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Trade Off Analysis of Technology Needs for Public Service Helicopters

Final Report

James S. Bauchspies, William R. Bryant, Jr.,
and William E. Simpson

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PREFACE

This report on "Tradeoff Analysis of Technology Needs for Public Service Helicopters" presents the results of an independent analysis of Public Service Helicopter technology needs identified by a Public Service Helicopter User's Workshop at the NASA Ames Research Center in July 1980. This work was conducted to provide further information to support continuing assessments by NASA's Office of Aeronautics and Space Technology (OAST) of factors affecting the need for aeronautical research applicable to rotorcraft technology and related planning of research and technology (R&T) programs.

The scope of work involved two phases of effort under Tasks 013 and 013A of NASA Contract NASW-3554. The initial effort classified the needs identified by Public Service Helicopter (PSH) users into those that would require new technology and those that could be met by application of existing technology. In addition, some needs impacted upon others, indicating some degree of tradeoff would be required. The results of that effort were presented in a separate ORI report in December 1983.[1] This report presents the results of the second phase of effort involving comparisons of the PSH technology needs with NASA's rotorcraft technology programs and relevant objectives of the Army's Light Helicopter (LHX) program. Further, the results of a preliminary analysis to quantify the tradeoffs required by combining selected performance needs into the same helicopter design, as well as discussion of new concepts, is reported.

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[1]Bauchspies, J. S. and Simpson, W. E., Analysis of Technology Program Needs for Public Service Helicopters, Final Report, Prepared by ORI, Inc. for the National Aeronautics and Space Administration, Washington, D.C., December 1983.

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SUMMARY

The expanding use of helicopters in the civil sector for public service missions (law enforcement, emergency medical services, search and rescue, and environmental control) has identified needed advances in rotorcraft technology to meet Public Service Helicopter (PSH) user requirements. These needs were identified at a NASA-sponsored Public Service Helicopter User's Workshop held at the NASA Ames Research Center on July 14-16, 1980.

In 1983, ORI, Inc. was requested to conduct an independent analysis of the PSH users' needs. Specifically, ORI was requested to classify the needs identified by the July 1980 Workshop as to whether the need is a design option by the manufacturer or would require new technology to meet the stated requirement. It was noted that some of the needs had an impact upon other stated needs and required a tradeoff analysis to determine the synergistic limitations in performance resulting from trying to meet two or more needs in the same aircraft. The results of ORI's initial analysis were reported in the ORI report, Analysis of Technology Program Needs for Public Service Helicopters.[1]

This study is an extension of the previous effort and reports the results of a preliminary trade-off analysis of selected performance needs conducted to identify options available to the PSH user. The study also compares the PSH technology needs with NASA's rotorcraft technology program objectives and relevant program objectives of the Army's Light Helicopter (LHX) program. The study concludes with recommendations for investigating innovative rotorcraft concepts and structuring a PSH technology program to meet the users' needs.

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[1]Bauchspies, J. S. and Simpson, W. E., Analysis of Technology Program Needs for Public Service Helicopters, Final Report, Prepared by ORI, Inc. for the National Aeronautics and Space Administration, Washington, D.C., December 1983.

COMPARISON WITH NASA'S ROTORCRAFT PROGRAM

As a first step, ORI compared the technology needs identified in the previous effort with ongoing rotorcraft R&T programs. The current NASA Rotorcraft Technology Program includes 14 projects divided equally between the R&T Base and Rotorcraft Systems Technology program areas. A review of the current NASA Research and Technology Objectives and Plans (RTOPS) indicates that, when considered individually, most of the PSH technology needs are being addressed by one or more of the NASA projects (see Table 1). However, since some of the stated needs impact on other needs, particularly in the vehicle design area, complete fulfillment of the PSH users' needs will not be met.

COMPARISON WITH ARMY'S LHX PROGRAM OBJECTIVES

In a similar manner, ORI investigated available information pertaining to the Army's proposed LHX program. Since the LHX program is currently in the concept formulation stage, specific aircraft configurations and design requirements have yet to be determined. However, available information indicates that there may be several areas where similarities exist in PSH and LHX objectives. These areas include weight class, speed profiles, propulsion and structures objectives, and improvements in performing cockpit functions. For example, it is estimated that the speed for the LHX could range from 160 to 200 knots for a pure helicopter concept to above 200 knots for high performance rotorcraft concepts.

DESIGN TRADEOFFS

As mentioned, trade-off analyses were conducted to investigate the synergistic impact of combining the PSH users' needs into a single aircraft. The most significant conflicts involved the relationships between the needs for high speed (200 kn continuous cruise; 300 kn 30-minute dash); single engine hover (HOGE -- 10,000 ft.; HIGE -- 20,000 ft.); a dimensional constraint of a 20 ft. maximum rotor diameter; a maximum gross weight of 10,000 pounds; and mission requirements of 6 passengers plus 4 hours fuel and 4,000 pound payload plus 2 hours fuel. The combination of these extraordinary requirements became the driver for this analysis, resulting in a need to make compromises in the stated requirements or develop unconventional rotorcraft concepts.

Specifically, the effects of varying rotor characteristics at selected gross weights were examined to determine the required power to meet PSH user performance requirements. A computer model for estimating helicopter performance capabilities developed for the U.S. Army Foreign Science and Technology Center (FSTC) was used to compute power required for specified performance criteria at three selected gross weight conditions considered representative of user needs:

- 6,000 lb.: Minimum Mission Weight;
- 8,000 lb.: Design Gross Weight;
- 10,000 lb.: Maximum Overload or Alternate Design Gross Weight.

TABLE 1

COMPARISON OF PSH TECHNOLOGY NEEDS AND NASA ROTORCRAFT TECHNOLOGY PROGRAM

Identified Technology Needs For Public Service Helicopters (By Technology Areas)	NASA RTOP Number/Title	NASA Research and Technology Projects														
		R&T Base Projects							Rotorcraft Systems Technology							
		505-40-12 Fan & Compressor Research	505-40-22 Combustors & Turbines	505-40-42 Power Transfer Research	505-42-11 R/C Aeromechanic & Perform	505-42-23 R/C Airframe Systems	505-42-32 R/C Operating Problems	505-42-61 R/C Flight Test Operations	532-01-11 R/C Flt Guidance Sys Tech	532-03-11 RSRA Flt Research/ Rotors	532-06-11 R/C System Integration	532-06-12 Convertible Engine Sys Tech	532-06-13 R/C Vibration & Noise	532-09-10 RSRA/X-Wing R/C Flt Invest	532-09-11 Adv Tilt Rotor Research
Vehicle Design																
Increase Speed, 200-300 kn					X		X	X		X	X	X		X	X
HIGE-20,000 ft (SE)															
HOGE-10,000 ft (SE)															
Rotor Diameter-20 ft															
No Tail Rotor															X
Reduce Int & Ext. Noise				X	X	X	X			X	X		X		
Propulsion															
Non-Petroleum Fuels			X												
Low Fuel Consumption		X					X								
Dual Power Band							X					X			
Increase Shaft HP							X								
Lightweight Powerplant		X													
Particle Separation*															
Safety & Reliability															
Crashworthy Structures						X									
Reduce Tail Rotor Haz.															X
Nav., Guidance & Flt Cont															
Combine Controls									X						
All-Weather Flt.					X			X	X						
Low Airspeed Measur									X						
Electronic Map Display									X						
Precise Loc./Nav									X						
Auxiliary Systems															
Night Vision System															
Car Identifier*															
Car Lock-On*															
Car Stopper*															
Human Factors															
Noise & Vibration					X	X							X		
Integrated Flt Instr					X				X		X				
Monitoring/Diagnostics															
Trend Warning											X				
Computerized Monitoring		X									X				
Head-Up Display															
Perf. Limit Monitor					X						X				

*Requires Further User Definition

In addition, the evaluation covered three rotor tip speed conditions and two rotor solidities, which constituted a family of rotor options representative of conventional single rotor designs:

- o Selected Tip Speeds: V_t = 615 ft/sec.
  = 670 ft/sec.
  = 725 ft/sec.
- o Selected Rotor Solidities: σ = 0.14
  = 0.16

In order to evaluate the relative power required relationships for high altitude hover on a single engine against the power required for forward flight on two engines at Sea Level, the calculated power requirements were resolved to a common basis of comparison -- the required engine installation based on the maximum power rating at Sea Level, Standard Day atmospheric conditions.

The adjusted power requirements were grouped in bands which bound the range of tip speeds and rotor solidities under consideration, and plotted as a function of rotor diameter for the various combinations of gross weight and performance criteria. This approach facilitated the cross-plotting of hover requirements against high speed forward flight requirements, in terms of the required engine installation versus rotor diameter for the various combinations of solidity and tip speed. The impact of rotor diameter on required engine sizing becomes obvious, as discussed in the paragraphs which follow.

Hover Capabilities

Figure 1 illustrates a consolidation of the achievable hover criteria at Sea Level, 10,000 ft. and 20,000 ft. for the three selected gross weights. It can be seen that the 10,000 ft. hover criteria at a Design Gross Weight of 8,000 lb. represents the driving requirement on rotor diameter versus power required. The power requirements begin to increase rapidly as the rotor diameter decreases below 30 ft. and the optimum diameter appears to be in the 35 ft. to 45 ft. range. The symbols (◆) illustrate three representative design alternatives which satisfy the hover criteria at a Design Gross Weight of 8,000 lb. from Sea Level to 10,000 ft., Standard Day conditions, and also satisfy the 20,000 ft. hover criteria at the Minimum Mission Weight of 6,000 lb. These three options trade off increases in rotor diameter from 30 ft. to 40 ft. to achieve reductions in required installed engine power from 4,000 to 3,000 shp. In each case, the design option is capable of meeting a single engine hover criteria at the 10,000 lb. overload or Alternate Design Gross Weight at Sea Level.

Forward Flight Capabilities

The relationship between rotor diameter and required installed power is illustrated in Figure 2 for a Design Gross Weight of 8,000 lb. for the two specified speed requirements of 200 knots maximum continuous cruise (both engines at 85 percent power) and 300 knots 30-minute dash speed (both engines at 95 percent power) at Sea Level, Standard Day atmospheric conditions. As in the hover tradeoffs, the power required for the given flight condition has been resolved to a required power installation based on Sea Level, Standard Day conditions. The forward speed requirements are based on flight with both

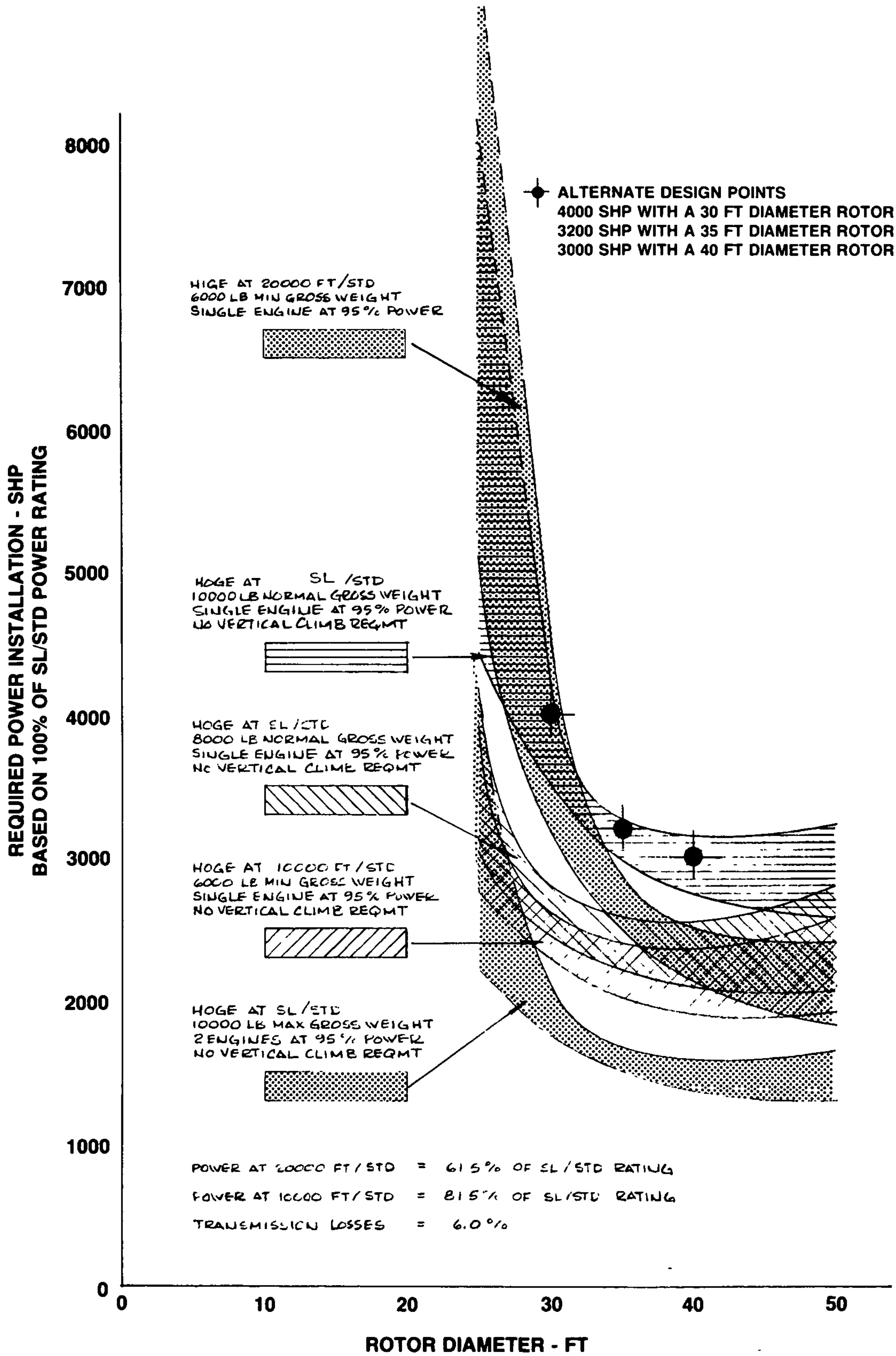

FIGURE 1. REQUIRED POWER INSTALLATION FOR HOVER CRITERIA

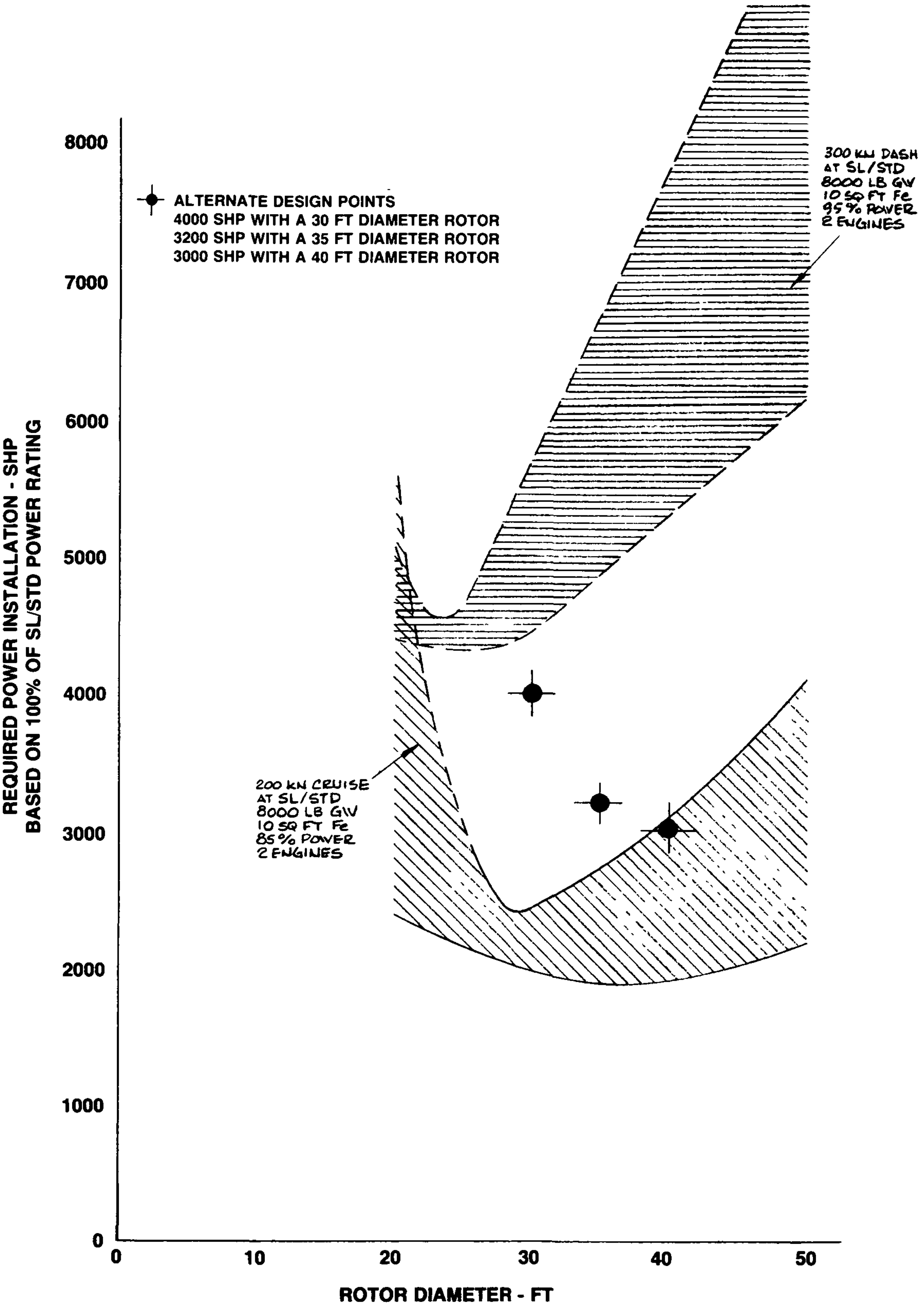

FIGURE 2. REQUIRED POWER INSTALLATION FOR FORWARD FLIGHT SPEED CRITERIA: 200 AND 300 KNOTS

engines in operation, whereas the hover criteria were based on a single engine requirement. Therefore, in the forward flight condition, the power required is corrected only for degradation between overhauls, power setting, engine installation and transmission losses to determine the required engine installation sizing.

The broad band for each speed represents the various combinations of rotor solidities and tip speeds considered representative of typical single rotor helicopter designs. A pronounced minimum in the required power installation occurs with a 29 ft. diameter rotor at 200 knots cruise and a 24 ft. diameter rotor for the 300 knots dash speed. The pronounced minimums in the required power installations for the high speed criteria are the combined effects of compressibility and blade stall on the rotor at high flight speeds. The optimum design points occur at the minimum values. In order to best satisfy both speed requirements, an intermediate rotor diameter lying between the two minimums should be selected.

Comparison of Hover and Forward Flight

Figure 3 illustrates a comparison of the required power installation necessary to satisfy the requirement of Hover-Out-of-Ground-Effect at 10,000 ft. on one engine at the selected Design Gross Weight of 8,000 pounds and the forward flight criteria as a function of rotor diameter. The regions on the chart defined by the overlap of the hover criteria and the forward flight criteria illustrate those combinations of minimum rotor diameter and multi-engine power installation required to satisfy both conditions. This figure indicates that a rotor diameter in the range of 40 to 50 ft. and approximately 3,000 shp of installed power satisfy the combined hover/cruise criteria, while a rotor diameter in the 27 to 30 ft. range and approximately 4,500 shp of installed power best satisfy the combined hover/dash criteria.

Effects of Maneuver Loading

The previously discussed tradeoff relationships were based on operations at a maximum rotor lift coefficient, C_T/σ , which allowed no margin for maneuver loadings. This results in a high degree of stall sensitivity to gusts and random variations in airflow. Table 2 summarizes the relationships between selected values of rotor tip speed, maximum permissible forward flight speed at an advancing blade tip Mach number of 0.95, blade load, and required blade area for an 8,000 lb. helicopter for maneuver load factors of 1.0 g, 2.0 g, and 3.0 g. The three maneuver load conditions were based on the load factor occurring at the maximum forward flight speed indicated and presumed that no auxiliary lift systems were employed to off-load the rotor.

Figure 4 illustrates the conversion of blade area into maneuver load factors for an 8,000 lb. helicopter at selected speeds ranging from 100 knots to 300 knots, and the conversion of blade area into rotor diameter for six values of rotor solidity ranging from 0.1 to 0.2. The figure was based on the advancing blade tip speed being limited to a Mach number of 0.95 and the required blade area computed based on operation at a maximum rotor lift coefficient (C_T/σ) as a function of the square of the advance ratio (μ).

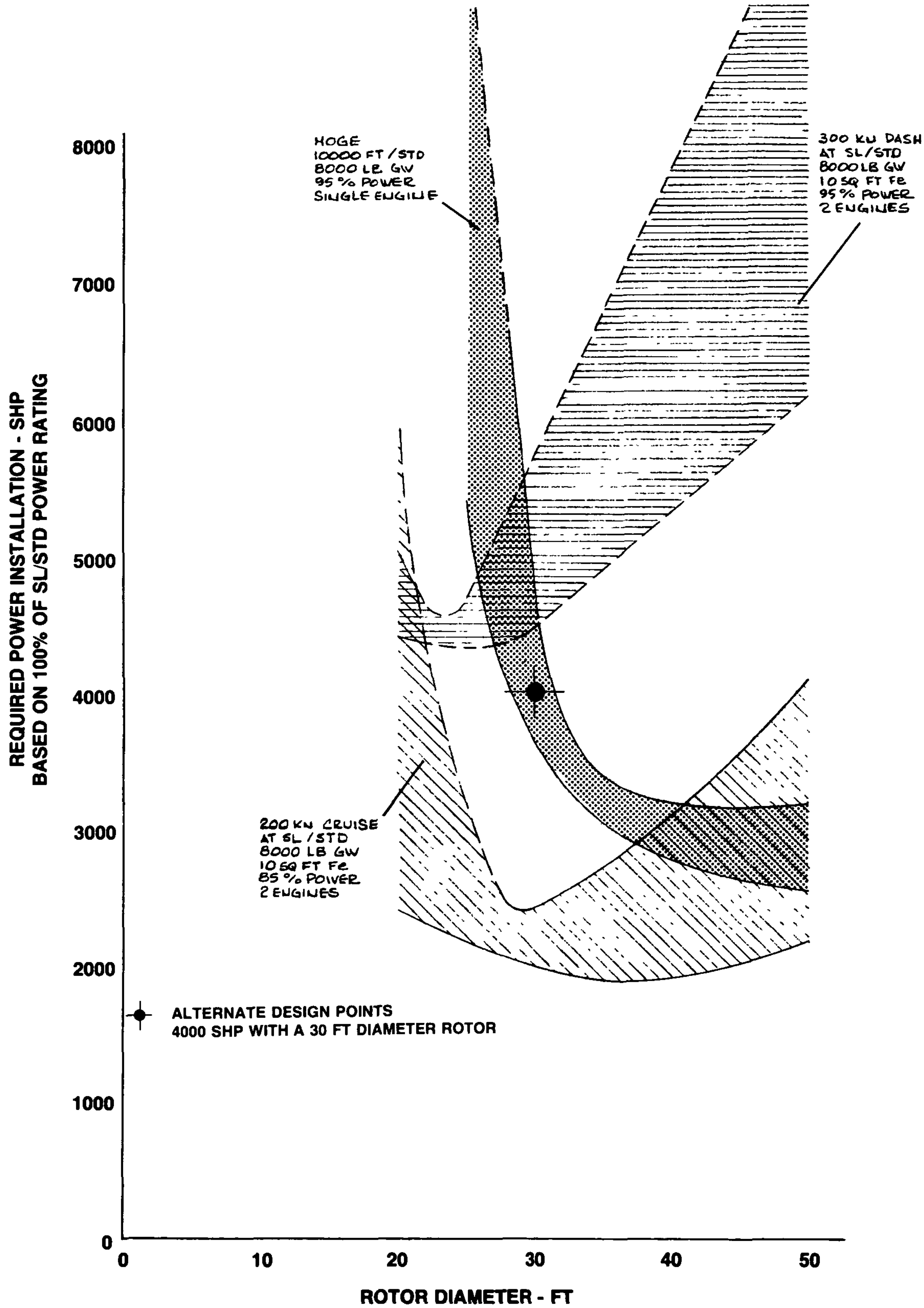

FIGURE 3. COMPARISON OF REQUIRED POWER INSTALLATION FOR HOVER AND FORWARD FLIGHT SPEED CRITERIA

TABLE 2

REQUIRED BLADE AREA TRADEOFFS

Rotor Tip Speed ft/sec	Advancing Blade Tip Mach Number	Maximum Forward Flight Speed - knots	Advance Ratio-μ	Maximum Rotor Lift Coefficient-$(Ct/\sigma)_{max}$	Maximum Allowable Blade Loading - lb/sq ft	Required Blade Area for an 8000 lb Aircraft at a 1 0g Maneuver at Maximum Speed	Required Blade Area for an 8000 lb Aircraft at a 2 0g Maneuver at Maximum Speed	Required Blade Area for an 8000 lb Aircraft at a 3 0g Maneuver at Maximum Speed
500 (3)	0 95	331 9	1 1204	0 00000	0 00	_ (4)	_ (4)	_ (4)
550 (3)	0 95	302 3 (1)	0 9276	0 01674	12 04	664 7	1329 4	1994 1
600 (3)	0 95	272 7	0 7670	0 04940	42 28	189 2	378 5	567 7
615 (2)	0 95	263 8	0 7239	0 05711	51 35	155 8	311 6	467 4
650 (3)	0 95	243 0	0 6310	0 07221	72 52	110 3	220 6	331 0
670 (2)	0 95	231 2	0 5824	0 07930	84 61	94 6	169 2	253 8
700 (3)	0 95	213 4	0 5146	0 08823	102 76	77 86	155 7	233 6
725 (2)	0 95	198 6 (1)	0 4623	0 09435	117 88	67 87	135 7	203 6
750 (3)	0 95	183 8	0 4136	0 09947	133 00	60 15	120 3	180 5

Notes

(1) Approximately Meets Specified User Requirement
(2) Used in Performance Assessment
(3) Included to Show Trend
(4) Approaches Infinity
(5) Sea Level Standard Day conditions

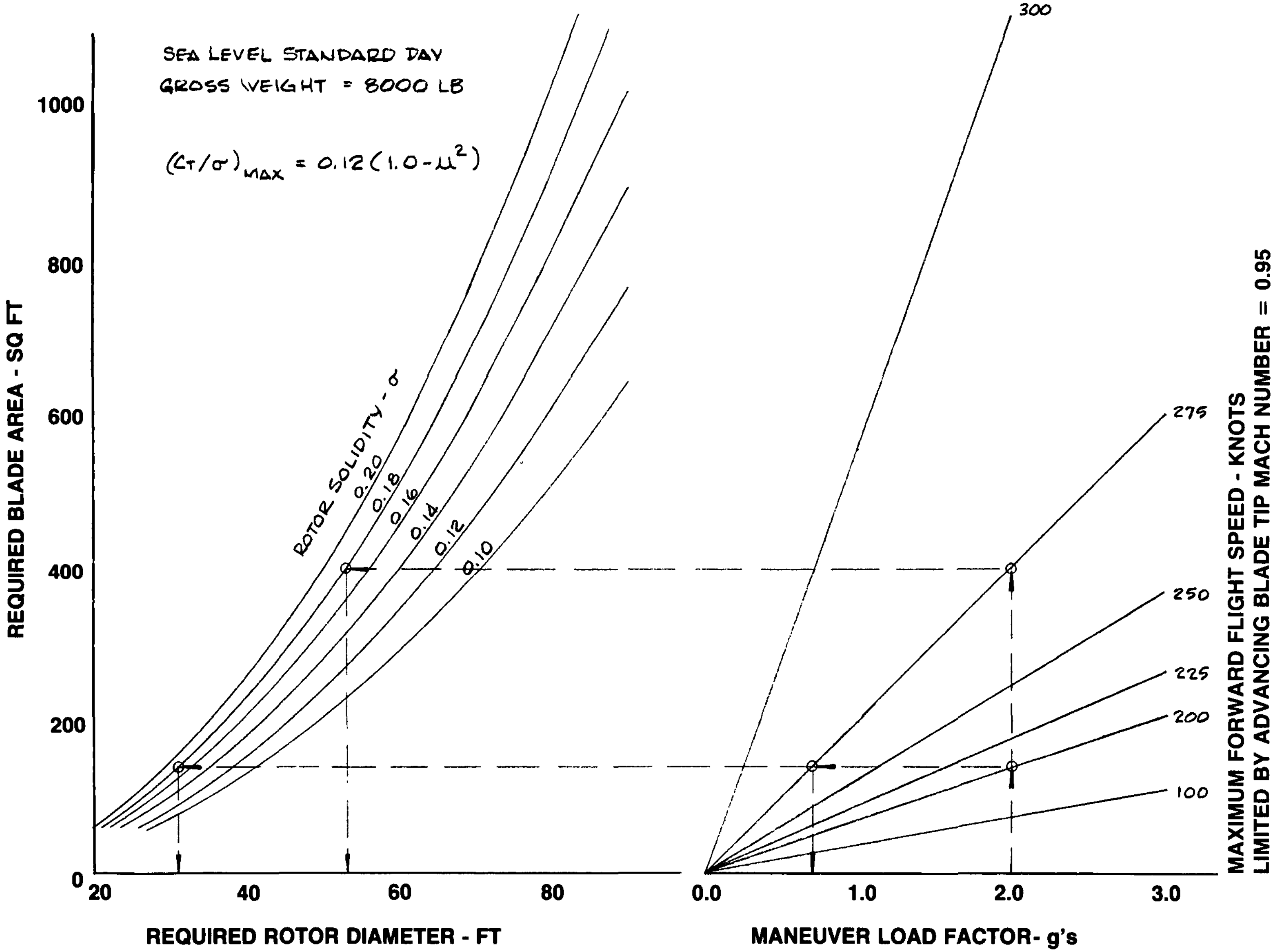

FIGURE 4. MANEUVER LOAD RELATIONSHIPS

The example shown in Figure 4 selects a 2.0 g maneuver capability at 200 and 275 knots, which requires 140 sq. ft. of blade area at 200 knots and 400 sq. ft. of blade area at 275 knots. If the rotor solidity is 0.18, the required rotor diameters are 31 ft. and 53 ft., respectively.

