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## THESIS

GUIDE FOR CONCEPTUAL  
HELICOPTER DESIGN

by

Stephen Glenn Kee

June 1983

Thesis Advisor:

Donald M. Layton

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## Guide for Conceptual Helicopter Design

by

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Submitted in partial fulfillment of the  
requirements for the degree of

MASTER OF SCIENCE IN AERONAUTICAL ENGINEERING

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## ABSTRACT

A conceptual helicopter design method utilizing closed form formulas and approximations from historical data is developed for use in a helicopter design course. The design manual is to be used for the conceptual design of a single main rotor, utility helicopter. The manual was written principally for use in AE4306--Helicopter Design.

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## I. INTRODUCTION

### A. BACKGROUND

The methods of conceptual helicopter design have changed dramatically over the past 30 years. Two principal reasons for this change are the greatly increased research and development costs and the scientific advances which have led to engineering specialization. Today almost all new production helicopters are to a great extent hybrids of their predecessors, with technological advances applied to existing systems rather than to an entirely new design. The helicopters of today are designed by committees composed of specialists in various fields who compromise and optimize to reach a final design. As a result the designer of today is presented with challenges not faced by his predecessors.

Traditionally, design courses have been used to equip the engineering student with the skills needed to meet the challenges in the real world of engineering [Ref. 1]. Even though the required skills have changed, design courses are still very important in the education process; however, design is no longer being taught in many aeronautical engineering schools [Ref. 2]. In the field of helicopter design this is due largely to the complexity of the subject matter.

When it was decided to incorporate a helicopter design course into the curriculum at the Naval Postgraduate School,

the search for a design methodology that could be used in this course led to the discovery that few texts on the helicopter design process exist. Many manufacturers have their own methods, but since they rely so heavily on previous in-house designs, they are not suitable for use in an educational framework. It became apparent that a conceptual design process for the course would have to be developed.

#### B. GOALS

The goals of this study were to:

1. provide a simple design process for use in a capstone helicopter design course,
2. provide an opportunity to exercise the skills of optimization and decision-making, not only in engineering but in managerial areas as well, and
3. provide a framework from which further research can be undertaken.

## II. APPROACH TO THE PROBLEM

The first step in the development of the helicopter design course was to analyze the complex nature of helicopter engineering and to simplify its concepts into a series of design steps. It became evident that the design methodology would require an iterative process due to the relationships among the major design variables. A decision was made not to use the computer directly to perform the iterations, so that the student, as he performed the required calculations, would develop a much greater understanding of the interrelationships of the variables. However, the decision not to utilize a computer increases the complexity and length of the process.

It was decided to use a guided design technique of instruction, in order to limit the time and complexity factors.

"Guided design is a relatively recent innovation in which students are first presented with a problem setting. Sequential written instructions . . . are then used to expose them to desired course material and to aid them in learning to use rational problem solving techniques. The guidance feature makes the approach best suited to elementary courses or for introductory projects in an advanced course." [Ref. 3]

Another advantage of the guided design process is that it frees the student from lecture sessions and allows the instructor to focus on critical areas. A disadvantage of



this structure is that student creativity and initiative is somewhat inhibited.

The second step was to incorporate the design methodology into a structure which would guide the student to a completed project and allow for creativity and initiative.

### III. THE SOLUTION

The design process uses closed form equations to provide a numerical analysis of helicopter theory. Where theory does not provide an exact solution, historical trends are used to approximate. The historical data that is obtained for use in the design process should reflect the type of helicopter that is to be designed. Specifications and mission requirements must be provided by the instructor.

The design process was incorporated into a design manual (Appendix A) for student use. Included in Appendix B are sample handouts which provide the specifications and historical data necessary to use this manual. These handouts limit the design to a single rotor, utility helicopter. The difficulty of the design process can be adjusted by varying the amount of information provided or the type of helicopter to be designed can be changed through modification of these handouts. Additionally, the handouts provide a means by which the course can be improved by continued research without the requirement of having to rewrite or modify the manual.

The student is expected to have a basic understanding of the theory of helicopter performance. Many of the equations used in the manual were taken from Helicopter Performance

[Ref. 4]. Those areas of theory that are not included in Reference 4 were covered in greater detail in the manual.

During the design process the student is required to make decisions and optimize several factors. It was decided to require the student to write several brief discussions at these steps in the design process so that the instructor can follow the student's reasoning in the critical decisions. These discussions allow the student an opportunity to exhibit initiative and creativity.

Several hand-held calculator programs [Ref. 5] and a computer plotting routine [Ref. 6] have been developed for use in the design process. These programs are mentioned in the design manual where they are applicable. They have been included to reduce the tedium of several repetitive calculations.

An introduction to cost estimating relationships have been included to broaden the student's perspective of real world considerations and to allow the use of some managerial skills.

#### IV. RESULTS

The design process has been included in a design manual (See Appendix A) which is suitable for use in a helicopter design course.

An example design is included in Appendix C. The specifications and historical data used in this example are included in Appendix B. The specifications for two additional designs are also contained in Appendix B (See HD-1).

## V. CONCLUSIONS AND RECOMMENDATIONS

The simplification of the field of helicopter engineering into the design process presented in Appendix A necessitated that some areas be slighted. Several major areas are either covered only superficially or omitted due to the time constraint that the student would face in a one quarter design course. Typical of these areas are airframe design, structural designs, cost, tail rotor and vertical fin effects, and power required to overcome retreating blade stall effects. Other areas which were included must use estimations or historical trends to reach a design due to a lack of an adequate theory to predict real world data. The weight estimating relationships and profile drag computations are examples. Future research in these areas could be included in the design manual or the handouts.

Many of the decisions that the student is required to make using the design process are based on a numerical analysis of the various factors. In the real world, many other factors may be present which would necessitate a different decision, so that in this sense the process is not a valid representation. However, the design process as found in the design manual should be useful as an educational tool.

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APPENDIX A

Helicopter Design Manual

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June 1983

# ABSTRACT

This design manual is to be used for the conceptual design of a single main rotor, utility helicopter. The design method utilizes closed form formulas and approximations from historical data. This manual was written for use in AE4306--Helicopter Design.



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## Chapter 1

### INTRODUCTION

#### 1.1 DESIGN OBJECTIVE

The goal of the design process is to optimize the mission effectiveness of the design item. There are two groups of factors which determine an item's mission effectiveness.

##### A) Operational factors.

1. Mission readiness which is a measure of the degree to which an item is operable at the start of a randomly selected mission. Mission readiness is measured by an item's availability, reliability, and maintainability.
  - a) Availability is a function of the mean time between maintenance actions and the maintenance down time.
  - b) Reliability is a function of an item's failure rate.
  - c) Maintainability is measured by the item's mean time to repair.
2. Survivability which is a measure of the item's ability to withstand a hostile man-made environment and still be mission ready.
3. Overall performance which is a measure of how well an item performs its designated mission.

B) Economic factors. The principle economic factor is cost, which is a measure in dollars of the amount required to design, produce, test, and operate an item during its life-cycle.

One cost concept generally used to select among competing designs is life-cycle cost, which is the sum of all expenditures required from design conception to operational phase-out. The life-cycle cost is comprised of research and development costs, initial investment, annual operating costs, annual maintenance costs, and salvage value. There are two major difficulties in using the life-cycle cost method of economic analysis. First, the task of assembling sufficient historical data on similar systems to create meaningful cost estimating relationships is quite large. Second, the various parties involved in the procurement process have a different perspective from that of the designer. For example, a manufacturer may be more concerned with an item's initial cost than with the user's maintenance costs.

Any economic analysis of a design process should contain the following elements:

- A) a statement of the design objective and the effectiveness measures which will be used to determine if the objective is reached,
- B) specification of the design choices (alternatives),
- C) costs associated with each choice,

D) a set of relationships (a model) that relates each choice to the effectiveness measures, and

E) the criteria or criterion which will be used to select one of the design alternatives.

There are two major sources of uncertainty in any cost analysis: inadequate or inaccurate specification of the system being analyzed, and statistical inaccuracies in the cost estimating relationships.

## 1.2 DESIGN PHASES

The design process can be divided into five basic phases. The first phase is a study of the historical data and the trends in the helicopter design. Second, a conceptual study is conducted using "rules of thumb" and experience to develop simple layouts. Third, preliminary designs are drawn which include volumetric sizing, airframe lines, mechanisms, and structural concepts. Fourth, the design enters a "proposal status" in which detailed subsystems are developed, structural sizing is refined, and mockups are constructed. Fifth, the final details are completed.

Table 1 shows the design phases in outline form.

TABLE 1

### The Phases of the Design Process

1. Trend study
  - (a) Provide direction for further study
  - (b) Obtain quick look answers



2. Conceptual study
  - (a) Compare configuration
  - (b) Estimate size and cost
  - (c) Establish feasibility
  - (d) Recommend follow-on
3. Preliminary design
  - (a) Insure design practicality
  - (b) Develop structural concepts
  - (c) Develop concepts for mechanisms
  - (d) Expand data base
4. "Proposal status" design
  - (a) Increase detail of structure, weight, etc.
  - (b) Increased confidence by risk reduction
  - (c) Support proposal commitment
5. Details

### 1.3 PROCEDURE

A design sequence has been delineated in a step-by-step manner in the subsequent chapters of this manual. Each step is numbered sequentially within each chapter so that ready reference can be made to any step, when necessary, in other chapters. Where calculations are required in the design sequence used in this manual, the necessary equations have been included for the convenience of the user.

The specifications and historical data required for use in the design sequence of this manual are listed in Appendix A-1. This data is in the form of handouts which should be obtained from the instructor. The symbology used in this manual has been defined in Appendix A-2.

#### 1.4 APPLICABILITY

Because this manual has been written for use in the Helicopter Design Course (AE4306) at the Naval Postgraduate School, it presumes a final report will be prepared and provides sample formats and guidance for the preparation of that report (see Step 9.1).

#### 1.5 ASSUMPTIONS

There have been several assumptions made in this manual in order to limit the scope of the design process which the student would have to undertake. These assumptions are mentioned as they occur in the design process.

Reference 1 was used as the principle source for the theories of helicopter performance. Several assumptions are made in order to quantify the theory. The assumptions made in Reference 1 are not listed in this manual.

## Chapter 2

### MAIN ROTOR DESIGN

#### 2.1 MAKE A ROUGH ESTIMATE OF THE MANUFACTURER'S EMPTY WEIGHT.

Use the specification maximum gross weight from HD-1 and the graph of historical weight ratios (HD-2) to estimate the manufacturer's empty weight.

#### 2.3 MAKE A ROUGH ESTIMATE OF GROSS WEIGHT.

Begin with an estimated gross weight which is 80% of the specification value of gross weight.

#### 2.4 CALCULATE THE MAXIMUM TIP VELOCITY.

The maximum tip velocity should be calculated at the 90 degree point at hover, standard sea level. At hover the main rotor tip Mach number should not exceed 0.65. Calculate the maximum main rotor tip velocity using Equations 1 and 2.

$$V_{\text{max tip-mr}} = M_{\text{max tip-mr}} * a \quad (1)$$

$$a = \sqrt{\gamma * g * R * T} \quad (2)$$

#### 2.4 DETERMINE THE ROTOR RADIUS.

The selection of a main rotor radius will affect the disk loading of the blade. The disk loading is a function of gross weight and main rotor radius. Historically, the value of disk loading increases with increasing gross weight (HD-3

shows this trend). Use Equation 3 to calculate a rotor radius which will optimize disk loading for the gross weight estimate. Note that the maximum allowable radius is given as a specification in HD-1.

$$R_{<mr>} = \sqrt{W_{<gross>} / (DL * \pi)} \quad (3)$$

## 2.5 DETERMINE A FIRST-CUT ROTATIONAL VELOCITY.

The maximum rotational velocity can be determined since the maximum tip velocity and the design rotor radius are known. Use Equation 4 to calculate the first-cut rotational velocity.

$$\Omega_{<mr-max>} = V_{<max\ tip-mr>} / R_{<mr>} \quad (4)$$

## 2.6 MAKE A FIRST-CUT DETERMINATION OF THRUST COEFFICIENT.

The first-cut value of thrust coefficient should be determined at the specification density altitude using Equation 5. The tip velocity is found using Equation 6 with the main rotor radius found in Step 2.4 and the rotational velocity found in Step 2.5. Thrust should be set equal to the value of gross weight found in Step 2.2.

$$C_{<thrust-mr>} = T_{<mr>} / [A_{<mr>} * \rho * V_{<tip-mr>}^2] \quad (5)$$

$$V_{<tip-mr>} = \Omega_{<mr>} * R_{<mr>} \quad (6)$$

## 2.7 DETERMINE THE BLADE SOLIDITY.

Equation 7 can be used to find the maximum advance ratio. The tip velocity was determined in Step 2.6 and the maximum forward velocity is given as a specification in HD-1. Once

the maximum advance ratio has been determined, HD-4 can be used to determine the maximum blade loading. With this maximum blade loading value and the main rotor thrust coefficient from Step 2.6, calculate the solidity using Equation 8.

$$\mu_{\langle \text{max-mr} \rangle} = V_{\langle \text{max fwd} \rangle} / V_{\langle \text{tip-mr} \rangle} \quad (7)$$

$$\sigma_{\langle \text{mr} \rangle} = C_{\langle \text{thrust-mr} \rangle} / BL \quad (8)$$

## 2.8 DETERMINE THE NUMBER OF MAIN ROTOR BLADES TO BE USED.

The number of rotor blades to be used is a function of rotor radius (as it affects solidity), vibration, and weight.

For a given solidity, more blades would be required if the radius and chord are to be kept small.

The vibration of the main rotor is a factor in the determination of the number of blades to be used. The airframe can be expected to vibrate at the rotor harmonic frequency and at integer multiples of the rotor frequency. This integer multiple corresponds to the number of blades. A vibration with a frequency of 1/REV (4 Hz) may occur which could be caused by rotor unbalance, a blade out of track, differences between blades, or a combination of these factors. A vibration will also occur at the blade passage frequency ( $b_{\langle \text{mr} \rangle} / \text{REV}$ ). Vibrations of the rotor blades induce loads which are summed at the hub and passed to the airframe. A perfect rotor with all blades identical will

act as a filter, so that only the  $(b_{mr} / REV)$  and multiples of  $(b_{mr} / REV)$  are passed to the airframe [Ref. 2]. Therefore, the effect of main rotor vibration on the airframe can be reduced by increasing the number of blades.

Note that the positive effects of reduced vibration and smaller blades are offset by an increase in weight and hub complexity. Select the number of blades for design and include a brief analysis of your decision in the final report.

## 2.9 DETERMINE THE CHORD AND THE ASPECT RATIO.

The chord can be calculated from Equation 9, since the solidity (Step 2.7), the number of blades (Step 2.8), and the rotor radius (Step 2.4) are known.

For a helicopter rotor, the aspect ratio is defined as the radius divided by the chord. Historically, the main rotor aspect ratio has been between 15 to 20. The aspect ratio can be found using Equation 10 and the radius found in Step 2.4. Adjust the value of rotational velocity (Step 2.5) as necessary, in order to obtain an aspect ratio of 15 to 20. If the rotational velocity is reduced, then Steps 2.6 and 2.7 must be recalculated.

$$c_{mr} = (\sigma_{mr} * \pi * R_{mr}) / b_{mr} \quad (9)$$

$$AR_{mr} = R_{mr} / c_{mr} \quad (10)$$

## 2.10 DETERMINE THE AVERAGE LIFT COEFFICIENT.

The average lift coefficient is a function of thrust coefficient and solidity as shown in Equation 11.

$$C_{\text{lift}} = ( 6 * C_{\text{thrust-mr}} ) / \sigma_{\text{mr}} \quad (11)$$

## 2.11 CHOOSE AN AIRFOIL SECTION FOR THE MAIN ROTOR BLADES.

The selection criteria for an airfoil are [Ref. 3]:

- A) high stall angle of attack to avoid stall on the retreating side,
- B) high lift curve slope to avoid operation at high angles of attack,
- C) high maximum lift coefficient to provide the necessary lift,
- D) high drag divergence Mach number to avoid compressibility effects on the advancing side,
- E) low drag at combinations of angles of attack and Mach numbers representing conditions at hover and cruise, and
- F) low pitching moments to avoid high control loads and excessive twisting of the blades.

Historically, the NACA 0012 airfoil has been used most often. The primary sources for making your selection should be HD-5 and Theory of Wing Sections [Ref. 4]. Choose one of the airfoils from HD-5 or select one from Reference 4. Your final report should include a brief discussion of the reasons for your selection.

2.12 DETERMINE AVERAGE LIFT CURVE SLOPE AND AVERAGE  
PROFILE DRAG COEFFICIENT.

If you selected one of the blades from HD-5, then use HD-5 to determine the profile drag coefficient and the lift curve slope ( $C_{L,\alpha}$ ). If another airfoil was chosen, then use Reference 4 to determine the profile drag coefficient and the lift curve slope. DATCOM [Ref. 5] can also be used to find the lift curve slope.



## Chapter 3

### PRELIMINARY POWER CALCULATIONS

#### 3.1 MAKE FIRST ESTIMATE OF THE POWER REQUIRED TO HOVER.

The total power required to hover out-of-ground effect at standard sea level is the induced power (out-of-ground effect with tiploss) added to the profile power. Compute the power required to hover out-of-ground effect at standard sea level using Equations 12 through 15. An HP-41CV program entitled HOVER [Ref. 6] can be also used to compute these values

$$B_{<mr>} = 1 - [ \sqrt{2 * C_{<thrust-mr>}} / b_{<mr>} ] \quad (12)$$

$$P_{i<mr-TL>} = (1/B_{<mr>}) * [ T_{<mr>}^{1.5} / \sqrt{2 * \rho * A_{<mr>}} ] \quad (13)$$

$$P_{o<mr>} = 0.125 * \sigma_{<mr>} * C_{do<mr>} * \rho * A_{<mr>} * V_{<Tip-mr>}^3 \quad (14)$$

$$P_{T<mr-hover>} = P_{i<mr-TL>} + P_{o<mr>} \quad (15)$$

#### 3.2 MAKE SECOND GROSS WEIGHT ESTIMATE.

Gross weight, empty weight, rotor disc area, solidity, and total power to hover (OGE, SSL) are variables in the weight estimation formulas. Preliminary values have been found for these variables in Steps 2.1, 2.2, 2.4, 2.7, and 3.1. Using HD-6, refine the first gross weight estimate used in the main rotor design process. To estimate the fuel weight required in HD-6, use the fuel capacity of a comparably sized helicopter in HD-7. The useful load is

defined as the internal load capacity plus the crew weight allowance. Use the format in Figure 1 for your report.

