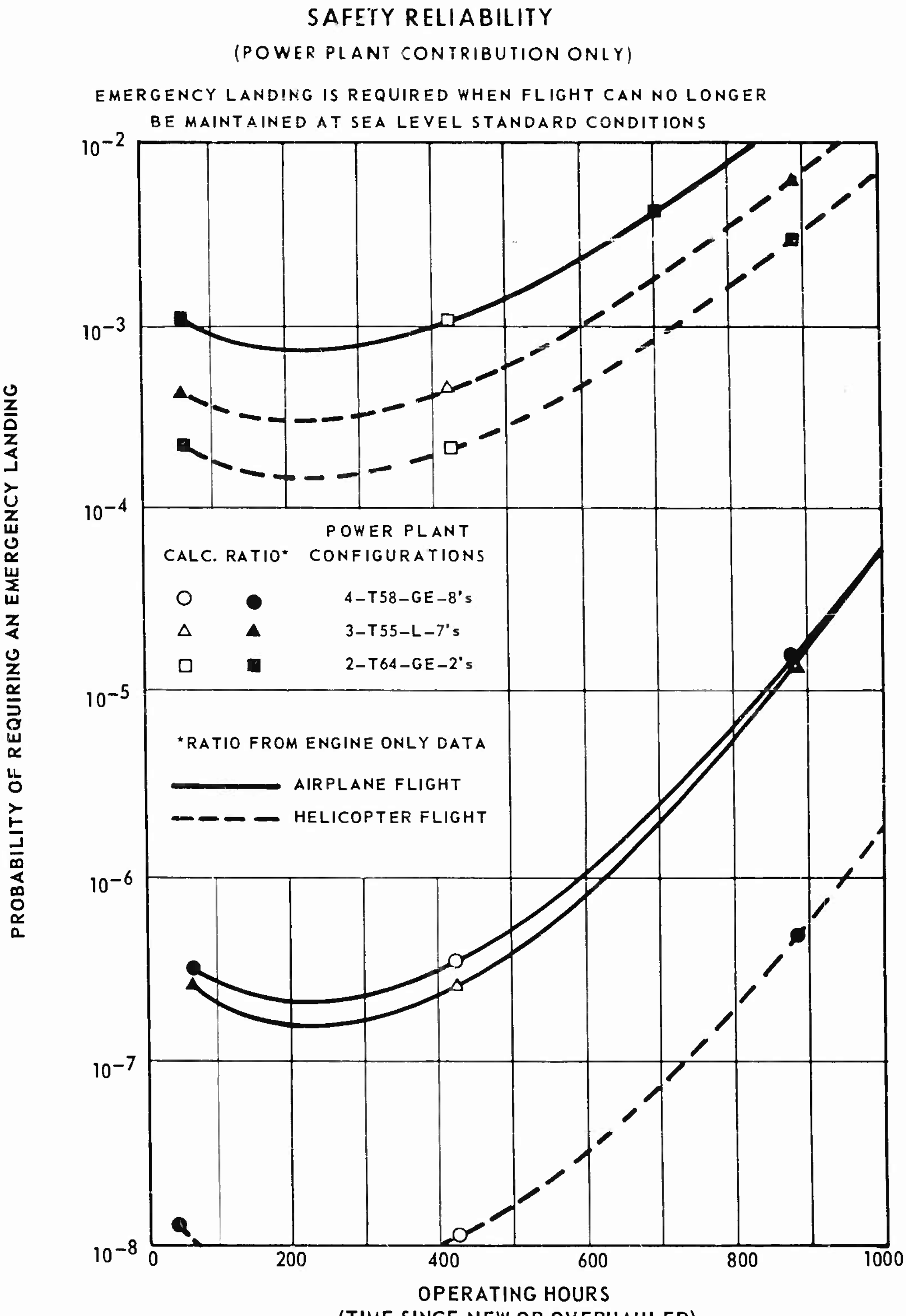

SAFETY RELIABILITY
(POWER PLANT CONTRIBUTION ONLY)
EMERGENCY LANDING IS REQUIRED WHEN FLIGHT CAN NO LONGER BE MAINTAINED AT SEA LEVEL STANDARD CONDITIONS
PROBABILITY OF REQUIRING AN EMERGENCY LANDING
10−2
10−3
10−4
10−5
10−6
10−7
10−8
CALC. RATIO*
POWER PLANT CONFIGURATIONS
4–T58–GE–8's
3–T55–L–7's
2–T64–GE–2's
*RATIO FROM ENGINE ONLY DATA
AIRPLANE FLIGHT
HELICOPTER FLIGHT
0
200
400
600
800
1000
OPERATING HOURS
(TIME SINCE NEW OR OVERHAULED)