ALTERNATE ROTORCRAFT DESIGNS

In addition to an assessment of a conventional single rotor helicopter, ORI considered other vehicle design alternatives such as an Unconventional Single Rotor Concept, an Advancing Blade Concept, and a Tilt Rotor Concept.

Unconventional Single Rotor Helicopter

The conventional single rotor helicopter design tends to reach the limit of its lift potential at forward speeds above 200 knots due to the combined effects of blade stall and compressibility. These effects converge with increasing flight speed to degrade the maximum lift potential of the rotor system or the maximum allowable blade loading. As a result, the maneuver margins of the aircraft steadily decrease with increasing flight speed due to retreating blade stall.

The typical design approach for solving this problem is to increase the blade area of the rotor or add a fixed wing to augment or unload the rotor in high speed flight. These options, however, tend to degrade the hover performance of the aircraft to some extent. However, each incremental increase in blade area required for maneuver margin or high speed flight beyond the optimum rotor design required for hover and low speed flight weighs approximately 17 times the aerodynamic equivalent wing area, assuming the weight per unit area of the wing area and the additional blade area are approximately equal. The comparative weight argument suggests sizing the rotor according to the hover and low speed flight requirements only and augmenting the rotor with a wing for all maneuver and high speed flight requirements. This approach can be used to some extent to improve speed capabilities, but frequently leads to a very lightly loaded rotor with most of the lift at high speeds generated by the wing. This condition leads to a complex set of control problems governing the spatial relationships between the lifting fuselage and the non-lifting rotor system at high flight speeds. Also, auxiliary propulsion systems may be required and selected on the basis of the maximum required speed and total aircraft drag. This added weight has an adverse effect on the aircraft empty weight fraction and hover capabilities.

The basic problem with the high speed flight requirement, however, is independent of the design of the wing and the appropriate selection of an auxiliary propulsion system. The problem evolves to the dilemma of how to reduce the loads on the rotor in order to avoid stall while simultaneously maintaining adequate loads to assure rotor control. A possible solution to this dilemma is to incorporate substantial or severe forward shaft tilt such that the rotor is operating at extreme angles of attack in high speed flight, as shown in Figure 5. Operation of the rotor at high angles of attack, on the order of 20 to 30 degrees, should tend to reduce the adverse effects of retreating blade stall and compressibility to some extent, while maintaining an increased load on the rotor to assure adequate control. The increased load on the rotor, in the form of the resultant thrust vector, is comprised of the forward thrust component and the vertical thrust component. At a 30 degree angle of attack, or shaft tilt, 8,000 lb. of total rotor thrust would provide a 4,000 lb. propulsive force and a 6,930 lb. lift force. While this tends to degrade the vertical lift force by 15.4 percent, the lift would be lost anyway due to blade stall at conventional angles of attack and can be provided by an appropriately designed wing. The forward propulsive thrust component of 4,000 lb. is sufficient to sustain a 300 knot speed capability for an aircraft with 13.1 sq. ft. of equivalent drag area with no requirement for auxiliary propulsion systems.

Such severe shaft tilt, however, would adversely affect hover performance, necessitating some form of variable tilt mechanism to restore a nearly vertical shaft orientation in hover. This, in turn, tends to complicate the control system throughout the conversion range from a vertical position to a 30 degree shaft tilt as the rotor transitions from conventional rotor control to a hybrid prop/rotor control system. In addition, the rotor control system would have to be integrated with a fixed wing control system to provide a coordinated maneuver capability.

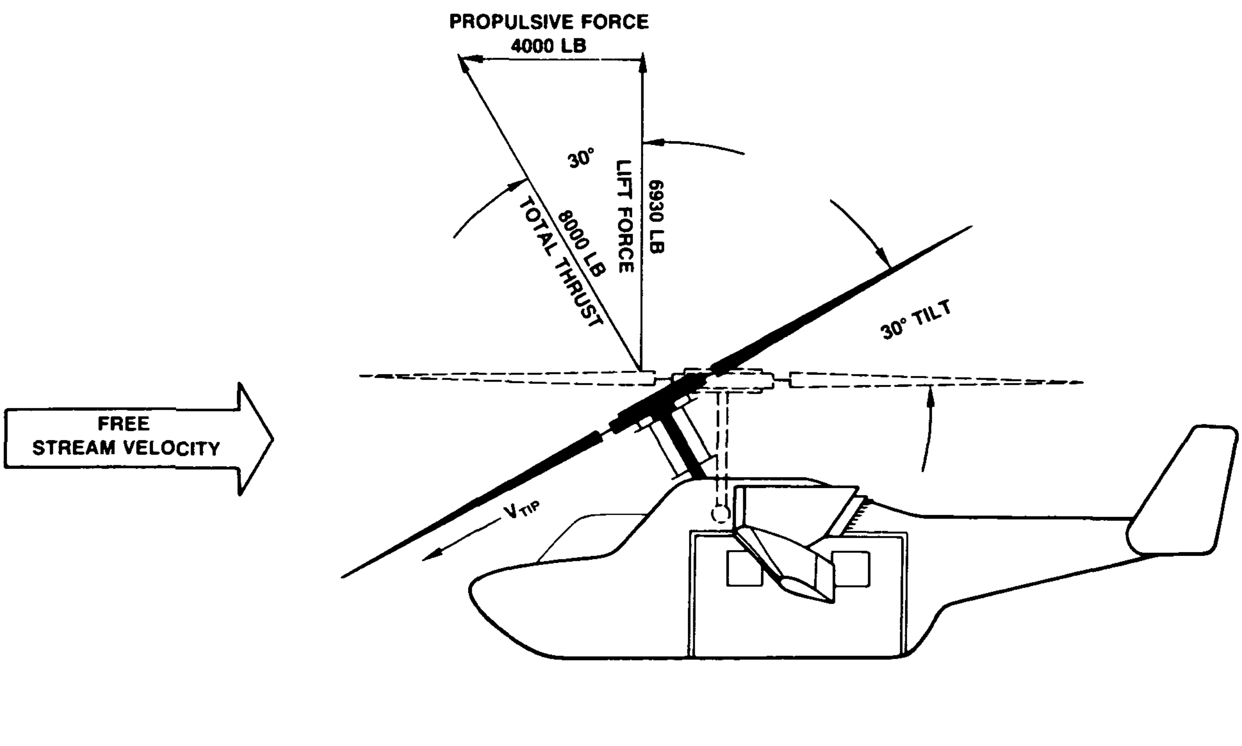

FIGURE 5. VARIABLE TILT SHAFT CONCEPT

The development of such a system, however, could offer the maximum speed potential of the Advancing Blade Concept and the tilt rotor designs without the attendant empty weight penalties, increased size, increased downwash velocities or auxiliary propulsion requirements. Based upon preliminary conceptualization of the design, the unconventional variable shaft tilt single rotor helicopter design appears to offer many solutions to the consolidated requirements of the PSH users.

Advancing Blade Concept

The Advancing Blade Concept (ABC) is a special type of coaxial rotor system comprised of two closely-spaced counter-rotating rigid rotors coupled together through a substantial rigid main rotor shaft assembly. The ABC functions as a pure coaxial helicopter in hover; but in forward flight, the ABC begins to feather the retreating blades of both the upper and lower rotor such that at some high forward flight speed, the retreating blades are fully feathered and lift is produced with only the advancing blades of both rotors. By this approach, the ABC rotor system effectively eliminates the problem of retreating blade stall at high forward flight speeds.

The limiting advancing blade tip Mach number for the XH-59 ABC research aircraft is 0.85. At the reported tip speed of 650 ft/sec, in the pure helicopter configuration, the advancing blade tip Mach number limits the maximum speed to less than 177 knots. In order to achieve higher flight speeds, the ABC must be equipped with auxiliary propulsion systems. The published speed capability for the ABC with auxiliary propulsion is 280 knots.

The ABC rotor design offers excellent rotor performance in both hover and high speed flight in terms of rotor efficiency, based on the power required at a given gross weight. The improved rotor efficiency in hover results from the improved efficiency of the coaxial rotor system compared to a conventional single rotor design and the elimination of the tail rotor for torque control. The improved high speed efficiency is due to the reduction of retreating blade stall, enabling the rotor to operate at a relatively constant lift coefficient (C_T/σ) at all forward flight speeds.

The most significant disadvantage of the ABC for potential PSH applications is the relatively high empty weight fraction, which severely limits the useful load. The empty weight of the XH-59A was reported as 8,060 lb. without auxiliary engines, which is 73.3 percent of the 11,000 lb. Design Gross Weight. The weight of the rigid rotor, flight control and drive system comprises 4,275 lb., or 53 percent of the aircraft empty weight.

In order to meet an empty weight objective of 4,500 lb. for a 10,000 lb. Design Gross Weight aircraft, the empty weight of the ABC aircraft must be reduced by 3,560 lb., or 44 percent, for a PSH production version. This indicates that the design comprises associated with an operational ABC design may be more severe than the tradeoffs associated with a conventional or compound single rotor helicopter design.

Tilt Rotor Aircraft

The tilt rotor aircraft, such as the XV-15, is a laterally displaced twin rotor convertiplane design. The current prototype employs two counter-rotating, 25-ft. diameter rotors mounted on rotating engine/transmission nacelles at the tips of a high mounted fixed wing with a span of approximately 30 ft. The rotors and nacelles rotate from a vertical shaft orientation to produce lift in hover to a horizontal shaft orientation to produce propulsive thrust in forward flight. Lift in forward flight is provided solely by the wing.

The principal advantage of the tilt rotor design relative to the specified PSH requirements is its speed capability. Since the rotors of the tilt rotor aircraft rotate from a vertical shaft orientation in hover to a horizontal shaft orientation in forward flight, the rotor systems are not subjected to high oblique flow conditions for extended periods of time. As a result, the tilt rotor system does not generally experience the speed limitations due to compressibility and retreating blade stall effects characteristic of conventional helicopters at high flight speeds.

The principal disadvantages of the tilt rotor design with respect to the PSH requirements are:

- The empty weight fraction and its impact on aircraft sizing, and thrust required at the useful loads specified by the users.
- Its physical size.
- The downwash velocity at the thrust levels dictated by the combination of empty weight and required useful load.

The empty weight of the current 13,000 lb. tilt rotor research aircraft is 9,600 lb., giving an empty weight fraction of 73.8 percent. In order to achieve a 10,000 lb. tilt rotor design with an empty weight of 4,500 lb., a 53 percent decrease in the empty weight of the current prototype would be required.

The laterally displaced rotors of the XV-15 research aircraft measure over 57 feet in width from blade tip to blade tip with rotors turning, which is significantly larger than the 20 foot diameter rotor span specified by the PSH users. While the tilt rotor design offers a potential solution to the 300 knot dash speed requirement, sizing the tilt rotor design to meet the useful load requirements will probably result in an unacceptably large aircraft and intolerable downwash velocities for many PSH missions.

CONCLUSIONS

Based on the above, it is concluded that:

1. When taken individually, the NASA Rotorcraft Technology Program is addressing nearly all the PSH stated needs. However, to combine two or more of the needs into the same aircraft requires either tradeoff in performance or new thrusts to advance the state of the art.

2. Most of the vehicle characteristics for a Public Service Helicopter are similar to the Army's program objectives for development of the LHX-Utility helicopter, especially in weight, size, and speed boundaries.

3. The major design conflicts involving the PSH users' stated needs arise from combining requirements for load capacity, performance and rotor size.

4. To meet the most stringent load requirement of 4,000 lb. payload plus 2 hours fuel, the aircraft must be sized at or near the 10,000 lb. gross weight limit. The next most stringent load requirement of 6 passengers plus 4 hours fuel can be met with an aircraft sized at 4,300 lb. Design Gross Weight.

5. To meet the hover and speed requirements, the minimum feasible rotor diameter is 30 feet; but the optimum design region is in the 35-45 foot range.

6. The high altitude single engine hover criteria appears to be achievable, but results in an unusually high installed power requirement which may not be acceptable for all PSH users.

7. The 200 knot cruise speed requirement appears to be feasible with a conventional single rotor helicopter design employing 3,000 to 3,200 shp of installed power and a 35 to 45 ft. diameter rotor, as required to meet the PSH single engine hover requirements.

8. The requirement for a 300 kn dash speed capability cannot be met with a conventional single rotor helicopter regardless of the installed power, due to the effects of compressibility and retreating blade stall. In order to satisfy the 300 kn dash speed requirement, unconventional single rotor and/or advanced multirotor design options must be employed.

9. Advanced multirotor rotorcraft designs, such as the Advancing Blade Concept coaxial rotor design and the Tilt Rotor convertiplane design may offer the dash speed specified by the PSH users' group, but both options present significant disadvantages in terms of empty weight fractions and high downwash velocities which render them unsuitable in terms of the specified loading requirements. In addition:

   a. The Advancing Blade Concept coaxial rotor design falls 7 percent short of the desired 300 kn dash speed capability, even with auxiliary propulsion systems. In this configuration, the ABC compound design offers only a 12 percent increase in maximum speed over the potential of an equivalent single rotor design, but at a substantial deficit in load carrying capacity.

   b. The Tilt Rotor design appears more than capable of meeting the 300 kn dash speed requirement, but substantially exceeds the dimensional requirements of the users.

10. The single rotor helicopter design may represent the best compromise in attempting to satisfy the various PSH user requirements. In order to achieve the 300 kn dash speed requirement, it may be necessary to:

   a. Develop rotor systems conducive to sustained operation at supersonic tip speeds to alleviate the problem of retreating blade stall at high forward flight speeds. This could be achieved with current technology, but at considerable expense in terms of compressibility effects, power required and noise. New research initiatives directed at developing optimum rotor designs for supersonic flow conditions could serve to reduce the adverse effects of compressibility at these speeds.

   b. Develop unconventional single rotor design alternatives, such as thrust and lift compounded configurations and/or variable shaft tilt designs, to minimize the problems of retreating blade stall and compressibility effects at the 300 knot speed. Continued development of convertible engines will serve to further support this requirement.

RECOMMENDATIONS

The following recommendations are made for NASA consideration:

1. The scope of the NASA Rotorcraft Technology Program should be expanded to include:

   a. Research in supersonic rotors for use on high speed helicopters.

   b. Investigation of advanced concepts, such as variable shaft tilt single rotor designs and potential combinations of auxiliary lift and propulsion systems, to provide an optimum design configuration for forward speeds up to 300 knots without excessively degrading other design parameters.

2. PSH users should be provided with the results of the needs evaluation and provided a forum for revising their stated needs.

3. A Public Service Helicopter Technology and Systems Plan should be developed to meet the revised technology needs of the PSH users.

I. INTRODUCTION

The Public Service Helicopter (PSH) User's Workshop held at the NASA Ames Research Center on July 14-16, 1980, examined the uses and benefits of public service helicopters in four mission areas -- Search and Rescue (SAR), Emergency Medical Services (EMS), Law Enforcement and Public Safety, and Environmental Control and Fire Fighting.[1] A working group for each mission area considered current problems, desired vehicle characteristics and performance capabilities, and related technology needs in seven technology areas -- Vehicle Design, Propulsion, Safety and Reliability, Navigation/ Guidance and Flight Controls, Auxiliary Systems, Human Factors, and Monitoring and Diagnostic Systems. The results of the working group sessions were used by workshop participants to identify a list of needs for each technology area. Finally, selected members of the individual working groups met to develop a consolidated list of needs for future public service helicopters.. These consolidated needs were reconfirmed during an Emergency Medical Services (EMS) Rotorcraft Technology Workshop held in Washington, D.C. on October 14-15, 1981. They are reproduced in Table 3.

Many of the needs listed in Table 3 appear to be within the state-of-the-art and could be met by application of available technology. Other needs require some advances in technology to achieve the desired capability. In some instances, the needs impact upon one another, requiring either a trade-off or significant advancement in technology to achieve a feasible system design.

In support of NASA planning activities in aeronautical research, ORI conducted an independent analysis of the list of needs developed at the

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[1]Helicopter Technology Needs, Public Service Helicopter User's Workshop, NASA Ames Research Center, July 14-16, 1980, Volume I - Summary, Volume II - Appendices.

TABLE 3

PUBLIC SERVICE HELICOPTER USER'S WORKSHOP LIST OF HELICOPTER TECHNOLOGY NEEDS

VEHICLE DESIGN
1. Increase speed (300 kn dash 30 min max/200 kn max continuous)
2. HIGE 20,000 ft (single engine)
3. HOGE 10,000 ft (single engine)
4. Twin engine
5. Endurance - 4 hours
6. 10,000 lb max G.W.
7. 20 ft rotor diameter
8. Eliminate tail rotor
9. Internal cabin area (5.00 ft high x 4.33 ft wide x 8.00 ft long)
10. Modularized cabin
11. Pressurization*
12. Autorotation capability
13. Internal & external noise reduction
14. Pilot operated hoist
15. Compatible electrical system
16. Shutdown power capability
17. Quick access maintenance
18. Water/retardant capability
19. Improved all-terrain landing gear
20. Improved visibility
21. Improved maneuverability
22. Sliding cargo door
23. Internal access to cargo cabin
24. Equipment storage
25. Cold interior lighting
26. Hot refueling capability

*Optional

PROPULSION
1. Non-petroleum fuels
2. Multiple fuel capability
3. Low fuel consumption
4. Dual power band
5. Increased shaft HP
6. Lightweight power plant
7. Emergency power capability
8. Particle separators (FOD proof)
9. Main rotor clutch
10. Minimal warm-up time

SAFETY & RELIABILITY
1. Crashworthy structure
2. Crashworthy seats
3. Crashworthy fuel system
4. Eliminate dynamic roll-over
5. Improved restraint system
6. Improved helmets
7. Improved egress system
8. Increased main rotor clearance
9. Reduced tail rotor hazard (remove tail rotor)
10. Birdstrike protection
11. Removable ballistics protection & detection
12. Fuel dumping capability
13. Fire protection
14. Hazardous material store

NAVIGATION GUIDANCE & FLIGHT CONTROLS
1. Automatic flight control
2. Combined controls
3. Stabilization
4. All-weather capability
5. Low airspeed measurement
6. Electronic map display
7. Precision location/navigation

AUXILIARY SYSTEMS
1. Hoist locations & capabilities
2. Rappel attachments
3. Improved litter
4. Litter suspension
5. Night vision system
6. Improved searchlight
7. Optical equipment
8. Photo/TV equipment
9. On-board APU
10. A/C visual identification
11. Car identifier
12. Car lock-on
13. Car stopper
14. Towing equipment

HUMAN FACTORS
1. Improved seats
2. Environmental control
3. Noise and vibration
4. Control standardization
5. Dual controls
6. Visibility
7. Integrated flight instruments

MONITORING & DIAGNOSTIC SYSTEMS
1. Trend warning
2. Computerized monitoring system
3. Warning/caution system
4. Color-coded annunciation
5. Aural warning
6. Head-up display
7. Performance limitations

July 1980 workshop.[2] This independent analysis was performed as two discrete subtasks -- an initial review of the needs identified by the workshop and then an examination of relationships and impacts between critical design needs. In the first subtask, ORI compared each of the stated needs with the current state-of-the-art to determine whether the requirement was a design need or technology need. The selection criteria was based upon the availability of technology, not user application of the technology. The results of that classification process are summarized in Table 4.

In the second subtask, ORI found that certain desired vehicle design characteristics impact adversely upon other desired characteristics. The most significant conflicts involve the stated needs for increased speed (300 kn. dash/200 kn. maximum continuous cruise), hover capabilities at 10,000 ft. and 20,000 ft., a 20 ft. rotor diameter, and sizing parameters of an aircraft with a maximum gross weight at or near 10,000 lbs.

The disaggregation of the PSH user needs into design needs and technology needs provides a focus on technology areas where NASA's research capabilities could contribute to achieving the PSH user needs. The next step involved a comparison of the technology needs for public service helicopters with the current NASA rotorcraft technology program, and related activities, to determine specific areas where new technology initiatives may be required.

ORI was requested to expand upon the scope of work covered in the December 1983 report to compare PSH technology needs with NASA's current rotorcraft research and with relevant program objectives of the U.S. Army's Light Helicopter/Experimental (LHX) program. In addition, ORI was requested to examine vehicle technical/performance trade-offs to identify various options available among critical design parameters. The comparison of PSH technology needs with NASA programs and Army Aviation development objectives are presented in Chapters II and III. Technical assessments of key vehicle design parameters, design and performance trade-offs, and alternative rotorcraft designs are provided in Chapters IV through VI. Conclusions and recommendations are shown in the final chapter.

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[2]Bauchspies, J.S. and Simpson, W.E., Analysis of Technology Needs for Public Service Helicopters, Final Report, Prepared by ORI, Inc. for the National Aeronautics and Space Administration, Washington, D.C., December, 1983.

TABLE 4

DISAGGREGATION OF DESIGN AND TECHNOLOGY NEEDS FOR PUBLIC SERVICE HELICOPTERS

Technology Area	Design Needs	Technology Needs
Vehicle Design	Twin Engine Endurance - 4 Hours 10,000 lb. Max. GW Internal Cabin Area (60″ High × 52″ Wide × 96″ long) Modularized Cabin Pressurization Autorotation Capability Pilot Operated Hoist Compatible Electrical System Shutdown Power Capability Quick Access Maintenance Water/Retardant Capability Improved All Terrain Landing Gear Improved Visibility* Improved Maneuverability* Sliding Cargo Door Internal Access to Cargo Equipment Storage Cold Interior Lighting Hot Refueling Capability	Increased Speed (300 kn Dash, 200 kn Max. Continuous) HIGE 20,000 ft. (Single-Engine) HOGE 10,000 ft. (Single-Engine) 20 ft. Rotor Diameter Eliminate Tail Rotor Internal & External Noise Reduction
Propulsion	Multiple Fuel Capability Emergency Power Capability Main Rotor Clutch Minimal Warm-Up Time	Non-Petroleum Fuels Low Fuel Consumption Dual Power Band Increased Shaft HP Lightweight Power Plant. Particle Separation (FOD Proof)*

*Needs Further Definition

TABLE 4 Continued

Technology Area	Design Needs	Technology Needs
Safety and Reliability	Crashworthy Seats (Adjustable/Swivel and 20 g Impact Resistance) Crashworthy Fuel System Eliminate Dynamic Roll-Over* Improved Restraint System Improved Helmets Improved Egress System Increased Main Rotor Clearance Birdstrike Protection Removable Ballistics Protection Fuel Dumping Capability Fire Protection* Hazardous Material Storage	Crashworthy Structures (Composite With 20 g Impact) Reduced Tail Rotor Hazard (Remove Tail Rotor Included Under Vehicle Design)
Navigation Guidance and Flight Controls	Automatic Flight Controls* Stabilization	Combine Controls All-Weather Capability Low Airspeed Measurement Electronic Map Display Precision Location/Navigation

*Requires Further Definition

TABLE 4 Continued

Technology Area	Design Needs	Technology Needs
Auxiliary Systems	Hoist Locations and Capabilities Rappel Attachments Improved Litter Litter Suspension Improved Searchlight Optical Equipment Photo/TV Equipment On-Board APU A/C Visual Identification Towing Equipment	Night Vision System* Car Identifier* Car Lock-On* Car Stopper*
Human Factors	Improved Seats Environmental Controls Control Standardization Dual Controls Visibility*	Noise and Vibration Integrated Flight Instruments
Monitoring and Diagnostic Systems	Warning/Caution System Color-Coded Annunciation Aural Warning	Trend Warning Computerized Monitoring System Head-Up Display Performance Limitation Monitor

*Requires Further Definiton

II. COMPARISON OF NASA PROGRAMS AND PUBLIC SERVICE HELICOPTER TECHNOLOGY NEEDS

The technology needs for public service helicopters were compared with NASA's current rotorcraft research and technology activities. These activities are grouped into two program areas: Research and Technology (R&T) Base and Systems Technology Programs.[1] The R&T Base program area comprises activities oriented toward establishing a solid base of aeronautical technology in all of the relevant disciplines. The Systems Technology Program seeks to advance and to accelerate the transfer of technology for user application in civil and military rotorcraft.

Specific activities are documented as work areas for the research centers in Research and Technology Objectives (RTOPS). The NASA Aeronautics Program for fiscal year 1984 includes 14 projects which focus on various aspects of rotorcraft technology. Seven of these projects are in the R&T Base area and seven are in the Rotorcraft Systems Technology program area. A comparison of the technology needs for public service helicopters (PSH) with NASA's approved objectives and plans indicates that some aspects of most of the PSH technology needs are being addressed by the NASA Aeronautics Program (Table 5). However, these current NASA programs do not focus on all of the technology needs for public service helicopters in the following areas:

- Vehicle Design
- Propulsion
- Auxiliary Systems
- Monitoring/Diagnostic

The pertinent aspects of these PSH technology needs and current NASA research objectives and plans are discussed in the subsections which follow.

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[1] Aeronautics Research and Technology Program and Specific Objectives, Fiscal Year 1984, Office of Aeronautics and Space Technology, National Aeronautics and Space Administration, Washington, D.C., March 28, 1983.

TABLE 5

COMPARISON OF PSH TECHNOLOGY NEEDS AND
NASA ROTORCRAFT TECHNOLOGY PROGRAM

Identified Technology Needs For Public Service Helicopters (By Technology Areas)	NASA RTOP Number/Title	NASA Research and Technology Projects: R&T Base Projects							NASA Research and Technology Projects: Rotorcraft Systems Technology						
		505-40-12 Fan & Compressor Research	505-40-22 Combustors & Turbines	505-40-42 Power Transfer Research	505-42-11 R/C Aeromechanic & Perform	505-42-23 R/C Airframe Systems	505-42-32 R/C Operating Problems	505-42-61 R/C Flight Test Operations	532-01-11 R/C Flt Guidance Sys Tech	532-03-11 RSRA Flt Research/ Rotors	532-06-11 R/C System Integration	532-06-12 Convertible Engine Sys Tech	532-06-13 R/C Vibration & Noise	532-09-10 RSRA/X-Wing R/C Flt Invest	532-09-11 Adv Tilt Rotor Research
Vehicle Design															
Increase Speed, 200-300 kn					X		X	X		X	X	X		X	X
HIGE-20,000 ft. (SE)															
HOGE-10,000 ft (SE)															
Rotor Diameter-20 ft.															
No Tail Rotor															X
Reduce Int. & Ext Noise				X	X	X	X			X	X		X		
Propulsion															
Non-Petroleum Fuels			X												
Low Fuel Consumption		X					X								
Dual Power Band							X					X			
Increase Shaft HP							X								
Lightweight Powerplant		X													
Particle Separation*															
Safety & Reliability															
Crashworthy Structures						X									
Reduce Tail Rotor Haz.															X
Nav., Guidance & Flt Cont.															
Combine Controls									X						
All-Weather Flt					X			X	X						
Low Airspeed Measur									X						
Electronic Map Display									X						
Precise Loc./Nav									X						
Auxiliary Systems															
Night Vision System															
Car Identifier*															
Car Lock-On*															
Car Stopper*															
Human Factors															
Noise & Vibration					X	X							X		
Integrated Flt Instr.					X				X		X				
Monitoring/Diagnostics															
Trend Warning											X				
Computerized Monitoring		X									X				
Head-Up Display															
Perf Limit Monitor					X						X				

*Requires Further User Definition

VEHICLE DESIGN NEEDS

There are eleven projects which include research objectives to reduce noise and improve performance. While each of the individually stated needs may be potentially achievable, the design of the desired vehicle would involve trade-off considerations. These trade-offs are discussed in Chapter IV. More importantly, Table 5 does not show any particular NASA projects with technical objectives which focus on the desired hover capabilities or rotor diameter as specific vehicle design parameters.