WEIGHT ESTIMATION TABLE			
	----- ITERATION -----		
	FIRST	SECOND	THIRD
SOLIDITY:			
RADIUS:			
HOVER POWER:			
ROTOR - BLADES:			
HUB/HINGES:			
TOTAL:			
PROPULSION:			
FUSELAGE:			
FLIGHT CONTROLS:			
ELECTRICAL:			
FIXED EQUIPMENT:			
EMPTY:			
FUEL:			
USEFUL LOAD:			
GROSS:			

Figure 1: Weight Estimation Format

### 3.3 ESTABLISH FIGURE OF MERIT AT APPROXIMATELY 0.75.

Historically, a Figure of Merit of 0.75 is considered average [Ref. 2]. If the induced power is between 70 to 80% of the total power, the Figure of Merit will be approximately 0.75. Recalculate the hover power (Step 3.1) using the new gross weight estimate calculated in Step 3.2. For small gross weight changes the value of disk loading may be adjusted provided it remains within the limits defined by

HD-3. If a significant gross weight change occurs, a radius change should be made in Step 2.3 to keep disk loading within limits. If the main rotor radius is changed, then Steps 2.5, 2.6, 2.7, and 2.10 must be recalculated. Verify that the new value of induced power is between 70 to 80% of the new total power. If not, the geometric parameters must be changed to obtain that relationship. Hint: If the induced power is greater than 80% of the total power, then increase the chord length. An increase in the value of chord length increases solidity and decreases blade loading. The maximum value of blade loading was used in Step 2.7 and lower blade loadings are desirable. If the chord length is increased, Steps 2.7, 2.9, and 2.10 must be recalculated. If the chord length cannot be increased without violating the limits of aspect ratio, then the radius or the rotational velocity should be changed. If the induced power is less than 70% of the total power, then reduce the rotational velocity. In Step 2.5, a maximum tip velocity was used which could be lowered slightly. A reduction in the rotational velocity increases the advance ratio which decreases the blade loading (see HD-4).

#### 3.4 REFINE SECOND GROSS WEIGHT ESTIMATE.

Since the hover power (and possibly solidity or area) was adjusted, refine the gross weight using HD-6. List the new

weight estimates (third iteration) in your report as per the format of Figure 1.

**3.5 MAKE THIRD ESTIMATE OF POWER REQUIRED TO HOVER.**

Refine the power required to hover out-of-ground effect (Step 3.2) using the revised gross weight (Step 3.4).

**3.6 REPEAT THE GROSS WEIGHT ESTIMATE AND THE HOVER POWER REQUIRED ITERATION.**

Repeat the iterations (Steps 3.3, 3.4, and 3.5) until successive steps converge at less than 10% difference.

**3.7 DETERMINE THE POWER REQUIRED TO HOVER IGE, SSL.**

The profile power stays the same as for hover OGE (Step 3.1), but the induced power gets smaller. Calculate the hover power in ground effect using Equations 15 through 17. An HP-41CV program entitled HOVER [Ref. 6] can also be used to compute these values. Use a hover height of 10 feet above ground level for your calculations.

$$P_{i<mr-TL+GE>} = (P/P_{<OGE>}) * P_{i<mr-TL>} \quad (15)$$

$$P/P_{<OGE>} = - 0.1276(h/D)^4 = 0.7080(h/D)^3 - 1.4569(h/D)^2 \quad (16) \\ + 1.3432(h/D) + 0.5147$$

$$P_{T<mr-hover>} = P_{i<mr-TL+GE>} + P_{o<mr>} \quad (17)$$

**3.8 DETERMINE THE PARASITE POWER REQUIRED IN FORWARD FLIGHT**

Equation 18 shows the relationship between the equivalent flat plate area loading and the parasite power required in forward flight. There are several methods that can be used

to determine the equivalent flat plate area in forward flight. One method is to use HD-8 and the latest estimate of gross weight to find the equivalent flat plate area loading, then to use Equation 19 to find the equivalent flat plate area in forward flight.

$$Pp<fwd> = 0.5 * \rho * V<fwd>^3 * EFPA<FF> \quad (18)$$

$$EFPA<FF> = W<gross> / Loading<EFPA> \quad (19)$$

### 3.9 DETERMINE MAIN ROTOR POWER REQUIRED AND THE MACH NUMBER OF THE ADVANCING BLADE TIP FOR FORWARD FLIGHT.

These calculations should be made at both standard sea level and at specification density altitude using Equations 20 through 26. Create tables with the velocity incremented at least every 20 knots. The cruise velocity should also be included. Two programs have been developed either of which could also be used to calculate these values: a FORTRAN program entitled Helicopter Power Computation Package [Ref. 7] for use on the IBM-3033, and a program entitled FLITE [Ref. 8] for use with the HP-41CV programmable calculator. Use the format shown in Figure 2 for your report.

$$\mu<mr> = V<fwd> / V<tip-mr> \quad (20)$$

$$Po<mr-fwd> = [1 + 4.3 * \mu<mr>^2] * Po<mr-hover> \quad (21)$$

$$Vi = \sqrt{T<mr> / [2 * \rho * \pi * R<mr>^2]} \quad (22)$$

$$Vi<T>^4 + 2 * V<vert> * Vi<T>^3 + \quad (23)$$

$$[V<fwd>^2 + V<vert>^2] * Vi<T>^2 - Vi^4 = 0$$

$$Pi<mr-TL-fwd> = (1/B<mr>) * T<mr> * Vi<T> \quad (24)$$

$$P_{T<mr-fwd>} = P_{i<mr-TL-fwd>} + P_{o<mr-fwd>} + P_{p<fwd>} \quad (25)$$

$$M_{<tip-mr>} = (V_{<fwd>} + V_{<tip-mr>}) / \sqrt{\gamma * g * R * T} \quad (26)$$

#### MAIN ROTOR POWER PROFILE

AIRSPPEED	TIP	----- POWER -----			TOTAL
(KNOTS)	MACH	INDUCED	PROFILE	PARASITE	(SHP)
		(SHP)	(SHP)	(SHP)	

Figure 2: Main Rotor Power Profile Format

## Chapter 4

### TAIL ROTOR DESIGN

#### 4.1 MAKE PRELIMINARY DETERMINATIONS OF TAIL ROTOR GEOMETRY.

Use HD-9 in your determination of radius, rotational velocity, drag coefficient, and the number of blades. The length of the fuselage from the center-of-gravity to the tail rotor hub should be calculated using Equation 27. The chord can be determined by Equation 28 and the use of a value of tail rotor aspect ratio within the historical range of 4.5 to 8.0. Use Equation 29 to calculate the tail rotor solidity.

$$L_{tr} = R_{mr} + R_{tr} + 0.5 \text{ ft} \quad (27)$$

$$c_{tr} = R_{tr} / AR_{tr} \quad (28)$$

$$\sigma_{tr} = b_{tr} * c_{tr} / (\pi * R_{tr}) \quad (29)$$

#### 4.2 DETERMINE TAIL ROTOR POWER REQUIRED AT HOVER OGE, SSL.

The total power of the tail rotor required to hover out-of-ground effect at standard sea level is the induced power, out-of-ground effect with tiploss, added to the profile power. Use Equations 30 through 36 to determine the tail rotor hover power. The HP-41CV programs entitled TR or Hover [Ref. 6] can be used to compute these values.

$$T_{tr} = P_{Tmr-hover} / (\Omega_{mr} * L_{tr}) \quad (30)$$

$$C_{thrust-tr} = T_{tr} / [A_{tr} * \rho * V_{tip-tr}^2] \quad (31)$$

$$V_{tip-tr} = \Omega_{tr} * R_{tr} \quad (32)$$

$$B_{tr} = 1 - [ \sqrt{2 * C_{thrust-tr}} / b_{tr} ] \quad (33)$$

$$P_{i$$

$$P_{o$$

$$P_{T$$

#### 4.3 DETERMINE TAIL ROTOR POWER REQUIRED AND TIP MACH EFFECTS FOR FORWARD FLIGHT.

These calculations should be made at both standard sea level and at specification density altitude using Equations 37 through 46. Create tables with the velocity incremented at least every 20 knots. The cruise velocity should also be included. The computer program [Ref. 7] and the HP-41CV program [Ref. 8] mentioned in Step 3.9 can also be used to calculate these values. Use the format shown in Figure 3 for your report.

$$L_{tr} = V_{fwd} / V_{tip-tr} \quad (37)$$

$$P_{o$$

$$T_{tr} = P_{T<mr-fwd} / (\Omega_{mr} * L_{tr}) \quad (39)$$

$$C_{thrust-tr} = T_{tr} / [A_{tr} * \rho * V_{tip-tr}^2] \quad (40)$$

$$B_{tr} = 1 - [ \sqrt{2 * C_{thrust-tr}} / b_{tr} ] \quad (41)$$

$$V_i = \sqrt{T_{tr} / [2 * \rho * \pi * R_{tr}^2]} \quad (42)$$

$$V_i < T >^4 + 2 * V_{vert} * V_i < T >^3 + \quad (43)$$

$$[V_{fwd}^2 + V_{vert}^2] * V_i < T >^2 - V_i^4 = 0$$

$$P_{i$$

$$P_{T<tr-fwd} = P_{i$$

$$M_{tip-tr} = (V_{fwd} + V_{tip-tr}) / \sqrt{\gamma * g * R * T} \quad (46)$$



# TAIL ROTOR POWER PROFILE

AIRSPEED (KNOTS)	TIP MACH	----- INDUCED (SHP)	POWER PROFILE (SHP)	----- TOTAL (SHP)
---------------------	-------------	---------------------------	---------------------------	-------------------------

Figure 3: Tail Rotor Power Profile Format

## Chapter 5

### POWER CALCULATION REFINEMENTS

#### 5.1 DETERMINE TOTAL POWER REQUIRED FOR HOVER AND FORWARD FLIGHT

These calculations should be made for flight out-of-ground effect at both standard sea level at the specification density altitude using Equations 47 and 48. Create tables with the velocity incremented at least every 20 knots. The cruise velocity should also be included. The FORTRAN computer program for the IBM-3033 and the HP-41CV program mentioned in Step 3.9 can be used to calculate these values. The computation package will also provide a graph of the induced, profile, parasite, and the total power curves. The total power curve will be required in Steps 7.4 and 7.5. Use the format shown in Figure 4 for your report.

$$PT<acft-hover> = PT<mr-hover> + PT<tr-hover> \quad (47)$$

$$PT<acft-fwd> = PT<mr-fwd> + PT<tr-fwd> \quad (48)$$

ROTOR POWER PROFILE				
AIRSPEED (KNOTS)	----- POWER -----			
	INDUCED (SHP)	PROFILE (SHP)	PARASITE (SHP)	TOTAL (SHP)
Figure 4: Total Rotor Power Profile Format				

## 5.2 DETERMINE COMPRESSIBILITY CORRECTION.

When an airfoil is operated at a Mach number of 1.0 or greater, pressure disturbances caused by the airfoil cannot propagate forward and shock waves will form ahead of the airfoil. The noise level produced by the rotor is greatly increased when the shock waves form [Ref. 9]. Weak oblique shock waves can form at local points on the airfoil even before the free stream reaches Mach 1.0. The free stream Mach number at which any local Mach number reaches 1.0 is called the critical Mach number. Above this critical Mach number, drag begins to increase. Therefore, additional power is required to overcome the effects of compressibility on the performance of a helicopter rotor. Use Equations 49 through 51 to calculate the compressibility correction, at both standard sea level and the specification density altitude. Note that Equation 50 has been adjusted by 0.06 in order to agree with experimental data for a NACA 0012 airfoil [Ref. 10]. Assume this equation applies for any airfoil. The compressibility correction should be determined only at the highest Mach number (usually at the maximum velocity). The values of critical Mach number can be found in HD-10. Use the lowest value of critical Mach number (at the highest angle of attack) for your calculations.

$$P_{\text{Comp}} = \rho * A_{\text{mr}} * V_{\text{tip-mr}}^3 * \sigma_{\text{mr}} * [0.012 * M_d + 0.10 * M_d^3] \quad (49)$$

$$M_d = M_{\langle \text{tip-mr} \rangle} - M_{\langle \text{Crit} \rangle} - 0.06 \quad (50)$$

$$M_{\langle \text{tip-mr} \rangle} = (V_{\langle \text{tip-mr} \rangle} + V_{\langle \text{fwd} \rangle}) / \sqrt{\gamma * g * R * T} \quad (51)$$

### 5.3 DETERMINE THE REQUIRED RSHP AT MAXIMUM VELOCITY.

Add the appropriate compressibility corrections from Step 5.2 to the values of total power at maximum velocity determined in Step 5.1 (for both standard sea level and the specification density altitude). After adding the compressibility corrections, the higher of the two values should be used as the required rotor shaft horsepower.

### 5.4 DETERMINE THE REQUIRED RSHP FOR HOVER (IGE) AT THE SPECIFIED CEILING.

Use the gross weight that has been used since Step 3.6 and note that the density will be different at the specification hover ceiling. The specification hover ceiling can be found in HD-1. Use Equations 52 through 59 to calculate the main rotor hover power and the equations provided in Step 4.2 to calculate the tail rotor hover power at the specification hover ceiling. Assume a hover height of 10 feet above ground level. The HP-41CV programs entitled HOVER and TR [Ref. 6] can be used to compute these values. The required RSHP for hover is the aircraft total hover power.

$$C_{\langle \text{thrust-mr} \rangle} = T_{\langle \text{mr} \rangle} / [A_{\langle \text{mr} \rangle} * \rho * V_{\langle \text{tip-mr} \rangle}^2] \quad (52)$$

$$B_{\langle \text{mr} \rangle} = 1 - [\sqrt{2 * C_{\langle \text{thrust-mr} \rangle}} / b_{\langle \text{mr} \rangle}] \quad (53)$$

$$P_{i\langle \text{mr-TL} \rangle} = (1/B_{\langle \text{mr} \rangle}) * [T_{\langle \text{mr} \rangle}^{1.5} / \sqrt{2 * \rho * A_{\langle \text{mr} \rangle}}] \quad (54)$$

$$P_{i<mr-TL+GE>} = (P/P<OGE>) * P_{i<mr-TL>} \quad (55)$$

$$P/P<OGE> = - 0.1276(h/D)^4 + 0.7080(h/D)^3 - 1.4569(h/D)^2 + 1.3432(h/D) + 0.5147 \quad (56)$$

$$P_{o<mr>} = 0.125 * \sigma_{<mr>} * C_{do<mr>} * \rho * A_{<mr>} * V_{<tip-mr>}^3 \quad (57)$$

$$P_{T<mr-hover>} = P_{i<mr-TL-GE>} + P_{o<mr>} \quad (58)$$

$$P_{T<acft-hover>} = P_{T<mr-hover>} + P_{T<tr-hover>} \quad (59)$$

#### 5.5 DETERMINE THE TOTAL ESHP REQUIRED.

Compare the values of RSHP found in Steps 5.3 and 5.4. The greater of these two values is the maximum required RSHP, which should be used to find the required ESHP. The engine shaft horsepower is defined as the rotor shaft horsepower adjusted for transmission and accessory losses. Use Equation 60 to calculate ESHP, assuming:

- A) ten horsepower for accessories,
- B) ten percent losses in SHP for multiple engine installation, and
- C) three percent transmission losses.

$$ESHP_{<Req>} = [0.10 * RSHP_{<max>} * (n-1)] + 1.03 * RSHP_{<max>} + 10.0 \text{ hp} \quad (60)$$

## Chapter 6

### ENGINE SELECTION

#### 6.1 SELECT TYPE AND NUMBER OF ENGINES.

The number of engines to be used should be determined by considering the factors of safety, survivability, and reliability. Select the number of engines based on these considerations and briefly explain your decision in your final report. Remember that in Step 5.4 multiple engine installation was considered. If you decide to use a different number of engines, go back to Step 5.4 and recompute ESHP.

The criteria for selection of the type of engine to be used are weight, life-cycle costs, availability, reliability, maintainability, and performance.

The weight of the engine should be as small as possible.

The life-cycle costs can be computed using the data found in HD-1 and HD-11. The life-cycle costs are the summation of the research and development costs, the initial cost, the yearly operating cost (average value, adjusted for inflation), the yearly maintenance cost (average value including overhauls, adjusted for inflation), the replacement cost (if the life of the engine is less than the life of the helicopter) and the salvage value. In order to compute the life-cycle costs, the expected time of service

of the helicopter in years, the expected number of flight hours per year of the helicopter, and the expected engine life (sometimes called mean time between replacements) in hours must be known. If the engine life in hours is less than the helicopter life in hours, then the number of engine replacements must be computed. To find the number of engine replacements divide the helicopter life in hours by the engine life in hours, round up to the next integer if there is a remainder, then subtract 1.0 (the initial engine). Equation 61 should be used to compute the life-cycle costs.

The availability is a function of the mean time between maintenance actions and the maintenance down time. The engine availability can be computed using the relationship given by Equation 62 and the appropriate values in HD-11.

The reliability of the engine is a function of the mean time between failures (failure rate) and the length of the average flight in hours (see HD-1). Equations 63 and 64 can be used to compute the reliability.

The maintainability is a function of the mean time to repair but is sometimes given as a fraction of the maintenance down time in hours divided by the total flight hours. Either measure can be used for comparison.

The performance of the engine can be measured in many ways, but for this course the measure of performance will be the shaft horsepower produced by the engine. Insure the engine you select has a military ESHP greater than the

ESHP<Req> you found in Step 5.4. A computer program entitled Helicopter Engine Program [Ref. 11] has been developed that computes the shaft horsepower available at any given altitude and airspeed for various engines; however, this computation is done for a "rubber" engine and should not be used if real engine data is available.