FIGURE 8.14

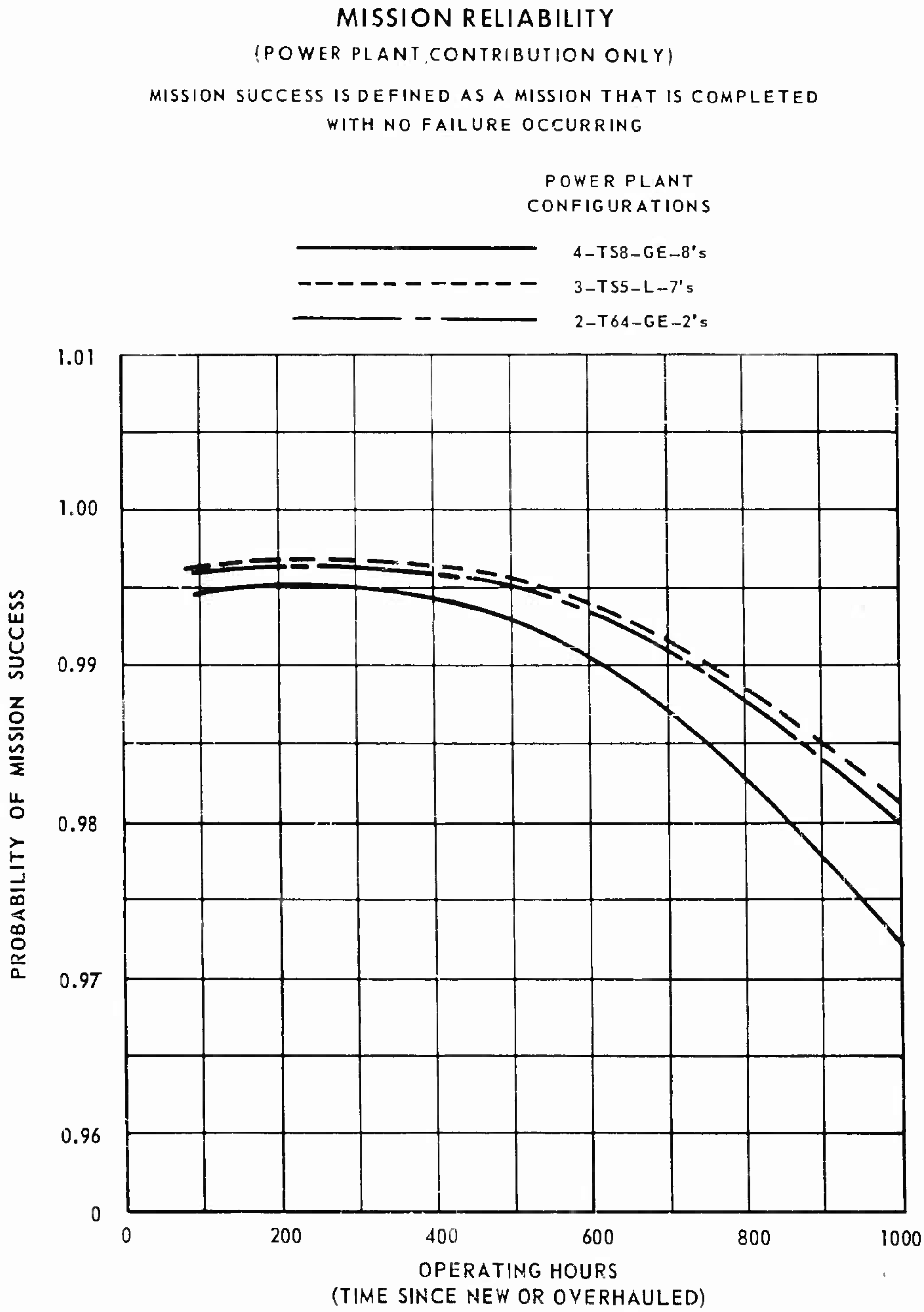

MISSION RELIABILITY
(POWER PLANT CONTRIBUTION ONLY)
MISSION SUCCESS IS DEFINED AS A MISSION THAT IS COMPLETED
WITH NO FAILURE OCCURRING
POWER PLANT
CONFIGURATIONS
4-TS8-GE-8's
3-TS5-L-7's
2-T64-GE-2's
PROBABILITY OF MISSION SUCCESS
1.01
1.00
0.99
0.98
0.97
0.96
0
0
200
400
600
800
1000
OPERATING HOURS
(TIME SINCE NEW OR OVERHAULED)

FIGURE 8.15

SECTION IX

9. OPERATING COSTS

9.1 Maintenance Costs

9.1.1 General - Good agreement has been found between system maintenance cost (parts plus labor) and system weight of similar designs. For maintenance cost estimation purposes the aircraft is divided into the following systems: Power Plant, Transmission and Drives, Rotor, Tip Jets, Instrument and Electronics, and the remainder termed Airframe. Each system is further sub-divided to account for major design differences. Those systems or groups that are not listed separately are included in the airframe group.

9.1.2 Power Plant - Most of the turbine engine experience to date has been gained on fixed wing aircraft. Since most of the flight time of the unloaded rotor helicopter will be spent in airplane flight, the maintenance cost is expected to be very similar to that of the airplanes having the same engines.

9.1.3 Transmission and Drives - In place of high torque transmissions the Model 113 has compressors, clutches, and tip jets. Tip jets are treated separately, and compressors and clutches are assumed to have the same specific maintenance as high torque gear boxes. This is considered conservative since these components are operated only in take-off and landing conditions.