NASA recently awarded study contracts to four helicopter manufacturers -- Bell Helicopter Textron, Boeing-Vertol, Hughes Helicopters, and Sikorsky -- to study helicopter application and equipment for EMS missions.[2] The scope of the studies includes mission requirements, potential market, benefit assessments, and identification of technologies ready or nearly ready for development, which are judged to be useful for EMS helicopter applications. The technologies to be examined by the manufacturers will emphasize precision guidance, all-weather capability, low altitude performance, internal and external noise reduction, enhanced low speed/hover performance, vibration reduction, contingency power, and advanced transmissions. Highlights of other

NASA R&T activities which could contribute to reducing internal and external noise and to improving performance are discussed below.

Power Transfer Research

This effort at Lewis Research Center is directed toward advancing the state-of-the-art in transmissions and mechanical components such as bearings and gears. Reduction of noise in the mechanical power transmission systems for helicopters is one of the research goals. The program also includes improvements in component performance, weight, reliability and efficiency in the high-temperature and high-speed environments associated with power transfer in turbine powered rotorcraft and turboprop applications.

Rotorcraft Aeromechanics and Performance

Current and planned research at the Ames Research Center includes a broad scope of activities to improve fundamental understanding and develop techniques to design rotors optimized for aerodynamic performance and noise reduction. These activities include work to investigate the basic aerodynamic and aeroacoustic phenomena of rotary wings, the aeromechanical phenomena of rotors and rotorcraft, and to devise and develop advanced rotor concepts and configurations.

[2]NASA Ames Research Center announcement in the Commerce Business Daily, Issue No. PSA-8069, Friday, November 4, 1983.

Rotorcraft Airframe Systems

Research work on airframe systems at the Langley Research Center includes investigations of fundamental aeroacoustics, main rotor/tail rotor acoustics and acoustic interactions. The rotorcraft airframe systems project includes work on composite helicopter structures, rotor aeroelasticity, rotor unsteady airloads, and helicopter aerodynamic experiments and analyses.

Rotorcraft Operating Problems

Part of the NASA Rotorcraft Technology program being conducted at the Lewis Research Center is focused on advancing critical technology needed to solve propulsion, power transfer, and icing problems associated with the operation of military and civil rotorcraft. Current activities include a conceptual design and feasibility study of torque converters for high-speed rotorcraft propulsion systems, research directed at reducing internal cabin noise and vibration by development of low-noise gearbox design methodologies, systematic analytical evaluations of various contingency power concepts with some verification testing, and continuation of research on rotorcraft icing aimed primarily at establishing and verifying analytical/empirical techniques for use in ice prediction and protection methods.

Rotorcraft Flight Test Operations

The Ames Research Center conducts flight research programs involving research aircraft operations and flight experiment support facilities. The center performs a wide variety of analytical and experimental research in aerodynamics, flight dynamics and control, guidance and navigation, and avionics systems, with emphasis on advanced rotorcraft and powered lift aircraft.

RSRA Flight Research/Rotors

In cooperation with the U.S. Army, the Ames Research Center is utilizing the Rotor Systems Research Aircraft (RSRA), and other testbed aircraft, as appropriate, for rotor system flight tests. The scope of activity involves system design studies, analytical prediction methods, simulation, ground testing, and flight testing of advanced concept rotor systems. Current activities include work to develop a comprehensive experimental data base on rotor system aeroacoustics, vibration, performance, and interactions; demonstration of the RSRA fixed-wing flight envelope, and work to develop and evaluate new rotor system technology. A joint NASA/Army program to design and evaluate an Integrated Technology Rotor/Flight Research Rotor (ITR/FRR) is currently in the preliminary design phase. Plans are being formulated for a focused program to design, fabricate and test an advanced low noise rotor system.

Rotorcraft System Integration

This research at the Ames Research Center is a focused, integrated systems technology effort in aerodynamics, acoustics, vibration reduction, flight controls, crew station concepts, and advanced high-speed vehicle technology. Current activities include full-scale wind tunnel tests of rotor systems and fuselage configurations, investigation of concepts for advanced flight controls, human factor studies, and investigation and technology assessments of advanced vehicle investigations. The efforts on advanced vehicle configurations include study of civil mission applications of LHX (Light Helicopter, Experimental) technology.

Convertible Engine System Technology

Propulsion system research at the Lewis Research Center includes an experimental program undertaken jointly with the Defense Advanced Research Projects Agency (DARPA) to investigate new engine concepts which can efficiently supply power for higher speeds than are now being achieved with conventional helicopters. One promising concept called the "convertible engine" is being tested. The experimental convertible engine is a modified turbofan which can selectively supply power to a rotor for vertical lift or provide forward thrust during high-speed forward flight.

Rotorcraft Vibration and Noise

The Langley Research Center is developing technology for reducing the interior noise of helicopters through transmission/airframe isolation and for improving rotor noise prediction methods to establish the capability to design toward specified noise criteria. The focus on vibration reduction is oriented toward exploiting the full potential of modern analytical techniques, such as finite element modeling analysis, for predicting and controlling the vibration characteristics of new rotorcraft during the design process. An objective of this work is to develop and transfer to industry the technology necessary to meet demands for quieter rotorcraft with smoother ride qualities.

RSRA/X-Wing Rotor Flight Investigation

NASA has established a system technology program at the Ames Research Center for flight investigation of X-Wing technology using the Rotor Systems Research Aircraft (RSRA) in a cooperative program with the Defense Advanced Research Projects Agency (DARPA). The X-Wing is an extremely stiff, four-bladed rotor which can be stopped in flight to provide "X" configured fixed wings to achieve higher forward speeds. The program goal is to adequately demonstrate specific X-Wing technology such that proof-of-concept flight investigations, coupled with successful completion of the DARPA/NASA Convertible Engine Program and the DARPA/Army NOTAR Program, would provide the necessary technology base to lower development risk for an X-Wing prototype aircraft. The Army is using a modified OH-6A helicopter as a testbed for investigating a NOTAR concept which uses a circulation control/jet thruster tail boom assembly as a possible alternative to the tail rotor for directional control.

Advanced Tilt Rotor Research

Proof-of-concept investigations for the tilt rotor were conducted in a joint program with the U.S. Army using the XV-15 research aircraft. The tilt rotor concept provides the higher speed capabilities of a fixed-wing turboprop aircraft combined with slow flight and hover capabilities achievable with the wing-mounted engines tilted vertically for operations in the rotorcraft mode. The Department of Defense program has moved into the development phase as the JVX development program.

Activities at NASA Ames are providing simulation, wind tunnel and XV-15 flight test data in support of the military JVX program and pursuing advanced tilt rotor technology for potential commercial as well as military applications for quiet, efficient VTOL aircraft. Planned advanced technology developments will focus on tilt rotor aeromechanics, advanced flight control concepts, and flight dynamics.

PROPULSION

Four NASA projects appear to be addressing various aspects of all PSH technology needs in the propulsion technology areas except for particle separation. In an analysis of technology needs for public service helicopters, the previous ORI report identifies "particle separation (FOD Proof)" as a potential technology need that requires further definition. Since particle separators are available on a variety of turboshaft engines now in production, the stated technology need was judged to be based on the effects of volcanic ash on rotorcraft engines experienced as a result of the volcanic eruption on Mt. St. Helen. There appears to be a need to further define this requirement in terms of the probability of potential risk of volcanic ash in the atmosphere and acceptable engine performance. Engine research may be required based upon further definition.

The objectives and plans for current NASA projects in the propulsion technology area are summarized below.

Fan and Compressor Research

Propulsion research at the Lewis Research Center includes activities directed toward improving efficiency, operating range, distortion tolerance, durability and reliability, and reducing the weight, volume and cost of fans and compressors. Research on small compressors is conducted jointly with the U.S. Army. Current focus is on experiments to better understand internal flow physics and to verify internal flow codes.

Combustors and Turbines

Research work on combustors and turbines at the Lewis Research Center is currently focused on fuel systems, combustion, turbine aerodynamics and turbine cooling. This work includes investigations of aviation fuels and fuel systems; effects of alternative aviation fuel composition on fuel system performance; generic combustor research to improve performance, durability and enhance fuel flexibility; size-related characteristics of small axial and radial turbines, and turbine cooling designs and temperature barriers.

Rotorcraft Operating Problems

Recent and planned activities on propulsion technology include systematic analytical evaluations of various contingency power concepts, conceptual design and feasibility study of torque converters for high-speed rotorcraft, and methods for reducing cruise fuel consumption. Several power augmentation concepts are being evaluated in terms of direct operating cost and life cycle cost.

Convertible Engine System Technology

This project, previously discussed under the Vehicle Design technology area, also includes systems technology efforts directed towards providing a totally integrated aircraft and propulsion control system, with the initial tests focused at conventional helicopters. These tests are planned to confirm the viability of total digital control and will be followed by an evaluation of a modern multi-variable control having dual input (shaft speed and torque) and dual output (fuel flow and compressor variable geometry) capability.

SAFETY AND RELIABILITY

The research work at the Langley Research Center on composite helicopter structures includes static and dynamic tests to determine the specific energy absorption capabilities of various composite materials and evaluation of the crash failure aspects of novel structural design concepts.

The Tilt Rotor research aircraft does not require a tail rotor for directional torque control, but does not represent a conventional helicopter design. The results from the previously mentioned NOTAR aircraft concept may provide an alternative design for eliminating the tail rotor.

NAVIGATION, GUIDANCE AND FLIGHT CONTROLS

Research and systems technology in this technology area are being pursued at the Ames Research Center. Research on rotorcraft flight dynamics is being carried out by analyses, pilot simulations, and flight tests to investigate rotorcraft handling qualities and advanced flight control/display concepts. Flight guidance systems technology activities are oriented towards providing the critical technology needed to significantly improve operational capabilities under low visibility conditions. The three main systems technology thrusts include: (1) development of design criteria and performance tradeoffs for promising guidance concepts at remote sites; (2) definition of operational limitations of various approach paths to a helipad in proximity to a microwave landing system (MLS) installation; and (3) development of crew station design criteria for advanced integrated guidance and control system concepts for all-weather flight operations.

AUXILIARY SYSTEMS

No NASA research and technology activities were identified as relating directly to PSH technology needs in the Auxiliary Systems area. Research efforts at the Ames Research Center to develop crew station and control design criteria for all-weather integrated cockpit concepts for single pilot operations under instrument conditions could address night vision capabilities as one aspect of the design criteria. Night vision goggles for night time, low level flight have been developed and used by the military services.

The specialized needs for law enforcement applications (car identifier, lock-on and stopper) appear to be related to functional needs which are outside the scope of the NASA aeronautics program.

HUMAN FACTORS

A broad base of research activities to reduce rotorcraft vibration and noise are being carried out at the Ames Research Center and Langley Research Center. Research efforts on noise at Ames are oriented toward investigations of rotor noise characteristics. Efforts at the Langley Research Center include investigations of main rotor and tail rotor acoustic interactions on rotorcraft vibration and noise. The objectives of this research are to develop the system technology for reducing interior noise through rotor/transmission/airframe isolation, to develop noise prediction methods, and to control rotorcraft vibration and noise characteristics through predictive design methods.

Human factors research at Ames is investigating the helicopter pilot's information needs, information processing capacities, and performance assessment techniques. An advanced fully integrated guidance and control system will be tested in a simulation environment for rotorcraft operations. Research on rotorcraft systems integration includes investigations of concepts for advanced flight controls, displays, fault-tolerant actuation systems, and methods for software verification and validation in the design of digital flight control systems.

MONITORING/DIAGNOSTICS

The Lewis Research Center is conducting research to develop a technology base for producing high temperature transducers and electronic systems that can operate on or in close proximity to a turbine engine for the purpose of control, condition monitoring, or experimentation. Major emphasis in the application area is the use of micro- and minicomputers for control and condition monitoring of propulsion systems.

Work on advanced crew station concepts at the Ames Research Center includes studies to determine the value of speech input/output technology for systems status interrogation, automatic warning, and guidance/navigation. This effort builds on previous studies on the value of speech input/output for display of vehicle flight parameters and external threat warnings. No current work appears to be focused on head-up displays which are now state-of-the-art equipment in military systems.

III. COMPARISON OF PSH VEHICLE NEEDS WITH ARMY AVIATION DEVELOPMENT OBJECTIVES

The U.S. Army conducts extensive technological developments necessary to meet Army Aviation's overall modernization strategy. A significant number of these technical activities are closely coordinated with NASA and some jointly funded by Army Laboratories and NASA Research Centers. A catalogue of ongoing technology developments and the potential enhancements to the Army's fleet of aircraft is provided in the Army Aviation Research, Development, Test and Evaluation (RDT&E) Plan.[1] The RDT&E Plan contains a Technology Plan and a Systems Plan. The Technology Plan emphasizes the major thrusts and objectives of programs to advance technology for Army aircraft systems. The Systems Plan describes potential improvements to mission performance for Army aircraft.

The Technology Plan includes several program objectives which have potential for technology transfer to meet Public Service Helicopter (PSH) technology needs for improvements in civil aircraft systems. Selected examples are:

- Develop practical alternatives to the tail rotor for directional control (including NOTAR);
- Investigate advanced rotor control concepts;
- Investigate approaches to reduce rotor noise 50 to 75 percent without reducing rotor performance or increasing rotor and drive system weight;
- Develop the 2000 shaft horsepower advanced technology engine with 25 percent more power than the T700;

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[1] Army Aviation RDT&E Plan, FY1984-2003, Twelfth Edition, U.S. Army Aviation Research and Development Command, St. Louis, MO, June 1984.

- Demonstrate 20 percent reduction in fuel consumption and 25-35 percent improvement in specific power on the 800 shaft horsepower advanced technology demonstrator engine;

- Develop definitive structure design criteria for helicopter composite structures;

- Using multifunction controls and displays, embedded microprocessors, and avionics system multiplexing, integrate selected cockpit management functions and displays; and

- Develop decelerated approach and landing (DSAL) capability for landing helicopters in all flyable weather conditions.

The System Plan includes a family of Light Helicopters (LHX) which the Army plans to develop for its next generation of light rotorcraft. Current plans envision two LHX versions; a scout/attack (SCAT) version and a utility (UTIL) version. The scout and attack versions would be based on a common airframe design and the utility version would use the same dynamic components (engines, transmission, rotor, etc.) and integrated cockpit subsystems. The LHX-Utility functions include tactical team carrier (4-6 combat troops), light cargo carrier (1,500 pounds) and command/liaison transport. The LHX program is now in the concept formulation stage, so specific aircraft configurations and design requirements have not yet been determined. Potential aircraft weights are estimated to be in the 8,000 pound class. Comparisons of PSH vehicle needs with LHX program objectives indicates that modified versions of the LHX-Utility aircraft may have potential for future application by many PSH users.

COMPARISON OF PSH AND LHX OBJECTIVES

Although the final configuration for the LHX has not been determined, review of published articles on preliminary designs indicates that there may be several areas where similarities exist in PSH and LHX objectives.[2,3] Table 6 presents a comparison of similarities in PSH and LHX objectives for several system characteristics. This comparison indicates that the LHX program objectives are similar to PSH objectives for a significant number of desired characteristics. According to the current LHX program schedule the Army plans to initiate full-scale engineering development of the LHX-SCAT in early fiscal year 1987 to provide an initial operational capability by 1992. Engineering development of the LHX-UTIL version is scheduled to start in fiscal year 1988. Manufacturers could possibly provide a new PSH version derived from LHX applied technology by the mid 1990s.

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[2]Schrage, D.P., "The LHX Preliminary Design Process", VERTIFITE, November/December 1983.

[3]Levine, L.S. and Zalesch, S.E., Commonality Potential of Future Public Service Helicopters and Army Light Utility Helicopters, AIAA Aircraft Design, Systems and Technology Meeting, Oct. 17-19, 1983 Ft. Worth. TX.

TABLE 6

COMPARISON OF SIMILARITIES IN PSH AND LHX OBJECTIVES

CHARACTERISTICS	PSH OBJECTIVES	LHX OBJECTIVES
Design Gross Weight	o 10,000 lb. Wt. Limit	o 8,000 lb. Class
Speed, Cruise	o 200 kn SL, ISA	o 160 kn-200 kn Range
Dash	o 300 kn SL, ISA	o High Perf. Rotorcraft Concept
Propulsion	o Twin-Engine	o Single vs Multi-Engine Tradeoff
	o Low Fuel Consumption	o Up to 25% Improvement in SFC
	o Increase Shaft Horsepower	o 25-35% Increase in Specific Power
	o Lightweight Power Plant	
Safety & Reliability	o Crashworthy Structures	o Adv. Composite Structure
	o Reduce Tail Rotor Hazard	o Advanced Helicopter Concept
Operations	o Combine Controls	o Advanced Flight Controls
	o All-weather Flight	o Night Navigation and Pilotage System
	o Low Airspeed Measurements	o Integrated Controls & Displays
	o Electronic Map Display	o Mission Equipment Package
	o Precise Location/Navigation	o Voice Interactive Controls
	o Trend Warning	o Very High Speed Integrated Circuits
	o Computerized Monitoring	o Advanced Target Acquisition System
	o Head-up Display	
	o Performance Limit Monitor	

LHX PROGRAM ACTIVITIES

In the fall of 1983, the Army awarded preliminary design contracts to helicopter manufacturers to perform preliminary design investigations. Contractors were asked to utilize technology consistent with a 1987 start date develop data on the integration and automation of cockpit functions to support preparation of the LHX specifications. Emerging technologies in ongoing technology base programs have demonstrated that the integration and automation of cockpit functions could reduce pilot workloads, increase navigation accuracies, and improve mission performance. However, several critical questions need to be resolved to develop the system design requirements and electronic architecture, and to determine the feasibility of a single pilot concept for possible application in the LHX. This work is being performed under a program entitled Advanced Rotorcraft Technology Integration (ARTI).[4]

A perceived production need of about 5,000 LHX aircraft has attracted the attention of large segments of the helicopter industry. Industry competition for participation in the development program includes teaming arrangements between aircraft manufacturers, computer manufacturers, and flight control and avionics system integrators. The five industry teams working on the ARTI contracts are led by Bell Helicopter Textron, Boeing Vertol, Hughes Helicopters, Sikorsky and IBM as prime contractors. Aerospatiale Helicopter Corp. at Grand Prairie, Texas is teamed as a subcontractor with IBM.[5]

The ARTI program is divided into two phases. Phase I is to provide inputs to the Army by December 31, 1985 to support documentation of LHX system specifications. Phase II expands the investigations to consider far-term technologies which could be included in a Pre-planned Product Improvement program in FY1992. The Phase II work is to be completed by December 31, 1987.

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[4] Aviation Daily, December 30, 1983, p. 318.
[5] Smith, L.K., "LHX: The Helicopter Program of the Century", Rotor & Wing International, January 1984.

IV. TECHNICAL ASSESSMENT OF PSH USER REQUIREMENTS

The Public Service Helicopter (PSH) requirements identified at the July 1980 workshop represent a consolidation of the mobility and transportation needs of four specific PSH user functions or responsibilities, as follows:

- o Emergency Medical Services;
- o Search and Rescue Operations;
- o Law Enforcement Operations; and
- o Environmental Control (Fire Fighting, Resource Management).

While each of these functions encompasses certain specific needs and some of the requirements are common to all or several of the areas of responsibility, the balance of the needs are unique to only one specific function. However, it was the consensus of the workshop panel chairmen that a single aircraft type could probably be developed with a detachable or interchangeable mission specific modular cabin to satisfy user needs for all four public service mission areas.

This report presents a technical analysis of the user requirements, and the design and performance trade-off alternatives available to the helicopter designer in attempting to satisfy the several Public Service Helicopter needs with a single aircraft design. The results of the previous ORI study effort found that certain desired vehicle characteristics impacted adversely on other desired characteristics.[1] The most significant conflicts involve the relationships between the needs for high speed performance, single engine hover capabilities at extreme altitudes, a dimensional constraint of 20 ft. maximum rotor diameter, a maximum gross weight of 10,000 lbs., and desired mission capabilities. This section reviews the user requirements and identifies those requirements most conducive to compromise or relaxation. The next section examines the feasibility of satisfying these requirements with a conventional, low development risk, single rotor helicopter design. Finally, Chapter VI briefly addresses the potential advantages and disadvantages of several alternate rotorcraft designs in meeting the stated operational needs.

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[1]Bauchspies, J. S. and Simpson, W.E., Op. Cit., pp. 4-1 and 4-2.

SPECIFIED USER REQUIREMENTS

In order to precisely define the objectives of this section, it is first necessary to review the list of consolidated Public Service Helicopter requirements, as shown below:

Maximum Rotor Diameter:	20 ft.
Maximum Gross Weight:	10,000 lbs.
Required Useful Load:	6 passengers + 4 hrs. fuel 4,000 lb. payload + 2 hours fuel
Required Hover Capabilities:	10,000 ft. HOGE (Single Engine) 20,000 ft. HIGE (Single Engine)
Required Speed	200 kn Continuous Cruise Capability 300 kn 30-minute Dash Capability
Required Endurance:	4 hrs. (mission not stated)
Required Propulsion	Twin Engine
Tail Rotor Configuration:	No Tail Rotor
Aircraft Configuration:	Modular Mission Specific Cabin
Crew:	2

While most of these requirements, if taken individually, are achievable within current rotary wing aircraft technology, the consolidation of all of these requirements into a single aircraft design represents a technologically demanding design problem. Furthermore, to require the resulting single aircraft design to be capable of satisfying any arbitrary combination of user requirements simultaneously, at a given point during the mission, represents an unsolvable problem at any level of current or forecasted rotorcraft technology. For example, it was shown in the previous ORI report that the power required to sustain a Hover-In-Ground-Effect (HIGE) capability at 10,000 lbs. gross weight and 20,000 ft., Standard Day conditions, with a 20 ft. diameter rotor was on the order of 32,300 shp. Allowing for a 38.5 percent decrease in power available at 20,000 ft./STD conditions, 6 percent transmission and drive system losses, 5 percent power loss due to engine degradation between overhauls and providing for the single engine requirement at this condition, results in a required power installation of approximately 120,000 shp based on Sea Level Standard Day rating. Applying even the most optimistic forecast of power to weight ratio of 10.0 shp/lb. of engine weight results in a 12,000 lb. required engine installation to satisfy this condition. Consequently, even the most optimistic estimate of engine weight required to meet this condition amounts to 2,000 lb. more than the total permissible aircraft weight.

The most severe conflicts that arise in attempting to satisfy all of the foregoing requirements simultaneously are not due to technological limitations which might be resolvable by technological advances; but, rather, result from the physics governing the aerodynamics of rotary lift systems.

The ultimate solution, therefore, to satisfy all of the specified user requirements, simultaneously in any arbitrary combination, with a single aircraft design lies beyond the realm of either current or forecasted rotary wing aircraft technology.

Furthermore, it is erroneous to assume that the workshop intended or desired that all of the user requirements be met simultaneously in a single aircraft design, particularly if such a goal led to unacceptable compromises and/or problems which the workshop did not address. Rather, it is interpreted that the intentions of the workshop findings were to direct preliminary design efforts toward a single aircraft design capable of performing as many of the specified optimistic design goals as possible, without introducing any new and/or untenable operational problems to the user. This rationale is supported, to a degree, by the panel conclusions that a single aircraft of modular design may be suitable for all public service missions, thus implying that certain requirements would be satisfied in one modular form, while other requirements would be satisfied in yet other modular forms.

Therefore, the principal objective of this report will be to examine design tradeoffs which satisfy as many of the user needs as possible, taking each requirement in isolation. Secondly, the analysis will illustrate, where applicable, the impact of attempting to satisfy multiple requirements simultaneously.

REDEFINING THE REQUIREMENTS

While the conclusions of the Public Service Helicopter workshop defined a set of generalized requirements specifying their operational needs, the list of criteria fails to specify all of the pertinent parameters relative to each performance requirement. As a result, it is necessary to redefine the user requirements based on assumed or estimated minimum levels of acceptable performance for those criteria not specified in the workshop findings.

In addition, it appears essential to introduce two additional performance requirements not addressed by the workshop in order to ensure operational safety. These include:

1. Specifying that the downwash velocity from the rotor not exceed 50 ft/sec or 34 mi/hr at Sea Level Standard Day conditions. While this figure is quite high, it is intended only as a maximum tolerable limit, and the actual design requirement for public service use should probably be much lower. In the course of establishing a realistic limit on rotor downwash, the users should consider the effects or impact of high downwash velocities in heliport design, on ground service equipment and personnel, in intracity/rooftop operations, operations over emergency and disaster debris, rooftop rescues and operations in snow and dust. High downwash velocities tend to cause accidents and disasters of their own, such as knocking over objects, including people, propelling debris, spreading fires, aggravating hover operations, impeding ground operations, and impairing pilot vision in snow, dust, sand and soot environments.

2. Specifying a requirement for single engine Hover-Out-of-Ground-Effect (HOGE) capability at the Design Gross Weight of the aircraft from Sea Level to 10,000 ft. under Standard Day atmospheric conditions. This requirement is recommended to ensure safety for intracity operations, takeoffs from rooftop heliports, etc. Although the workshop specified a requirement for single engine Hover-Out-of-Ground-Effect capability at 10,000 ft., it did not specify the gross weight condition associated with this requirement. If the workshop intended the 10,000 ft. HOGE capability on a single engine as an emergency condition only, this requirement could be satisfied at a gross weight substantially less than the Design Gross Weight of the aircraft. If, on the other hand, the workshop intended to ensure a single engine HOGE capability at the Design Gross Weight at Sea Level Standard Day conditions by specifying the requirement at 10,000 ft., then the single engine HOGE requirement at 10,000 ft. should specify a gross weight condition equivalent to the desired single engine HOGE capability at Sea Level. Furthermore, this requirement should be sufficient to provide for a single engine HOGE capability at the Design Gross Weight of the aircraft to allow operational safety in the event of an engine failure at takeoff. Specifying a single engine Hover-Out-of-Ground-Effect capability at the Design Gross Weight of the aircraft from Sea Level to 10,000 ft., Standard Day conditions, should cover most, if not all, engine failure conditions.

In the course of establishing design criteria based on user needs, it is necessary to segregate the design criteria from the performance criteria and to define those terms that will be used to describe the various requirements. The terms are defined in Appendix A, and a list of abbreviations is provided in Appendix B. The design requirements as specified by the user, along with comments on potential areas of conflict and possible solutions, are shown in Table 7. The revised performance requirements listed in Table 8 include a complete set of specified or assumed conditions governing the performance criteria.

AIRCRAFT SIZING CONSIDERATIONS

In order to translate the specific load carrying requirements of the user, in terms of useful load, into an equivalent gross weight for comparison with potential rotor thrust capabilities, it is necessary to develop an empty weight fraction as a design objective. Since light helicopters, in the range of 3,000-12,000 lb. maximum takeoff weight, generally exhibit empty weight fractions of 0.4 to 0.6, as shown in Table 9, an optimistic empty weight fraction of 0.45 was assumed throughout this report as a design objective for a conventional single rotor helicopter design. Using 0.45 as the empty weight fraction yields a useful load fraction of 0.50, as shown in Table 10.