Once values have been determined for each criteria for each competing engine, the designer must choose the engine that best optimizes the six criteria mentioned above. In order to select the "best" engine, the six criteria must be weighted according to their importance. For example, high performance and low weight may be more desirable than low life-cycle cost. In order to establish uniform selection criteria, the weighting factors for the six criteria are given in HD-11. Your final report should contain a completed chart using the format of Figure 5 and a brief discussion of the decision-making process you used in your selection of an engine type. Compute the availability, reliability, and maintainability on a per engine basis.

$$LCC = n*[RDC + IC + HL<yrs>*(YOC+YMC) + n<rpl>*(RC-SV)] \quad (61)$$

$$AVAIL = MTBMA / ( MTBMA + MDT ) \quad (62)$$

$$RELY = e^{(-\lambda<FR> * LAF<hrs>)} \quad (63)$$

$$\lambda<FR> = 1 / MTBF \quad (64)$$



## ENGINE SELECTION CRITERIA

	----- ENGINE -----					
	A	B	C	D	E	F
POWERPLANT WEIGHT:						
LIFE-CYCLE COST:						
ENGINE LIFE (HRS):						
NUMBER OF REPLACEMENTS:						
R & D COSTS:						
INITIAL COST:						
YEARLY MAINTENANCE COST:						
YEARLY OPERATING COST:						
REPLACEMENT COST:						
SALVAGE VALUE:						
AVAILABILITY (per engine):						
RELIABILITY (per engine):						
MAINTAINABILITY (per engine):						
PERFORMANCE (SHP):						

Figure 5: Engine Selection Format

### 6.2 REVISE GROSS WEIGHT AND POWER REQUIRED.

Check engine weight (including installation, oil, and transmission weights) against previous estimates of powerplant weight (called propulsion weight in HD-6). Compute a new gross weight using the engine weight. Compare the new gross weight with the value you have been using from Step 3.6. If it is not within 10% use the new gross weight and go back to Step 3.3.

### 6.3 DETERMINE FUEL FLOW RATES AT VARIOUS POWER SETTINGS.

The specific fuel consumption can be obtained from the manufacturer's data on the engine at military, normal, and cruise power settings. The fuel flow rates at the specified

power settings can then be determined on a per engine basis  
from Equation 65.

$$W_f = SFC * SHP$$

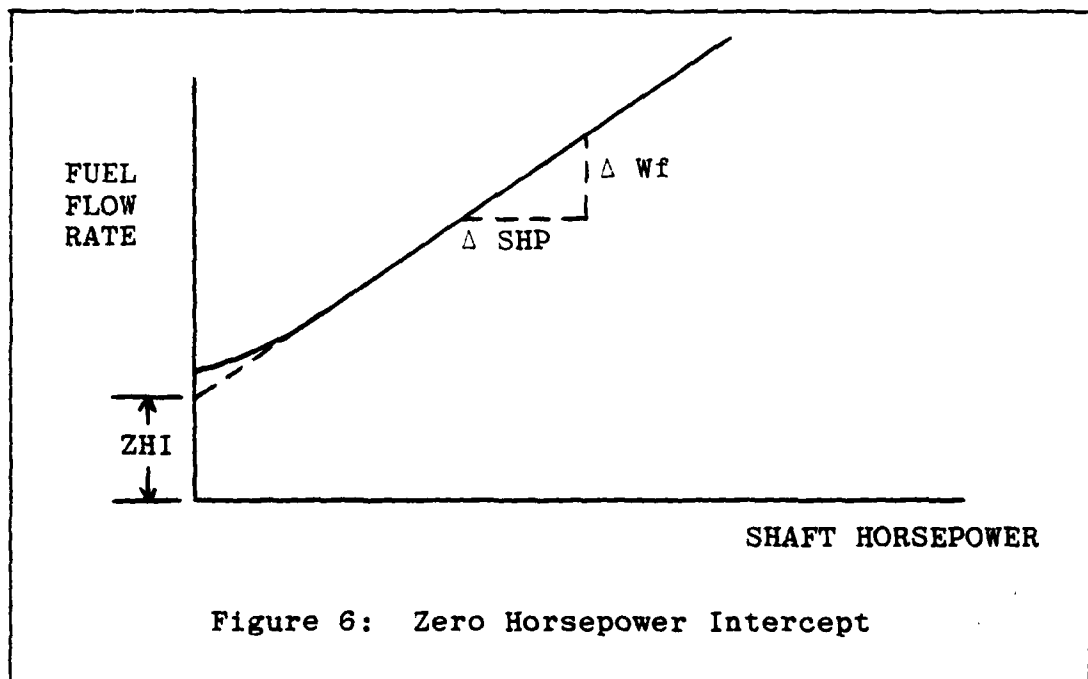
(65)

## Chapter 7

### RANGE AND ENDURANCE CALCULATIONS

#### 7.1 DETERMINE THE SLOPE OF THE FUEL FLOW VERSUS SHP LINE AND THE ZERO HORSEPOWER INTERCEPT.

These calculations should be made at standard sea level. The relationship between fuel flow rate and shaft horsepower for a turboshaft engine is fairly linear except at low values of shaft horsepower. The average slope of the fuel flow versus SHP line can be found mathematically using the values of fuel flow rate at the military, normal, and cruise power settings calculated in Step 6.3. The intercept of the fuel flow versus SHP line with the ordinate axis is the zero horsepower intercept.



$$\beta_{\text{ave}} = \Delta W_f / \Delta \text{SHP} \quad (66)$$

## 7.2 COMPUTE THE ZERO HORSEPOWER INTERCEPT AT SPECIFICATION CONDITIONS.

At standard sea level, both  $\theta$  and  $\delta$  are equal to 1.0; therefore, the value of  $\alpha$  can be determined using the zero horsepower intercept found in Step 7.1. Find  $\theta$  and  $\delta$  at the specification pressure altitude using Equations 67 and 68, then calculate the zero horsepower intercept using Equation 69. The pressure at the specification pressure altitude can be found using an ICAO standard atmospheric table. The temperature at the specification pressure altitude is given in HD-1.

$$\theta_{\text{spec}} = T_{\text{spec}} / T_{\text{SSL}} \quad (67)$$

$$\delta_{\text{spec}} = P_{\text{spec}} / P_{\text{SSL}} \quad (68)$$

$$\text{ZHI} = \alpha_{\text{ave}} * \delta * \sqrt{\theta} \quad (69)$$

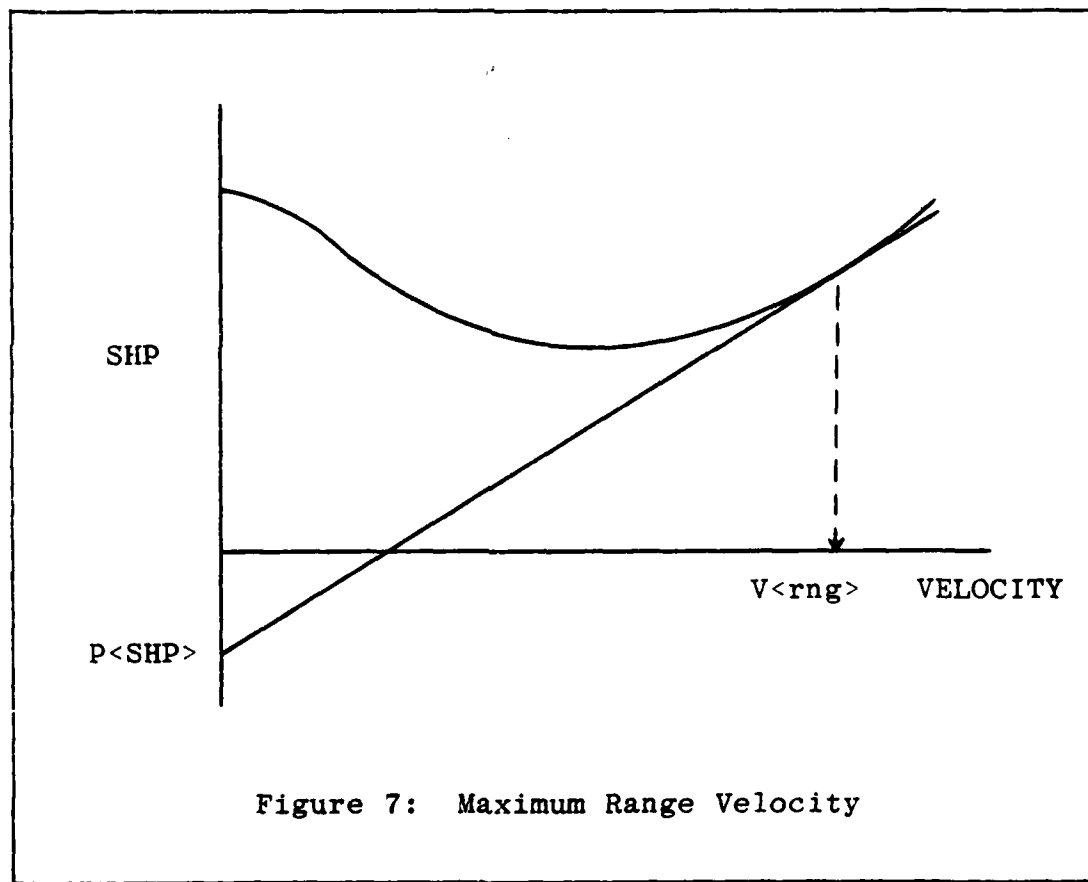
## 7.3 DETERMINE THE ZERO HORSEPOWER INCREMENT.

The value of the zero horsepower increment is based on the number of engines, the zero horsepower intercept, and the fuel flow rate per horsepower. The slope of the fuel flow versus SHP line is the same at all density altitudes. These calculations should be made at the specification density altitude. The zero horsepower increment is sometimes called the phantom SHP. Use Equation 70 to calculate the phantom SHP.

$$P_{\text{SHP}} = [n * \alpha_{\text{ave}} * \delta * \sqrt{\theta}] / \beta_{\text{ave}} \quad (70)$$

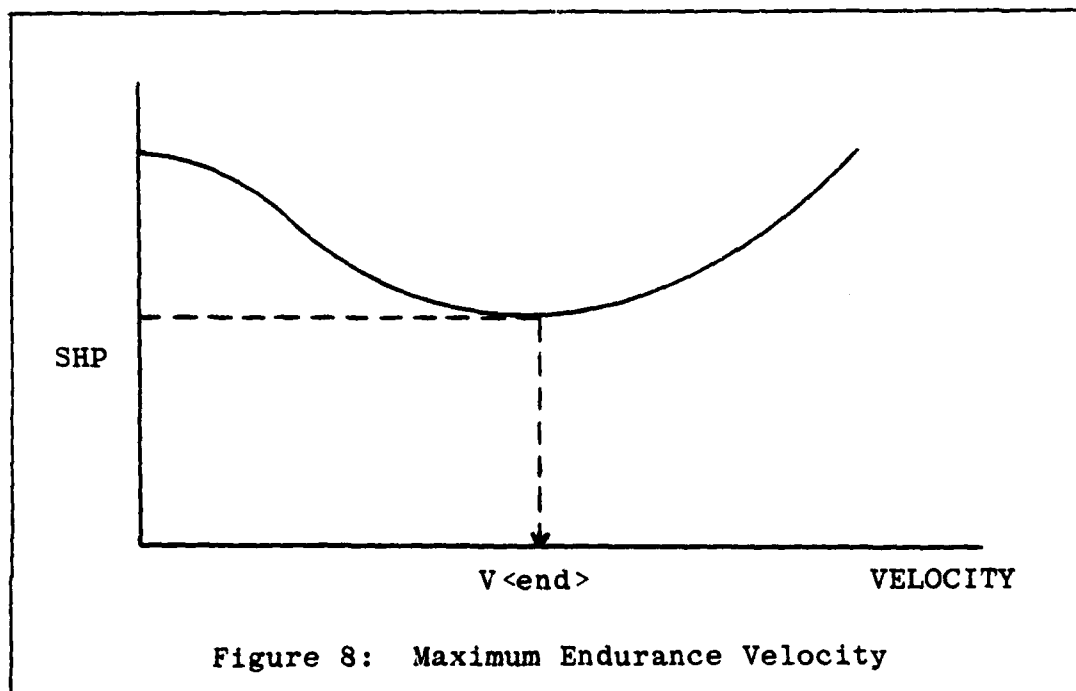
#### 7.4 DETERMINE THE MAXIMUM RANGE VELOCITY.

Use the total power versus velocity curve at the specification density altitude. Draw a line through the zero horsepower intercept and tangent to the total power versus velocity curve found in Step 5.1. Draw a second line perpendicular to the abscissa which passes through the point of tangency. The intercept of the second line with the abscissa is the maximum range velocity (see Figure 7).



7.5 DETERMINE THE MAXIMUM ENDURANCE VELOCITY AND THE REQUIRED FUEL FLOW RATE AT THIS VELOCITY.

Draw a line parallel to the abscissa on the total power versus velocity curve which is tangent to the lowest value on the curve. Use the total power versus velocity curve at the specification density altitude. Draw a second line perpendicular to the abscissa which passes through the point of tangency. The intercept of the second line with the abscissa is the maximum endurance velocity (see Figure 8). Find the RSHP at the maximum endurance velocity from the total power curve (see Step 5.1) and the phantom SHP from Step 7.3, then use Equation 71 to find the required RSHP at the maximum endurance velocity. Use Equation 72, the required RSHP, and the value of  $\beta_{ave}$  found in Step 7.1 to find the required fuel flow rate at the maximum endurance velocity.



$$RSHP<Req-max\ end> = P<SHP> + RSHP<max\ end> \quad (71)$$

$$Wf<max\ end> = RSHP<Req-max\ end> * \beta<ave> \quad (72)$$

7.6 DETERMINE THE POWER REQUIRED AT SPECIFICATION CRUISE VELOCITY AND THE REQUIRED FUEL FLOW RATE AT THIS VELOCITY.

Find the RSHP at cruise velocity from the total power curve at the specification density altitude (see Step 5.1) and add the phantom SHP found in Step 7.3. This sum (see Equation 73) is the required RSHP at cruise velocity. After computing the required RSHP, use Equation 74 to find the fuel flow rate at cruise velocity.

$$RSHP<Req-cruise> = P<SHP> + RSHP<cruise> \quad (73)$$

$$Wf<cruise> = RSHP<Req-cruise> * \beta<ave> \quad (74)$$

7.7 DETERMINE THE TOTAL FUEL REQUIREMENTS FOR SPECIFIED MAXIMUM RANGE.

The total fuel requirements are based on the following assumptions:

- A) warm-up and take-off requires three minutes of fuel at normal rated power,
- B) cruise at specification velocity,
- C) approach and landing requires three minutes of fuel at normal rated power, and
- D) reserve requires fifteen minutes at maximum endurance velocity.

The cruise velocity and the maximum range are specifications and can be obtained from HD-1. Use Equation 75 to find the fuel weight. Assume the fuel flow rates are constant.

$$FW = 0.05*n*Wf<NRP> + [(Wf<cruise>*RNG<max>)/V<cruise>] (75)$$

$$+ 0.05*n*Wf<NRP> + 0.25*Wf<max end>$$



## Chapter 8

### MISCELLANEOUS CALCULATIONS

#### 8.1 COMPUTE DESIGN GROSS AND EMPTY WEIGHT.

At this point, your power calculations have been made with an estimated gross weight from Step 3.6 which is different from the design gross weight. The design gross weight includes the actual powerplant weight from Step 6.1 and the actual fuel weight from Step 7.7. If the design gross weight is greater than the estimated gross weight, return to Step 3.3. If the design gross weight is less than the estimated gross weight, increase the fuel weight or the internal load capacity so that the design gross weight equals the estimated gross weight. Note that specifications may be exceeded if the change is advantageous. If the fuel weight is increased in this manner, the maximum range must be recomputed (See Step 7.7).

#### 8.2 DETERMINE RETREATING BLADE STALL VELOCITY.

The retreating blade stall velocity can be calculated using a technique developed in Reference 10. A program for the HP-41CV programmable calculator entitled BS [Ref. 12] has been developed utilizing this technique which can be used to calculate the retreating blade stall velocity. The program uses the geometric design parameters of the main

rotor and the aircraft's forward velocity as input, then outputs the maximum angle of attack at the 270 degree point, the amount of collective set, and the amount of cyclic that is set. The cyclic will be negative for forward flight. The program assumes that the maximum unstalled angle of attack is 12.5 degrees, that the effective dimensionless radius is 0.97 and that there is no lateral flapping. Note that the slope of the lift curve was found in Step 2.12 and that a value of the twist of the main rotor blade is required. (Assume a linear twist of -10 degrees.)

To determine the retreating blade stall velocity at standard sea level, run the program using various forward velocities until the program indicates that the retreating blade has stalled (use the greatest unstalled velocity) or until the specification maximum velocity is reached.