9.1.4 Rotor - The rotor maintenance cost of rotary wing aircraft to date has been a substantial part of the total. The cost per flight hour of the Model 113 rotor represents a considerable reduction due to the specific design of the McDonnell semi-articulated rotor system used in the unloaded rotor concept. Some major factors contributing to this improvement are:

a. The blade retention system has no bearings and no dampers.

b. The rotor is designed to have infinite life. Blade changes would be necessary only in the event of random damage.

c. All oscillating bearings (14) employing a Teflon material are designed to have a life of 2500 hours without any lubrication or attention.

d. The tail rotor is small in comparison to that of conventional helicopters of the same gross weight and has a thrust output of only one-fifth that of an anti-torque tail rotor.

The rotor maintenance cost is conservative since no credit has been taken for the reduction in rotor rpm in the airplane flight regime to one-half that of hovering rpm. The airplane flight regime represents approximately 90 percent of the time. Furthermore, the tail rotor operates only in helicopter flight.

9.1.5 Tip Jets - The only data on the tip pressure jet maintenance stems from MAC experience. Three different McDonnell tip burner programs - the XV-1, the Navy 75-foot rotor, and Model 120 which comprise a total of 7500 hours of burner operation - provides the basis for predicting maintenance of operational tip burners used in the Model 113. The maintenance cost is based on a 150-hour flameholder life; i.e., a flameholder replacement every 1500 hours of flight, assuming helicopter flight is 10 percent of total flight time.