Aircraft Weight

The loading criteria specified by the workshop defined the required useful load in terms of payload plus hours of flight endurance. As a result, it is necessary to estimate the fuel load required to achieve the desired

TABLE 7

DESIGN REQUIREMENTS

Requirement	Rationale	Problems/Conflict	Reference	Options/Solutions
20 ft Maximum Rotor Diameter	Minimize Aircraft Clearance Problems at Emergency Landing Sites and to Allow Minimum Ground Time and Transportation Requirements for Injured by Landing in Close Proximity to Accidents	1 High Downwash Velocities at Gross Weights Exceeding 2500 lb 2 Unacceptable Downwash Velocities at Gross Weights Exceeding 6000 lb 3 Poor Hover Performance at Gross Weights Exceeding 5000 lb 4 No Hover Performance at Gross Weights Exceeding 6000 lb 5 High Power Requirements at Gross Weights Exceeding 5000 lb 6 Inability to Satisfy Specified Load Requirements at Any Level of Installed Power 7 High Noise Levels	Previous ORI Report on Contract NASW-3554 Dated December 1983	1 Increase Rotor Diameter to Reduce Downwash Velocity & Improve Lift Capacity 2 Increase Rotor Tip Speed to Improve Rotor Lift Capacity 3 Increase Rotor Solidity to Improve Rotor Lift Capacity 4 Reduce Load Requirements to Meet Rotor Sizing 5 Accept High Downwash Velocities at Minimum Rotor Sizing 6 Accept High Noise Levels at Minimum Rotor Sizing 7 Install More Power to Yield Lift at Minimum Rotor Sizing
10000 lb Maximum Gross Weight	Weight Limit for Rooftop Heliport and Remote Area Operations	1 Assuming an Optimum Empty Weight Fraction of 50% Fails to Satisfy 1 Loading Requirement 4000 lb Payload +2 hrs Fuel	Previous ORI Report on Contract NASW-3554 Dated December 1983	1 Minimize Empty Weight Fraction to Achieve Loading Requirement 2 Compromise the Most Severe Loading Requirement
50 ft/Sec Maximum Downwash Velocity	To Permit Operations in the Vicinity of Ground Personnel & Equipment, and to Permit Operations in Snow, Sand & Dust Environments	1 Limits Maximum Gross Weight to 4000 lb With the 20 ft Rotor 2 Fails to Meet Loading Requirements With a 20 ft Rotor 3 Requires at Least a 33 Ft Diameter Rotor to Lift a 10000 lb Gross Weight	None	1 Increase Rotor Diameter to Lift Specified Load Requirements 2 Reduce Load Requirements With a 20 ft Rotor Diameter to Meet Downwash Velocity Requirement
No Tail Rotor	To Insure Safe Operations Around Ground Personnel and Equipment and to Protect Aircraft From Loss Due to Tail Rotor Strikes	No Conflict Identified	None	1 Counter Torque Via Notar Circulation Control Tail Boom 2 Counter Torque Via Fenestron/Ducted Fan Tail Rotor 3 Counter Torque Via Differential Auxiliary Thrust 4 Counter Torque Via Multirotor Design Coaxial, ABC, Tilt Rotor 5 Torqueless Rotors Such as Tip Jets, Hot or Cold Cycle Reaction Rotors
Twin Engine	To Provide for Engine Failure Contingency	None	None	1 Provide Convertible Engines for Thrust Augmentation in Forward Flight
Modular Mission Specific Cabin	To Allow One Aircraft Design to Satisfy Multiple User Needs	No Conflict Identified	None	1 Cabin Configuration Options Based on Various User Specifications

TABLE 8

PERFORMANCE REQUIREMENTS

Operational Requirement	Airspeed	Altitude	Ambient Condition	Payload	Fuel	Useful Load	Gross Weight	Condition
Sea Level HOGE (Single Engine)	Hover	Sea Level	Standard Day	Normal	Normal	Normal	Design Gross Weight	Emergency Engine Out
10000 ft HOGE (Single Engine)	Hover	10,000 ft	Standard Day	Normal	Normal	Normal	Design Gross Weight	Emergency Engine Out
20000 ft. HIGE (Single Engine)	Hover	20,000 ft	Standard Day	0 Lbs	Minimum	Minimum	Minimum Gross Weight	Emergency Engine Out
200 kn Cruise Speed Continuous	200 kn	Sea Level	Standard Day	Normal	Normal	Normal	Design Gross Weight	Standard Operations on Two Engines
300 kn Dash Speed 30 Minutes	300 kn	Sea Level	Standard Day	Normal	Normal	Normal	Design Gross Weight	Standard Operations on Two Engines
6 Passengers +4 hrs Fuel	Best Endurance(3)	Sea Level	Standard Day	1,200 lbs(1)	950 lb Est(2)	2,150 lb Est	Design Gross Weight	Standard Operations on Two Engines
4000 lb Payload +2 hrs Fuel	Best Endurance(3)	Sea Level	Standard Day	4,000 lbs	1,200 lb Est(2)	5,200 lb Est	Maximum Gross Weight	Overload Operations on Two Engines
4 hrs Endurance	Best Endurance(3)	Sea Level	Standard Day	0 Lbs	950 lb Est(2)	950 lb Est	Design Gross Weight	Standard Operations on Two Engines

NOTES

(1) Based on 200 lbs Per Passenger

(2) Based on an Average Mission Fuel Flow of 6% of Gross Weight per Hour

(3) Based on the Specification by the User in Terms of Hours of Flight Time, as Opposed to Best Range Speed Which Would Apply If the User Had Specified a Required Distance

TABLE 9

EMPTY WEIGHT FRACTIONS

Aircraft Type	Empty Weight	Max Gross Weight	Empty Weight Fraction
Hughes 500	1,088 lb.	3,000 lb.	0.363
Bell 206B Jetranger III	1,615 lb.	3,200 lb.	0.505
Bell 206L Longranger	2,156 lb.	4,150 lb.	0.520
Bell 212	6,143 lb.	11,200 lb.	0.548
Bell 222	4,828 lb.	7,850 lb.	0.615
Bell 412	6,070 lb.	11,500 lb.	0.528
Sikorsky S-76	5,475 lb.	10,000 lb.	0.548
MBB/Kawasaki BK117	3,351 lb.	6,614 lb.	0.507
MBB BO105	2,622 lb.	5,291 lb.	0.496
Aerospatiale SA 315B	2,251 lb.	5,070 lb.	0.444
Aerospatiale SA 316B	2,520 lb.	4,850 lb.	0.520
Aerospatiale SA 319B	2,527 lb.	4,960 lb.	0.509
Aerospatiale SA 341	2,022 lb.	3,970 lb.	0.509
Aerospatiale SA 342	2,105 lb.	4,190 lb.	0.502
Aerospatiale AS 350	2,304 lb.	4,630 lb.	0.498
Aerospatiale AS 355	2,712 lb.	5,070 lb.	0.535
Aerospatiale SA 360	3,609 lb.	6,614 lb.	0.546
Aerospatiale SA 365	4,136 lb.	7,495 lb.	0.552
Westland WG13A	5,860 lb.	10,000 lb.	0.586
Westland WG13N	6,680 lb.	10,500 lb.	0.636
Westland WG30	6,680 lb.	12,000 lb.	0.557

TABLE 10

WEIGHT FRACTION ASSUMPTIONS

Weight Fraction	Percentage of Maximum Gross Weight
Empty Weight Fraction	45.0
Crew, Mission Equipment, Trapped Fluids	5.0
Operational Empty Weight Fraction	50.0
Useful Load Fraction	50.0
Fuel Fraction (Normal)	22.0
Payload Fraction (Normal)	28.0

flight endurance as a function of aircraft gross weight. The typical helicopter in the 3,000-10,000 lb. weight class requires 0.08 shp per pound of flight weight to sustain straight and level flight at minimum power, which equates to optimum flight endurance conditions. Assuming that the installed specific fuel consumption of the engines is unusually high at about 0.75 lb./shp-hr., the resulting fuel consumption at endurance speed is approximately 0.06 lb. of fuel per hour per pound of aircraft flight weight, or 6 percent of flight weight per hour. The 4 hour endurance specified by the workshop requires a 22 percent fuel fraction, while the 2 hour endurance condition requires an 11.5 percent fuel fraction. Alternatively, typical fuel fractions ranging from 15-20 percent yield 2.6-3.6 hrs. of flight endurance, respectively. The relationships between gross weight, useful load and required fuel load are shown in Figure 6 for the specified endurance requirements. The high value assumed for specific fuel consumption was selected to represent the high altitude, high speed and single engine criteria specified by the workshop. These factors tend to drive the installed power requirements of the aircraft up, which results in reduced partial power settings, particularly at minimum power for optimum endurance, and correspondingly high specific fuel consumption rates.

The combination of the estimated 50 percent useful load fraction and the estimated 22 percent fuel fraction allows a 28 percent payload fraction with 4 hrs. endurance fuel. Imposing the maximum payload requirement corresponding to the 4 hr. endurance specification, results in 6 passengers at an estimated weight of 200 lb. per person or a 1,200 lb. payload requirement. The minimum gross weight capable of carrying a 1,200 lb. payload at a 28 percent payload fraction is 4,300 lb. While the estimate of 200 lb. per person used to compute the required payload capacity may seem unreasonably high, it tends to compensate for the low value for crew and mission equipment at low gross weights due to the fixed 5 percent crew and mission equipment fraction.

Combining the 50 percent estimated useful load fraction with the 11.5 percent estimated fuel fraction allows a 38.5 percent payload fraction with 2 hrs. endurance fuel. Applying the 4,000 lb. payload requirement corresponding to the 2 hr. endurance specification, at the 38.5 percent payload fraction results in a minimum gross weight of 10,400 lbs., which exceeds the specified maximum gross weight limit of 10,000 lbs. by 400 lbs. or 4 percent. These relationships are shown graphically in Figure 7 and summarized for the two payload conditions in Table 11.

It should be noted that the estimated fuel loads shown in this section serve to illustrate the variance in user needs based on 100 percent operation at endurance speed or minimum power only, as suggested in the specified requirements. Flight operations at speeds other than optimum endurance speed will degrade flight endurance estimates. The estimated fuel loads shown in this section should, therefore, be increased by some percentage or factor to account for time spent in hover and high speed flight if the specified flight endurance is to be preserved. Since the loading requirements specified by the workshop relate to different mission requirements, the time spent in hover and high speed flight may differ substantially from one loading condition to the other. Consequently, any fuel allowance for hover and high speed flight may need to be allocated disproportionately to each loading condition to account for mission related differences in flight requirements.

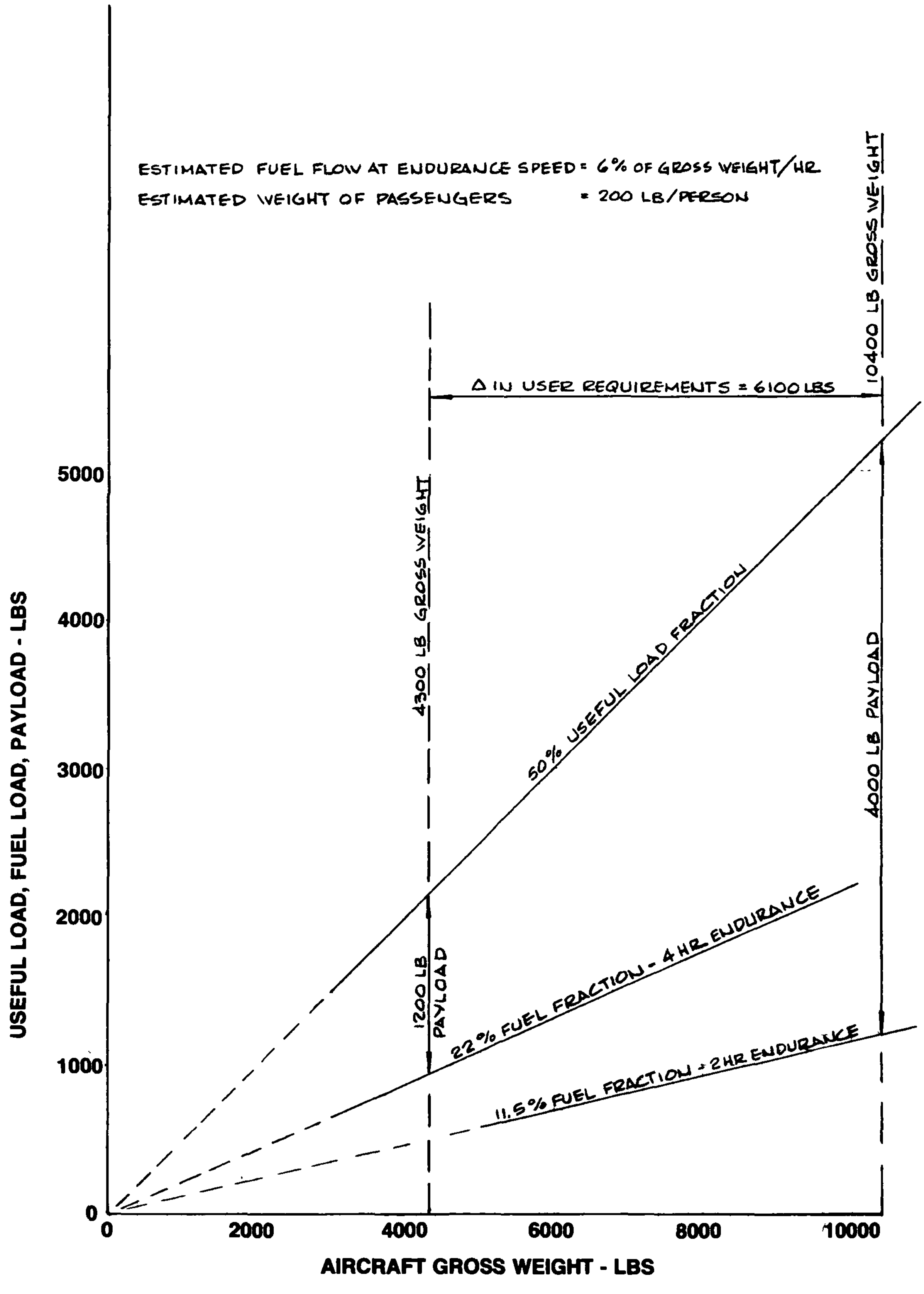

FIGURE 6. COMPARISON OF REQUIRED LOADINGS

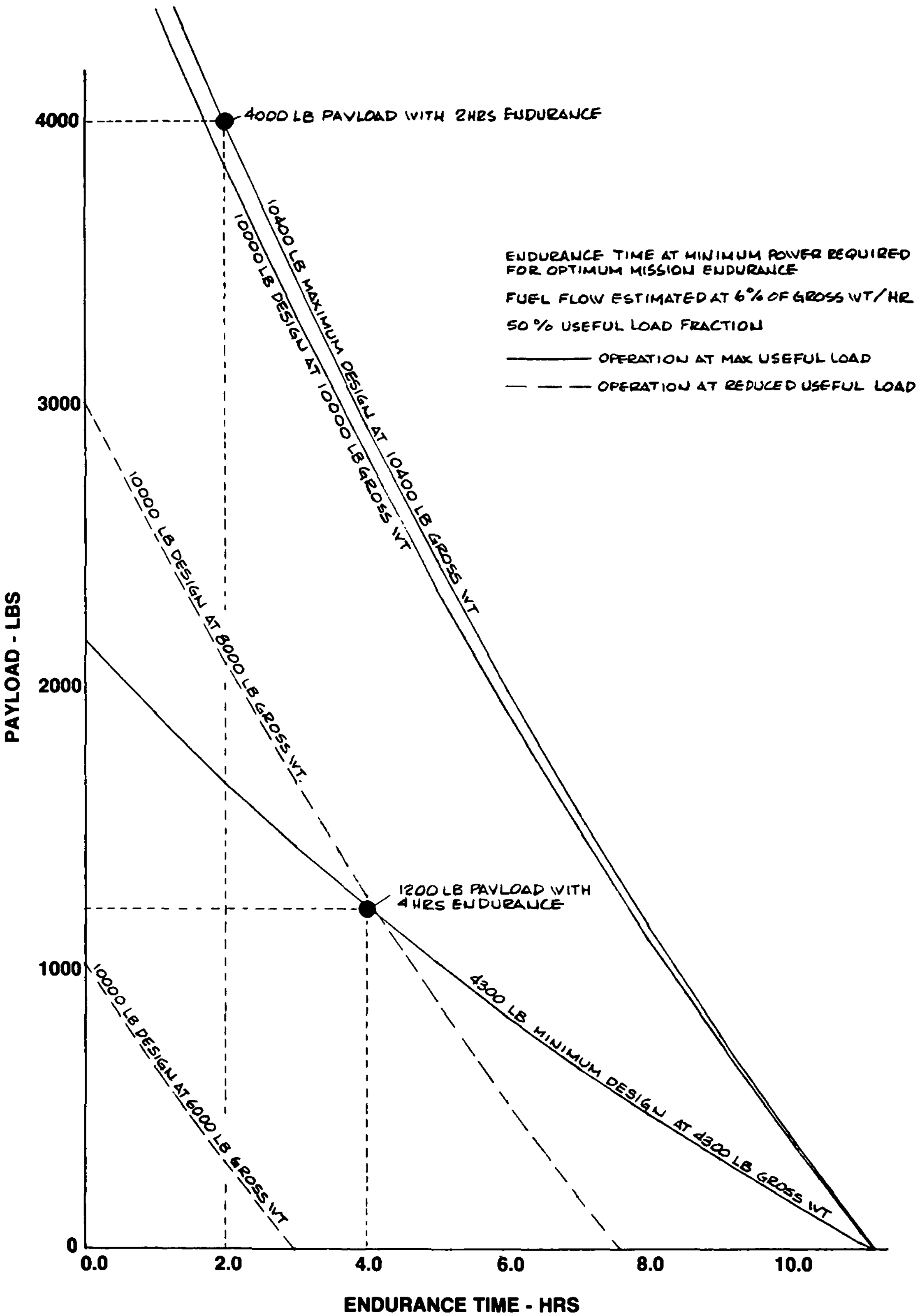

FIGURE 7. PAYLOAD VS. ENDURANCE

TABLE 11

COMPARISON OF USER LOAD REQUIREMENTS

Gross Weight	4,300 lb.	100%	10,400 lb.	100%
Empty Weight	1,935 lb.	45.0%	4,680 lb.	45.0%
Crew, Mission Equip. & Trapped Fluids	215 lb.	5.0%	520 lb.	5.0%
Operational Empty Weight	2,150 lb.	50.0%	5,200 lb.	50.0%
Useful Load	2,150 lb.	50.0%	5,200 lb.	50.0%
Payload	1,200 lb.	28.0%	4,000 lb.	38.5%
Fuel Load	950 lb.	22.0%	1,200 lb.	11.5%
Endurance	4 hrs.		2 hrs.	
Impact	Fails to meet 4,000 lb. payload requirement.		Exceeds gross weight limit by 400 lbs.	

The conclusions of the workshop were insufficient to determine whether the users required the specified flight endurance in addition to mission related hover and high speed flight requirements, or whether the users are willing to compromise the specified endurance criteria to achieve hover and high speed flight time. In addition, the findings of the workshop provide no insight into the criteria for disproportionate fuel loadings to allow for differences in mission related flight requirements. For these reasons, the sizing comparisons in this report are based on 100 percent operation at minimum power to achieve optimum flight endurance. In order to correct for this assumption and to evaluate the impact of disproportionate hover and high speed flight requirements on mission fuel loadings, it is suggested that the fuel loads be increased by 3 percent for each 10 percent of mission time spent in hover and high speed flight. Consequently, a mission comprised of 10 percent hover and high speed flight and 90 percent operation at minimum power for optimum flight endurance would require 103 percent of the estimated fuel load, while a mission comprised of 100 percent operation at hover and high speed flight would require 130 percent of the estimated fuel load. Accordingly, if the fuel loadings are not increased, the specified flight endurance will suffer.

As Table 11 illustrates, the useful load criteria specified by the workshop result in a span of required aircraft gross weight from 4,300 to 10,400 lbs., or a factor of 2.42. It is obvious that the driving requirement is the 4,000 lb. payload with 2 hrs. endurance. This requirement alone drives the gross weight up to 10,400 lbs. or 2.42 times the next most severe loading condition of 6 passengers and 4 hrs. endurance. While it is possible to design a single aircraft which meets, or nearly meets, all of the Public Service Helicopter requirements, it does not seem to be practical to impose the larger aircraft design with its attendant acquisition costs, operating costs and design compromises on the balance of users that do not require the 4,000 lb. payload capacity. If the requirement for a 4,000 lb. payload is universal among the users, this requirement is then the driving factor dictating development of a 10,000 lb. aircraft to meet the needs of the majority of users. The minority of users not requiring the 4,000 lb. payload capability would be advised to seek an alternative, smaller aircraft, at reduced cost which better meets their needs. Conversely, if the 4,000 lb. payload requirement is unique to only a small percentage of Public Service Helicopter users, strong consideration should be given to eliminating the 4,000 lb. payload requirement in favor of a substantially smaller aircraft which meets the needs of the majority of users.

In view of the workshop's proposed concept to pursue a single aircraft design to meet all of the users' needs, the balance of this report assumes that the 4,000 lb. payload requirement is universal to the needs of the Public Service Helicopter users. Sizing the aircraft to meet the specified 4,000 lb. payload criteria with 2 hrs. endurance yields a gross weight of 10,400 lbs. which exceeds the specified maximum gross weight of 10,000 lbs. by 400 lbs. Alternatively, imposing the gross weight limit of 10,000 lbs. yields a 5,000 lb. useful load or a 3,800 lb. payload with approximately 1,200 lbs. of fuel for 2 hrs. of endurance. Restoring the 4,000 lb. payload requirement results in a fuel load of 1,000 lbs. or approximately 1.7 hrs. endurance. These options are shown in Table 12.

Designing the aircraft to the 10,000 lb. specified gross weight limit comes very close to meeting the most severe loading requirement specified by the workshop and results in an operational empty weight design objective of 5,000 lb. This equates to an empty weight of 4,500 lb. plus a 500 lb. allowance for crew, mission equipment and trapped fluids. Table 13 provides a comparison between the 4,300 lb. gross weight design and the 10,000 lb. gross weight design operating at the reduced useful load.

Having nearly satisfied the assumed universal user requirement for a 4,000 lb. payload with 2 hrs. endurance by setting the design objective at the 10,000 lb. maximum allowable gross weight, the remaining loading requirements can all be satisfied at gross weights of 8,000 lb. or less. This suggests the option of designing the aircraft structurally to the 10,000 lb. gross weight in order to meet the 4,000 lb. payload requirement in an overload or Alternate Design Gross Weight condition, while the remaining, less severe, loadings are met at an 8,000 lb. Design Gross Weight. The overload or Alternate Design Gross Weight condition generally involves some form of reduction or relaxation of performance criteria such as:

- o Elimination of single engine hover capability in the overload condition;
- o Elimination of the Hover-Out-of-Ground-Effect capability in the overload condition;
- o Elimination of high altitude hover capability in the overload condition;
- o Elimination of the maximum speed capability in the overload condition.

In this case, the best alternative appears to be elimination of the single engine Hover-Out-of-Ground-Effect capability in the overload or Alternate Design Gross Weight condition, while preserving the Hover-Out-of-Ground-Effect capability on two engines and a single engine capability in forward flight cruise at minimum power or endurance speed. Since this option will assure twin engine reliability and safety in all modes of operation except the vertical takeoff at the 10,000 lb. overload condition, consideration should be given to limiting such missions to sling load operations with provisions for payload or fuel jettison in order to instantaneously recover from the overload weight to a single engine Hover-Out-of-Ground-Effect capability in the event of an engine failure on takeoff. Table 14 illustrates the recommended loading options at the 10,000 lb. Alternate Gross Weight condition.

In order to ensure a single engine Hover-Out-of-Ground-Effect capability at the Alternate Design Gross Weight of 10,000 lbs., at least 2,000 lbs. of payload must be carried externally via sling, with provisions for jettison in the event of an engine failure. To provide optimum loading flexibility within these constraints, the design should offer the following accommodations:

- o Minimum internal payload capacity of 1,200 lb. to meet the remaining load criteria specified by the workshop;

TABLE 12

MAXIMUM DESIGN LOADING OPTIONS

Gross Weight	10,400 lb.	100%	10,000 lb.	100%	10,000 lb.	100%
Empty Weight	4,680 lb.	45.0%	4,500 lb.	45.0%	4,500 lb.	45.0%
Crew, Mission Equip.	520 lb.	5.0%	500 lb.	5.0%	500 lb.	5.0%
Operational Empty Weight	5,200 lb.	50.0%	5,000 lb.	50.0%	5,000 lb.	50.0%
Useful Load	5,200 lb.	50.0%	5,000 lb.	50.0%	5,000 lb.	50.0%
Payload	4,000 lb.	38.5%	3,800 lb.	38.0%	4,000 lb.	40.0%
Fuel Load	1,200 lb.	11.5%	1,200 lb.	12.0%	1,000 lb.	10.0%
Endurance	2.0 hrs.		2.0 hrs.		1.7 hrs.	
Impact	Exceeds gross weight limit by 4%.		5% less payload than required.		15% less endurance than required.	

TABLE 13

COMPARISON OF MINIMUM AND MAXIMUM DESIGNS

Parameter	Minimum Design G.W. 4,300 lb.		Maximum Design G.W. 10,000 lb.		Relative Impact
Mission Gross Weight	4,300 lb.	100.0%	8,000 lb.	100.0%	86% greater gross weight.
Design Empty Weight	1,935 lb.	45.0%	4,500 lb.	56.2%	133% greater empty weight.
Crew, Mission Equip. Trapped Fluids	215 lb.	5.0%	500 lb.	6.3%	Not significant.
Operational Empty Weight	2,150 lb.	50.0%	5,000 lb.	62.5%	Not significant.
Useful Load	2,150 lb.	50.0%	3,000 lb.	37.5%	Not significant
Payload	1,200 lb.	28.0%	1,200 lb.	15.0%	Equal payloads
Fuel Load	950 lb.	22.0%	1,800 lb.	22.5%	90% more fuel required.
Endurance	4 hrs.		4 hrs.		Equal endurance
Impact	Not capable of 4,000 lb. payload with 2 hrs. endurance.		Nearly capable of all user loading requirements.		90% more fuel required for equal missions.

TABLE 14

ALTERNATE DESIGN GROSS WEIGHT LOADING OPTIONS

Parameter	Minimum Mission Fuel	Normal Fuel	Maximum Fuel
Alternate Design Gross Weight at Takeoff	10,000 lb.	10,000 lb.	10,000 lb.
Design Empty Weight	4,500 lb.	4,500 lb.	4,500 lb.
Crew, Mission Equipment	500 lb.	500 lb.	500 lb.
Operational Empty Weight	5,000 lb.	5,000 lb.	5,000 lb.
Useful Load	5,000 lb.	5,000 lb.	5,000 lb.
Payload	4,500 lb.	3,800 lb.	2,000 lb.
Maximum Fixed or Internal Payload	2,500 lb.	1,800 lb.	0 lb.
Minimum Jettisonable Payload	2,000 lb.	2,000 lb.	2,000 lb.
Fuel Load: Internal	500 lb.	1,200 lb.	3,000 lb.
Gross Weight After Emergency Load Jettison at Takeoff	8,000 lb.	8,000 lb.	8,000 lb.
Endurance	0.8 hrs.	2.0 hrs.	5.75 hrs.
Maximum Internal Load (Fuel + Payload)	3,000 lb.	3,000 lb.	3,000 lb.
Minimum Jettison Requirement	2,000 lb.	2,000 lb.	2,000 lb.
Minimum Required Sling Load	2,000 lb.	2,000 lb.	2,000 lb.
Maximum Sling Load Capability	4,500 lb.	3,800 lb.	2,000 lb.
Maximum HOGE Capability: Single Engine	8,000 lb.	8,000 lb.	8,000 lb.
Maximum HOGE Capability: Both Engines	10,000 lb.	10,000 lb.	10,000 lb.
Maximum Gross Weight for Single Engine Cruise at Minimum Power	10,000 lb.	10,000 lb.	10,000 lb.

- Maximum internal payload capacity of 2,500 lb. as an option with minimum fuel;
- A normal internal fuel tank capacity of 1,800 lb. to meet the 4 hr. endurance requirement;
- Internal auxiliary fuel tank option to provide an additional 1,200 lb. of internal fuel for a maximum internal fuel capacity of 3,000 lb.;
- Jettisonable external auxiliary fuel tank option to provide an additional 2,000 lb. of external fuel;
- Jettisonable sling cargo with a 4,500 lb. capacity.

Based on the specified user needs, in terms of design requirements and performance requirements and the conclusions of the previous report, it is obvious that the principal conflict centers around the users' desire for a 20 ft. diameter rotor system. The rationale for this requirement is based on the necessity of landing the emergency services vehicle in close proximity to the scene of the accident or disaster, thereby facilitating ground operations and minimizing service response time. This requirement is clearly most essential in the medical services role; important, but to a lesser degree, in the law enforcement and search and rescue roles; and probably even less important in the environmental control role.

In terms of required gross lift capacity or payload capability, it appears that the driving criteria on aircraft weight result principally from the emergency medical services and environmental control requirements. This suggests that, while the requirements of the law enforcement and search and rescue agencies could probably be satisfied with a smaller aircraft on the order of 4,000-6,000 lb. gross weight and a 20 ft. rotor, the environmental control missions require a 10,000 lb. or larger aircraft and could operate effectively with a much larger rotor system in the range of 40-50 ft. in diameter. The needs of the emergency medical services agencies, however, appear to fall in the 6,000-8,000 lb. range of gross lift and present the most demanding operational need for a minimum diameter rotor on the order of 20 ft.

Assuming the emergency medical service requirements dictate the need for an aircraft in the 6,000-8,000 lb. gross weight range with a useful load capability of 2,500-3,500 lbs. and a 20 ft. rotor diameter to minimize aircraft clearance problems, thus permitting operations in close proximity to accident and disaster areas; is such a system feasible? Ignoring, for the moment, the lift limitations of the rotor and the related power requirements, a rotor system of this size and loading would generate a downwash velocity in the range of 45-50 mph. Downwash velocities of this magnitude would effectively preclude any possibility of operating such an aircraft in close proximity to the scene of an accident or disaster, regardless of rotor diameter or aircraft clearance accommodations; thus, effectively invalidating the original basis and rationale for the 20 ft. rotor diameter requirement.