Equations 76 through 92 may also be used to determine the retreating blade stall velocity at standard sea level in lieu of the HP-41CV program. (The assumptions mentioned above also apply to these equations.) First, determine if the blade is stalled by comparing the angle of attack of the main rotor at the 270 degree position with the maximum angle of attack. Second, reduce the forward velocity if the maximum angle of attack (12.5 degrees) is exceeded and iterate to find the greatest unstalled forward velocity. Note that Equations 90 and 91 must be solved simultaneously for the collective and cyclic angles in radians and that

Equation 92 yields the angle of attack at the 270 degree position in degrees.

$$T<1> = 0.4705 + 0.5000 * \mu<mr>^2 \quad (76)$$

$$T<2> = 0.3042 + 0.4850 * \mu<mr>^2 \quad (77)$$

$$T<3> = 0.2213 + 0.2352 * \mu<mr>^2 \quad (78)$$

$$T<4> = 0.2352 * \mu<mr> + 0.1250 * \mu<mr>^2 \quad (79)$$

$$A<D> = 0.9409 - 0.5000 * \mu<mr>^2 \quad (80)$$

$$A<1> = [2.0000 * \mu<mr> - 0.5314 * \mu<mr>^3] / A<D> \quad (81)$$

$$A<2> = [2.5867 * \mu<mr>] / A<D> \quad (82)$$

$$A<3> = [1.9400 * \mu<mr>] / A<D> \quad (83)$$

$$A<4> = [0.9409 - 1.5000 * \mu<mr>^2] / A<D> \quad (84)$$

$$\alpha = [0.0012 * EFPA<FF> * V<fwd>^3] / W<gross> \quad (85)$$

$$w = W<gross> / [0.0149 * R<mr>^2 * V<fwd>] \quad (86)$$

$$\lambda = (V<fwd> * \alpha - w) / (\Omega<mr> * R<mr>) \quad (87)$$

$$C<thrust> = W<gross> / [0.0075 * R<mr>^4 * \Omega<mr>^2] \quad (88)$$

$$Z = (2 * C<thrust>) / (C_{L,\alpha} * \sigma<mr>) \quad (89)$$

$$0.0 = \alpha * A<1> + \theta<o> * A<2> \quad (90)$$

$$+ \theta<t> * A<3> + \theta<y> * A<4>$$

$$Z = \alpha * T<1> + \theta<o> * T<2> \quad (91)$$

$$+ \theta<t> * T<3> + \theta<y> * T<4>$$

$$\alpha<270> = [(\lambda / 1 + \mu<mr>) - \theta<y> \quad (92)$$

$$+ \theta<o> + \theta<t>] * 57.3$$

### 8.3 DETERMINE BEST RATE OF CLIMB.

The best rate of climb occurs in forward flight in the vicinity of the minimum total power on the total main rotor

power curve. The following procedure should be used in order to find the best rate of climb:

- A) select a velocity near the lowest point on the rotor total power curve (SSL),
- B) compute the rotor total power available using Equation 93 (ESHP<Avail> is the military rated power at SSL multiplied by the number of engines),
- C) guess a value of vertical velocity,
- D) solve for the thrust component of induced velocity using Equations 94 and 95,
- E) calculate the induced, profile, parasite, and climb power using Equations 96 through 100,
- F) compute main rotor total power using Equation 101, then add the tail rotor total power calculated using the equations in Step 4.3 and compare with the total rotor power available, and
- G) repeat the process until the total rotor power is equal to the total rotor power available or until the main rotor induced power equals zero.

The HP-41CV programs called FLIGHT and TR [Ref. 5] can be used to compute the main rotor power and the tail rotor power. The final value of vertical velocity is the best rate of climb.

$$PT<Avail> = (ESHP<Avail>-10.0) / (0.10*(n-1)+1.03) \quad (93)$$

$$V_i = \sqrt{T<mr> / [2 * \rho * \pi * R<mr>^2]} \quad (94)$$

$$V_i<T>^4 + 2*V<vert>*V_i<T>^3 + \quad (95)$$

$$[V<fwd>^2 + V<vert>^2]*V_i<T>^2 - V_i^4 = 0$$

$$P_p<mr> = [0.5*\rho*EFPA<VF>*V<vert>^3] \quad (96)$$

$$+ [0.5*\rho*EFPA<FF>*V<fwd>^3]$$

$$P_o<mr> = 0.125*\rho*C_{do}<mr>*C_{\sigma}<mr>*A<mr>*V<tip-mr>^3 \quad (97)$$

$$* [1 + 4.3*\mu<mr>^2]$$

$$EFPA<VF> = 2 * EFPA<FF> \quad (98)$$

$$P_c = T<mr> * V<vert> \quad (99)$$

$$P_i<mr-TL> = (1/B<mr>) * T<mr> * V_i<T> \quad (100)$$

$$P_T<mr-fwd climb> = P_o<mr> + P_i<mr-TL> + P_c + P_p<mr> \quad (101)$$

#### 8.4 COMPUTE MAXIMUM HOVER ALTITUDE (IGE).

Calculate the highest altitude at which the helicopter can hover in ground effect using the following method:

- A) assume a hover height of 10 feet above ground level,
- B) guess an altitude,
- C) compute the total power required to hover in ground effect using the equations given in Step 5.1,
- D) compare the total power required with the total power available found in Step 8.3 (assume power available remains constant with increasing altitude) and repeat the process if they are not equal.

The altitude at which the total power required equals the total power available is the maximum hover altitude in ground effect.

### 8.5 COMPUTE SERVICE CEILING.

Service ceiling is defined as the maximum altitude at which the helicopter exhibits a 100 fpm rate of climb capability at a given temperature. It is normally defined at the best rate of climb velocity using the normal engine power rating. The service ceiling may be calculated using the following method:

- A) guess an altitude,
- B) compute the total power required for climbing forward flight at best rate of climb velocity with a vertical velocity of 100 fpm using the equations in Step 8.2,
- C) compute the value of total rotor power available at the normal engine power rating using Equation 102 (assume power available remains constant with increasing altitude),
- D) compare the power available with power required and repeat the process if they are not equal.

The altitude at which the total power required equals the total power available (NRP) is the service ceiling.

$$PT<avail> = (ESHP<NRP>-10.0) / (0.10*(n-1)+1.03) \quad (102)$$

## Chapter 9

### FINAL CHECK

#### 9.1 MAKE FINAL CHECK FOR COMPLIANCE WITH SPECIFICATIONS.

Check for compliance with specifications as given in HD-1 and insure all work has been redone with your final design parameters.

Prepare your final report and check to insure it contains the following items:

- A) For each step either show your calculations or explain how you determined the required data. Summarize the results of Steps 3.2, 3.9, 4.3, 5.1, and 6.1 using the given formats.
- B) If a computer or calculator program is used, include a program listing unless the program has been referenced in this manual.
- C) Be sure to include the brief discussions required in Steps 2.8, 2.11, and 6.1
- D) Prepare a final summary using the format found in Appendix A-3.

## Appendix A-1

### LIST OF HELICOPTER DESIGN COURSE HANDOUTS

HD-1	Specifications
HD-2	Weight Ratios Chart
HD-3	Disk Loading Trend
HD-4	Blade Loading Limits
HD-5	Rotor Airfoil Data
HD-6	Weight Estimation
HD-7	Current Helicopter Data
HD-8	Trends In Equivalent Flat Plate Loading
HD-9	Tail Geometry Factors
HD-10	Critical Mach Number
HD-11	Powerplant Selection



## Appendix A-2

### LIST OF SYMBOLS

a	- Speed of sound
A<mr>	- Disk area of main rotor
A<tr>	- Disk area of tail rotor
AR	- Aspect ratio
AVAIL	- Availability
b<mr>	- Number of main rotor blades
b<tr>	- Number of tail rotor blades
B<mr>	- Main rotor tiploss factor
B<tr>	- Tail rotor tiploss factor
BL	- Blade loading
C<lift>	- Coefficient of lift
C <sub>L,α</sub>	- Slope of the lift curve
c<mr>	- Main rotor chord
C<thrust-mr>	- Main rotor coefficient of thrust
C<thrust-tr>	- Tail rotor coefficient of thrust
c<tr>	- Tail rotor chord
Cdo<mr>	- Profile drag coefficient of the main rotor
Cdo<tr>	- Profile drag coefficient of the tail rotor
D	- Main rotor diameter
DL	- Disk loading
EFPA<FF>	- Equivalent flat plate area in forward flight
EFPA<VF>	- Equivalent flat plate area in vertical flight
EL<hrs>	- Engine life in hours
ESHP	- Engine shaft horsepower
FW	- Fuel weight
g	- Gravitational constant
GE	- In ground effect
h	- Height above ground
HL<yrs>	- Helicopter life in years
HRS	- Flight hours
IC	- Initial cost
L<tr>	- Distance from center-of-gravity to tail rotor hub

LAF<hrs>	- Length of average flight in hours
Loading<EFPA>	- Equivalent flat plate area loading
M<crit>	- Critical Mach number of advancing blade at the 90 degree point
M<tip>	- Mach number at rotor tip velocity
M<tip-max>	- Mach number at maximum tip velocity
Md	- Drag divergence Mach number
MDT	- Maintenance Down Time
MTBF	- Mean Time Between Failures
MTBR	- Mean Time Between Replacements
MTBMA	- Mean Time Between Maintenance Actions
MTTR	- Mean Time to Repair
n	- Number of engines
n<rpl>	- Number of engine replacements
N	- Main rotor RPM
NRP	- Normal Rated Power
P<Comp>	- Power required to overcome compressibility effects
P<SHP>	- Phantom SHP
Pc<mr>	- Climb power of the main rotor
Pi<mr-TL>	- Induced power of the main rotor with tiploss
Pi<mr-TL-GE>	- Induced power of the main rotor with tiploss in ground effect
Po<mr>	- Profile power of the main rotor
Pp<fwd>	- Parasite power of the main rotor in forward flight
PT<mr-hover>	- Total power of main rotor at a hover
PT<tr-hover>	- Total power of the tail rotor at a hover
R	- Individual gas constant
R<max>	- Maximum allowable main rotor radius
R<min>	- Minimum allowable main rotor radius
R<mr>	- Design main rotor radius
R<tr>	- Tail rotor radius
RC	- Replacement cost
RDC	- Research and development cost
RELY	- Reliability
REV	- Revolution
RSHP	- Rotor shaft horsepower
SFC	- Specific fuel consumption
SHP	- Shaft horsepower
SV	- Salvage value
T	- Temperature
T<mr>	- Thrust due to the main rotor
T<tr>	- Thrust due to the tail rotor

V<fwd>	- Forward velocity
Vi	- Induced velocity of the main rotor
Vi<T>	- Thrust component of the induced velocity of the main rotor
V<max>	- Maximum forward velocity
V<max end>	- Maximum endurance velocity
V<max rng>	- Maximum range velocity
V<max tip>	- Maximum tip velocity
V<tip-mr>	- Design tip velocity of the main rotor
V<tip-tr>	- Design tip velocity of the tail rotor
w	- Downwash angle
W<gross>	- Gross weight
Wf	- Fuel flow rate
Wf<cruise>	- Fuel flow rate at cruise velocity
Wf<max end>	- Fuel flow rate at maximum endurance velocity
Wf<CRP>	- Fuel flow rate per engine at Cruise Rated Power
Wf<MRP>	- Fuel flow rate per engine at Military Rated Power
Wf<NRP>	- Fuel flow rate per engine at Normal Rated Power
YMC	- Yearly maintenance cost
YOC	- Yearly operating cost
$\alpha$	- Angle of attack
$\alpha<270>$	- Angle of attack of the retreating main rotor blade at the 270 degree point
$\alpha<ave>$	- Zero usable SHP fuel flow rate at standard sea level
$\beta$	- Rate of change of fuel flow with horsepower
$\beta<ave>$	- Average rate of change of fuel flow with horsepower
$\delta$	- Ratio of pressure at a specific altitude to standard sea level pressure
$\gamma$	- Ratio of specific heats
$\lambda$	- Main rotor inflow rate
$\lambda<FR>$	- Failure rate
$\mu<mr>$	- Advance ratio of the main rotor
$\mu<tr>$	- Advance ratio of the tail rotor
$\Omega<mr>$	- Main rotor rotational velocity
$\Omega<tr>$	- Tail rotor rotational velocity
$\rho$	- Density
$\sigma$	- Solidity
$\theta$	- Square of the ratio of speed of sound at an altitude to the speed of sound at SSL
$\theta<y>$	- Angular longitudinal cyclic position
$\theta<o>$	- Angular collective position
$\theta<t>$	- Main rotor blade twist

### Appendix A-3

#### SAMPLE FINAL SUMMARY FORMAT

##### Performance

Specification

Design

Crew:  
Internal Load [lbs]:  
Max Hover Altitude (IGE):  
Service Ceiling:  
Disk Loading:  
Velocity [kts] - Cruise:  
    Maximum:  
    Max Endurance:  
    Max Range:  
    Retreating Blade Stall:  
Best Rate of Climb [fpm]:  
Figure of Merit:  
Maximum Advance Ratio:  
Coefficient of Thrust:  
Coefficient of Lift:  
Engine - Number / Type:  
    SHP (Military):  
Maximum Range [nmi]:

##### Geometry

Specification

Design

Weight [lbs] - Max Gross:  
    Empty:  
Fuel Capacity [lbs] - Internal:  
Main Rotor - Airfoil Section used:  
    Chord [ft]:  
    Radius [ft]:  
    Number of Blades:  
    Drag Coefficient:  
    Rotational Velocity [rad/sec]:  
    Solidity:  
    Aspect Ratio:  
Tail Rotor - Airfoil Section used:  
    Chord [ft]:  
    Radius [ft]:  
    Number of Blades:  
    Drag Coefficient:

Rotational Velocity [rad/sec]:  
Solidity:  
Aspect Ratio:  
Equivalent Flat Plate Area [ft<sup>2</sup>] - Forward:  
Vertical:  
Distance from CG to Tail Rotor Hub [ft]:

#### Discussion

(a) Comment on deficiencies from specification requirements, including reasons, if known.

(b) If the design is better than specification requirements, indicate possible advantages, e.g., increased cargo could be carried by reducing the range to the specification limit.

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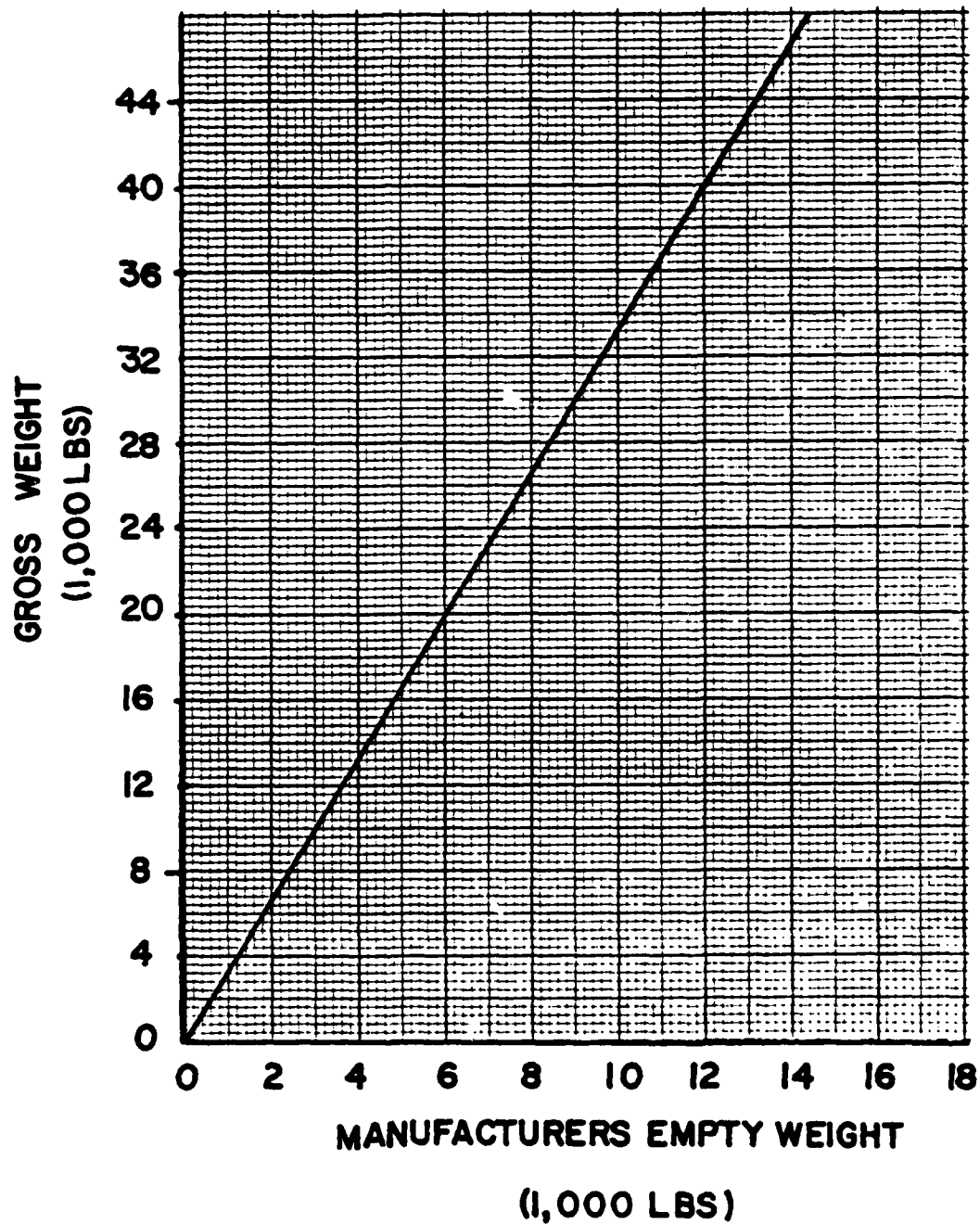
APPENDIX B  
HELICOPTER DESIGN COURSE HANDOUTS  
SPECIFICATIONS  
for a  
Single Rotor Helicopter