9.1.6 Instruments and Electronics - The specific maintenance cost per pound of instruments and electronics is essentially the same for any transport of this size providing the mission requirements are similar.

9.1.7 Airframe - The airframe, consisting of fuselage, wing, tail, landing gear, hydraulics, electrical, fuel system, and controls, is assumed to have specific maintenance cost per pound that is derived from a mean value for all aircraft. As airframe weight increases, the maintenance cost per pound decreases.

9.1.8 Maintenance Cost Summary - Using the component weights from Table 7.2 and combining these with maintenance equations (Reference 12.23) the maintenance costs of the Model 113 with four T58-GE-8 engines are as shown:

MAINTENANCE COST SUMMARY
(Dollars)

Maintenance(1)	
Power Plant	22.30
Transmission and Drives	8.00
Rotor	10.70
Tip Jets	.80
Electronics and Instruments	4.35
Airframe	8.00
Total Cost per Flight Hour	54.15 x 2.5 = 135.35

(1) For 4 T58-GE-8's - would be slightly different for the T55-L-7's and T64-GE-2's.

9.2 Flight Operations Cost

9.2.1 Military Flight Crew Cost - The average rank of the Army aircraft pilot per Reference 12.29 is a Captain with seven years' service; the average copilot is a Warrant Officer, Grade W-1, with nine years' service; and the average flying crew chief is a Corporal with three years' service. The average work month is 173.3 hours which is broken down into 50 hours flying times, 50 hours related duties, and 73.3 hours additional duties. The cost rate for the average flight crew, chargeable to the flight time only, is $6.31 per hour for Grade O-3, $4.58 per hour for Grade W-1, and $2.66 for Grade E-4, making a total of $13.55 per flight hour for a 30,000-pound class VTOL transport.

9.2.2 Fuel and Lubricants (POL) Cost - POL costs are rather insensitive to the radius of operation, except when extended hovering periods are employed. The POL cost for turbine engines amounts to 12.2 cents per gallon in CONUS. Using this value with a typical VTOL light transport mission definition, the POL cost for the Model 113 amounts to $32.30 per flight hour.

9.3 Direct Cost Summary - The total operating cost for the Model 113 is as follows:

OPERATING COST SUMMARY
(Dollars)

Maintenance	135.35
POL	32.30
Flight Crew	13.55
Total Cost per Flight Hour	181.20

The total maintenance cost for the 30,000-pound Model 113 is of the same order of magnitude as that of present operational helicopters in the 12,000- to 14,000-pound gross weight class. The direct operating cost is only slightly greater, but the cruise velocity is doubled and the payload is quadrupled, resulting in a much lower cost per ton mile value.

SECTION X

10. DEVELOPMENT AND PRODUCTION ESTIMATES

10.1 General - Costs and deliveries for the recommended configurations are provided herein for planning purposes. Since estimates for the various configurations do not differ too greatly, average data is shown and a tolerance of plus or minus 25 percent is considered applicable.

10.2 Development Schedule and Cost - The development time schedule for both five and ten prototype aircraft is shown in Figure 10.1. The development cost for five aircraft is estimated to be $29,000,000. The development cost for ten aircraft is estimated to be $36,000,000.

10.3 Production Cost - Annual expenditures for follow-on production programs, building up to peak production rates of 25 and 100 aircraft per year, are shown in 10.3.1 and 10.3.2, respectively.