Therefore, the principal conflict inherent in the user requirements demands a tradeoff between required useful load, maximum allowable rotor diameter and an established limitation on the maximum acceptable downwash velocities in public service applications. These tradeoffs are discussed in the next section of this report.

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V. DESIGN TRADEOFFS FOR THE CONVENTIONAL SINGLE ROTOR HELICOPTER

The preceding section of this report addressed the sizing of the aircraft, in terms of gross weight, required to meet the specified loading criteria. This section will examine the effects of tradeoffs in rotor characteristics on the power required to meet performance requirements at specified aircraft gross weights.

The performance estimates which constitute the basis for this assessment were derived using a computer model for estimating helicopter performance capabilities, designated HELPER IV, developed for the U.S. Army Foreign Science and Technology Center. The performance model was used to compute the power required in straight and level flight as a function of airspeed and altitude at the three gross weight conditions considered representative of the user needs, as follows:

- 6,000 lb.: Minimum Mission Weight
- 8,000 lb.: Design Gross Weight
- 10,000 lb.: Maximum Overload or Alternate Design Gross Weight.

In addition, the evaluation covered three tip speed conditions and two rotor solidities which constitutes a family of rotor options representative of conventional single rotor helicopter designs. The rotor solidities selected are approximately 50 percent higher than typical design practices to reduce the problem of retreating blade stall at high forward speed; the tip speeds selected are generally representative of current design practices. The selected values are:

- Rotor Solidities -- σ = 0.14 and 0.16
- Tip Speeds -- V_t = 615 ft/sec M = 0.55 at SL/STD
  = 670 ft/sec M = 0.60 at SL/STD
  = 725 ft/sec M = 0.65 at SL/STD

The performance requirements specified by the workshop stipulated hover performance, in and out of ground effect, on a single engine at altitudes of 20,000 ft. and 10,000 ft., respectively. In order to evaluate the relative power required relationships of these criteria against the power required in forward flight on two engines at Sea Level, the calculated power requirements were resolved back to a common basis for comparison, the required engine installation based on the maximum power rating at Sea Level Standard Day conditions. Table 15 illustrates the correction factors applied to the calculated power requirements to achieve a common base for comparison.

The corrected power requirements were grouped in bands which bound the range of tip speeds and rotor solidities under consideration, and plotted as a function of rotor diameter for the various combinations of gross weight and performance criteria. This approach facilitated cross-plotting of hover requirements against the high speed forward flight requirements, in terms of the required engine installation versus rotor diameter for the various combinations of solidity and tip speed. As a result, the impact of rotor diameter on required engine sizing becomes obvious from the discussions which follow. This will enable the reader to draw approximate conclusions on aircraft cost, fuel economy (or lack thereof), and the resulting operating costs of the system.

PERFORMANCE METHODOLOGY

The helicopter performance estimation program, designated HELPER IV, employed in these analyses computes power required to the main rotor as a function of aircraft gross weight, air speed and altitude. The program computes profile power required using test data for a generic rotor system comprised of a 12 percent thick asymmetric constant chord blade section out to 90 percent of the rotor radius with a 9 percent thick, symmetrical high-speed tip section. The program compensates for nonuniform inflow, tip losses and vertical drag effects according to generally accepted design practices. While the program provides for variations in parasite drag area as a function of velocity and aircraft trim in forward flight, these provisions were not incorporated in this evaluation and the assumed drag area was held constant at 10 square feet.

The effects of blade stall are accounted for as a function of the maximum rotor lift coefficient, $(C_T/\sigma)_{max}$, with a limiting value of 0.12 for the blade stall boundary in hover and decreasing in forward flight based on the square of the advance ratio, . Power requirements for flight conditions exceeding the stall limits are generated based on the test data for the unstalled rotor and labeled with a blade stall code, thus enabling power requirements to be plotted beyond the stall boundary.

The program also incorporates a math model to predict the power required to overcome the effects of compressibility based on published test data. The compressibility effects are calculated as an additional compressibility power requirement as a function of the required horizontal and vertical thrust coefficients, the tip Mach number in hover and the forward flight advance ratio. The compressibility power is computed at all flight conditions under study and included in the total power requirement, with the provision

TABLE 15

CORRECTION FACTORS FOR POWER

Requirement	Correct For Engine Power Available at Altitude	Correct For Power Setting	Correct For Transmission Losses	Correct For Single Engine Requirement	Total Correction Factor $= \frac{1.0}{\text{Correction Factors Product}}$
Hover OGE at SL/STD Single Engine	100% of Max at SL/STD	95% of Max[2]	94%[4]	50%	2.25 × Power Required
Hover OGE at 10,000 ft/STD Single Engine	81.5% of Max at SL/STD[1]	95% of Max[2]	94%[4]	50%	2.75 × Power Required
Hover IGE at 20,000 ft/STD Single Engine	61.5% of Max at SL/STD[1]	95% of Max[2]	94%[4]	50%	3.64 × Power Required
Cruise at 200 Knots at SL/STD Both Engines Continuous	100% of Max at SL/STD	85% of Max[3]	94%[4]	100%	1.25 × Power Required
Dash at 300 Knots at SL/STD Both Engines 30 Min	100% of Max at SL/STD	95% of Max[2]	94%[4]	100%	1.12 × Power Required

Notes

(1) Based on Figure 3-95 of AMCP 706-201, "Engineering Design Handbook, Helicopter Engineering, Part One Preliminary Design," U S Army, August 1974, Pg 3-93.

(2) Allows for Engine Degradation in Power Output Between Overhauls and Provides for 30 Minute Contingency

(3) Allows for Continuous Operation

(4) Allows for 4% Transmission and Drive Train Losses and 2% for Accessories

that flight conditions resulting in compressibility power requirements exceeding 20 percent of installed power bear a compressibility code. As a result, total power requirements, including the effects of compressibility can be plotted well into the compressibility region. This represents a potential problem area in the results presented in this section. While the compressibility power model has proven to yield reasonable results in hover and forward flight conditions out to advancing blade tip Mach numbers of 0.95, the accuracy of the compressibility power model at advancing blade tip speeds in excess of the critical Mach number of the blade tip is questionable. At present, there is insufficient data on supersonic rotors or tip speeds with which to evaluate or improve upon the available compressibility power model. This caveat applies principally to predicted power requirements at 300 knots where all of the tip speeds considered result in advancing blade tip Mach numbers of 1.0 to 1.1, and to the predicted power requirements at 200 knots with a high tip speed of 725 ft/sec for which the advancing blade tip Mach number exceeds 0.95. The subject of supersonic rotors and the compressibility power requirements associated therewith is an area that probably requires further research and investigation.

The results presented in this section illustrate blade stall and compressibility boundaries with a dashed line to distinguish these regions of marginal performance from those conditions of acceptable performance. Generally speaking, those regions lying beyond the blade stall and compressibility boundaries should be avoided in the comparative evaluation of design tradeoffs.

Finally, the performance model provides for tail rotor power, or directional control power, as a function of forward flight speed and power required by the main rotor. Although the users expressed a desire to eliminate the tail rotor, the requirement for tail rotor power was retained in the analysis to account for some form of directional control power. The assumed relationship of control power to main rotor power required is shown below as the percentage of main rotor power and a parabolic function of flight speed with a minimum at 150 knots.

$$P_{control} = P_{main\ rotor} \times ((0.00012 \times (250 - V_{ff})^2 + 5.0) \times 0.01)$$

ASSUMPTIONS AND DESIGN OBJECTIVES

The performance analysis presented in this section is based on a number of performance assumptions and design objectives. For reasons of clarity and simplicity, these values are shown in Table 16.

HOVER CAPABILITIES

Figures 8, 9 and 10 illustrate the relationship between rotor diameter and required power installation as a function of aircraft gross weight for each of the hover conditions under consideration. These conditions are:

- o Hover OGE at SL/STD: Single Engine (preliminary design assumption); shown in Figure 8

TABLE 16

ASSUMPTIONS AND DESIGN OBJECTIVES

Parameter	Assumption or Design Objective	Rationale
Aircraft Gross Weight	6000 lb Minimum Mission Weight 8000 lb Design Gross Weight 10000 lb Alternate Design Gross Weight	As Per Aircraft Sizing Considerations
Parasite Drag Area	10 sq ft Constant For All Conditions, Velocities and Aircraft Trim States	As Per Typical Parasite Drag Trends With 50% Improvement
Rotor System	Conventional Single Rotor Design	Assumed
	4 Blades	Assumed
	Constant Chord	As Per Estimation Methodology
	23012 or 230 M Airfoil Section From 0 to 0 9 R (12% Thick)	As Per Estimation Methodology
	9% Thick Symmetrical High Speed Tip From 0.9 to 1 0 R	As Per Estimation Methodology
	7% of Linear Twist	As Per Estimation Methodology
	Diameter 20 0, 25 0, 30 0, 35 0, 40 0, 45 0, 50 0 ft	Trade Off
	Solidity 0 14 and 0 16	Trade Off
	Tip Speed 615, 670, 725 ft/sec	Trade Off
	RPM According to Diameter and Tip Speed	Trade Off
	Chord According to Diameter and Solidity	Trade Off
	Tip Losses According to Chord	As Per Estimation Methodology
	Non Uniform Inflow Correction Factor on Induced Power 1 13	As Per Estimation Methodology
	Rotor Height Above ℄ of Landing Gear 10 ft	Assumed
Tail Rotor/Control Power	Calculated as a Percentage of Main Rotor Power Required Percentage Computed as a Symmetric Parabolic Function of Velocity Minimum Percentage Equal to 5% at 150 knots Percentage Equal to 12 5% at Hover and 296 knots Percentage Equal to 5 9% at 200 knots Percentage Equal to 12 9% at 300 knots	As Per Estimation Methodology
Required Power Installation	Based on 100% Power Rating at Sea Level Standard Day	Required To Equalize Comparisons
	Degraded by 5% for All Conditions at 30 Minute Contingency Rating	Assumed
	Degraded by 15% for Continuous Power Rating	Assumed
	Degraded by 6% for Installation and Transmission Losses	Assumed
	Lapse Rates With Altitude Under Standard Day Conditions	
	Degraded by 18 5% at 10,000 ft or 81 5% of Sea Level Rating Available	As Per Fig 3-93 AMCP 706-201 Pg 3-95*
	Degraded by 31 5% at 20,000 ft or 61.5% of Sea Level Rating Available	As Per Fig 3-93 AMCP 706-201 Pg 3-95*
	Degraded by 50% To Provide Single Engine Capability All Losses Cumulative	Required To Equalize Comparison
Vertical Drag	3.2% of Gross Weight in Hover	As Per Typical Vertical Drag Trends

* Engineering Design Handbook, Helicopter Engineering, Part One, Preliminary Design, AMC Pamphlet No 706-201, Headquarters, U.S Army Materiel Command, August 1974

- o Hover OGE at 10,000 ft./STD: Single Engine (user requirement); shown in Figure 9.
- o Hover IGE at 20,000 ft/STD: Single Engine (user requirement); shown in Figure 10.

The power required to achieve each of the sets of specified hover criteria has been referred to a required power installation based on 100 percent of the maximum Sea Level Standard Day power rating. In the course of resolving these various criteria to a common basis for comparison, the power required has been corrected for engine degradation between overhauls, power setting, transmission and installation losses, degradation of power output with altitude and single engine performance requirements.

As these figures illustrate, the required power installation increases drastically for rotor systems much less than 30 ft. in diameter, and a minimum acceptable limit on rotor diameter appears to be in the range of 27 to 28 ft. A rotor system of this size requires a total power installation of at least 4,300 shp to meet the most severe requirement to Hover-Out-of-Ground-Effect at 10,000 ft., Standard Day atmospheric conditions, at the Design Gross Weight of 8,000 lb. on one engine, as shown in Figure 9. The 4,300 shp required power installation is actually twice the required power installation necessary to achieve this flight condition in order to provide the capability to meet these criteria with a 50 percent reduction in power in the event of a power failure in one engine.

The combination of a 27-28 ft. diameter rotor and 4,300 shp easily meets the requirement to Hover-Out-of-Ground-Effect at Sea Level at the Design Gross Weight of 8,000 lb. on one engine, as shown in Figure 8. In fact, this combination further affords the capability to meet these conditions at the 10,000 lb. Alternate Design Gross Weight.

The combination of the 27-28 ft. diameter rotor and 4,300 shp of installed power also satisfies the requirement to Hover-In-Ground-Effect at 20,000 ft., Standard Day conditions, at the Minimum Mission Weight of 6,000 lb. on one engine. As Figure 10 illustrates, attempting to meet the 20,000 ft. hover criteria at the Design Gross Weight of 8,000 lb. with a 27-28 ft. diameter rotor requires the installation of 7,500 shp and results in a design requirement that should be avoided for reasons of excessive blade stall. In order to meet the 20,000 ft. hover requirement at a Design Gross Weight of 8,000 lb., a minimum rotor diameter of 30 ft. is required with a 6,000 shp power installation. Alternatively, satisfying this requirement at the 8,000 lb. Design Gross Weight with the 4,300 shp power installation requires a rotor diameter of at least 33 to 34 ft.

The effects of blade stall are independent of power installation and can only be resolved by increasing rotor diameter and, hence, blade area or reducing loading. Table 17 illustrates the minimum allowable rotor diameter necessary to prevent stall for each of the three gross weight conditions and hover criteria.

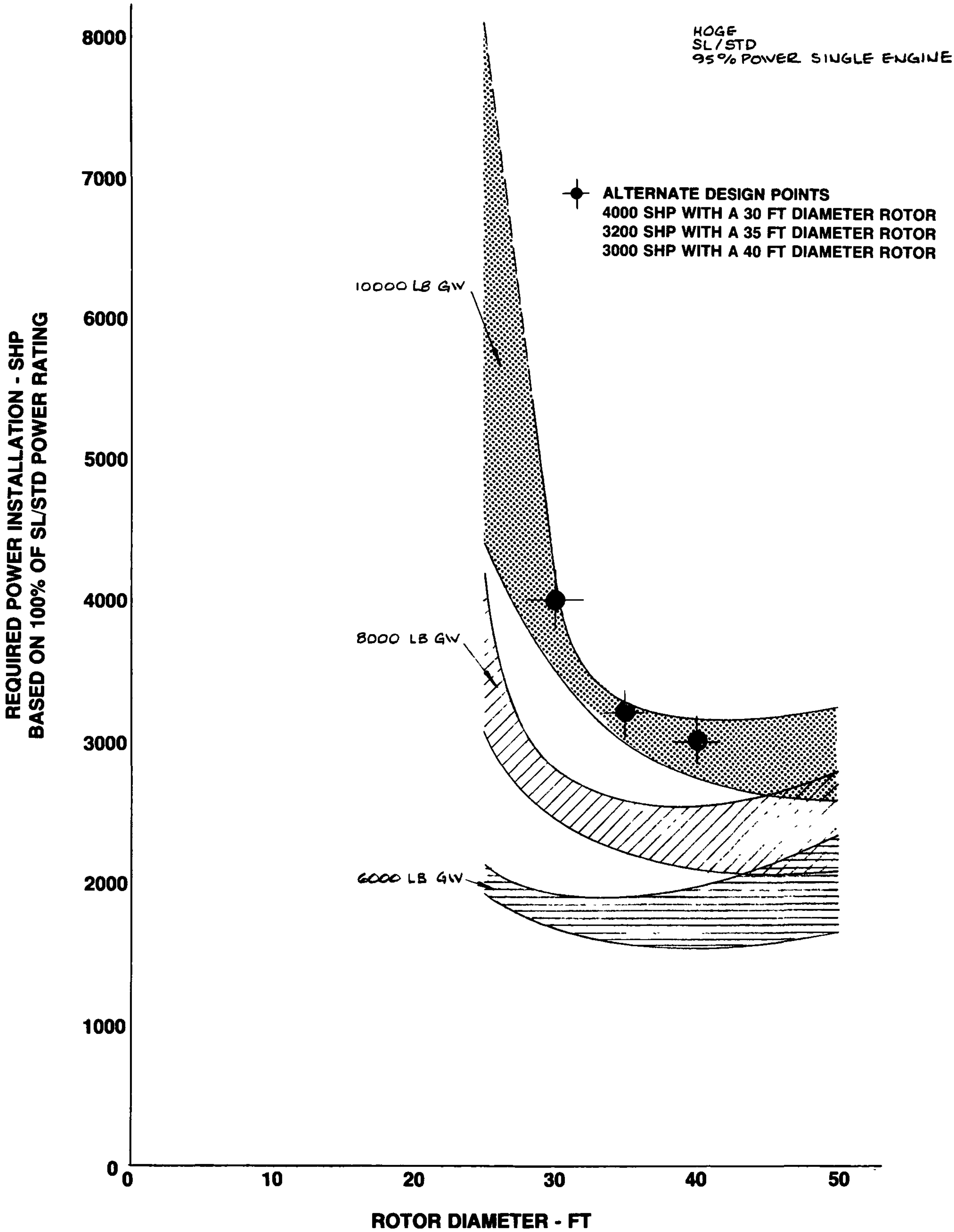

FIGURE 8. HOVER OUT OF GROUND EFFECT AT SEA LEVEL

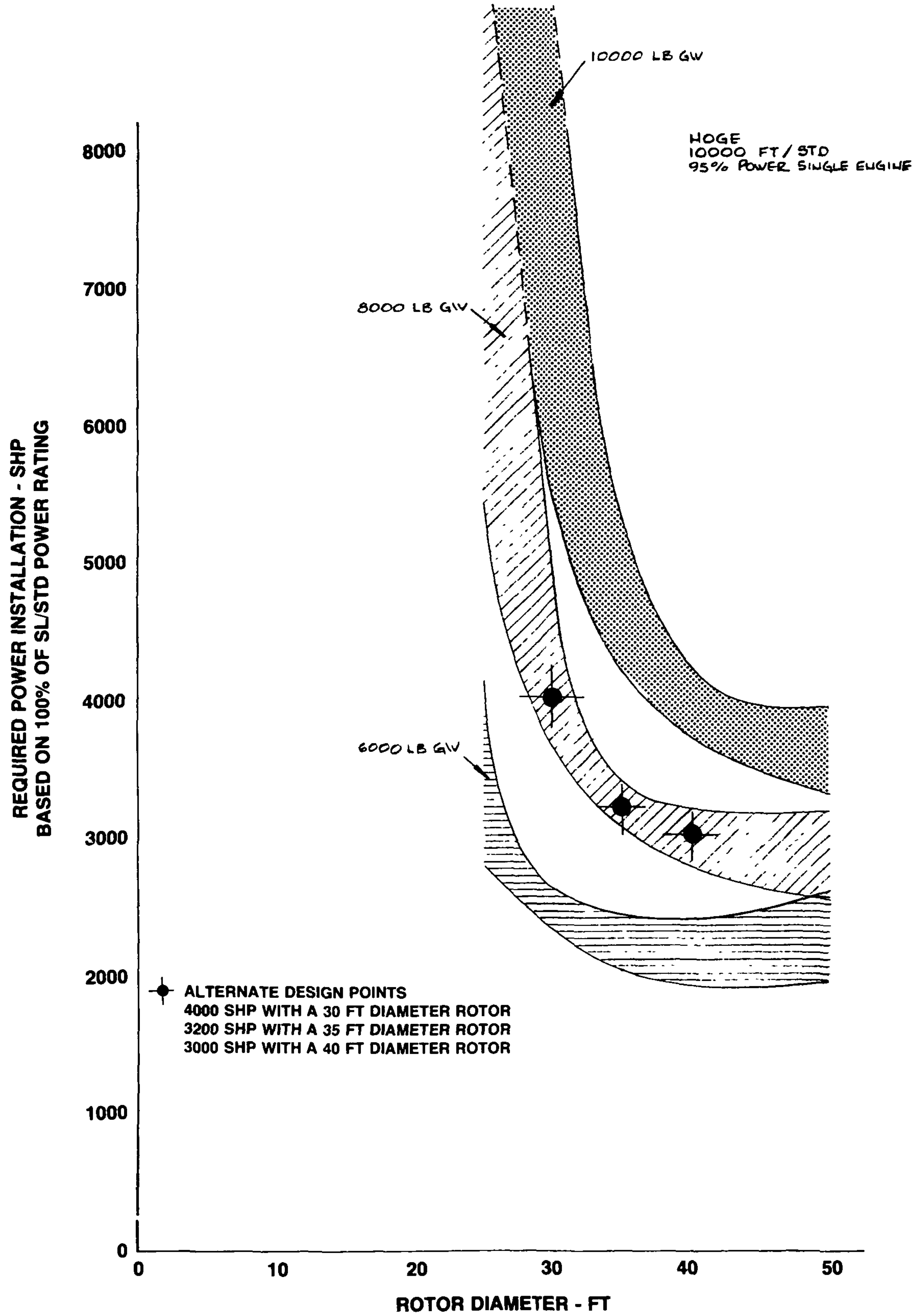

FIGURE 9. HOVER OUT OF GROUND EFFECT AT 10,000 FEET

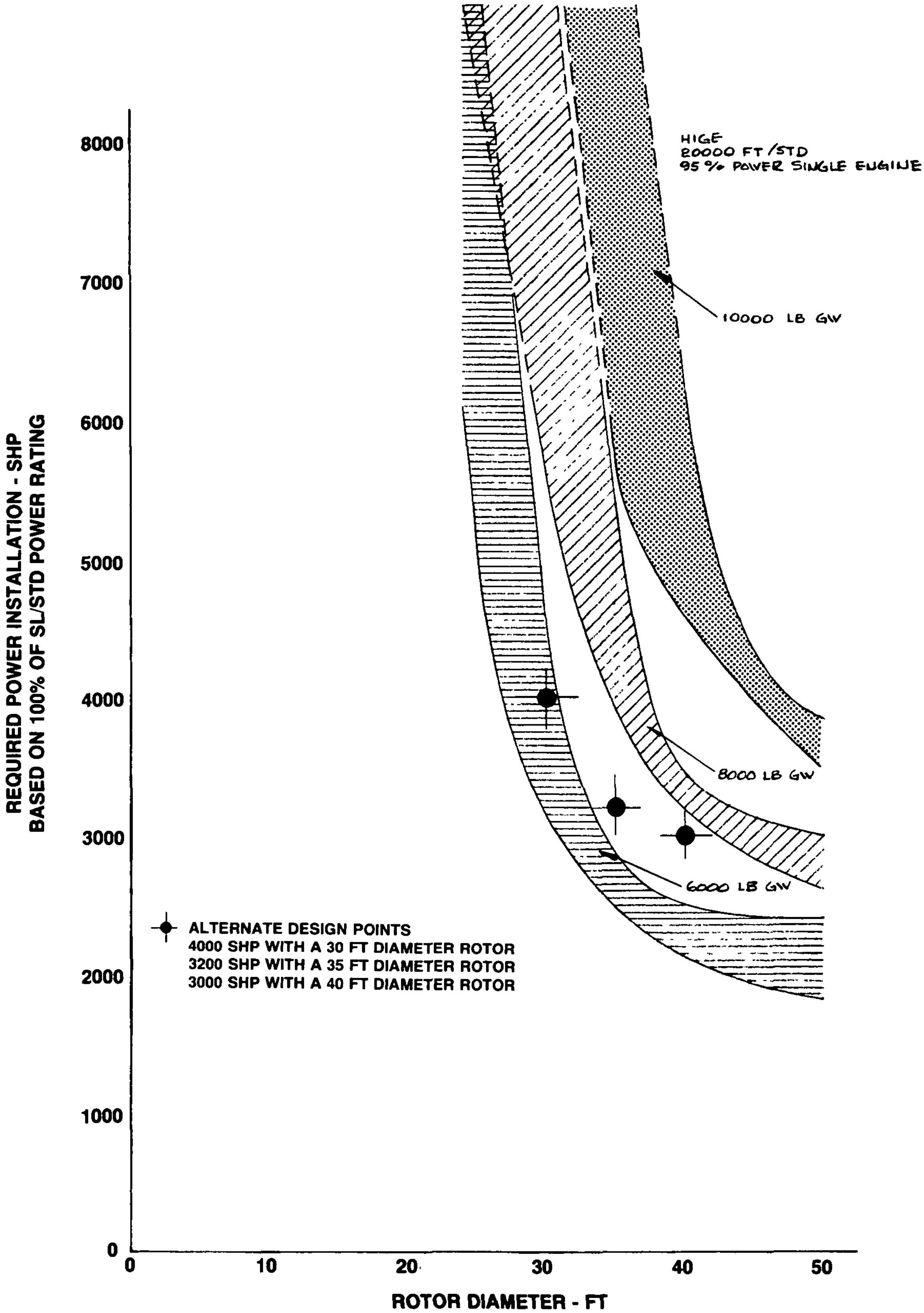

FIGURE 10. HOVER IN GROUND EFFECT AT 20,000 FEET

The symbols (◆) in Figures 8, 9 and 10 illustrate three representative design alternatives which satisfy the hover criteria at a Design Gross Weight of 8,000 lb. from Sea Level to 10,000 ft., Standard Day conditions, and also satisfy the 20,000 ft., Standard Day conditions, hover criteria at a Minimum Mission Weight of 6,000 lb. These three options trade off increases in rotor diameter from 30 ft. to 40 ft. to achieve reductions in required power from 4,000 to 3,000 shp. In each case, the design option is capable of exceeding the 8,000 lb. Design Gross Weight at Sea Level, Standard Day conditions and, in fact, capable of meeting the single engine hover criteria at the 10,000 lb. overload or Alternate Design Gross Weight at Sea Level. The effect of increased diameter offers a substantial improvement in rotor lift capabilities based on the hover criteria at 20,000 ft. It can be concluded from Figure 10 that increasing the rotor diameter to 40 ft. eliminates blade stall problems in hover and offers a 20-25 percent improvement in gross lift capabilities from the 6,000 lb. Minimum Mission Weight to approximately 7,500 lb.

Figure 11 illustrates a consolidation of the achievable hover criteria. It can be seen from Figure 11 that the 10,000 ft. hover criteria at a Design Gross Weight of 8,000 lb. represents the driving requirement on rotor diameter versus required power installation.

FORWARD FLIGHT CAPABILITIES

Figure 12 illustrates the relationship between rotor diameter and required power installation at a Design Gross Weight of 8,000 lb. for the two specified forward flight conditions, as follows:

- Continuous cruise speed of 200 knots at SL/STD: Both engines at 85 percent power.

- 30 minute dash speed of 300 knots at SL/STD: Both engines at 95 percent power.

As in the case of hover discussed in the preceding section, the power required for the given flight condition has been resolved back to a required power installation based on 100 percent of maximum Sea Level Standard Day power rating in order to provide a common basis for comparison and conform with the methodology used to assess the hover criteria. Since the forward flight requirements are being evaluated at Sea Level Standard Day in routine flight with both engines operative, the power required is corrected only for engine degradation between overhauls, power setting, and installation and transmission losses.

Figure 12 illustrates a pronounced minimum in the required power installation as a function of rotor diameter. The minimum required power installation is observed to occur with a 29 ft. diameter rotor in cruise at 200 knots and a 24 ft. diameter rotor in the case of the 300 knot dash speed requirement. The strong minimums evident at both the 200 and 300 knot speeds are a result of the combined effects of compressibility and blade stall. Obviously, the optimum design point for either condition occurs at the rotor diameter corresponding to the minimum value for the required power installation. In attempting to satisfy both design points simultaneously, an intermediate rotor diameter lying between the minimum points should be selected.

TABLE 17

BLADE STALL LIMITS ON MINIMUM ROTOR DIAMETER

Hover Criteria	6,000 lb. Min. Mission Weight	8,000 lb. Design Gross Weight	10,000 lb. Alternate Design Gross Weight
Hover OGE at SL/STD	25 ft.	25 ft.	25 ft.
Hover OGE at 10,000 ft/STD	25 ft.	25 ft.	29 ft.
Hover IGE at 20,000 ft/STD	25 ft.	29 ft.	35 ft.