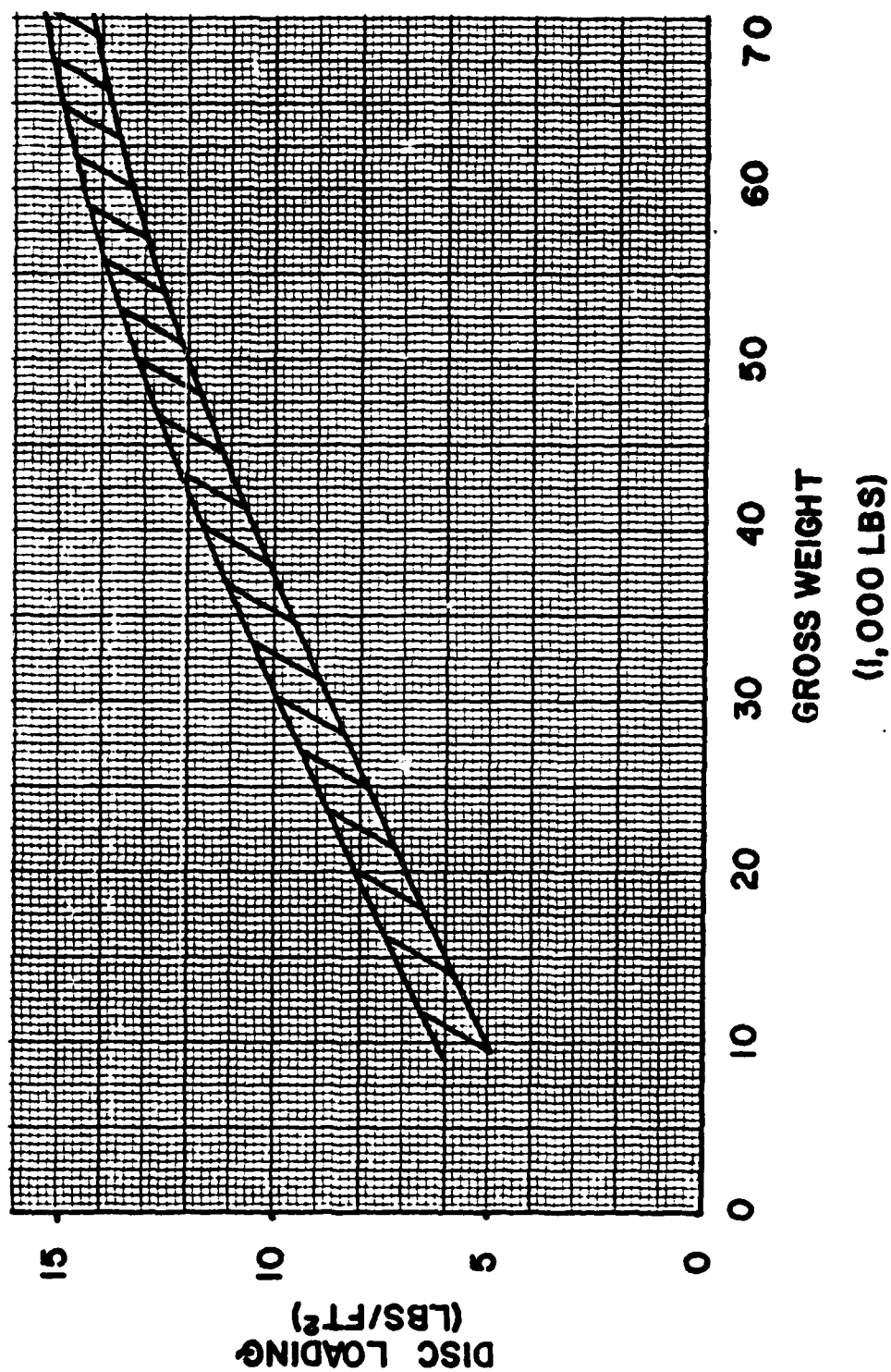
Design Group	I	II	III
Crew	2	3	3
Useful Load (lbs)	1,000	3,750	9,000
Hover IGE (ft)	11,000	12,000	13,500
Service ceiling (ft)	14,500	17,500	21,000
V max* (kts)	120	160	180
V cruise* (kts)	105	135	150
Range max (nmi)	225	250	225
Max Rate of Climb (ft/min)	1,750	2,500	1,100
Max Gross Weight# (lbs)	11,000	18,000	38,000
Max Rotor Diameter# (ft)	54	58	76
Max Fuselage Length# (ft)	50	56	68
Average flight hours per year per airframe	120	300	480
Average flight hours per flight	0.7	2.0	4.1
Average airframe service life, years	8	10	12

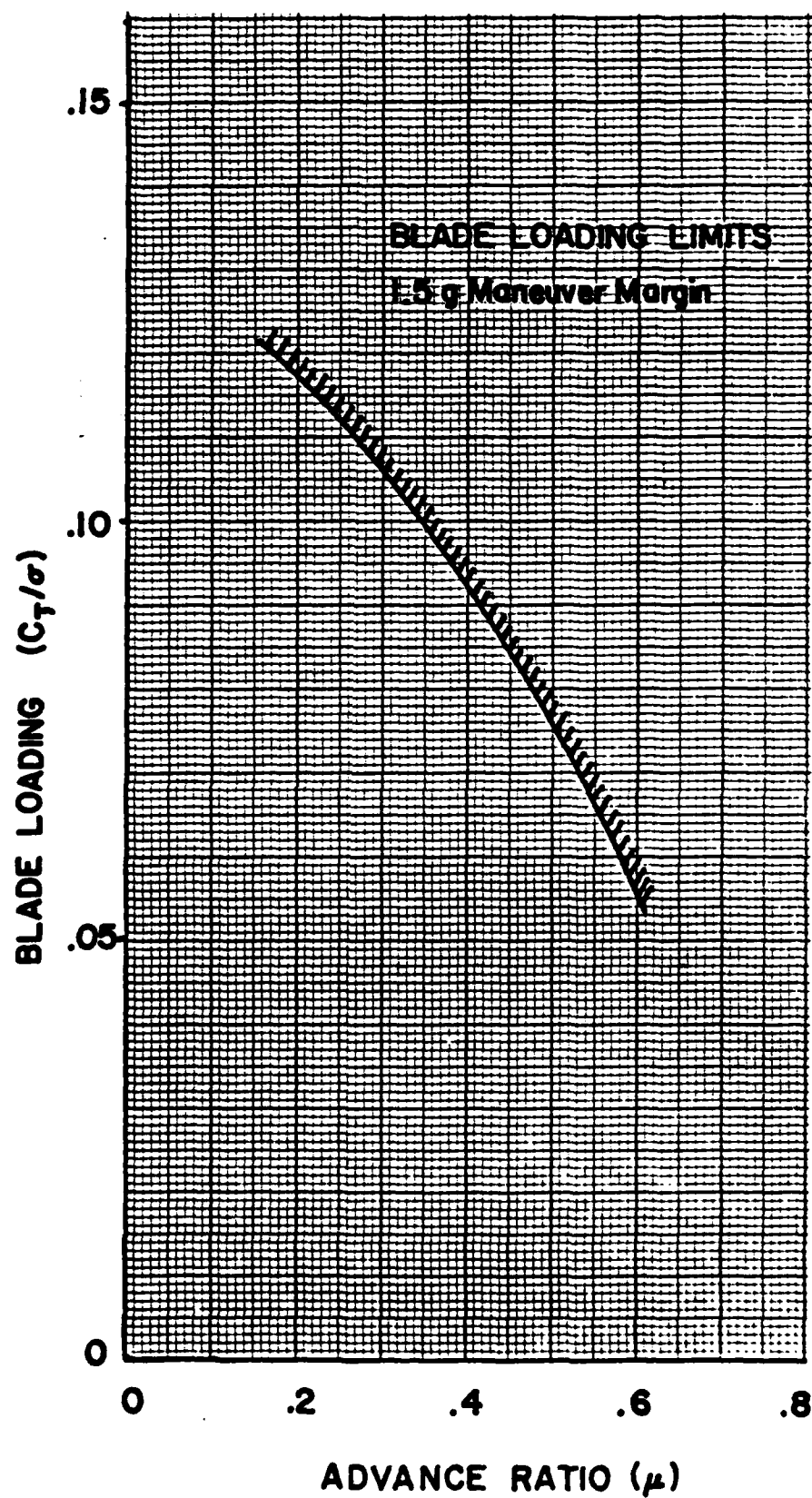
\* 4,000 pressure altitude, OAT 95 deg F

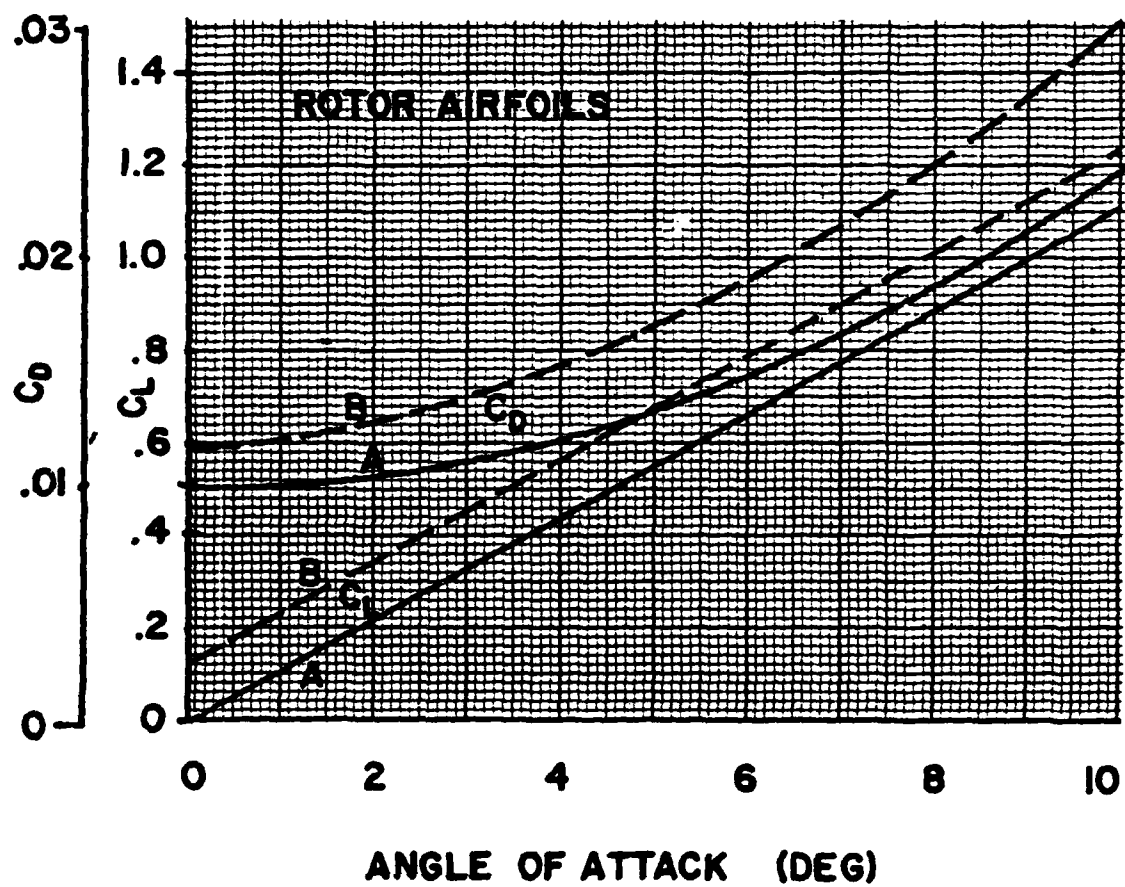
# Not-to-be-exceeded values











## HELICOPTER DESIGN COURSE HANDOUT

## Weight Estimation

## Main Rotor -

$$W<\text{blades}> = 0.06 * W<\text{empty}> * R<\text{mr}>^{0.4} * \sigma<\text{mr}>^{0.33}$$

$$W<\text{hub}> = 0.0135 * W<\text{empty}> * R<\text{mr}>^{0.42}$$

$$W<\text{mr}> = W<\text{blades}> + W<\text{hub}>$$

## Propulsion -

$$W<\text{prp}> = 1.2 * \text{SHP}<\text{req}>$$

## Fuselage -

$$W<\text{fu}> = 0.21 * W<\text{empty}>$$

## Controls -

$$W<\text{ctl}> = 0.06 * W<\text{empty}>$$

## Electrical -

$$W<\text{el}> = 0.06 * W<\text{empty}>$$

## Fixed Equipment -

$$W<\text{fe}> = 0.28 * W<\text{empty}>$$

## Empty Weight (second cut) -

$$W<\text{empty}> = W<\text{mr}> + W<\text{prp}> + W<\text{fu}> + W<\text{ctl}> + W<\text{el}> + W<\text{fe}>$$

## Gross Weight (second cut) -

$$W<\text{gross}> = W<\text{empty}> + W<\text{fuel}> + W<\text{load}>$$

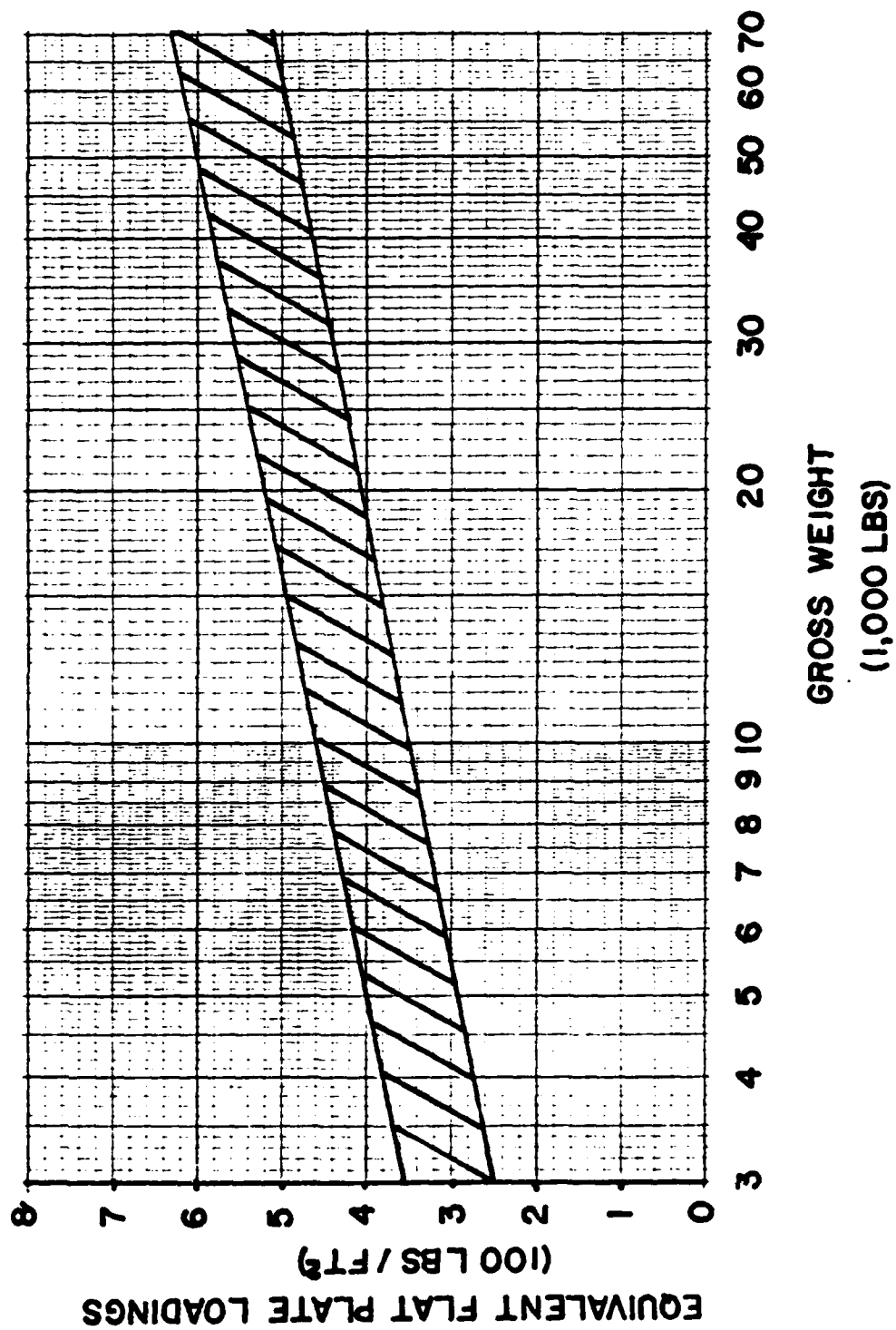
## HELICOPTER DESIGN COURSE HANDOUT

## Helicopter Parameters

	AH-64	UH-1N	OH-6A	SH-3H	S-76
Main Rotor					
Radius (ft):	24.0	24.0	26.3	31.0	22.0
Number of blades:	4	2	4	5	4
Height AGL (ft):	12.6	13.0	7.0	14.3	10.0
$\Omega$ (rad/sec):	30.3	30.8	49.2	21.3	30.7
Chord (ft):	1.75	1.95	0.57	1.52	1.29
Span (ft):	18.8	21.4	11.5	29.3	--
Twist (deg):	-9	--	-9	-8	-10
Cdo:	0.009	0.009	0.009	0.0095	0.009
DL:	8.1	5.8	4.68	6.96	6.58
Tail Rotor					
Radius (ft):	4.6	4.3	4.3	5.3	4.0
Number of blades:	4	2	2	5	4
$\Omega$ (rad/sec):	147.0	174.0	315.0	130.0	168.0
Chord (ft):	0.83	0.95	0.40	0.61	0.54
Span (ft):	3.1	3.3	1.4	4.0	3.3
Twist (deg):	-8.8	--	-8	0	-8
Cdo:	0.009	0.009	0.009	0.0105	0.015
Fuselage					
Width (ft):	3.96	9.08	4.57	7.08	7.0
Length (ft):	49.1	45.9	23.0	31.3	43.4
EFPA<FF> (sqft):	34.7	25.0	5.0	31.3	11.6
EFPA<VF> (sqft):	45.8	37.5	10.8	36.0	30.0
V<max>:	154	132	116	120	155
Range (nmi):	246	238	330	505	404
Rate of Climb (fpm):	2490	1810	500	--	425
Hover Ceiling (IGE):	14200	14200	7100	3700	62000
Hover Ceiling (OGE):	11000	10000	4200	--	28000
Tail length (ft):	29.7	25.8	15.2	36.6	26.5
Weight					
Airframe (lbs):	11010	6430	1160	13600	5600
Load (lbs)	2020	2450	960	1760	2500
Fuel (lbs):	1620	1600	400	5640	1880
Total (lbs):	14660	14200	2550	21000	10000