10.3.1 Expenditures for 25 Per Year Production

FY 1962	$ 380,000
1963	5,000,000
1964	10,000,000
1965	13,000,000
1966	13,000,000
1967	13,000,000

10.3.2 Expenditures for 100 Per Year Production

FY 1962	$ 300,000
1963	5,000,000
1964	19,000,000
1965	37,000,000
1966	44,000,000
1967	41,000,000

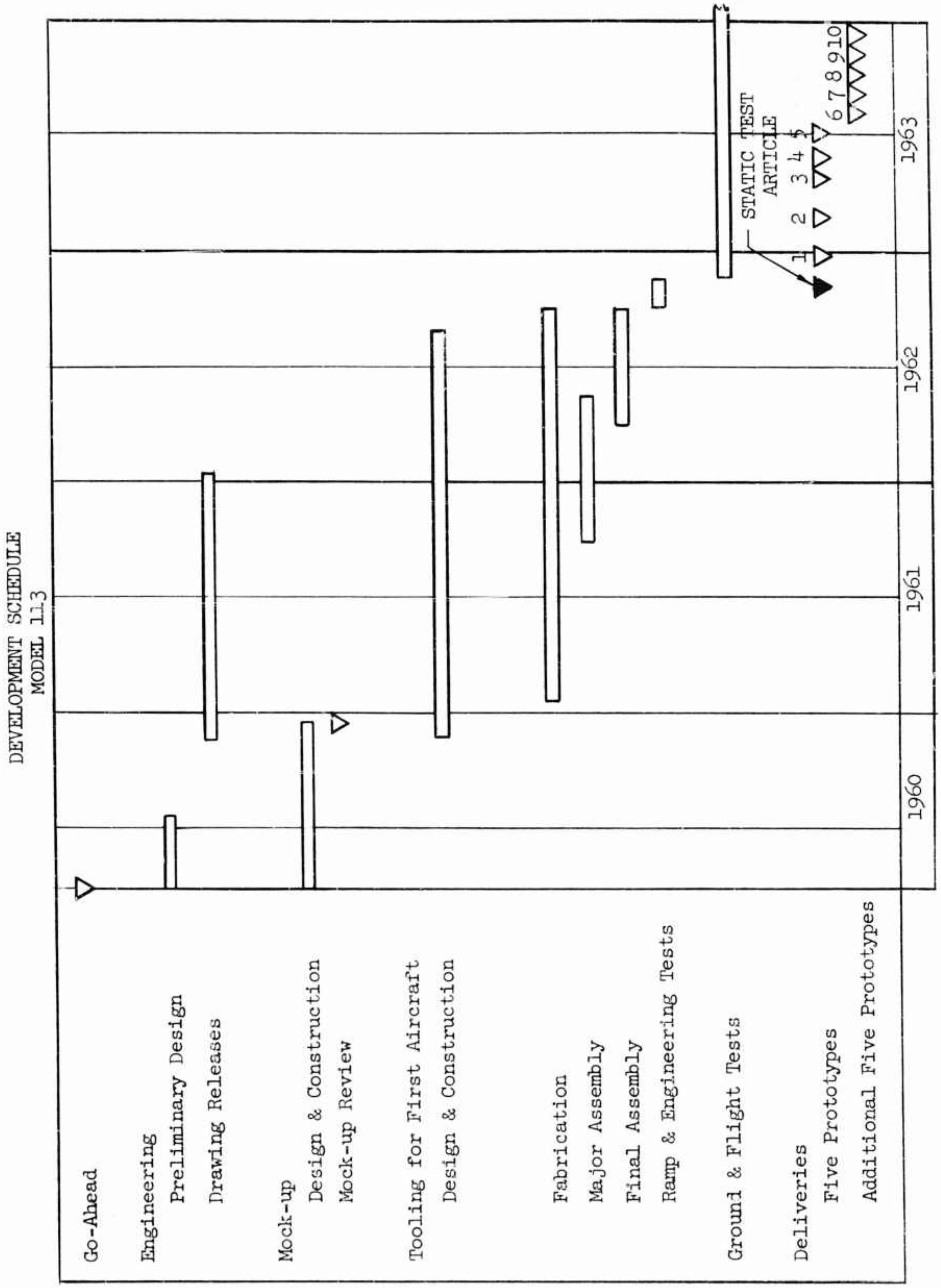

DEVELOPMENT SCHEDULE
MODEL 113
Go-Ahead
Engineering
Preliminary Design
Drawing Releases
Mock-up
Design & Construction
Mock-up Review
Tooling for First Aircraft
Design & Construction
Fabrication
Major Assembly
Final Assembly
Ramp & Engineering Tests
Ground & Flight Tests
Deliveries
Five Prototypes
Additional Five Prototypes
STATIC TEST ARTICLE
1 2 3 4 5
6 7 8 9 10
1960
1961
1962
1963