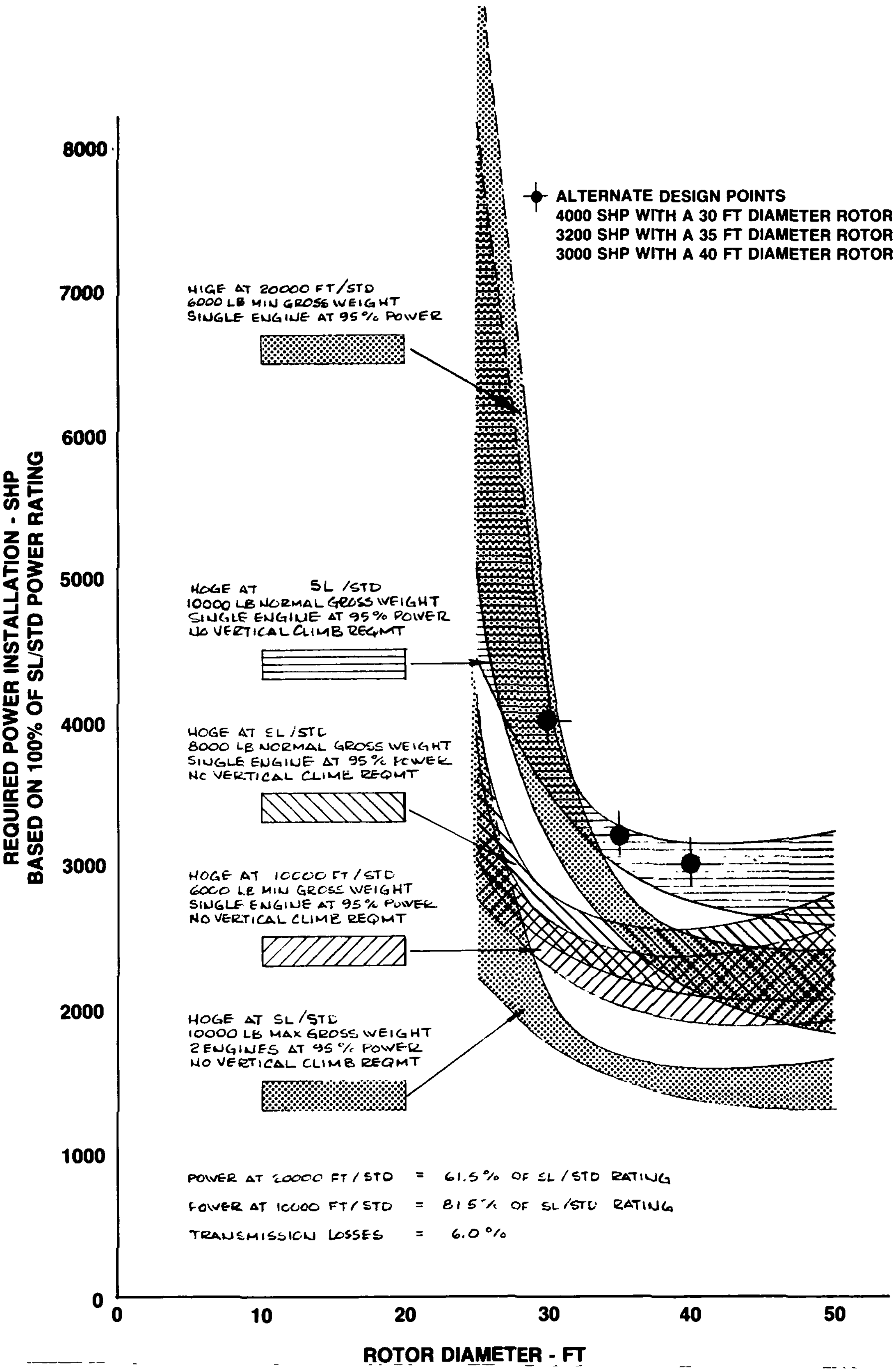

FIGURE 11. REQUIRED POWER INSTALLATION TRADEOFFS FOR HOVER CRITERIA

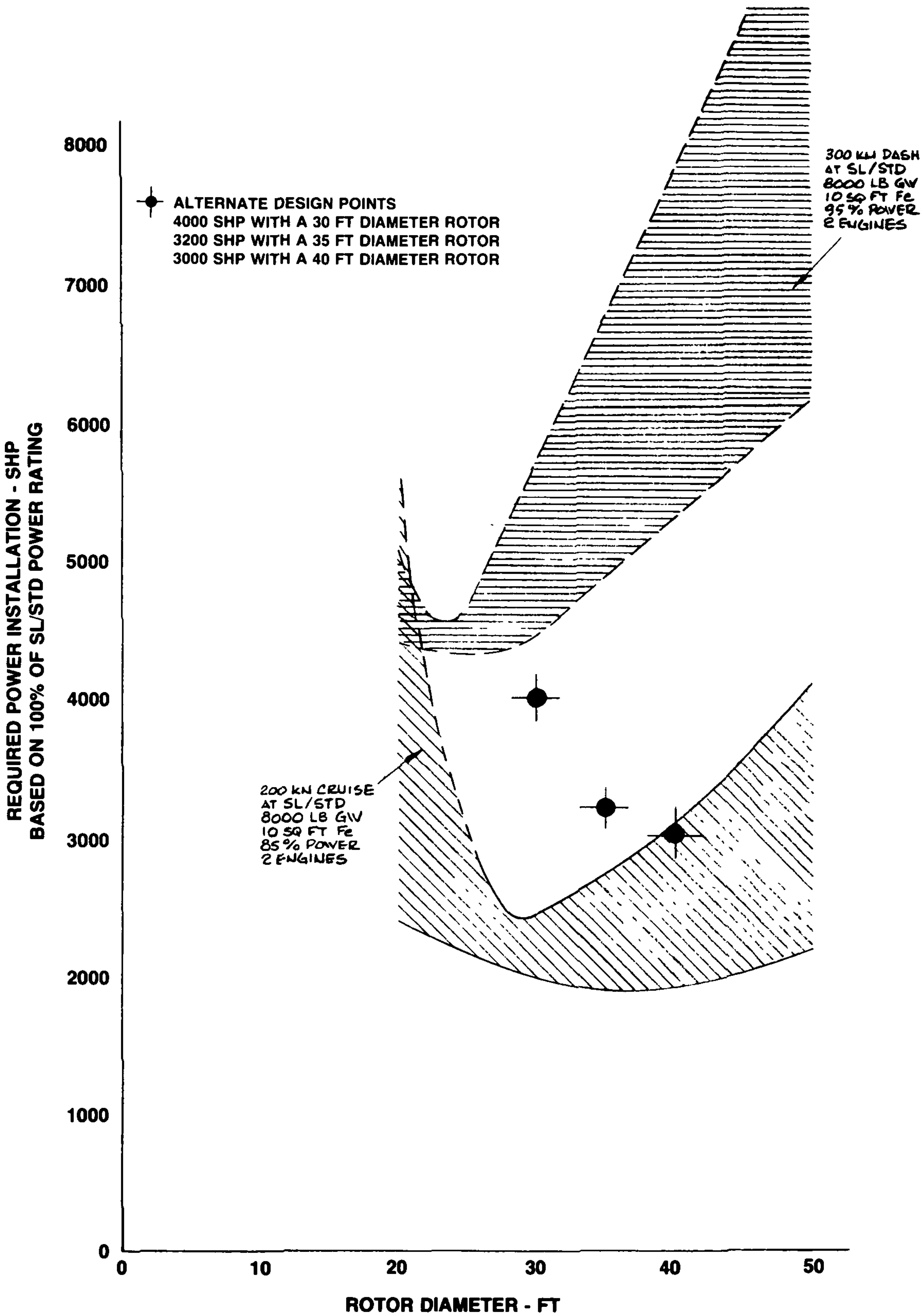

FIGURE 12. REQUIRED POWER INSTALLATION TRADEOFFS FOR FORWARD SPEEDS OF 200 AND 300 KNOTS

The broad band for each speed represents the various combinations of rotor solidities and tip speeds considered representative of typical single rotor helicopter designs. At the 200 knot cruise speed, the tip speeds under consideration (615, 670, 725 ft/sec) yield values for the advancing blade tip Mach number which range from 0.85 to 0.95. Since the range of advancing blade tip Mach numbers is less than our assumed limiting value of 0.95, the combination of tip speeds and the 200 knot cruise speed requirement can be tolerated without compromise. Furthermore, the compressibility power model incorporated in the performance estimation program has been shown to yield reasonable accuracy at tip Mach numbers up to 0.95, which suggests that the required power to rotor diameter relationships shown for the 200 knot cruise speed in Figure 12 are fairly reliable.

The predicted values for the required power installation at the 300 knot dash speed condition should be viewed with some caution, however, due to the severe effects of compressibility at this high forward flight speed. The combination of the 300 knot forward speed and tip speeds of 615 to 725 ft/sec results in advancing blade tip Mach numbers ranging from 1.0 to 1.1, indicating that certain portions of the rotor are operating supersonically. While the compressibility power model has proven to be reasonably accurate at tip Mach numbers up to 0.95, the reliability of the model at supersonic tip speeds is extremely uncertain. It is certain, however, that the entire 300 knot dash speed envelope represents a region of compressibility problems at the tip speeds under consideration. As a result, steps should be taken in the course of preliminary design to avoid this region of operation. The only alternatives at this point are to either reduce the tip speed, reduce the required dash speed capability or accept sustained operation of the rotor at the resulting supersonic advancing blade tip Mach numbers.

Reducing the rotor speed aggravates the problem of retreating blade stall and results in further degradation of the maximum rotor lift coefficient, $(C_T/\sigma)_{max}$, such that the rotor is incapable of providing sufficient lift for sustained forward flight. This necessitates either increasing the rotor size or solidity to provide increased blade area, or adding auxiliary lift systems such as a wing or an additional rotor system.

In order to reduce the advancing blade tip Mach number at the 300 knot dash speed to a value less than or equal to 0.95, the rotor speed must be reduced to a tip speed of less than 554 ft/sec. This results in an advance ratio of 0.914 at 300 knots and a maximum rotor lift coefficient of 0.0197. This corresponds to a maximum blade loading of 14.37 lb./sq. ft. at Sea Level Standard Day, or a requirement for 556.7 sq. ft. of blade area to produce 8,000 lbs. of lift at 300 knots with no auxiliary lift devices. Table 18 illustrates the minimum rotor blade area required to produce various amounts of lift (with no auxiliary lift systems) at an advancing blade tip Mach number of 0.95, as a function of maximum forward flight speed.

TABLE 18

BLADE AREA REQUIRED AT AN ADVANCING BLADE TIP MACH NUMBER OF 0.95

Maximum Lift	Maximum Flight Speed			
	Hover	100 Knots	200 Knots	300 Knots
6,000 lb.	18.7 sq. ft.	27.5 sq. ft.	51.5 sq. ft.	417.5 sq. ft.
8,000 lb.	25.0 sq. ft.	36.6 sq. ft.	68.7 sq. ft.	556.7 sq. ft.
10,000 lb.	31.2 sq. ft.	45.8 sq. ft.	85.9 sq. ft.	695.9 sq. ft.

Based on the values shown in Table 18, the 300 knot condition requires 810 percent more blade area than the 200 knot cruise speed. This results in a substantial increase in rotor blade and hub weight and, consequently, the empty weight of the aircraft. It should be emphasized that the required blade areas shown in Table 18 are minimum required blade areas to produce the given values of thrust at a maximum rotor lift coefficient. Since the thrust values shown in Table 18 correspond to the aircraft gross weights under consideration, the blade areas shown relate to a maximum 1.0 g maneuver load factor at the flight speeds shown. Technological developments in blade materials and hub design may offer some benefits in this respect, but it is unlikely that such developments will be sufficient to offer incremental rotor lift efficiencies better than a wing for high forward flight speeds.

These considerations lead to several alternatives as potential solutions to the combined effects of compressibility and blade stall at the 300 knot dash speed, as follows:

- Operating the rotor at supersonic tip speeds in the range of Mach 1.0 to 1.1.
- Providing a variable speed transmission and rotor system to reduce tip speeds at forward flight speeds in excess of 200 knots. This may require auxiliary lift systems such as a wing to augment the rotor system in high speed flight.
- Providing a high solidity rotor at fixed reduced speed. This necessitates an assessment of the increased rotor weight and its impact on empty weight versus the weight penalties associated with the other options.
- Incorporating substantial forward tilt of the main rotor shaft or providing for variable shaft tilt as a function of forward flight speed. This provision serves to reduce the combined effects of compressibility and blade stall at high forward flight speeds, while providing a significant increase in forward propulsive thrust at little increase in rotor loading. For example, at 30° of forward shaft tilt, the rotor can develop a forward thrust component equal to 58 percent of the vertical thrust, or lift, with less than a 16 percent increase in total thrust and blade loading.

- Developing a compromise design involving all of the above.

Having addressed the hover criteria and the forward flight criteria separately in terms of the capabilities offered by the conventional single rotor helicopter, the next subsection will consolidate these requirements to examine the impact of rotor diameter on the combined needs of the user.

COMPARISON OF HOVER AND FORWARD FLIGHT REQUIREMENTS

Figures 13 through 17 illustrate a comparison of the required power installation necessary to satisfy both the hover and forward flight criteria as a function of rotor diameter. While the various requirements are specified under substantially different operating conditions, the power required for each set of criteria has been referred to a required power installation based on 100 percent of maximum Sea Level Standard Day power rating.

The forward flight conditions shown in the figures are based on operation at a design gross weight of 8,000 lb. on two engines at Sea Level Standard Day conditions with an equivalent parasite drag area, Fe, equal to 10 sq. ft. The 200 knot cruise speed is assumed to occur at 85 percent power for continuous operation, while the 300 knot dash speed uses 95 percent power, assumed to correspond to a 30 minute time limit, as specified by the workshop. The three symbols (◆) in each figure illustrate the three representative rotor diameters discussed earlier.

The regions on each chart defined by the overlap of the hover criteria and the forward flight criteria illustrate those combinations of minimum rotor diameter and engine installation required to satisfy both conditions. It was concluded in the discussion of hover capabilities that the most severe hover condition was defined by the 10,000 ft. requirement to Hover-Out-of-Ground-Effect at the 8,000 lb. design gross weight on one engine. This requirement is shown in Figure 15. This figure demonstrates that rotors in the range of 40 to 50 ft. with approximately 3,000 shp of installed power best satisfy the combined hover/cruise criteria, while rotors in the range of 27 to 30 ft. in diameter with approximately 4,500-5,500 shp of installed power best satisfy the combined hover/dash criteria.

As pointed out in the previous section, the combination of the 300 knot dash speed and the typical tip speeds employed in this assessment result in advancing blade tip Mach numbers ranging from 1.0 to 1.1, which means that certain portions of the rotor are operating supersonically, causing severe compressibility problems. To avoid these problems, the advancing blade tip Mach number must be limited to values less than or equal to 0.95. Imposing this limitation results in maximum forward flight speeds much less than the desired 300 knot dash capability, as shown in Table 19.

In order to achieve the desired 300 knot dash capability at an advancing blade tip Mach number of 0.95, the tip speed must be reduced to 554 ft/sec. This results in a requirement for increased blade area in order to restore rotor lift to the required values, or the addition of auxiliary lift devices as discussed in the previous section.

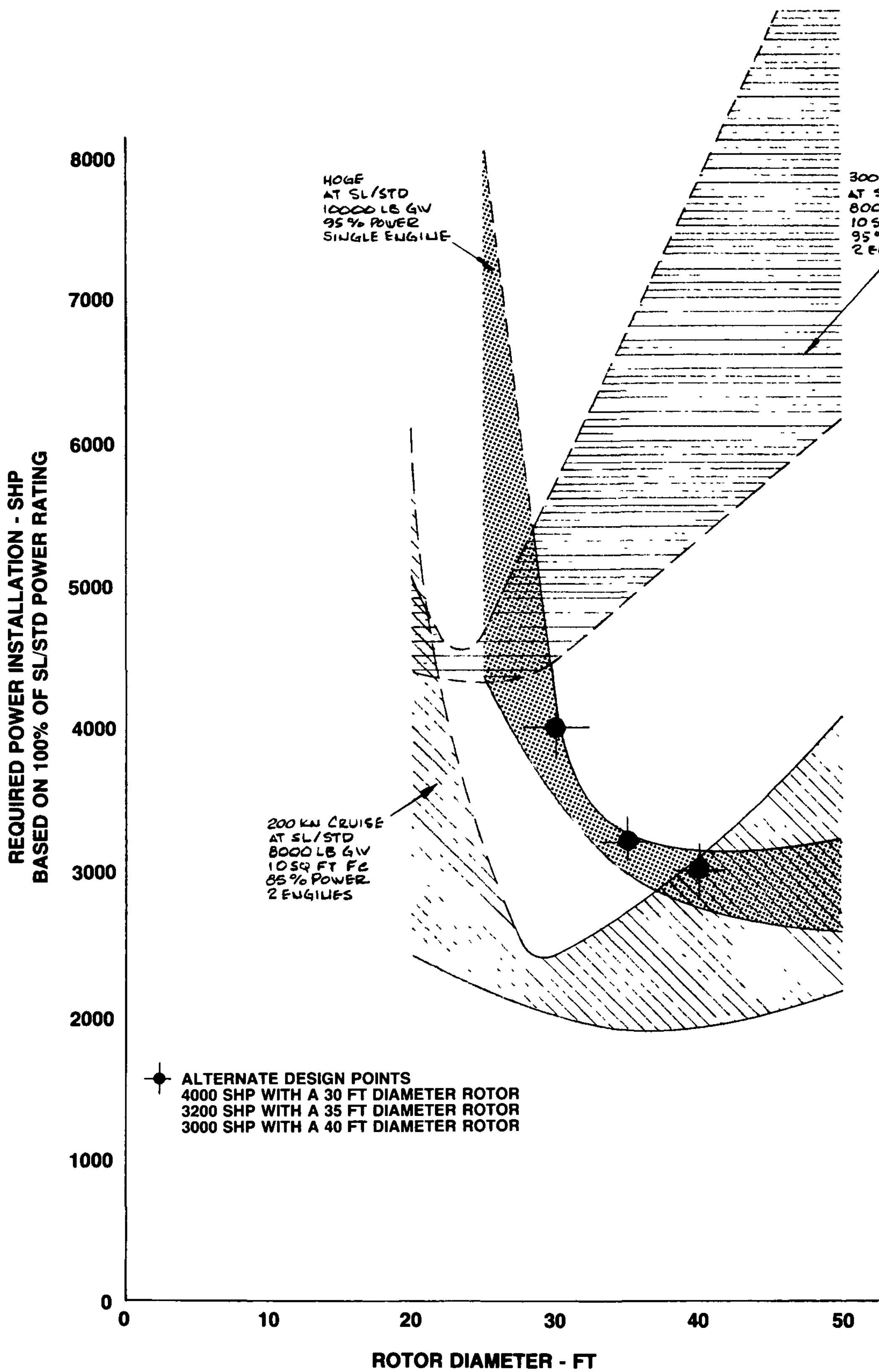

FIGURE 13. COMPARISON OF REQUIRED POWER INSTALLATION FOR HIGH SPEED FLIGHT AND OGE HOVER AT SEA LEVEL, 10,000 LB. GROSS WEIGHT

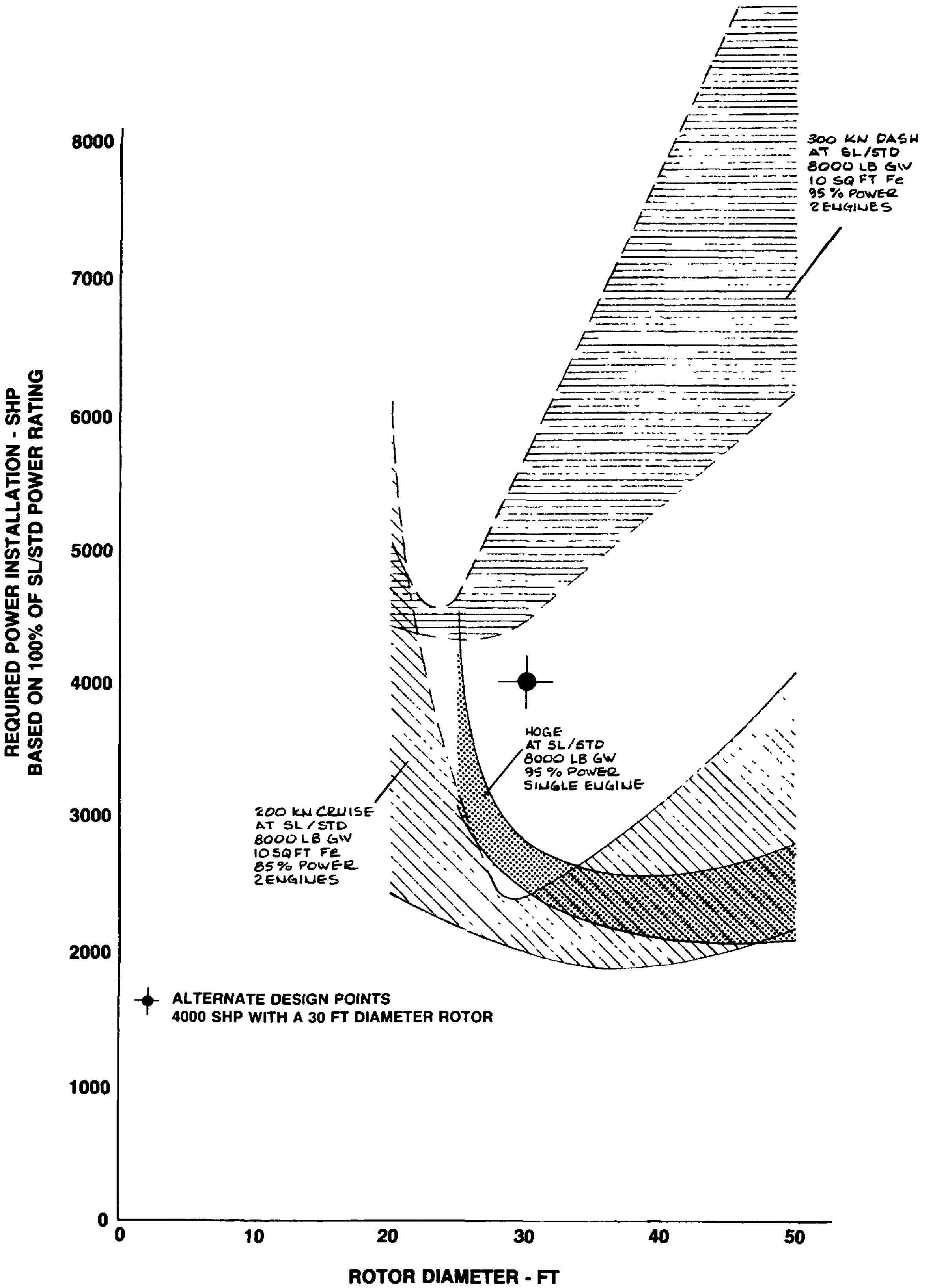

FIGURE 14. COMPARISON OF REQUIRED POWER INSTALLATION FOR HIGH SPEED FLIGHT AND OGE HOVER AT SEA LEVEL, 8,000 LB. GROSS WEIGHT

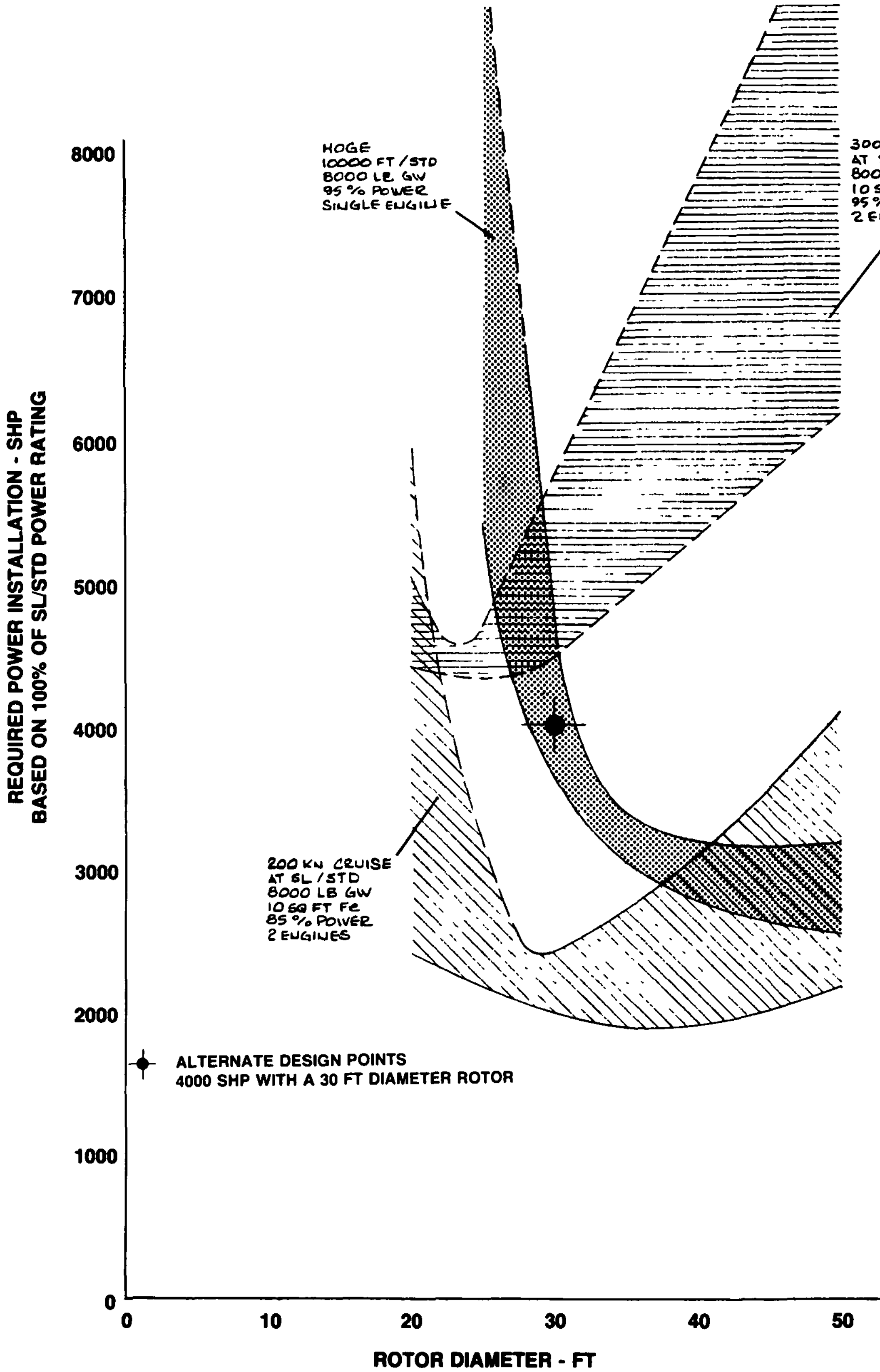

FIGURE 15. COMPARISON OF REQUIRED POWER INSTALLATION FOR HIGH SPEED FLIGHT AND OGE HOVER AT 10,000 FEET, 8,000 LB. GROSS WEIGHT

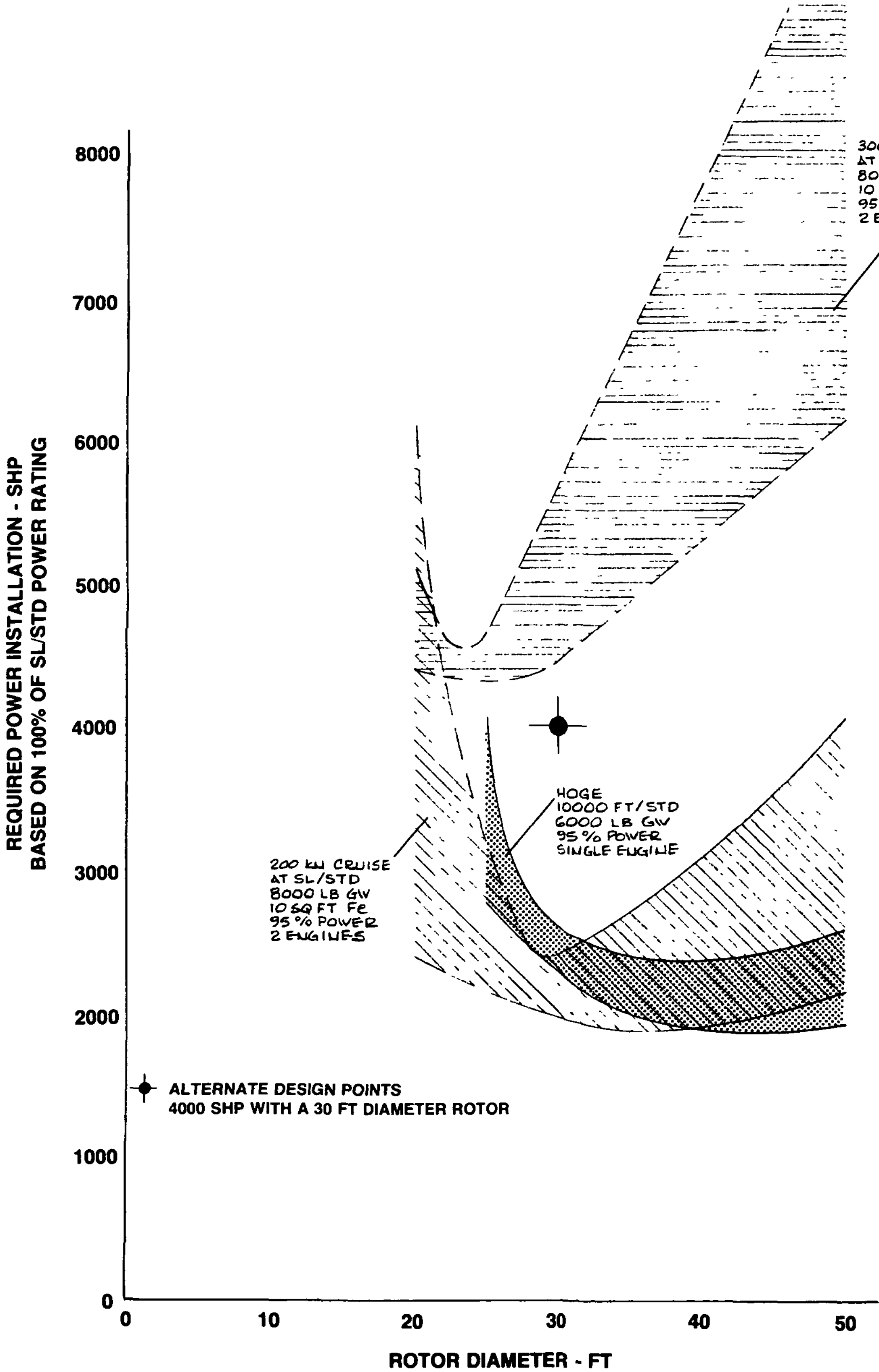

FIGURE 16. COMPARISON OF REQUIRED POWER INSTALLATION FOR HIGH SPEED FLIGHT AND OGE HOVER AT 10,000 FEET, 6,000 LB. GROSS WEIGHT