# Helicopter Parameters

	UH-60A	CH-54B	CH-53D	CH-53E
Main Rotor				
Radius (ft):	26.8	36.0	36.1	38.5
Number of blades:	4	6	6	7
Height AGL (ft):	11.2	17.6	15.8	16.0
$\Omega$ (rad/sec):	27.2	19.4	19.4	18.7
Chord (ft):	1.75	1.97	2.17	2.44
Span (ft):	29.3	29.8	28.9	28.6
Twist (deg):	-18	-8	-6	-13.6
Cdo:	0.008	0.0095	0.0095	0.009
DL:	8.95	10.3	10.3	15.0
Tail Rotor				
Radius (ft):	5.5	8.0	8.0	10.0
Number of blades:	4	4	4	4
$\Omega$ (rad/sec):	125.0	66.0	83.0	73.0
Chord (ft):	0.81	1.28	1.28	1.28
Span (ft):	4.25	6.45	6.45	8.53
Twist (deg):	-18	-8	-8	-8
Cdo:	0.008	0.0105	0.0095	0.0095
Fuselage				
Width (ft):	7.75	7.08	8.83	8.83
Length (ft):	50.1	70.2	67.2	99.0
EFPA<FF> (sqft):	25.7	65.0	47.3	120.0
EFPA<VF> (sqft):	30.8	99.4	90.0	63.6
V<max>:	156	110	164	146
Range (nmi):	275	200	242	400
Rate of Climb (fpm):	200	189	625	325
Hover Ceiling (IGE):	7800	6400	14000	6000
Hover Ceiling (OGE):	3900	2400	8000	1400
Tail length (ft):	31.5	44.5	44.5	48.0
Weight				
Airframe (lbs):	10680	19230	23630	24790
Load (lbs):	7270	14190	14030	15480
Fuel (lbs):	2350	8580	4340	25480
Total (lbs):	20250	42000	42000	73500





## HELICOPTER DESIGN COURSE HANDOUT

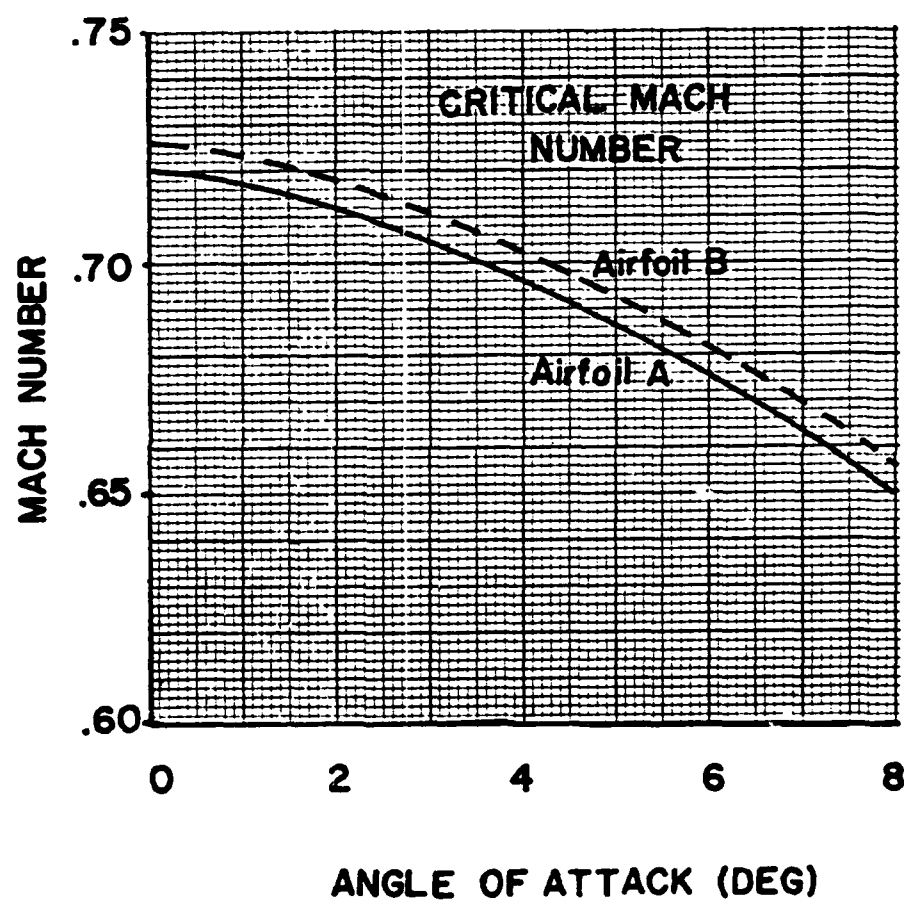
## Tail Geometry Factors

The following factors are based on analysis of several typical helicopters.

## Generalized Data

Tail radius (ft):  $R_{tr} = 1.3 * \sqrt{W_{gross} / 1000}$   
 Tail RPM:  $\Omega_{tr} = 4.5 * \text{Main rotor RPM}$   
 Tail Cdo:  $Cdo_{tr} = 0.0138 * \text{Main rotor Cdo}$

Group:	I	II	III
Number of blades:	2	3 - 4	4 - 7



## HELICOPTER DESIGN COURSE HANDOUT

## Engine Selection Parameters

The following turboshaft powerplants are available for the helicopter design. Data are presented on a per engine basis.

Engine:	A	B	C	D	E	F
Dry Weight (lbs):	136	290	338	709	723	720
SHP (SSL) Military:	317	725	1400	1800	2910	4110
Normal:	270	630	1250	1530	2450	3670
Cruise:	243	550	1125	1150	2205	3300
SFC (SSL) Military:	0.697	0.620	0.610	0.595	0.490	0.472
Normal:	0.706	0.639	0.620	0.616	0.511	0.476
Cruise:	0.725	0.658	0.640	0.661	0.525	0.484
Initial Cost:	\$90K	\$100K	\$200K	\$580K	\$640K	\$700K
Yearly Operating Cost (per hour):	\$8	\$16	\$20	\$35	\$40	\$60
Yearly Maintenance Cost (per hour):	\$25	\$50	\$100	\$125	\$160	\$220
MTBMA (hrs):	3.5	3.0	2.0	3.0	4.0	3.5
MDT (hrs):	0.7	0.6	0.5	1.3	2.0	2.6
MTBF (hrs):	185	210	205	285	280	320
MTBR (hrs):	600	750	800	800	1500	750
Replacement Cost:	----- 1.35 * Initial -----					
Salvage Cost:	----- 0.80 * Initial -----					

Note: These are all existing engines; therefore, the research and development costs have been absorbed into the initial cost.

## Engine Selection Parameters

### Engine Installed Weight:

Engine installed weight includes the dry weight of the engine(s) plus the installation fraction (exhaust system, cooling, controls, starting system and lubrication) multiplied by the dry weight. The installation fraction can be determined from the following data.

Dry weight (lbs)	Installation Fraction
100 - 300	0.29
301 - 700	0.27
701 - 1100	0.24
> 1100	0.20

### Transmission and Oil Weight:

$$W_{\text{transn-oil}} = 0.35 * \text{SHP}_{\text{avail}}$$

### Weighting Factors:

All parameters factors in the engine selection process are to be weighted equally.

## APPENDIX C

### EXAMPLE OF CONCEPTUAL HELICOPTER DESIGN

#### CONTENTS

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## Chapter 1

### INTRODUCTION

#### 1.1 PROCEDURE

This design example is for the conceptual design of a single main rotor, utility helicopter. The design method utilizes closed form formulas and symbology obtained from the design manual [Ref. 1] written for use in AE4306. It also uses approximations from historical data given in the various AE4306 handouts (see Appendix B). This example was written using the specifications contained in HD-1 for Design Group I.

#### 1.2 ASSUMPTIONS

There have been several assumptions made in the design process as per the instructions given in Reference 1. These assumptions are mentioned as they occur in the design process.

## Chapter 2

### MAIN ROTOR DESIGN

#### 2.1 MAKE A ROUGH ESTIMATE OF THE MANUFACTURER'S EMPTY WEIGHT.

The specification maximum gross weight from HD-1 is 11,000 lbs. The graph of historical weight ratios (HD-2) indicates that the manufacturer's empty weight is 60% of the gross weight. Therefore, the first estimate of manufacturer's empty weight is 6,600 lbs.

#### 2.2 MAKE A ROUGH ESTIMATE OF GROSS WEIGHT.

An estimated gross weight which is 80% of the specification value of gross weight should be used. The specification value of gross weight from HD-1 is 11,000 lbs.

$$W_{\text{gross}} = 0.8 * 11,000. = 8,800. \text{ lbs}$$

#### 2.3 CALCULATE THE MAXIMUM TIP VELOCITY.

At a hover, the main rotor tip Mach number should not exceed 0.65; therefore,  $M_{\text{max tip-mr}} = 0.65$ . At standard sea level the value of the ratio of specific heats is 1.4, the local gravitational constant is 32.2 ft/sec<sup>2</sup>, the gas constant is 53.3 and the temperature is 518.688 degrees R.

$$a = \sqrt{\gamma * g * R * T}$$

$$= \sqrt{1.4 * 32.2 * 53.3 * 518.688} = 1116.3713 \text{ ft/sec}$$

$$V_{\text{max tip-mr}} = M_{\text{max tip-mr}} * a$$

$$= 0.65 * 1116.3713 = 725.6413 \text{ ft/sec}$$

#### 2.4 DETERMINE THE ROTOR RADIUS.

Using HD-3 a disk loading value of 6.0 was chosen. The maximum value of main rotor radius given in HD-1 is 27 ft.

$$\begin{aligned} R_{\langle mr \rangle} &= \sqrt{W_{\langle gross \rangle} / (DL * \pi)} \\ &= \sqrt{8800.0 / (6.0 * 3.1416)} = 21.6068 \text{ ft} \end{aligned}$$

#### 2.5 DETERMINE A FIRST-CUT ROTATIONAL VELOCITY.

The maximum tip velocity is 725.6413 ft/sec from Step 2.3. The specification maximum forward velocity is 202.6667 ft/sec from HD-1. The design rotor radius is 21.6068 ft from Step 2.4.

$$\begin{aligned} \Omega_{\langle mr-max \rangle} &= V_{\langle max \text{ tip-mr} \rangle} / R_{\langle mr \rangle} \\ &= 725.6413 / 21.6068 = 33.5839 \text{ rad/sec} \end{aligned}$$

#### 2.6 MAKE A FIRST-CUT DETERMINATION OF THRUST COEFFICIENT.

The specification density altitude is 7122.13 ft from HD-1, at which the density is 0.0019196 slugs/ft. The rotational velocity is 33.5839 rad/sec from Step 2.5. Thrust equals the value of gross weight found in Step 2.2. Using the main rotor radius found in Step 2.4, the main rotor area is 1466.6645 ft<sup>2</sup>.

$$\begin{aligned} V_{\langle tip-mr \rangle} &= \Omega_{\langle mr \rangle} * R_{\langle mr \rangle} \\ &= 33.5839 * 21.6068 = 725.6413 \text{ ft/sec} \\ C_{\langle thrust-mr \rangle} &= T_{\langle mr \rangle} / [A_{\langle mr \rangle} * \rho * V_{\langle tip-mr \rangle}^2] \\ &= 8800.00 / [1466.6645 * 0.0019196 * (725.6413)^2] = 0.0059 \end{aligned}$$

## 2.7 DETERMINE THE BLADE SOLIDITY.

The maximum forward velocity is 202.6667 ft/sec and the tip velocity is 725.6413 ft/sec from Step 2.6. Using the maximum advance ratio and HD-4, the blade loading is 0.1065.

$$\begin{aligned}\mu_{\langle mr \rangle} &= V_{\langle max \rangle} / V_{\langle tip-mr \rangle} \\ &= 202.6667 / 725.6413 = 0.2793\end{aligned}$$

$$\begin{aligned}\sigma_{\langle mr \rangle} &= C_{\langle thrust-mr \rangle} / BL \\ &= 0.0059 / 0.1065 = 0.0554\end{aligned}$$

## 2.8 DETERMINE THE NUMBER OF MAIN ROTOR BLADES TO BE USED.

Four rotor blades will be used. (Discuss reasons.)

## 2.9 DETERMINE THE BLADE CHORD.

The solidity is 0.0554 from Step 2.7. The number of blades (Step 2.8) is 4 and the rotor radius (Step 2.4) is 21.6068 ft.

$$\begin{aligned}c_{\langle mr \rangle} &= (\sigma_{\langle mr \rangle} * \pi * R_{\langle mr \rangle}) / b_{\langle mr \rangle} \\ &= (0.0554 * 3.1416 * 21.6068) / 4 = 0.9401 \text{ ft}\end{aligned}$$

$$AR_{\langle mr \rangle} = R_{\langle mr \rangle} / c_{\langle mr \rangle} = 21.6068 / 0.9401 = 22.9827$$

Since the aspect ratio was not within the limits of 15 to 20, the rotational velocity (Step 2.5) was reduced to 31.00 rad/sec. Steps 2.6 and 2.7 were recalculated with the following results.

$\Omega_{\langle mr \rangle}$ :	31.00 rad/sec	$V_{\langle tip-mr \rangle}$ :	669.8108 ft/sec
$C_{\langle thrust-mr \rangle}$ :	0.0070	$\mu_{\langle mr \rangle}$ :	0.3026
BL:	0.1038	$\sigma_{\langle mr \rangle}$ :	0.0674
$c_{\langle mr \rangle}$ :	1.1444 ft	$AR_{\langle mr \rangle}$ :	18.8803

2.10 DETERMINE THE AVERAGE LIFT COEFFICIENT.

The thrust coefficient is 0.0070 from Step 2.10. The solidity is 0.0674 from Step 2.10

$$\begin{aligned} C_{\text{lift}} &= ( 6 * C_{\text{thrust-mr}} ) / \sigma_{\text{mr}} \\ &= ( 6 * 0.0070 ) / 0.0674 = 0.6231 \end{aligned}$$

2.11 CHOOSE AN AIRFOIL SECTION FOR THE MAIN ROTOR BLADES.

The selected airfoil section is airfoil "A" from HD-5.  
(Discuss reasons.)

2.12 DETERMINE AVERAGE LIFT CURVE SLOPE AND AVERAGE PROFILE DRAG COEFFICIENT.

From HD-5, the profile drag coefficient for airfoil "A" is 0.010 and the lift curve slope ( $C_{L,\alpha}$ ) is 6.4458 per radian.

## Chapter 3

### PRELIMINARY POWER CALCULATIONS

#### 3.1 MAKE FIRST ESTIMATE OF THE POWER REQUIRED TO HOVER.

The HP-41CV program entitled HOVER [Ref. 3] was used to compute these values. The current value of all design parameters are listed below.

DL:	6.00	W<gross>:	8800.00 lbs
R<mr>:	21.61 ft	$\Omega$ <mr>:	31.00 rad/sec
V<tip>:	669.81 ft/sec	C<thrust>:	0.0070
$\mu$ <mr>:	0.3026	BL:	0.1038
$\sigma$ <mr>:	0.0674	b<mr>:	4
c<mr>:	1.1444 ft	AR<mr>:	18.88
C<lift>:	0.6231	Cdo<mr>:	0.0100
DA:	0.00		
Pi<mr-TL>:	583.91 hp		
Po<mr>:	160.56 hp		
PT<mr-hover>:	744.48 hp		

#### 3.2 MAKE SECOND GROSS WEIGHT ESTIMATE.

Using HD-3, the first gross weight estimate used in the main rotor design process was refined. The useful load was computed using 200 lbs per crewmember. Table 1 shows the result of the second gross weight estimate.

TABLE 1

## Weight Estimation Table

	----- ITERATION -----			
	FIRST	SECOND	THIRD	FOURTH
SOLIDITY:	-	0.0674	0.0674	0.0745
RADIUS:	-	21.6068	21.6068	20.7192
TOTAL POWER:	-	744.4755	734.3833	681.3628
ROTOR-BLADES:	-	555.89	488.44	444.46
HUB/HINGES:	-	323.90	284.59	250.34
TOTAL:	-	879.79	773.03	694.80
PROPULSION:	-	893.37	881.26	817.64
FUSELAGE:	-	1386.00	1217.82	1095.28
FLIGHT CONTROLS:	-	396.00	347.96	311.51
ELECTRICAL:	-	396.00	347.96	311.51
FIXED EQUIPMENT:	-	1848.00	1623.76	1453.70
EMPTY:	6600.00	5799.16	5191.80	4679.43
FUEL:	-	1500.00	1500.00	1500.00
USEFUL LOAD:	-	1400.00	1400.00	1400.00
GROSS:	8800.00	8699.16	8091.80	7579.43

## 3.3 ESTABLISH FIGURE OF MERIT AT APPROXIMATELY 0.075.

The induced power from Step 3.1 was between 70 to 80% of the total power from Step 3.1. In order to keep the same radius, the disk loading was reduced to 5.93 (see Step 2.4). The thrust coefficient (Step 2.6), the solidity (Step 2.7), the chord (Step 2.9), the aspect ratio (Step 2.9), and the lift coefficient (Step 2.10) were adjusted and the hover power was recalculated (Step 3.1) using the second gross weight estimate with the following results.



DL:	5.93	W<gross>:	8699.16 lbs
R<mr>:	21.61 ft	$\Omega$ <mr>:	31.00 rad/sec
V<tip>:	669.81 ft/sec	C<thrust>:	0.0059
$\mu$ <mr>:	0.3026	BL:	0.1038
$\sigma$ <mr>:	0.0674	b<mr>:	4
c<mr>:	1.1444 ft	AR<mr>:	18.88
C<lift>:	0.6231	Cdo<mr>:	0.0100
DA:	0.00		
Pi<mr-TL>:	573.73 hp		
Po<mr>:	160.66 hp		
PT<mr-hover>:	734.38 hp		

The induced power from Step 3.1 was between 70 to 80% of the total power from Step 3.1; therefore the Figure of Merit was approximated.

$$\text{Figure of Merit} = 573.7261 / 734.3833 = 0.7812$$

#### 3.4 REFINE SECOND GROSS WEIGHT ESTIMATE.

See Table 1 for the values of the third iteration.

#### 3.5 MAKE THIRD ESTIMATE OF POWER REQUIRED TO HOVER.

Since the third gross weight estimate was much smaller than the second estimate, the radius (Step 2.4) was changed so that the disk loading would not be too small.

$$\begin{aligned} R_{\text{mr}} &= \sqrt{W_{\text{gross}} / (DL * \pi)} \\ &= \sqrt{8091.8012 / (6 * 3.1416)} = 20.7192 \text{ ft} \end{aligned}$$

Since the radius was changed, a new maximum allowable rotational velocity was calculated (Step 2.5) and the

current value of rotational velocity did not exceed the maximum allowable value. Then Steps 2.6, 2.7, 2.9, 2.10, and 3.1 were adjusted using the third gross weight estimate with the following results.