Figure 10.1

SECTION XI

11. CONCLUSIONS

11.1 Among the various VTOL aircraft concepts the XV-1 type of unloaded rotor compound helicopter with rotor jet drive has the lowest empty weight and highest transport productivity if the design is based on the TRECOM criterion of hovering out of ground effect at 6000 feet 95°F with two-ton payload, and fuel for a radius of action between 200 and 500 nautical miles.

11.2 A combination of the 6000-foot 95°F hovering requirement with a radius of action in the order of 500 nautical miles results in an uneconomic, large size for the two-ton Army light transport. Therefore, limiting the design radius of action to a value of the order of 250 nautical miles is strongly recommended.

11.3 The recommended Army light VTOL transport of the XV-1 type has about 30,000 pounds design gross weight and can be powered by any of three different gas turbines: four T58-GE-8 engines, or three T55-L-7 engines, or two T64-GE-2 engines, which are all available in the 1960-63 period.

11.4 The recommended VTOL aircraft satisfies the 6000-foot 95°F hovering criterion for the design gross weight while carrying two tons of payload for a radius of action of 235 to 340 nautical miles, depending on engine selection.

11.5 The recommended VTOL aircraft powered by two T64-GE-2 engines (which allows hovering out of ground effect at standard sea level condition at a gross weight of 40,000 pounds) carries two tons of payload for a radius of action of 925 nautical miles, or a payload of 14,800 pounds for a radius of action of 250 nautical miles.

11.6 The recommended VTOL aircraft possesses the handling characteristics of a stable helicopter in low speed flight and the flying qualities of a conventional fixed wing transport in cruise and high speed flight; the aircraft inherent stability levels relegate automatic stabilization equipment to secondary systems.

11.7 The recommended VTOL aircraft possesses the safety and reliability level of a multi-turbine helicopter characterized by the ability to autorotate, by ground cushion effects, high rotor rotational inertia, exceptional engine out performance, and centrally located lift systems that can create no uncontrollable lateral or longitudinal moment unbalance in event of power failure.

11.8 Maintenance of the recommended VTOL aircraft is appreciably reduced over that of current helicopters by combining the McDonnell type of semi-articulated rotor system with the tip jet drive system. Military maintenance costs are estimated to be 135 dollars per flight hour.

11.9 The military total direct operating cost of the recommended VTOL aircraft, excluding depreciation, is estimated to be 181 dollars per flight hour.

11.10 The recommended VTOL aircraft uses a small diameter-high solidity rotor rather than a large diameter-low solidity rotor resulting in minimum silhouette and aircraft size compatible with tripartite operation. The high solidity rotor permits the establishment of rotor dynamic characteristics that inherently eliminate the possibility of ground resonance or mechanical instability. Therefore, no compromise involving nature of take-off or landing terrain, STOL operation, or aircraft maneuverability is required.