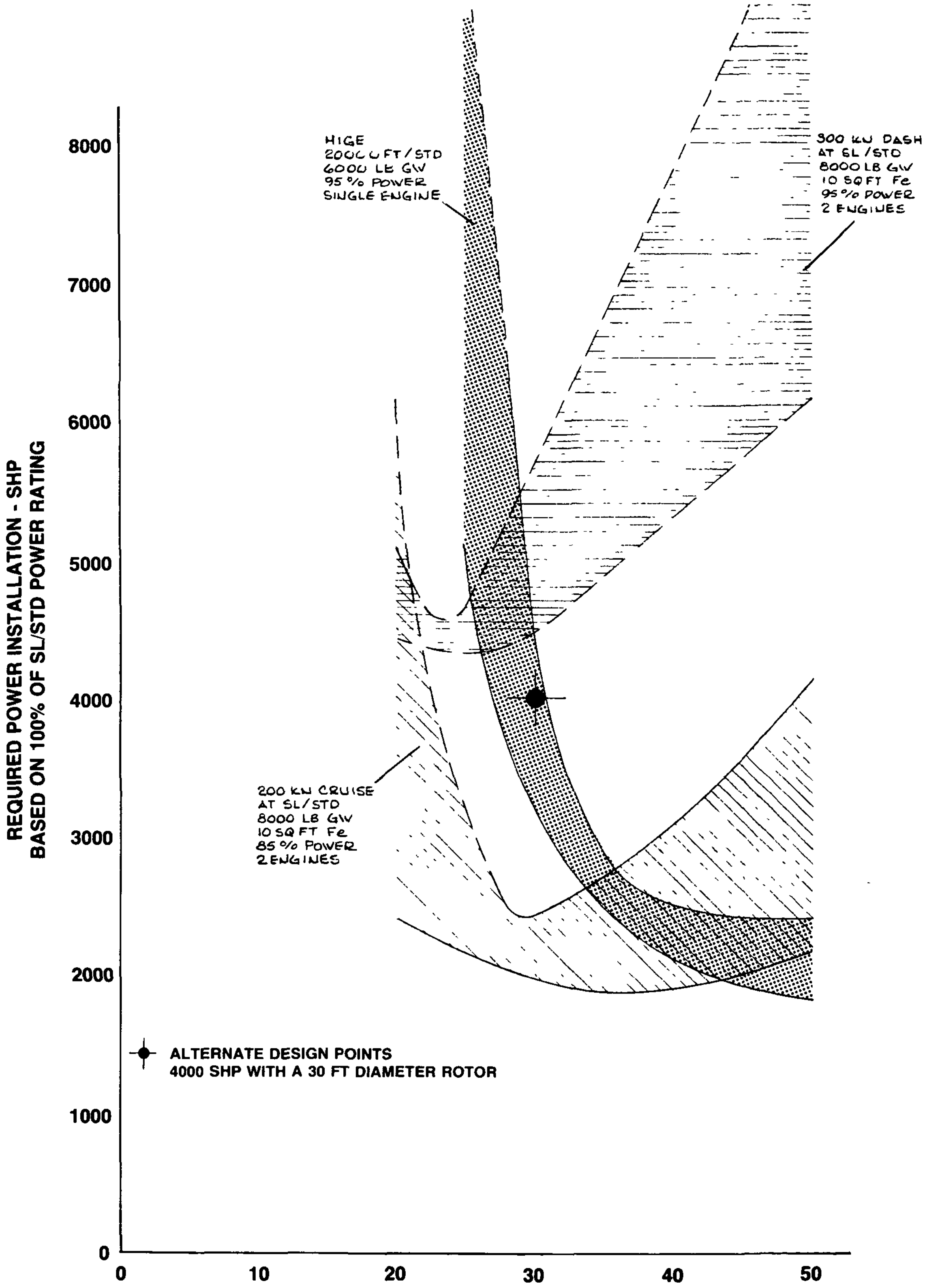

FIGURE 17. COMPARISON OF REQUIRED POWER INSTALLATION FOR HIGH SPEED FLIGHT AND IGE HOVER AT 20,000 FEET, 6,000 LB. GROSS WEIGHT

TABLE 19

MAXIMUM FORWARD FLIGHT SPEEDS AT AN ADVANCING BLADE TIP MACH NUMBER OF 0.95

Rotor Tip Speed ft/sec	Maximum Allowable Advancing Blade Tip Mach Number	Maximum Forward Flight Speed in Knots
615	0.95	263.8
670	0.95	231.2
725	0.95	198.6

EFFECTS OF MANEUVER LOADING

All of the relationships between rotor size or required blade area and rotor thrust presented thus far have been based on operation at a maximum rotor lift coefficient, C_T/σ , which allows no margin for forward thrust or maneuver loadings and results in a high degree of stall sensitivity to gusts and random variations in airflow. Table 20 illustrates the relationships between rotor tip speed, maximum forward flight speed, blade loading and required blade area for a helicopter at the 8,000 lb. design gross weight and three different maneuver load factors ranging from 1.0 g to 3.0 g. The values of required blade area shown in Table 20 for the three maneuver load conditions are based on the load factor occurring at the maximum forward flight speed indicated and presume that no auxiliary lift systems are employed.

Figure 18 facilitates the conversion of maximum forward flight speed, maneuver load factor and required blade area into an equivalent required rotor diameter for an 8,000 lb. helicopter at six values of rotor solidity ranging from 0.1 to 0.2 The figure was based on the maximum allowable forward flight speed limited by an advancing blade tip Mach number of 0.95 and the required blade area computed based on operation at a maximum rotor lift coefficient, C_T/σ, as a function of the advance ratio, μ , squared. The combination of required maximum forward flight speed and desired maneuver load factor specifies the required blade area for the 8,000 lb. helicopter. The example shown in Figure 18 selects a 2.0 g maneuver capability at 275 knots, which requires approximately 400 sq. ft. of blade area. Selecting a solidity of 0.18 yields a required rotor diameter of 53 ft. Selecting a 2.0 g maneuver capability at 200 knots requires only 140 sq. ft. of blade area, which equates to a 31 ft. diameter rotor at a solidity of 0.18. However, by reversing this process, it can be seen that the 140 sq. ft. of blade area limits the maximum maneuver load factor to less than 0.7 g's at the 275 knot dash speed, which is insufficient to support sustained straight and level flight performance, requiring some form of auxiliary lift system to provide the remaining 30 percent of required lift.

It can be seen that for the 8,000 lb. helicopter at load factors between 1.0 and 2.0 g, the 200 knot cruise speed requirement can be satisfied with a rotor in the range of 21 to 41 ft. in diameter; while the 300 knot speed requires a rotor on the order of at least 60 ft. in diameter at 1.0 g, and as much as 120 ft. in diameter at 2.0 g.

TABLE 20

REQUIRED BLADE AREA TRADEOFFS

Rotor Tip Speed ft/sec	Advancing Blade Tip Mach Number	Maximum Forward Flight Speed - knots	Advance Ratio-μ	Maximum Rotor Lift Coefficient-$(Ct/\sigma)_{max}$	Maximum Allowable Blade Loading - lb/sq ft	Required Blade Area for an 8000 lb Aircraft at a 1 0g Maneuver at Maximum Speed	Required Blade Area for an 8000 lb Aircraft at a 2 0g Maneuver at Maximum Speed	Required Blade Area for an 8000 lb Aircraft at a 3 0g Maneuver at Maximum Speed
500 (3)	0 95	331 9	1 1204	0 00000	0 00	— (4)	— (4)	— (4)
550 (3)	0 95	302 3 (1)	0 9276	0 01674	12 04	664 7	1329 4	1994 1
600 (3)	0 95	272 7	0 7670	0 04940	42 28	189 2	378 5	567 7
615 (2)	0 95	263 8	0 7239	0 05711	51 35	155 8	311 6	467 4
650 (3)	0 95	243 0	0 6310	0 07221	72 52	110 3	220 6	331 0
670 (2)	0 95	231 2	0 5824	0 07930	84 61	94 6	169 2	253 8
700 (3)	0 95	213 4	0 5146	0 08823	102 76	77 86	155 7	233 6
725 (2)	0 95	198 6 (1)	0 4623	0 09435	117 88	67 87	135 7	203 6
750 (3)	0 95	183 8	0 4136	0 09947	133 00	60 15	120 3	180 5

Notes

(1) Approximately Meets Specified User Requirement

(2) Used in Performance Assessment

(3) Included to Show Trend

(4) Approaches Infinity

(5) Sea Level Standard Day conditions

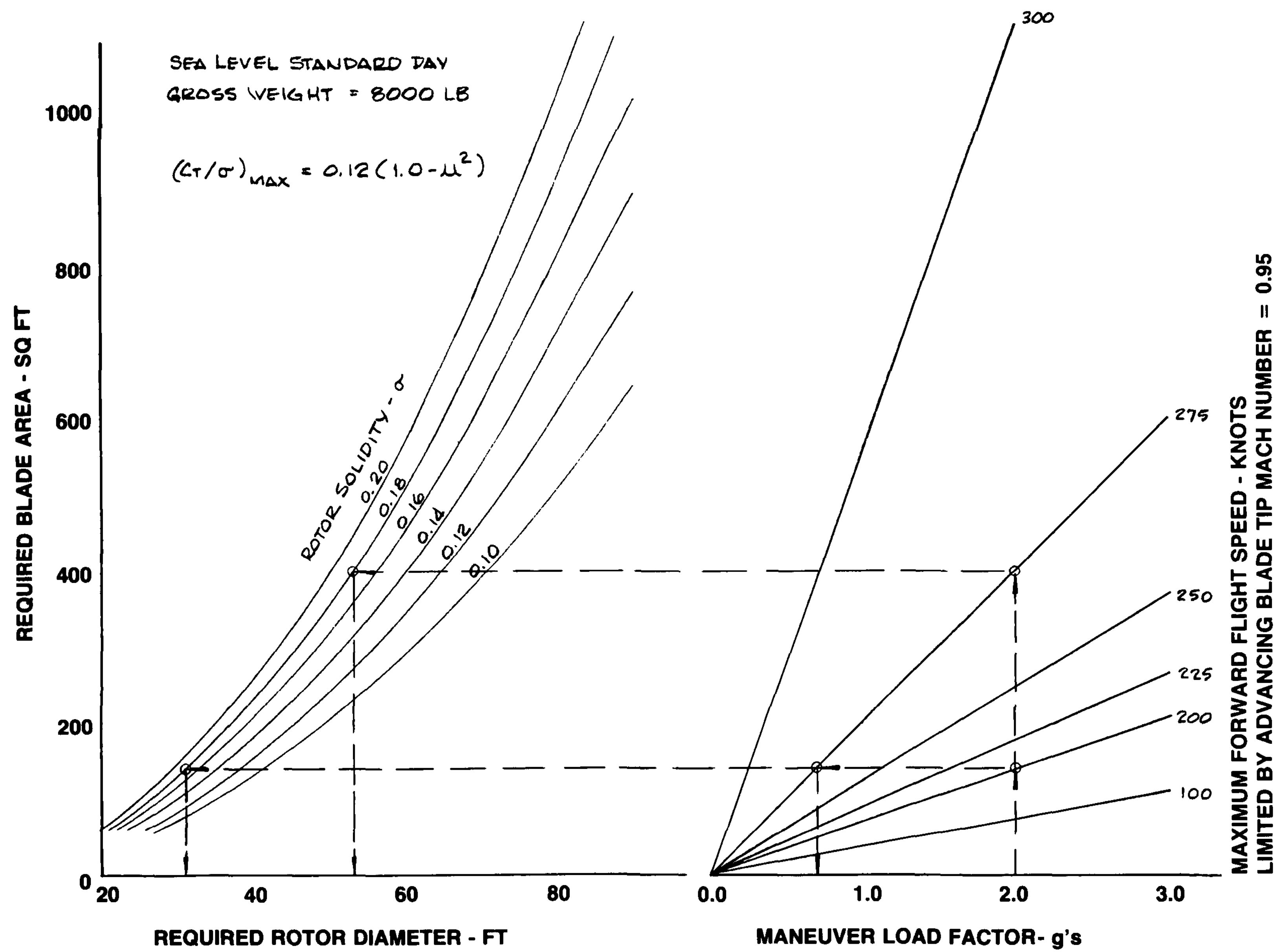

FIGURE 18. MANEUVER LOAD RELATIONSHIPS

Therefore, the maximum forward flight speed requirement defines the maximum allowable rotor tip speed, as limited by the Mach number of the advancing blade tip, according to:

$$V_t = a \times M_{90} - V_{ff}$$

where V_t = rotor tip speed in ft/sec

a = speed of sound: 1,116 ft/sec at SL/STD

M_{90} = Mach number of the advancing blade tip = 0.95

V_{ff} = maximum required forward flight speed in ft/sec

The combination of the maximum required forward flight speed and maximum allowable rotor tip speed, defined above, define the maximum potential rotor thrust coefficient as degraded by retreating blade stall, according to:

$$(C_T/\sigma)_{max} = 0.12\ (1 - \mu^2)$$

where $(C_T/\sigma)_{max}$ = maximum rotor lift coefficient

0.12 = maximum C_T/σ in hover

μ = advance ratio = V_{ff}/V_t

The combination of the maximum rotor lift coefficient, desired maneuver load factor and the required rotor thrust at a 1.0 g load factor for straight and level flight defines the minimum blade area necessary to meet these conditions, according to:

$$\text{Blade Area}_{min} = \frac{g \times T_0}{\rho (V_t)^2\ (C_T/\sigma)_{max}}$$

where Blade Area $_{min}$ = minimum required blade area

g = desired maneuver load factor

T_0 = required rotor thrust at 1.0 g

ρ = air density

σ = rotor solidity

Finally, the combination of required blade area and rotor solidity define the minimum rotor area and, hence, the rotor diameter necessary to meet the required maximum forward flight speed criteria, according to:

$$\text{Rotor Area}_{min} = \frac{\text{Blade Area}_{min}}{\sigma}$$

where Rotor Area $_{min}$ = required rotor disk area;

σ = rotor solidity.

If the blade area required to meet the maximum forward flight speed criteria results in an unacceptably large rotor either in terms of size or weight, a more efficient high speed lift device such as a fixed wing must be employed to meet the high speed lift requirements.

VI. ALTERNATIVE ROTORCRAFT DESIGNS

The results of the preliminary assessment presented in the preceding chapter indicate that a conventional single rotor helicopter design may nearly satisfy most of the Public Service Helicopter user requirements, allowing that:

- o The rotor diameter is increased substantially beyond the 20-ft. specified limit into the range of 40 ft., as dictated by the limits on allowable downwash velocity and desired or required maneuver capability at the high speed flight conditions.
- o The desired dash speed requirement is reduced by some 10 to 20 percent, with a possible requirement to add auxiliary lift devices to provide the necessary maneuver capability at the high speed flight conditions.
- o The conventional tail rotor assembly generally used on single rotor helicopter designs is replaced with some alternative form of countertorque control.

This chapter will briefly address other design alternatives and the potential benefits these options offer in the PSH role. The alternatives considered in this chapter include an unconventional single rotor design, the Advancing Blade Concept (ABC) coaxial rotor design, and the tilt rotor design. While the principal advantage of these alternatives is an increase in maximum flight speed, they tend to have increased empty weight, size and downwash velocities when compared with a conventional single rotor helicopter design of equal load carrying capability.

THE UNCONVENTIONAL SINGLE ROTOR HELICOPTER DESIGN

The conventional single rotor helicopter reaches the limit of its lift potential in the range of 200 to 260 knots due to the combined effects of blade stall and compressibility. These effects converge with increasing flight speed to degrade the maximum lift potential of the rotor system or the maximum allowable blade loading. As a result, the maneuver margins of the aircraft steadily decrease with increasing flight speed due to retreating blade stall.

Eventually, at some forward flight speed, the rotor is barely capable of providing sufficient lift to support the aircraft weight at a 1.0 g maneuver load condition, at which point no maneuver capability exists. Further increases in speed result in further degradation of rotor lift such that the rotor is incapable of providing sufficient lift for sustained straight and level flight and the aircraft descends, either controllably or uncontrollably depending on the design, due to retreating blade stall.

Typically, the most immediate solutions to this problem have been either to increase the blade area of the rotor or add a fixed wing to augment or unload the rotor in high speed flight. These options, however, tend to degrade the hover performance of the aircraft to some extent, as follows:

- The increased blade area required to offset blade stall in maneuvers and at high speeds results in increased profile power in hover and forward flight.
- The addition of a wing increases the vertical drag of the aircraft in hover, requiring increased total rotor thrust in hover to offset the download on the wing.
- Both options tend to increase the empty weight of the aircraft, requiring a further increase in rotor thrust to lift a specified load.

It can be seen from Figure 18 in the preceding chapter that the blade area required to produce 8,000 lb. of lift at 200 knots is 70.0 sq. ft. This area must be increased by 490.0 sq. ft., or 8 times, to 560.0 sq. ft. to produce 8,000 lb. of lift at 300 knots. Conversely, the original 70.0 sq. ft. of blade area is only capable of developing 1,000 lb. of lift at 300 knots, or 0.125 g for the 8,000 lb. aircraft. The lift deficit of 7,000 lb. at 300 knots must be provided by either the 490.0 sq. ft. of additional blade area or by adding a fixed wing. The 7,000 lb. lift deficit of the rotor requires the addition of less than 30 sq. ft. of wing area at 300 knots and an assumed lift coefficient of 0.8 to restore a sustained 1.0 g lift capability for the 8,000 lb. aircraft. Comparing the requirement for the addition of a 30 sq. ft. wing to the additional 490.0 sq. ft. of blade area necessary to yield equal lift at 300 knots, the benefit of the wing is readily apparent. Table 21 illustrates the comparative advantages of a wing versus increased blade area at the 300 knot speed.

Assuming the weight per unit area of wing area and additional blade area are approximately equal, each incremental increase in blade area required for maneuver margins or high speed flight beyond the blade area required for hover and low speed flight weighs 17.1 times the aerodynamically equivalent wing area. The comparative weight argument suggests sizing the rotor according to the hover and low speed flight requirements only and augmenting the rotor with a wing for all maneuver and high speed flight requirements. This approach, however, frequently leads to a very lightly loaded rotor with the majority of lift at high speeds generated by the wing. This condition leads to a complex set of control problems governing the spatial relationships between the lifting fuselage and the non-lifting rotor system at high flight speeds.

It appears, however, that an optimum distribution of lift between the wing and the rotor should be achievable such that at least 50 to 60 percent of the required lift is provided by the rotor and the remaining 40 to 50 percent of lift is generated by the wing. Assuming the users require a 1.5 g maneuver capability at 300 knots, 60 percent of this load, 0.9 g, or 7,200 lb., could be carried by the rotor, with the remaining 0.6 g, or 4,800 lb. load, on the wing. Figure 18 suggests that 500 sq. ft. of blade area is required to develop the 0.9 g of thrust for the 8,000 lb. aircraft at 300 knots. Over the range of solidities shown in Figure 18, this blade area equates to rotor diameters of 56 to 80 ft. The remaining 0.6 g load, or 4,800 lb., on the wing requires only 19.7 sq. ft. of wing area at 300 knots and a wing lift coefficient of 0.8.

The combination of 500 sq. ft. of blade area and a 19.7 sq. ft. wing results in an unacceptably large rotor system in terms of both size and weight. Increasing the wing area at these conditions would allow a substantial reduction of blade area, but would result in further unloading of the rotor and a redistribution of lift between the rotor and wing. This leads back to the control problems of the lightly loaded rotor that the lift distribution was intended to avoid. Reducing the dash speed requirement by 8.3 percent from 300 to 275 knots with a 1.5 g maneuver capability and 60 percent of the load on the rotor reduces the required blade area to 180 sq. ft., which equates to rotor diameters in the range of 33 to 46 ft. At this condition, the remaining 0.6 g load, or 4,800 lb., on the wing requires only 23.5 sq. ft. of wing area. Reducing the speed even further to 250 knots, or 16.7 percent, yields an even more favorable combination of approximately 115 sq. ft. of blade area and 28.4 sq. ft. of wing area at a lift coefficient of 0.8. Further reductions in blade area with corresponding increases in wing area not only require a reduction in maximum forward flight speed, but also result in a degradation of hover capabilities for two reasons:

- The increased vertical drag due to the wing.

- The reduced blade area of the rotor.

This suggests that the rotor size and solidity should be selected on the basis of the hover and low speed flight requirements and the wing should be selected on the basis of maximum allowable rotor blade loading at high speed due to blade stall. However, the wing design is also governed by a requirement to maintain a minimum level of load on the rotor to prevent control problems mentioned earlier. Auxiliary propulsion systems may be required and selected on the basis of the maximum required speed and total aircraft drag to include the induced and profile drag of the wing.

The basic problem with the high speed flight requirement, however, is independent of the design of the wing and the appropriate selection of an auxiliary propulsion system. The problem evolves to the dilemma of how to reduce the loads on the rotor in order to avoid stall while simultaneously maintaining adequate loads to assure rotor control. A possible solution to this dilemma is to incorporate substantial or severe forward shaft tilt such

TABLE 21

RELATIVE LIFT EFFICIENCY OF A WING VERSUS A ROTOR AT 300 KNOTS

Required Lift or Rotor Thrust lb	Maneuver Load Factor for an 8000 lb Aircraft g	Blade Area Required To Produce Rotor Thrust at 200 knots sq ft	Blade Area Required To Produce Rotor Thrust at 300 knots sq ft	Required Increase in Blade Area sq ft	Rotor Thrust at 200 knots lbs	Rotor Thrust at 300 knots lbs	Thrust Deficit at 300 knots lb	Wing Area Required To Produce Thrust Deficit at 300 knots at an Assumed Lift Coefficient of 0.8 sq ft	Ratio of Increase in Blade Area to Required Wing Area
8000	1 0	70 0	560 0	490 0	8000	1000	7000	28 7	17 1
16000	2 0	140 0	1120 0	980 0	16000	2000	14000	57.4	17 1
24000	3 0	210 0	1680 0	1470 0	24000	3000	21000	86 1	17 1

that the rotor is operating at extreme angles of attack in high speed flight, as shown in Figure 19. Operation of the rotor at high angles of attack, on the order of 20 to 30 degrees, should tend to reduce the adverse effects of retreating blade stall and compressibility to some extent, while maintaining an increased load on the rotor to assure adequate control. The increased load on the rotor, in the form of the resultant thrust vector, is comprised of the forward thrust component and the vertical thrust component. At a 30 degree angle of attack, or shaft tilt, 8,000 lb. of total rotor thrust would provide a 4,000 lb. propulsive force and a 6,930 lb. lift force. While this tends to degrade the vertical lift force by 15.4 percent, the lift would be lost anyway due to blade stall at conventional angles of attack and can be provided by an appropriately designed wing. The forward propulsive thrust component of 4,000 lb. is a significant advantage sufficient to sustain a 300 knot speed capability for an aircraft with 13.1 sq. ft. of equivalent drag area with no requirement for auxiliary propulsion systems.

Such severe shaft tilt, however, would adversely affect hover performance, necessitating some form of variable tilt mechanism to restore a nearly vertical shaft orientation in hover. This, in turn, tends to complicate the control system throughout the conversion range from a vertical position to a 30 degree shaft tilt as the rotor transitions from conventional rotor control to a hybrid prop/rotor control system. In addition, the rotor control system would have to be integrated with a fixed wing control system to provide a coordinated maneuver capability.

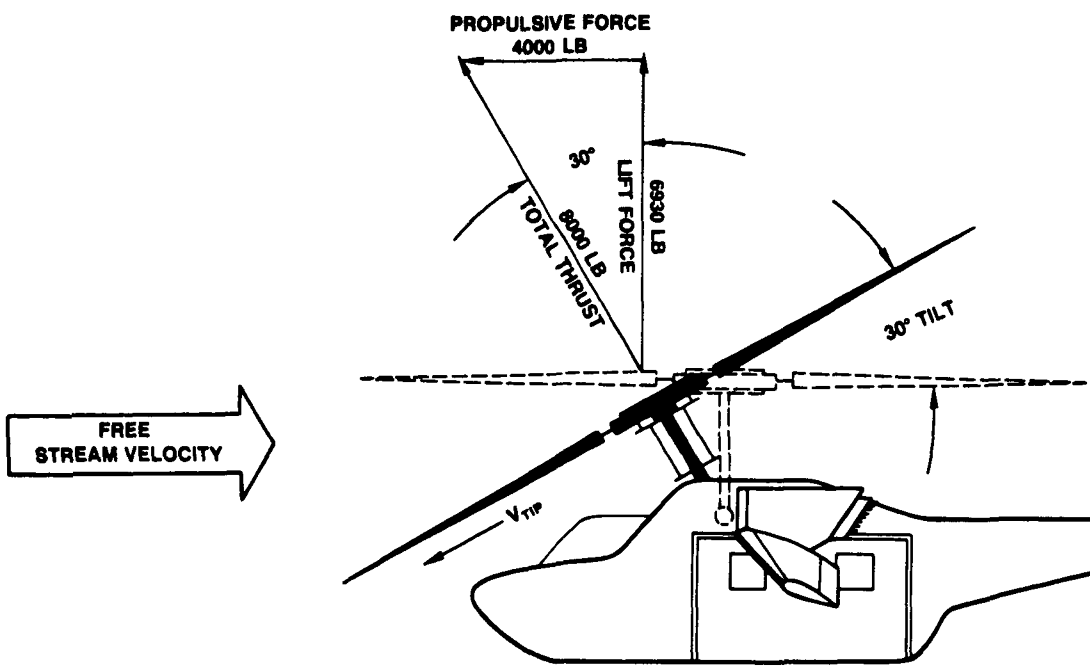

FIGURE 19. VARIABLE TILT SHAFT CONCEPT

Such a design would draw heavily upon NASA's established technology base and simultaneously offer new avenues of VTOL-related aeronautical research. For example, many of the solutions and technology required for such a variable shaft tilt single rotor compound helicopter design are currently available in NASA's XV-15 tilt rotor aircraft research program. Furthermore, while such a design may require a rigid or semi-rigid rotor system, technology and data from the XH-51 rigid rotor, XH-59 ABC rotor and XV-15 tilt rotor research programs may have significant application to a variable shaft tilt design. Incorporating 90 degree nose drive gear boxes from laterally displaced twin engines into the main gear box on a lateral centerline, as in the U.S. Army's AH-64, Apache, would simplify and facilitate the variable shaft tilt drive system, such that the main transmission and shaft tilt collectively as a unit on differential fore and aft hydraulic servos which constitute the transmission mounting struts. As a result, the empty weight penalty associated with the shaft tilt mechanisms would be minimized to yield empty weight fractions more consistent with conventional single rotor helicopter designs, requiring only the addition of hydraulic servos in place of the transmission mounting struts and wings for lift augmentation in high speed flight. This weight penalty should be more than offset by the elimination of the requirement for auxiliary propulsion typical of compound helicopter designs. The requirement for rigid or semi-rigid rotor systems and integrated flight controls may impose additional empty weight penalties on the design, but at a rate equal to half that of similar penalties associated with the current ABC and tilt rotor options.