DL:	6.00	W<gross>:	8091.80 lbs
R<mr>:	20.7192 ft	$\Omega$ <mr>:	31.00 rad/sec
V<tip>:	642.30 ft/sec	C<thrust>:	0.0076
$\mu$ <mr>:	0.3155	BL:	0.1020
$\sigma$ <mr>:	0.0745	b<mr>:	4
c<mr>:	1.2123 ft	AR<mr>:	17.09
C<lift>:	0.6121	Cdo<mr>:	0.0100
DA:	0.00		
Pi<mr-TL>:	537.55 hp		
Po<mr>:	143.82 hp		
PT<mr-hover>:	681.36 hp		

The induced power was between 70 to 80% of the total power so the Figure of Merit was recalculated.

Figure of Merit =  $537.5475 / 681.3628 = 0.7889$

### 3.6 REPEAT THE GROSS WEIGHT ESTIMATE AND THE HOVER POWER REQUIRED ITERATION.

Table 1 shows the result of the fourth gross weight estimate which is 93.67% of the previous estimate. Since the fourth gross weight estimate was close to the previous weight estimate, the disk loading was reduced to 5.62 (see Step 2.4) rather than calculating a new radius. The thrust coefficient (Step 2.6), the solidity (Step 2.7), the chord

(Step 2.9), the aspect ratio (Step 2.9), and the lift coefficient (Step 2.10) were adjusted and the hover power (Step 3.1) was recomputed using the fourth gross weight estimate with the following results.

DL:	5.62	W<gross>:	7579.43 lbs
R<mr>:	20.7192 ft	$\Omega$ <mr>:	31.00 rad/sec
V<tip>:	642.30 ft/sec	C<thrust>:	0.0071
$\mu$ <mr>:	0.3155	BL:	0.1020
$\sigma$ <mr>:	0.0696	b<mr>:	4
c<mr>:	1.1327 ft	AR<mr>:	18.29
C<lift>:	0.6121	Cdo<mr>:	0.0100
DA:	0.00		
Pi<mr-TL>:	486.86 hp		
Po<mr>:	134.37 hp		
PT<mr-hover>:	621.24 hp		

This hover power estimate is 91.18% of the previous estimate. The induced power from Step 3.1 was between 70 to 80% of the total power from Step 3.1; therefore the Figure of Merit was approximated.

Figure of Merit =  $486.8628 / 621.2352 = 0.7837$

### 3.7 DETERMINE THE POWER REQUIRED TO HOVER IGE, SSL.

An HP-41CV program entitled HOVER [Ref. 3] was used to compute the hover power in ground effect at standard sea level. A hover height of 10 feet above ground level was used.

Pi<mr-TL+GE>: 371.7278 hp

Po<mr>: 134.3724 hp

PT<mr-hover-IGE>: 506.1002 hp

3.8 DETERMINE THE PARASITE POWER REQUIRED IN FORWARD FLIGHT.

Using HD-8 and the latest estimate of gross weight (7579.43 lbs), the equivalent flat plate area loading is 330.0 psf.

$$\begin{aligned} \text{EFPA}<\text{FF}> &= W<\text{gross}> / \text{Loading}<\text{EFPA}> \\ &= 7579.43 / 330.0 = 22.9680 \text{ ft}^2 \end{aligned}$$

The values of parasite power have been listed in Table 2.

3.9 DETERMINE MAIN ROTOR POWER REQUIRED AND THE MACH NUMBER OF THE ADVANCING BLADE TIP FOR FORWARD FLIGHT.

A program entitled FLIGHT [Ref. 3.] for use with the HP-41CV programmable calculator was used to calculate the values given in Table 2.

TABLE 2

## Main Rotor Power Profile

## STANDARD SEA LEVEL

AIRSPEED (KNOTS)	TIP MACH	POWER			
		INDUCED (SHP)	PROFILE (SHP)	PARASITE (SHP)	TOTAL (SHP)
0	0.5753	486.86	134.37	0.00	621.23
20	0.6056	385.76	135.94	1.91	523.62
40	0.6359	240.33	140.66	15.30	396.29
60	0.6661	164.13	148.52	51.64	364.29
80	0.6964	123.64	159.52	122.41	405.57
100	0.7266	99.03	173.67	239.08	511.78
105	0.7342	94.33	177.70	276.77	548.80
120	0.7569	82.57	190.96	413.13	686.66

## SPECIFICATION DENSITY ALTITUDE

0	0.5564	543.44	108.52	0.00	651.96
20	0.5856	449.30	109.79	1.54	560.64
40	0.6149	294.25	113.60	12.36	420.21
60	0.6441	203.17	119.95	41.71	364.83
80	0.6734	153.40	128.83	98.86	381.09
100	0.7027	122.95	140.26	193.09	456.30
105	0.7100	117.12	143.51	223.52	484.16
120	0.7319	102.53	154.22	333.65	590.40

## Chapter 4

### TAIL ROTOR DESIGN

#### 4.1 MAKE PRELIMINARY DETERMINATIONS OF TAIL ROTOR GEOMETRY.

HD-9 was used to determine the tail rotor geometry. The following current main rotor values were required by HD-9.

W<gross>: 7579.43 lbs                      R<mr>: 20.7192 ft  
Ω<mr>: 31.00 rad/sec (296.0282 RPM)    Cdo<mr>: 0.010

The results of the calculations of HD-9 are shown below.

AR<tr>: 6.75                      b<mr>: 2                      Cdo<tr>: 0.0138

Ω<tr>: 139.5 rad/sec (1332.1269 RPM)    R<tr>: 3.5790 ft

$$\begin{aligned} L<tr> &= R<mr> + R<tr> + 0.5 \text{ ft} \\ &= 20.7192 + 3.5790 + 0.5 = 24.7982 \text{ ft} \end{aligned}$$

$$\begin{aligned} c<tr> &= R<tr> / AR<tr> \\ &= 3.5790 / 6.75 = 0.5302 \text{ ft} \end{aligned}$$

$$\begin{aligned} \sigma<tr> &= b<tr> * c<tr> / (\pi * R<tr>) \\ &= 2 * 0.5302 / 3.14 * 3.5790 = 0.0943 \end{aligned}$$

#### 4.2 DETERMINE TAIL ROTOR POWER REQUIRED AT HOVER OGE, SSL.

The total power of the tail rotor required to hover out-of-ground effect at standard sea level was calculated using the HP-41CV calculator program entitled Hover [Ref. 3]. The tail rotor thrust calculated below should be used for the value of weight called for in the program.

PT<mr>: 621.2346 hp (341,679.03 ft-lbs)

L<tr>: 24.7982 ft

$\Omega$ <mr>: 31.00 rad/sec

$T<tr> = PT<mr-hover> / ( \Omega<mr> * L<tr> )$

$= 341,679.03 / (31.00 * 24.7982) = 444.4639 \text{ lbs}$

The other input values required by the calculator program are listed below.

AR<tr>: 6.75

R<tr>: 3.5790 ft

b<mr>: 2

Cdo<tr>: 0.0138

$\Omega$ <tr>: 139.5 rad/sec

c<tr>: 0.5302 ft

The following values were output.

Pi<TL-tr>: 43.1150 hp

Po<tr>: 3.5211 hp

PT<tr-hover>: 46.6360 hp

#### 4.3 DETERMINE TAIL ROTOR POWER REQUIRED AND TIP MACH EFFECTS FOR FORWARD FLIGHT.

These calculations have been made at both standard sea level and at specification density altitude. The HP-41CV programs entitled Flight and TR [Ref. 3] were used to calculate the values listed in Table 3.

TABLE 3

## Tail Rotor Power Profile

## STANDARD SEA LEVEL

AIRSPEED (KNOTS)	TIP MACH	POWER		
		INDUCED (SHP)	PROFILE (SHP)	TOTAL (SHP)
0	0.4472	43.12	3.52	46.64
20	0.4775	28.65	3.59	32.24
40	0.5077	11.70	3.79	15.50
60	0.5380	6.82	4.13	10.96
80	0.5683	6.40	4.61	11.01
100	0.5985	8.25	5.23	13.48
105	0.6061	9.07	5.40	14.47
120	0.6288	12.57	5.98	18.55

## SPECIFICATION DENSITY ALTITUDE

0	0.4325	52.36	2.84	55.21
20	0.4617	37.11	2.90	40.01
40	0.4910	16.06	3.06	19.12
60	0.5202	8.51	3.34	11.85
80	0.5495	7.04	3.72	10.76
100	0.5788	8.15	4.22	12.38
105	0.5861	8.77	4.36	13.14
120	0.6080	11.55	4.83	16.37



## Chapter 5

### POWER CALCULATION REFINEMENTS

#### 5.1 DETERMINE TOTAL POWER REQUIRED FOR HOVER AND FORWARD FLIGHT.

These calculations were made for flight out-of-ground effect at both standard sea level and at the specification density altitude. The FORTRAN computer program for the IBM-3033 [Ref. 4] was used to calculate these values. The values for total power in Table 4 can also be calculated by adding the values for main rotor power (Step 3.9) and for tail rotor power (Step 4.3). The required RSHP for hover equals the total hover power.

#### 5.2 DETERMINE COMPRESSIBILITY CORRECTION.

The additional power required to overcome the effects of compressibility on the performance of the helicopter rotor is calculated below at both standard sea level and the specification density altitude.

##### Standard Sea Level Computations

$M_{\text{tip-mr}}: 0.7569$  (from Step 3.9)     $\sigma_{\text{mr}}: 0.0696$   
 $\rho: 0.0023769$  slugs/ft<sup>3</sup>     $A_{\text{mr}}: 1348.6394$  ft<sup>2</sup>  
 $M_{\text{crit}}: 0.650$  (from HD-10)     $V_{\text{tip-mr}}: 642.30$  ft/sec  
 $M_d = M_{\text{tip-mr}} - M_{\text{crit}} - 0.06$   
 $= 0.7569 - 0.650 - 0.06 = 0.0469$

$$\begin{aligned}
 P_{\text{Comp}} &= \rho * A_{\text{mr}} * V_{\text{tip-mr}}^3 * \sigma_{\text{mr}} \\
 &\quad * [0.012 * M_d + 0.10 * M_d^3] \\
 &= (0.0023769 * 1348.6394 * (642.30)^3 * 0.0696 \\
 &\quad * [0.012 * 0.0469 + 0.10 * (0.0469)^3]) / 550 = 61.6041 \text{ hp}
 \end{aligned}$$

#### Specification Density Altitude Computations

$$\begin{aligned}
 M_{\text{tip-mr}} &: 0.7319 \text{ (from Step 3.9)} & \sigma_{\text{mr}} &: 0.0696 \\
 \rho &: 0.0019196 \text{ slugs/ft}^3 & A_{\text{mr}} &: 1348.6394 \text{ ft}^2 \\
 M_{\text{crit}} &: 0.650 \text{ (from HD-10)} & V_{\text{tip-mr}} &: 642.30 \text{ ft/sec} \\
 M_d &= M_{\text{tip-mr}} - M_{\text{crit}} - 0.06
 \end{aligned}$$

$$= 0.7319 - 0.650 - 0.06 = 0.0219$$

$$\begin{aligned}
 P_{\text{Comp}} &= \rho * A_{\text{mr}} * V_{\text{tip-mr}}^3 * \sigma_{\text{mr}} \\
 &\quad * [0.012 * M_d + 0.10 * M_d^3] \\
 &= 0.0019196 * 1348.6394 * (642.30)^3 * 0.0696 \\
 &\quad * [0.012 * 0.0219 + 0.10 * (0.0219)^3] / 550 = 22.9047 \text{ hp}
 \end{aligned}$$

#### 5.3 DETERMINE THE REQUIRED RSHP AT MAXIMUM VELOCITY.

At standard sea level, the total power required is 766.8158 hp, which is the sum of the total power from Step 5.1 and the compressibility correction from Step 5.2. Similarly, the total power required at the specification density altitude is 629.6808 hp.

TABLE 4

## Total Rotor Power Profile

## STANDARD SEA LEVEL

AIRSPEED (KNOTS)	POWER			
	INDUCED (SHP)	PROFILE (SHP)	PARASITE (SHP)	TOTAL (SHP)
0	529.98	137.89	0.00	667.87
20	414.41	139.53	1.91	555.85
40	252.03	144.45	15.30	411.79
60	170.95	152.65	51.64	375.24
80	130.04	164.14	122.41	416.59
100	107.28	178.90	239.08	525.26
105	103.40	183.10	276.77	563.26
120	95.14	196.94	413.13	705.21

## SPECIFICATION DENSITY ALTITUDE

0	595.80	111.36	0.00	707.17
20	486.41	112.69	1.54	600.64
40	310.31	116.66	12.36	439.33
60	211.68	123.29	41.71	376.67
80	160.44	132.56	98.86	391.86
100	131.11	144.48	193.09	468.67
105	125.90	147.87	223.52	497.29
120	114.08	159.05	333.65	606.78

## 5.4 DETERMINE THE REQUIRED RSHP FOR HOVER (IGE) AT THE SPECIFIED CEILING.

The HP-41CV programs entitled HOVER and TR [Ref. 3] were used to compute these values. The values of the input parameters are listed below.

R<mr>:	20.7192 ft	$\Omega$ <mr>:	31.00 rad/sec
b<mr>:	4	c<mr>:	1.1327 ft
W<gross>:	7579.4305 lbs	Cdo<mr>:	0.0100
DA:	11000.00 ft	Hover height:	10 ft

The computed power values are listed below:

Pi<mr-TL-GE>:	441.6548 hp	Pi<tr-TL>:	41.4976 hp
Po<mr>:	96.1516 hp	Po<tr>:	2.5195 hp
PT<mr-hover>:	537.8065 hp	PT<tr>:	44.0171 hp
PT<acft>:	581.8236 hp		

#### 5.5 DETERMINE THE TOTAL ESHP REQUIRED.

The maximum power required from Step 5.3 is 766.8158 hp and the maximum power required from Step 5.4 is 581.8236 hp. The greater of these two values is 766.8158 hp, which is the maximum required RSHP. The required ESHP is calculated below.

$$\begin{aligned}\text{ESHP}<\text{Req}> &= 1.13 * \text{RSHP}<\text{max}> + 10.0 \\ &= 1.13 * 766.8158 + 10.0 = 876.5019 \text{ hp}\end{aligned}$$

## Chapter 6

### ENGINE SELECTION

#### 6.1 SELECT TYPE AND NUMBER OF ENGINES.

It was decided to use two engines. (Discuss reasons.)

The parameters for engine selection have been calculated for two engines and included in Table 5. Note that Engine "A" did not have the necessary available power. Engine "B" was selected for the design. (Discuss reasons.)

#### 6.2 REVISE GROSS WEIGHT AND POWER REQUIRED.

The value of gross weight using the new powerplant weight from Step 6.1 is 8017.50 lbs, which is within 10% of the estimated value of 7579.43 lbs from Table 1.

#### 6.3 DETERMINE FUEL FLOW RATES AT VARIOUS POWER SETTINGS.

The specific fuel consumption was obtained from the manufacturer's data on Engine "B" at military, normal, and cruise power settings. The fuel flow rates at these power settings are calculated below.

$$\begin{aligned} Wf\langle MRP \rangle &= SFC\langle MRP \rangle * SHP\langle MRP \rangle \\ &= (0.620) * (725 \text{ hp}) = 449.50 \text{ lbs/hr} \\ Wf\langle NRP \rangle &= SFC\langle NRP \rangle * SHP\langle NRP \rangle \\ &= (0.639) * (630 \text{ hp}) = 402.57 \text{ lbs/hr} \\ Wf\langle CRP \rangle &= SFC\langle CRP \rangle * SHP\langle CRP \rangle \\ &= (0.658) * (550 \text{ hp}) = 361.90 \text{ lbs/hr} \end{aligned}$$

TABLE 5

## Engine Selection Criteria

	ENGINE		
	A	B	C
POWERPLANT WEIGHT (lbs):	572.78	1255.70	1838.52
LIFE-CYCLE COST (per engine):	171.18k	218.36k	425.20k
ENGINE LIFE (hrs):	600.00	750.00	800.00
NUMBER OF REPLACEMENTS:	1	1	1
R & D COSTS:	0	0	0
INITIAL COST:	90.00k	100.00k	200.00k
YEARLY MAINTENANCE COST:	3.00k	6.00k	12.00k
YEARLY OPERATING COST:	0.960k	1.920k	2.400k
REPLACEMENT COST:	121.50k	135.00k	270.00k
SALVAGE VALUE:	72.00k	80.00k	160.00k
AVAILABILITY (per engine):	0.8333	0.8333	0.8000
RELIABILITY (per engine):	0.9962	0.9967	0.9966
MAINTAINABILITY (per engine):	0.0007	0.0006	0.0005
PERFORMANCE (SHP):	634	1450	2800
	ENGINE		
	D	E	F
POWERPLANT WEIGHT (lbs):	3018.32	3830.04	4662.60
LIFE-CYCLE COST (per engine):	1052.60k	1184.00k	1353.80k
ENGINE LIFE (hrs):	800.00	1500.00	750.00
NUMBER OF REPLACEMENTS:	1	1	1
R & D COSTS:	0	0	0
INITIAL COST:	580.00k	640.00k	700.00k
YEARLY MAINTENANCE COST:	15.00k	19.20k	26.40k
YEARLY OPERATING COST:	4.200k	4.800k	7.200k
REPLACEMENT COST:	783.00k	864.00k	945.00k
SALVAGE VALUE:	464.00k	512.00k	560.00k
AVAILABILITY (per engine):	0.6977	0.6667	0.5738
RELIABILITY (per engine):	0.9975	0.9995	0.9978
MAINTAINABILITY (per engine):	0.0014	0.0021	0.0027
PERFORMANCE (SHP):	3600	5820	8220