SECTION XII

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SECTION XIV

14. NOTATION AND SYMBOLS

a	Lift curve slope
a_1	Backward tilt of rotor, deg.
a_o	Rotor coning angle, deg.
AR	Aspect ratio, b^2/S_W
B	Tip loss factor
b	Wing span, ft. Number of rotor blades
c	Specific fuel consumption, lbs/HP/hr. Wing mean aerodynamic chord, ft. Rotor blade chord, ft.
C_D	Drag coefficient, D/q x area
C_d	Drag coefficient of element
C_L	Lift coefficient, L/q x area
C_ℓ	Rolling moment coefficient, Moment/q S_W b
C_m	Pitching moment coefficient, Moment/q S_W c
C_N	Yawing moment coefficient, Moment/q S_W b
C_p	Propeller power coefficient, $(HP/1000)/2\sigma\left(\frac{n}{1000}\right)^3\left(\frac{D}{10}\right)^5$
C_Q	Rotor torque coefficient, $Q/\rho\pi R^2 (\Omega R)^2 R$
C_T	Rotor thrust coefficient, $T/\rho\pi R^2 (\Omega R)^2$
C_T/σ	Aerodynamic blade loading
D	Drag force, lbs. Diameter, ft.
D.L.	Hovering download, percent gross weight
D/L	Drag-lift ratio
e	Airplane efficiency factor
f	Equivalent parasite area (C_D = 1.0), ft.^{2}
fps	Feet per second
F_J	Pressure jet thrust, lbs.

g	Acceleration of gravity, ft/sec.^{2}
G.R.	Engine-propeller gear ratio
HP	Horsepower
i	Incidence, deg.
J	Propeller advance ratio, V/n D
K	Rotor slipstream contraction ratio
K_E	Ratio of empty weight to design gross weight
L	Lift force, lbs.
"L"	Slope of C_D vs C_L^2 curve, $1/\pi AR_e$
M	Mach number
N_E	Turbine speed, rpm
n	Revolutions per second
NRP	Normal rated power
P.C.	Rotor pitch-cone ratio
q	Dynamic pressure, lbs/ft.^{2}
r	Rotor radius, ft., at a particular blade element
R	Rotor radius, ft. Resistance force, lbs.
R.N.	Reynolds number
rpm	Revolutions per minute
R/C	Rate of climb, fpm
S	Area, ft.^{2} Distance, ft.
T	Thrust, lbs.
TAF	Total activity factor, propeller
THP	Thrust horsepower
T/F	Hovering merit factor, (C_T/C_Q)
u_t	Blade element tangential velocity component $(r/R + \mu \sin \psi)$

μ	Rotor advance ratio, $V/\Omega R$
μ_f	Coefficient of friction
V	Flight path velocity, fps
V_B	Block speed, knots
V_{kn}	Flight path velocity, knots
v_i	Rotor induced velocity, fps
V_v	Vertical rate of climb, fpm
W	Gross weight, lbs.
W_E	Weight empty, lbs.
x	Ratio of blade element radius to rotor blade radius, r/R
X	Longitudinal rotor force, lbs.
Z	Distance below rotor, ft.
α	Angle of attack, deg.
β	Propeller blade angle at 3/4 radius, deg.
γ	Angle of climb, deg.
ϵ	Downwash angle, deg.
ρ	Mass density of air, slugs/ft.^{3}
σ	Rotor solidity, $\frac{b\,c}{\pi R}$ Density ratio at altitude
η_p	Propeller efficiency
η_I	Installation efficiency
η	Over-all propulsive efficiency Rotor control angle, deg.
θ_o	Rotor blade angle at 3/4 radius for zero cone, deg.
$\theta_{3/4}$	Rotor blade angle at 3/4 radius, deg.
θ_{root}	Rotor blade angle at blade root, deg.
θ_1	Rotor blade twist, deg.
Ω	Angular velocity, radians per second

λ	Inflow ratio, $\bar{V} \sin \alpha - v_i/\Omega R$ Wing taper ratio
ψ	Rotor blade azimuth angle, deg.
$\Lambda_{c/4}$	Quarter chord sweepback, deg.
δ_3	Arctan of pitch-cone ratio

<u>Subscripts</u>

A	Airplane and air distance
b	Biplane
C	Climb condition
g	Ground
H	Horizontal tail
i	Induced
J	Jet
M	Mach number
o	Profile and initial condition
p	Parasite and propeller
r	Blade element
R	Rotor
t	Tip
v	Vertical climb
W	Wing
1	Final condition

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