## Chapter 7

### RANGE AND ENDURANCE CALCULATIONS

#### 7.1 DETERMINE THE SLOPE OF THE FUEL FLOW VERSUS SHP LINE AND THE ZERO HORSEPOWER INTERCEPT.

These calculations were made at standard sea level. A linear regression curve fitting routine [Ref. 5] for the HP-41CV programmable calculator was used to calculate the slope and intercept of the fuel flow versus SHP line using the fuel flow rates from Step 6.3. The intercept of the fuel flow versus SHP line with the ordinate axis is the zero horsepower intercept.

$$\beta_{ave} = 0.5004$$

$$ZHI_{SSL} = 86.9229 \text{ hp}$$

#### 7.2 COMPUTE THE ZERO HORSEPOWER INTERCEPT AT THE SPECIFIED DENSITY ALTITUDE.

The pressure at the specification pressure altitude is 1827.69 psf obtained from an ICAO standard atmospheric table.

$$\theta_{spec} = T_{spec} / T_{SSL}$$

$$= (554.69 \text{ degrees R}) / (518.69 \text{ degrees R}) = 1.0694$$

$$\delta_{spec} = P_{spec} / P_{SSL}$$

$$= (1827.69 \text{ psf}) / (2216.22 \text{ psf}) = 0.8637$$

$$ZHI = \alpha_{ave} * \delta * \sqrt{\theta}$$

$$= (86.9229 \text{ hp}) * (0.8637) * \sqrt{1.0694} = 77.6369 \text{ hp}$$

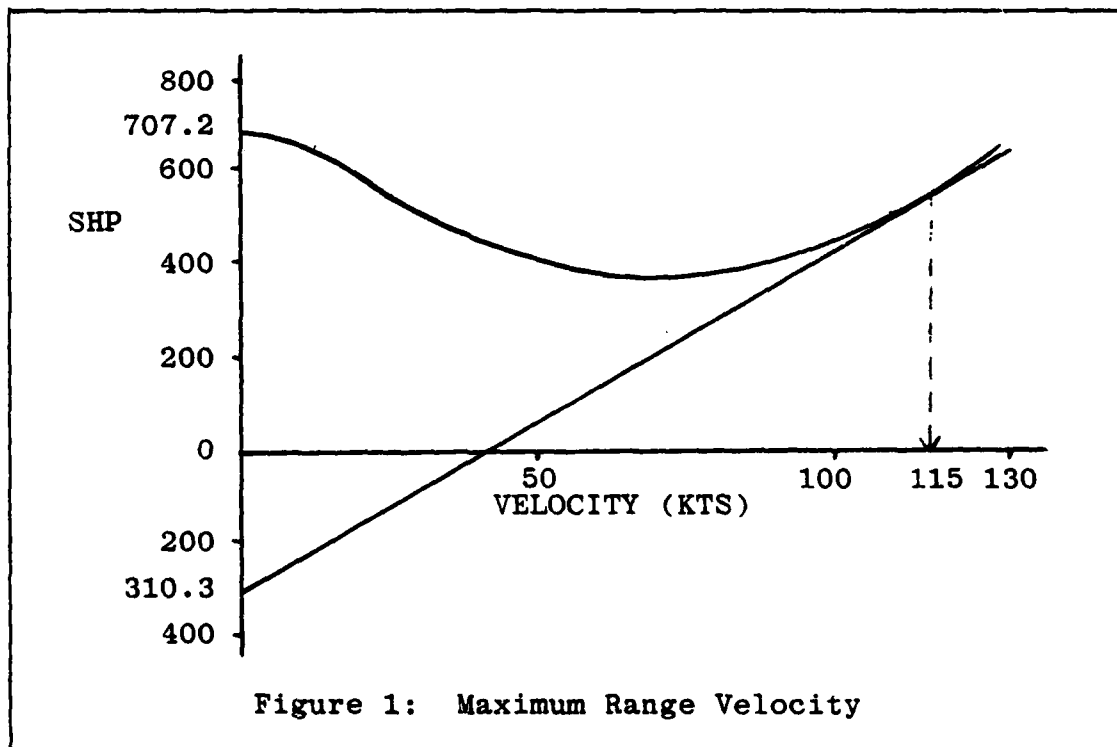
### 7.3 DETERMINE THE ZERO HORSEPOWER INCREMENT.

The calculation of the phantom SHP is shown below.

$$\begin{aligned} P\langle\text{SHP}\rangle &= [n * \alpha\langle\text{ave}\rangle * \delta * \sqrt{\theta}] / \beta\langle\text{ave}\rangle \\ &= 2 * (86.9229 \text{ hp}) * (0.8637) * \sqrt{1.0694} / (0.5004) \\ &= 310.2987 \text{ hp} \end{aligned}$$

### 7.4 DETERMINE THE MAXIMUM RANGE VELOCITY.

The maximum range velocity of 115 knots was determined graphically. See Figure 2.



### 7.5 DETERMINE THE MAXIMUM ENDURANCE VELOCITY AND THE REQUIRED FUEL FLOW RATE AT THIS VELOCITY.

The maximum endurance velocity of 65 knots was determined graphically. See Figure 3.



The RSHP at maximum endurance velocity at the specification density altitude is 374.2234 hp from Figure 3, the slope of the fuel flow versus SHP line is 0.5004 from Step 7.1 and the phantom SHP is 310.2987 hp from Step 7.3. The fuel flow rate is calculated below.

$$\text{RSHP}<\text{Req-max end}> = P<\text{SHP}> + \text{RSHP}<\text{max end}>$$

$$= 310.2987 \text{ hp} + 374.2234 \text{ hp} = 684.5221 \text{ hp}$$

$$\text{Wf}<\text{max end}> = \text{RSHP}<\text{Req-max end}> * \beta<\text{ave}>$$

$$= 684.5221 \text{ hp} * 0.5004 = 342.5349 \text{ lbs/hr}$$

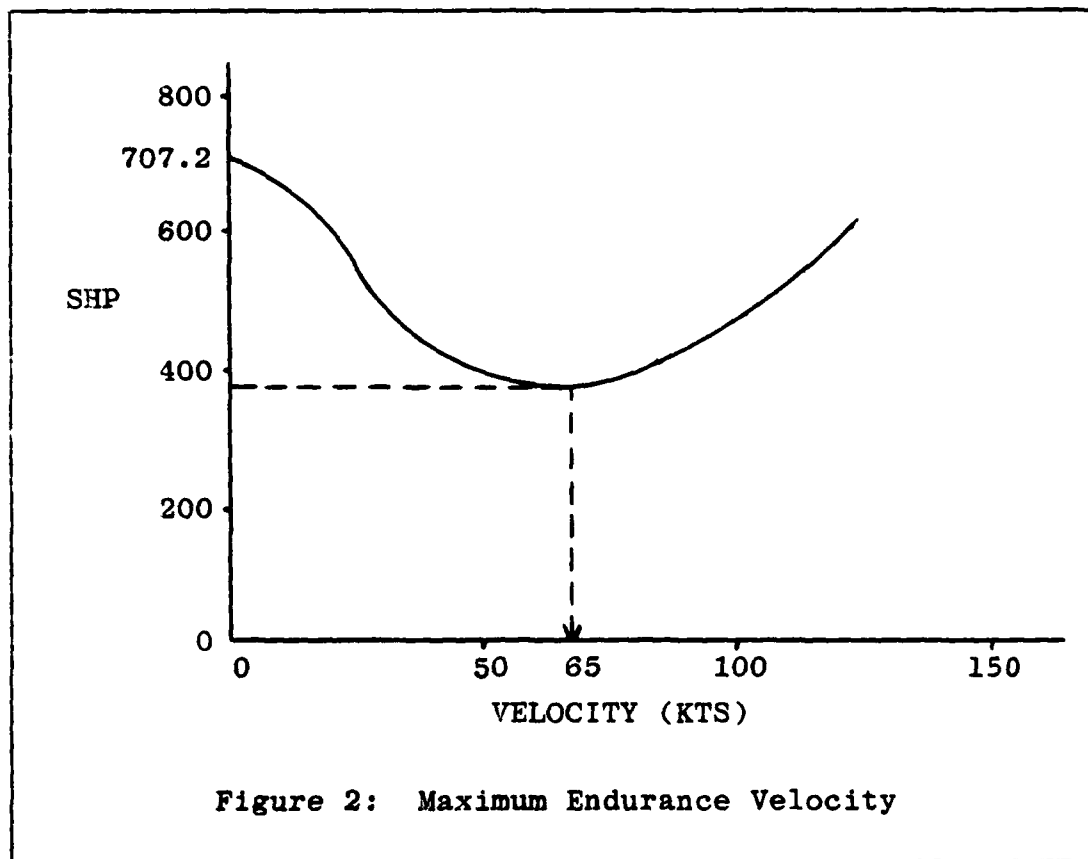


Figure 2: Maximum Endurance Velocity

7.6 DETERMINE THE POWER REQUIRED AT SPECIFICATION CRUISE VELOCITY AND THE REQUIRED FUEL FLOW RATE AT THIS VELOCITY.

The RSHP at cruise velocity at the specification density altitude is 497.2908 hp from Step 5.1, the slope of the fuel flow versus SHP line is 0.5004 from Step 7.1 and the phantom SHP is 310.2987 hp from Step 7.3. The fuel flow rate at cruise velocity is calculated below.

$$\begin{aligned} \text{RSHP}<\text{Req-cruise}> &= \text{P}<\text{SHP}> + \text{RSHP}<\text{cruise}> \\ &= 310.2987 \text{ hp} + 497.2908 \text{ hp} = 807.5895 \text{ hp} \\ \text{Wf}<\text{cruise}> &= \text{RSHP}<\text{Req-cruise}> * \beta<\text{ave}> \\ &= 807.5895 \text{ hp} * 0.5004 = 404.1178 \text{ lbs/hr} \end{aligned}$$

7.7 DETERMINE THE TOTAL FUEL REQUIREMENTS FOR SPECIFIED MAXIMUM RANGE.

The cruise velocity and maximum range are specification items from HD-1. The fuel flow rate at normal rated power for one engine is 402.57 lbs/hr from Step 6.3. The fuel flow rate at maximum endurance velocity is 342.5349 lbs/hr from Step 7.5. The fuel flow rate at cruise velocity is 404.1178 lbs/hr from Step 7.6. The total fuel requirements are based on the given assumptions and are calculated below.

$$\begin{aligned} \text{FW} &= 0.05 * n * \text{Wf}<\text{NRP}> + [(\text{Wf}<\text{cruise}> * \text{RNG}<\text{max}>) / \text{V}<\text{cruise}>] \\ &\quad + 0.05 * n * \text{Wf}<\text{NRP}> + 0.25 * \text{Wf}<\text{max end}> \\ &= (0.05 * 2 * 402.57) + [(404.1178 * 225) / 105] \\ &\quad + (0.05 * 2 * 402.57) + (0.25 * 342.5349) \\ &= 1032.1144 \text{ lbs} \end{aligned}$$

## Chapter 8

### MISCELLANEOUS CALCULATIONS

#### 8.1 COMPUTE DESIGN GROSS AND EMPTY WEIGHT.

The estimated fuel weight from Step 3.6 is 1500.0 lbs and the actual fuel weight from Step 7.7 is 1032.1144 lbs. The estimated powerplant weight from Step 3.6 is 817.64 lbs and the actual powerplant weight from Step 6.1 is 1255.70 lbs. The gross weight is therefore 29.8256 lbs lighter than estimated due to the error in the above estimates. It was decided to use this shortage for extra fuel. By increasing the fuel weight, the final design gross weight equals the weight estimate which was used in the power calculations; therefore, the power calculations are valid at the final design gross weight. The additional fuel weight allows for an increase in the maximum range of the helicopter. The new maximum range is calculated below.

$$\begin{aligned} \text{RNG} < \text{max} > &= [V < \text{Cruise} > * (\text{FW} + \text{FW} < \text{Added} > - 0.05 * n * W_f < \text{NRP} > \\ &\quad - 0.05 * n * W_f < \text{NRP} > - 0.25 * W_f < \text{max end} >)] / W_f < \text{Cruise} > \\ &= [105.0 * (1032.1144 + 29.8256 - (0.05 * 2 * 402.57) \\ &\quad - (0.05 * 2 * 402.57) - (0.25 * 342.5349))] / 404.1178 = \\ &= 232.7494 \text{ nmi} \end{aligned}$$

#### 8.2 DETERMINE RETREATING BLADE STALL VELOCITY.

The input values for the HP-41CV program entitled BS [Ref. 6] were:



power for comparison with the power available calculated above. The iterative values are shown in Table 6.

The best rate of climb is 3656 FPM.

TABLE 6			
Best Rate of Climb Iterations			
	----- Iterations -----		
	FIRST	SECOND	THIRD
Vertical velocity (fpm):	5000.0	3700.00	3656.00
Main Rotor -			
Induced Power (hp):	124.49	138.09	138.56
Profile Power (hp):	147.66	147.66	147.66
Parasite Power (hp):	104.09	69.93	69.11
Climb Power (hp):	1148.40	849.81	839.71
Total Power (hp):	1524.63	1205.49	1195.03
Tail Rotor -			
Induced Power (hp):	120.38	76.57	75.28
Profile Power (hp):	4.10	4.10	4.10
Total Power (hp):	124.49	80.67	79.38
Aircraft -			
Total Power (hp):	1649.12	1286.15	1274.41

#### 8.4 COMPUTE MAXIMUM HOVER ALTITUDE (IGE).

The total RSHP available from Step 8.3 is 1274.3363 hp. The height above ground was 10 feet. The HP-41CV programs entitled Hover and TR [Ref. 3] were used to compute the total required RSHP at hover for various altitudes. The program was repeated until the required RSHP equaled the RSHP available. The iterations and final values are shown in Table 7.

TABLE 7

## Maximum Hover Altitude (IGE) Iterations

	----- Iterations -----			
	FIRST	SECOND	THIRD	FOURTH
Altitude (ft):	50000.0	51000.0	50900.0	50910.0
Main Rotor -				
Induced Power (hp):	948.84	971.89	969.55	969.78
Profile Power (hp):	22.37	21.39	21.49	21.48
Total Power (hp):	971.21	993.28	991.04	991.26
Tail Rotor -				
Induced Power (hp):	255.99	284.37	282.35	282.55
Profile Power (hp):	0.59	0.56	0.56	0.56
Total Power (hp):	256.58	284.94	282.92	283.12
Aircraft -				
Total Power (hp):	1236.79	1278.22	1273.95	1274.38

Thus the maximum hover altitude (IGE) is 50910 ft.

## 8.5 COMPUTE SERVICE CEILING.

The calculation of total RSHP available at Normal Rated Power is shown below.

$$\begin{aligned}
 PT<avail> &= [(ESHP<NRP> - 10.0) / (0.10*(n-1) + 1.03)] \\
 &= (1260.0 - 10.0) / 1.13 = 1106.1947 \text{ hp}
 \end{aligned}$$

The best rate of climb velocity from Step 8.2 is 58 knots.

The HP-41CV program entitled Flight [Ref. 3] was used to compute the total RSHP required at the best rate of climb velocity with a 100 fpm rate of climb for various altitudes. The program was repeated until the required RSHP equaled the RSHP available. The iterations and final values are shown in Table 8.

TABLE 8

## Service Ceiling Iterations

	----- Iterations -----			
	FIRST	SECOND	THIRD	FOURTH
Altitude (ft):	20000.0	50000.0	48500.0	48400.0
Main Rotor -				
Induced Power (hp):	314.95	900.29	852.60	849.52
Profile Power (hp):	78.67	24.59	26.27	26.39
Parasite Power (hp):	24.86	7.77	8.30	8.34
Climb Power (hp):	22.97	22.97	22.97	22.97
Total Power (hp):	441.45	955.62	910.14	907.21
Tail Rotor -				
Induced Power (hp):	19.53	230.71	200.06	198.19
Profile Power (hp):	2.19	0.68	0.73	0.73
Total Power (hp):	21.72	231.40	200.79	198.92
Aircraft -				
Total Power (hp):	463.17	1187.01	1110.94	1106.13

Thus the service ceiling is 48400 ft.

## Chapter 9

### FINAL CHECK

#### 9.1 MAKE FINAL CHECK FOR COMPLIANCE WITH SPECIFICATIONS.

Table 9 is a final summary of this example.

TABLE 9		
Final Summary		
	<u>Performance</u>	
	Specification	Design
Crew:	2	2
Internal Load [lbs]:	1000	1000
Max Hover Altitude (IGE):	11000	50910
Service Ceiling:	14500	48400
Disk Loading:	--	5.62
Velocity [kts] - Cruise:	105	105
Maximum:	120	120
Max Endurance:	--	65
Max Range:	--	115
Retreating Blade Stall:	1750	> 120
Best Rate of Climb [fpm]:	--	3656
Figure of Merit:	--	0.7837
Engine - Number / Type:	--	2 "B"
SHP per engine (Military):	--	725
Coefficient of Thrust:	--	0.0071
Coefficient of Lift:	--	0.6121
Max Advance Ratio:	--	0.3155
Maximum Range [nmi]:	225	232.7
	<u>Geometry</u>	
Weight [lbs] - Max Gross:	11000	7579.43
Empty:	--	5117.49
Fuel Capacity [lbs] - Internal:	--	1061.94
Main Rotor - Airfoil section used:	--	"A"
Chord [ft]:	--	1.1327
Radius [ft]:	< 27	20.7192
Number of blades:	--	4
Drag Coefficient:	--	0.010
Rotational velocity [rad/sec]:	--	31.0
Solidity:	--	0.0696
Aspect Ratio:	--	18.29
Tail Rotor - Airfoil section used:	--	"A"
Chord [ft]:	--	0.5302



	Specification	Design
Radius [ft]:	--	3.5790
Number of blades:	2	2
Drag Coefficient:	--	0.0138
Rotational velocity [rad/sec]:	--	139.5
Solidity:	--	0.0943
Aspect Ratio	--	6.75
Equivalent Flat Plate Area [ft <sup>2</sup> ] - Fwd:	--	22.9680
	Vert: --	45.9359
Distance from CG to TR Hub [ft]:		24.7982

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