The development of such a system could offer the maximum speed potential of the ABC and the tilt rotor designs without the attendant empty weight penalties, increased size, increased downwash velocities or auxiliary propulsion requirements. Based upon preliminary conceptualization of the design, the unconventional variable tilt shaft design for a single rotor helicopter appears to offer many solutions to the consolidated requirements of the PSH users.

THE ADVANCING BLADE CONCEPT COAXIAL ROTOR DESIGN

The Advancing Blade Concept (ABC) is a special type of coaxial rotor system comprised of two closely-spaced counter-rotating rigid rotors coupled together through a substantial and rigid main rotor shaft assembly. The ABC functions as a pure coaxial helicopter in hover with equal blade pitch at all azimuths. In forward flight, however, the ABC begins to feather the retreating blades of both the upper and lower rotor such that at some high forward flight speed, the retreating blades are fully feathered and lift is produced with only the advancing blades of both rotors. While this results in an asymmetrical lift distribution on each rotor in isolation, the rolling moment of each rotor counters the rolling moment of the opposing rotor through the rigid main rotor shaft assembly.

By this approach, the ABC rotor system effectively eliminates the problem of retreating blade stall at high forward flight speeds. Unlike the conventional single rotor and coaxial rotor designs, which suffer severe degradation of the maximum rotor lift coefficient at high speeds due to the effects of retreating blade stall, the ABC rotor system maintains a fairly constant lift coefficient with increasing speed since the stall effects have been eliminated. The effects of blade stall in hover and low forward flight speeds, however, are not eliminated and stall may still occur in the low speed regions at high gross weights and density altitude conditions. In addition, since the ABC feathers the retreating blades at high speed, all of the thrust at these conditions must be generated by the advancing blades. This consideration may require design compromises in the rotor system which adversely affect hover performance and necessitate operating the advancing blades at increased angles of attack, which tends to lower the critical Mach number of the blade tip, leading to premature onset of compressibility effects when compared to a conventional single rotor helicopter. Table 22 illustrates the general characteristics of the XH-59 ABC research aircraft.

The limiting advancing blade tip Mach number for the ABC is advertised as 0.85. At the reported tip speed of 650 ft./sec, in the pure helicopter configuration, the advancing blade tip Mach number limits the maximum speed to less than 177 knots, which is currently achievable with conventional single rotor helicopters but fails to meet the users' requirement for a 200 knot cruise speed, much less the 300 knot dash speed requirement. The published maximum airspeed of the ABC in the pure helicopter configuration is 160 knots, which equates to an advancing blade tip Mach number of 0.824.

In order to achieve flight speeds in excess of 160 knots and more consistent with the 200 to 300 knot criteria specified by the users, the ABC must be equipped with auxiliary propulsion systems. According to the published data on the ABC, the tip speed must be reduced to 450 ft./sec in the compound configuration in order to prevent the advancing blade tip Mach number from exceeding the limit of 0.85. Based on these values, the maximum speed of the ABC with auxiliary propulsion would be limited to 295 knots by compressibility effects. The published speed capability for the ABC with auxiliary propulsion is 280 knots, which equates to an advancing blade tip Mach number of 0.826.

The ABC employs 1,825 shp to drive the rotors in the pure helicopter mode. This results in a fairly typical power loading of 6.03 lb./shp at the maximum reported gross weight of approximately 11,000 lbs. In the compound configuration, two 3,000-lb. thrust podded turbojets are added for auxiliary propulsion. Converting the 6,000 lb. thrust to power, at the 280 knot design speed for the compound version results in an equivalent 5,150 shp of installed power, in addition to the 1,825 shp rotor drive system. The total equivalent power installation for the compound configuration is nearly 7,000 shp, or a power loading of roughly 1.57 lb./shp at the 11,000 lb. weight.

It appears, therefore, that in the pure helicopter configuration, the ABC offers little or no benefit over the conventional single rotor helicopter design in terms of maximum speed capability. While the conventional single rotor design is speed-limited due to retreating blade stall and compressibility effects, the ABC in the pure helicopter configuration is speed-limited due to the inability of the substantially rigid rotor system to produce forward thrust and the increase in compressibility effects stemming from the reduced critical Mach number. In the compound configuration, however, the ABC appears capable of providing a 5 to 20 percent increase in maximum speed compared to the conventional single rotor helicopter design. While the maximum speed capability of the ABC nearly meets the 300 knot requirement specified by the users,

TABLE 22

CHARACTERISTICS OF THE XH-59 A & B ABC RESEARCH AIRCRAFT

Parameter	XH59A	XH59B*
Gross Weight	11,000 lb.	11,000 lb.
Empty Weight	8,060 lb.	9,060 lb.
Crew and Mission Equipment	500 lb.	500 lb.
Operational Empty Weight	8,560 lb.	9,560 lb.
Useful Load (Est.)	2,440 lb.	1,440 lb.
Payload (Est.)	1,170 lb.	170 lb.
Fuel Load (Est.)	1,270 lb.	1,270 lb.
Maximum Speed: Limited by Advancing Blade Tip Mach Number	177 knots	295 knots
Maximum Speed: Published	160 knots	280 knots
Rotor Diameter	36.0 ft.	36.0 ft.
Rotor Disk Area	1,017.9 sq. ft.	1,017.9 sq. ft.
Total Blade Area	129.3 sq. ft.	129.3 sq. ft.
Solidity	0.127	0.127
Rotor Tip Speed	650 ft/sec	450 ft/sec
Max Advancing Blade Tip Mach Number	0.85	0.85
Maximum Blade Loading	85.1 lb/sq. ft.	85.1 lb/sq. ft.
Maximum Disk Loading	10.8 lb/sq. ft.	10.8 lb/sq. ft.
Maximum Downwash Velocity	47.7 ft/sec	47.7 ft.sec
Installed Power: Rotor System	1,825 shp	1,825 shp
Installed Power: Auxil. Propulsion	0 shp	5,150 shp
Total Installed Power	1,825 shp	6,975 shp
Power Loading	6.03 lb/shp	1.57 lb/shp
Aircraft Maximum Width	36.00 ft.	36.00 ft.
Aircraft Maximum Length	41.42 ft.	41.42 ft.

* Compound engine configuration

this capability requires a substantial increase in installed power and remains to be demonstrated. Furthermore, equipping the conventional single rotor helicopter with a wing to augment the rotor in high speed flight and providing a power installation equivalent to the compound ABC design could enable the single rotor design to achieve nearly equivalent maximum flight speeds at a substantially lower empty weight fraction.

The most significant disadvantage of the ABC is the empty weight of the aircraft. The empty weight of the prototype ABC in the pure helicopter configuration was reported as 8,060 lb. or 73.3 percent of the 11,000 lb. gross weight. Allowing for a crew and mission equipment, the operational empty weight would be approximately 8,560 lb., or 77.8 percent. This allows a useful load fraction of only 22.2 percent. Imposing the 11.5 percent fuel fraction for 2 hrs. endurance developed in Chapter IV allows a payload fraction of only 10.6 percent. Based on this payload fraction, the ABC would have to be designed to a gross weight of over 39,000 lb. in order to achieve the 4,000 lb. payload requirement with 2 hrs. of endurance. This unacceptably high gross weight, however, was based on the empty weight fraction of the ABC helicopter configuration in prototype form. Allowing a 25 percent reduction in empty weight as a reasonable production objective yields an empty weight of approximately 6,045 lb. for a production version of the pure helicopter variant of the ABC, or an empty weight fraction of 55 percent. Adding a crew and mission equipment allowance results in an operational empty weight of approximately 6,545 lb., or an operational empty weight fraction of 59.5 percent. The resulting 40.5 percent useful load fraction and the 11.5 percent fuel fraction for 2 hrs. endurance yields a maximum payload fraction of only 29.0 percent, or 3,185 lbs. The production version of the ABC in the pure helicopter configuration must, therefore, weigh at least 13,800 lbs. in order to meet the 4,000 lb. payload requirement at a payload fraction of 29.0 percent with 2 hrs. of endurance. Although the current prototype ABC rotor system may be capable of a 25 percent increase in gross lift capacity from 11,000 lb. to 13,800 lb., the design still exceeds the 10,000 lb. maximum gross weight limit specified by the users and would require an empty weight fraction of 55.9 percent as a design objective.

Furthermore, the foregoing weight relationships were based on the ABC in the pure helicopter version without auxiliary propulsion. As a result, the potential speed benefits are not available, and the aircraft fails to meet the required 200 and 300 knot speed conditions specified by the users. In order to gain the maximum speed benefit, auxiliary propulsion engines must be employed which increase the empty weight of the aircraft with no offsetting increase in lift. Assuming the auxiliary propulsion system weighs 1,000 lbs. in production form, the empty weight of the production version of the ABC compound is 7,045 lbs., with an operational empty weight of 7,545 lbs. and an operational empty weight fraction of 68.6 percent. The resulting 31.4 percent useful load fraction and the 11.5 percent fuel fraction for 2 hrs. endurance yields a 19.9 percent payload fraction, or a 2,185 lb. payload. In order to carry the required 4,000 lb. payload with 2 hrs. of endurance at a 29.9 percent payload fraction and achieve the maximum speed potential of the ABC, the compound design must weigh at least 20,100 lbs., or 101 percent more than the maximum gross weight of 10,000 lb. specified by the users. Table 23 summarizes the estimated weight relationships of both the pure helicopter and compound versions of the ABC coaxial rotor system.

TABLE 23

ABC WEIGHTS AND FRACTIONS

Weight Component	XH-59A ABC Pure Helicopter					
	Current Prototype		Production Version With a 25% Reduction of Empty Weight		Production Version With a 25% Increase in Gross Weight and Empty Weight Reduction	
Gross Weight (Est)	11000 lbs(1)	100 0%	11000 lbs(1)	100.0%	13800 lbs	100.0%
Empty Weight (Est)	8060 lbs	73.3%	6045 lbs	55.0%	7710 lbs	55.9%
Crew and Mission Equipment	500 lbs	4.5%	500 lbs	4 5%	500 lbs	3.6%
Operational Empty Weight	8560 lbs	77 8%	6545 lbs	59.5%	8210 lbs	59.5%
Useful Load	2440 lbs	22 2%	4455 lbs	40.5%	5590 lbs	40.5%
Payload	1170 lbs	10.6%	3185 lbs	29.0%	4000 lbs	29.0%
Fuel Load	1270 lbs	11 5%	1270 lbs	11.5%	1590 lbs	11.5%

Weight Component	XH-59B ABC Compound					
	Current Prototype		Production Version With a 25% Reduction of Empty Weight		Production Version With a 83% Increase in Gross Weight and Empty Weight Reduction	
Gross Weight (Est)	11000 lbs(1)	100 0%	11000 lbs(1)	100.0%	20100 lbs	100.0%
Empty Weight (Est)	9060 lbs	82 3%	7045 lbs	64.1%	13290 lbs	66.1%
Crew and Mission Equipment	500 lbs	4 5%	500 lbs	4.5%	500 lbs	2.5%
Operational Empty Weight	9560 lbs	86 9%	7545 lbs	68.6%	13790 lbs	68 6%
Useful Load	1440 lbs	13 1%	3455 lbs	31.4%	6310 lbs	31 4%
Payload	170 lbs	1 6%	2185 lbs	19 9%	4000 lbs	19 9%
Fuel Load	1270 lbs	11 5%	1270 lbs	11 5%	2310 lbs	11 5%

Notes:

(1) *Reported Gross Weights Range From 9000 lbs to 11100 lbs The 11000 lb Gross Weight Was Assumed as Representative*

(2) Includes an Assumed 1000 lbs for the Weight of the Auxiliary Propulsion Systems on a Basis of 6000 lbs Thrust in the XH-59B at 6 lb Thrust/lb Engine Weight Installed, Including Engine Nacelles and Mounting But No Additional Fuel or Fuel Tanks

The ABC rotor design offers excellent rotor performance in both hover and high speed in terms of rotor efficiency, based on the power required at a given gross weight. The improved rotor efficiency in hover results from the improved efficiency of the coaxial rotor system compared to a conventional single rotor design and the elimination of the tail rotor for torque control. The improved high speed efficiency is due to the elimination of retreating blade stall, enabling the rotor to operate at a relatively constant lift coefficient (C_T/σ) at all forward flight speeds.

However, if the efficiency of the system is measured on a basis of the power required to lift a specified load, the relative inefficiency of the ABC design becomes immediately apparent. This measure of efficiency introduces the relative impact of empty weight into the comparative evaluation of alternate design configurations. The significant empty weight penalty of the ABC results from the weight of the rotor, flight control and drive systems, which comprise 4,275 lbs., or over 53 percent of the empty weight, as shown in Table 24. Assuming the remaining 47 percent of the empty weight represents conventional aircraft design not related to the ABC rotor system, then the 25 percent or 2,015 lb. reduction in empty weight assumed for the production version of this system must occur in the ABC related components. This suggests that the weight of the ABC related components must be reduced by 47 percent, from 4,275 lb. to 2,260 lb. Alternatively, the 8,060 lb. empty weight of the overall ABC aircraft must be reduced by 3,560 lbs., or 44 percent, to meet the 4,500 lb. empty weight design objective established in Chapter IV for the 10,000 lb. gross weight design.

TABLE 24

ABC COMPONENT WEIGHTS

Weight Component	Component Weight (lb.)	Empty Weight Basis	Gross Weight Basis
Rotor System	1,896	23.5%	17.2%
Flight Control System	1,260	15.6%	11.5%
Drive System	1,119	13.9%	10.2%
Total Aero Group	4,275	53.0%	38.9%
Empty Weight	8,060	100.0%	73.3%
Gross Weight	11,000	136.5%	100.0%

As a result of the unfavorable empty weight penalties of the ABC rotor system, the gross weight of the aircraft, or net rotor thrust, must be increased to lift a specified useful load. This results in increased requirements on installed power and/or rotor diameter. If additional power is provided at a constant rotor diameter, the downwash velocity increases. If the rotor diameter is increased at constant power, the physical dimensions of the aircraft increase. Both options are in conflict with the specified user requirements. Table 25 illustrates a comparison of the ABC coaxial rotor aircraft and the conventional single rotor helicopter design, in terms of rotor size and downwash velocity based on equal gross weight and payload conditions.

Consequently, empty weight is the principal disadvantage of the ABC coaxial rotor design. While this alternative may be capable of maximum speeds on the order of 5 to 20 percent better than the conventional single rotor helicopter, the empty weight fraction will tend to increase the aircraft gross weight, or required rotor thrust, by some 38 to 100 percent. The resulting impact and required operational and design compromises associated with the ABC design will probably be more severe than those associated with a conventional or compound single rotor helicopter design.

THE TILT ROTOR AIRCRAFT DESIGN

The tilt rotor aircraft, such as the XV-15, is a laterally displaced twin rotor convertiplane design. The current prototype employs two counter-rotating, 25-ft. diameter rotors mounted on rotating engine/transmission nacelles at the tips of a high mounted fixed wing with a span of approximately 30 ft. The rotors and nacelles rotate from a vertical shaft orientation to produce lift in hover to a horizontal shaft orientation to produce forward propulsive thrust in forward flight. Lift in forward flight is provided solely by the wing.

The principal advantage of the tilt rotor design relative to the specified PSH requirements is its speed capability. Table 26 summarizes salient features of the XV-15 tilt rotor system. Since the rotors of the tilt rotor aircraft rotate from a vertical shaft orientation in hover to a horizontal shaft orientation in forward flight, the rotor systems are not subjected to high oblique flow conditions for extended periods of time. As a result, the tilt rotor system does not generally experience the speed limitations due to compressibility effects and retreating blade stall characteristic of conventional helicopters at high flight speeds. As Table 26 illustrates, the tilt rotor design easily meets the speed requirements outlined by the PSH Workshop.

The principal disadvantages of the tilt rotor design with respect to the PSH requirements are:

- o The empty weight fraction and its impact on aircraft sizing, and thrust required at the useful loads specified by the users.
- o Its physical size.
- o The downwash velocity at the thrust levels dictated by the combination of empty weight and required useful load.

TABLE 25

COMPARISON OF ABC COAXIAL ROTOR AND SINGLE ROTOR DOWNWASH VELOCITIES

Design Rationale	ABC Coaxial Design with Auxiliary Propulsion: Gross Weight lb	Payload lb	Rotor Diameter ft	Rotor Area sq ft	Downwash Velocity-γ Neglecting Vertical Drag ft/sec	Single Rotor Design: Gross Weight lb	Payload lb	Rotor Diameter ft	Rotor Area sq ft	Downwash Velocity-γ Neglecting Vertical Drag ft/sec	Basis for Comparison	Impact
Estimated Production Version of Current ABC Design with 101% Over User Specified Gross Weight Limit To Meet Useful Load Requirement	20100	4000	36.00	1017 9	64 5	20100	7740	36.00	1017 9	64 5	Equal Gross Weight and Equal Rotor Diameter	Equal Downwash Velocity for ABC at 52% of Single Rotor Payload
	20100	4000	36.00	1017 9	64 5	10400	4000	36.00	1017 9	48 4	Equal Payload and Equal Rotor Diameter	39 % Increase in Downwash Velocity for ABC at Equal Payload
User Specified Rotor Diameter or Maximum Physical Dimension	20100	4000	20.00	314 2	116 0	20100	7740	20.00	314 2	116 0	Equal Gross Weight and Equal Rotor Diameter	Equal Downwash Velocity for ABC at 52% of Single Rotor Payload
	20100	4000	20.00	314 2	116 0	10400	4000	20.00	314 2	83 4	Equal Payload and Equal Rotor Diameter	39% Increase in Downwash Velocity for ABC at Equal Payload
At Minimum Feasible Single Rotor Diameter	20100	4000	30.00	706 8	77 3	20100	7740	30.00	706 8	77 3	Equal Gross Weight and Equal Rotor Diameter	Equal Downwash Velocity for ABC at 52% of Single Rotor Payload
	20100	4000	30 00	706 8	77 3	10400	4000	30.00	706 8	55 6	Equal Payload and Equal Rotor Diameter	39% Increase in Downwash Velocity for ABC at Equal Payload
At Recommended Single Rotor Diameter	20100	4000	40.00	1256 6	58 0	20100	7740	40.00	1256 6	58 0	Equal Gross Weight and Equal Rotor Diameter	Equal Downwash Velocity for ABC at 52% of Single Rotor Payload
	20100	4000	40 00	1256 6	58 0	10400	4000	40.00	1256 6	41 7	Equal Payload and Equal Rotor Diameter	39% Increase in Downwash Velocity for ABC at Equal Payload
Maximum Downwash Velocity Less Than 40 ft/sec	20100	4000	58 00	2642 1	40.0	20100	7740	58 00	2642 1	40.0	Equal Gross Weight and Equal Downwash Velocity	Equal Rotor Diameter for ABC at 52% of Single Rotor Payload
	20100	4000	58 00	2642 1	40 0	10400	4000	41 72	1367 3	40.0	Equal Payload and Equal Downwash Velocity	39% Increase in Rotor Diameter for ABC at Equal Payload

Notes
1 Shaded Areas Indicate Parameters Held Equal as a Basis for Comparison
2 Gross Weights Based on the Requirement To Carry at Least a 4000 lb Payload with 2 hrs Endurance
3 A Fuel Fraction of 11 5% of Gross Weight Is Assumed To Provide a 2 hr Endurance Capability
4 Effects of Vertical Drag and Tip Losses Have Been Neglected To Equalize Rotor Comparisons
5 Some of the Design Options Shown Are Not Feasible in Terms of Power Required or Blade Stall but Are Shown To Illustrate the Comparative Downwash Velocity Versus Rotor Size Relationships

TABLE 26

CHARACTERISTICS OF THE XV-15 TILT ROTOR RESEARCH AIRCRAFT

Maximum Design Vertical Takeoff Weight	13,000 lb.
Empty Weight	9,600 lb.
Crew Weight	500 lb.
Operational Empty Weight	10,100 lb.
Useful Load	2,900 lb.
Normal Fuel	1,500 lb.
Normal Payload	1,400 lb.
Maximum Speed	332 knots
Maximum Cruising Speed	303 knots
Economical Cruising Speed	200 knots
Rotor Diameter	25.00 ft.
Rotor Disk Area: Each Rotor	490.00 sq. ft.
Blade Area: Each Rotor	43.80 sq. ft.
Solidity	0.0892
Lateral Clearance Between Blade Tips	7.16 ft.
Lateral Distance Between Rotor Hubs	32.16 ft.
Aircraft Maximum Width Over Turning Rotors	57.16 ft.
Aircraft Maximum Length	42.08 ft.
Wing Span	35.16 ft.
Wing Chord: Constant	5.25 ft.
Wing Area	169.00 sq. ft.
Wing Aspect Ratio	6.12
Maximum Blade Loading	155.25 lb./sq. ft.
Maximum Disk Loading	13.85 lb./sq. ft.
Maximum Downwash Velocity	54.00 ft./sec.

As Table 27 illustrates, the empty weight of the current tilt rotor prototype is approximately 9,600 lb. This equates to a 73.8 percent empty weight fraction at the design takeoff weight of 13,000 lb. Allowing 500 lb. for a crew and mission equipment yields an operational empty weight of 10,100 lb., with a corresponding useful load fraction of 22.3 percent. Subtracting the 11.5 percent fuel fraction for 2 hrs. endurance leaves a 10.8 percent payload fraction. Therefore, the requirement to carry a 4,000 lb. payload at a 10.8 percent payload fraction with 2 hrs. endurance demands an aircraft gross weight in excess of 37,000 lb., or approximately the same as the ABC discussed earlier.

Assuming that the empty weight of the tilt rotor prototype could be reduced by 25 percent in production, the empty weight of the production version would be 7,200 lb., with a corresponding operational empty weight of 7,700 lb., or a 59.2 percent operational empty weight fraction. Using the estimated 11.5 percent fuel fraction for 2 hrs. endurance, as before, yields a 29.3 percent payload fraction, requiring a 13,600 lb. tilt rotor aircraft to carry a 4,000 lb. payload with 2 hrs. endurance. These relationships are summarized in Table 27.

While the tilt rotor design appears capable of meeting the users' requirements for speed and loading, the resulting gross takeoff weight required to meet these criteria appears to be approximately 13,600 lb., or 36 percent more than the maximum gross weight limit of 10,000 lb. specified by the users. Conversely, designing the tilt rotor aircraft to the 10,000 lb. gross weight limit with a 4,000 lb. payload and 2 hrs. endurance requires a reduction of 5,100 lb. in empty weight from 9,600 lb. to the 4,500 lb. empty weight design objective, or a 53 percent decrease in empty weight.

In addition to the weights and loading problem just discussed, the tilt rotor design is substantially larger than the specified user requirements. The laterally displaced rotors measure over 57 ft. in width from blade tip to blade tip with the rotors turning, which is 186 percent larger than the maximum 20 ft. diameter rotor specified by the users. Furthermore, the width is 90 percent greater than the minimum diameter of 30 ft. considered feasible for a conventional single rotor design, and 43 percent larger than the 40 ft. recommended single rotor diameter. The weight analysis suggests that the size of the tilt rotor aircraft must be increased slightly to meet the requirement for a 4,000 lb. payload with 2 hrs. endurance, even with an allowance for a 25 percent improvement in empty weight of the production version. As a result, the prospects for any significant reduction in physical dimensions at the required loadings specified by the users is remote.

Furthermore, any efforts to reduce the size of the tilt rotor design must involve a reduction in rotor diameter, since the lateral spacing of the rotors requires fuselage clearance when operating in the prop mode. Reducing the rotor diameter of the tilt rotor design at the estimated minimum gross weight of 13,600 lb. necessary to meet the 4,000 lb. payload requirement results in a substantial increase in downwash velocity, as shown in Table 28. Table 28 illustrates a comparison of the tilt rotor design and the conventional single rotor helicopter design in terms of weight, size and downwash velocity. The eight comparisons are shown on both a gross weight and useful

TABLE 27

TILT ROTOR WEIGHTS & FRACTIONS

Weight Component	Current XV-15 Prototype		Assumed Production Version with a 25% Reduction in of Empty Weight		Assumed Production Version with a 5% Increase in Gross Weight and a 21.5% Reduction of Empty Weight	
Maximum Gross Weight	13,000 lb.	100.0%	13,000 lb.	100.0%	13,600 lb.	100.0%
Empty Weight	9,600 lb.	73.8%	7,200 lb.	55.4%	7,540 lb.	55.5%
Crew and Mission Equipment	500 lb.	3.9%	500 lb.	3.9%	500 lb.	3.7%
Operational Empty Weight	10,100 lb.	77.7%	7,700 lb.	59.2%	8,040 lb.	59.2%
Useful Load	2,900 lb.	22.3%	5,300 lb.	40.8%	5,560 lb.	40.8%
Payload	1,400 lb.	10.8%	3,800 lb.	29.3%	4,000 lb.	29.3%
Fuel Load	1,500 lb.	11.5%	1,500 lb.	11.5%	1,560 lb.	11.5%

TABLE 28

COMPARISON OF TILT ROTOR AND SINGLE ROTOR DOWNWASH VELOCITIES

Design Rationale	Tilt Rotor Design: Gross Weight lb	Payload lb	Rotor Diameter ft	Rotor Tip Clearance ft	Width Over Rotors ft	Rotor Area sq ft	Downwash Velocity-γ Neglecting Vertical Drag ft/sec	Single Rotor Design: Gross Weight lb	Payload lb	Rotor Diameter ft	Rotor Area sq ft	Downwash Velocity-γ Neglecting Vertical Drag ft/sec	Basis for Comparison	Impact
Current Tilt Rotor Prototype With Improved Useful Load	13600	4000	25 00	7 16	57 16	490 9	54 0	13600	5305	57 16	2566 1	33 4	Equal Gross Weight and Equal Physical Dimension	62% Increase in Downwash Velocity for Tilt Rotor with 75% of Single Rotor Payload
	13600	4000	25 00	7 16	57 16	490 9	54 0	10400	4000	57 16	2566 1	29 2	Equal Payload and Equal Physical Dimension	85 % Increase in Downwash Velocity for Tilt Rotor at Equal Payload
User Specified Rotor Diameter	13600	4000	20 00	7 16	47 16	314 2	67 5	13600	5305	47 16	1746 8	40 5	Equal Gross Weight and Equal Physical Dimension	67% Increase in Downwash Velocity for Tilt Rotor with 75% of Single Rotor Payload
	13600	4000	20 00	7 16	47 16	314 2	67 5	10400	4000	47 16	1746 8	35 4	Equal Payload and Equal Physical Dimension	91% Increase in Downwash Velocity for Tilt Rotor at Equal Payload
User Specified Maximum Physical Dimension	13600	4000	6 42	7 16	20 00	32 4	210 2	13600	5305	20 00	314 2	95 4	Equal Gross Weight and Equal Physical Dimension	120% Increase in Downwash Velocity for Tilt Rotor with 75% of Single Rotor Payload
	13600	4000	6 42	7 16	20 00	32 4	210 2	10400	4000	20 00	314 2	83 4	Equal Payload and Equal Physical Dimension	152% Increase in Downwash Velocity for Tilt Rotor at Equal Payload
At Minimum Feasible Single Rotor Diameter as Dimensional Constraint	13600	4000	11 42	7 16	30 0	102 4	118 2	13600	5305	30 00	706 8	63 6	Equal Gross Weight and Equal Physical Dimension	86% Increase in Downwash Velocity for Tilt Rotor with 75% of Single Rotor Payload
	13600	4000	11 42	7 16	30 0	102 4	118 2	10400	4000	30 00	706 8	55 6	Equal Payload and Equal Physical Dimension	113% Increase in Downwash Velocity for Tilt Rotor at Equal Payload
At Recommended Single Rotor Diameter as the Dimensional Constraint	13600	4000	16 42	7 16	40 00	211 8	82 2	13600	5305	40 00	1256 6	47 7	Equal Gross Weight and Equal Physical Dimension	72% Increase in Downwash Velocity for Tilt Rotor with 75% of Single Rotor Payload
	13600	4000	16 42	7 16	40 00	211 8	82 2	10400	4000	40 00	1256 6	41 7	Equal Payload and Equal Physical Dimension	97% Increase in Downwash Velocity for Tilt Rotor at Equal Payload
Maximum Downwash Velocity Less Than 40 ft/sec	13600	4000	33 75	7 16	74 66	894 6	40 0	13600	5305	47 71	1788 0	40 0	Equal Gross Weight and Equal Downwash Velocity	56% Increase in Physical Dimension Over a Single Rotor at Equal Gross Weight and Equal Downwash Velocity
	13600	4000	33 75	7 16	74 66	894 6	40 0	10400	4000	41 72	1367 3	40 0	Equal Payload and Equal Downwash Velocity	79% Increase in Physical Dimension Over a Single Rotor at Equal Payload and Equal Downwash Velocity

Notes
1 Shaded Areas Indicate Parameters Held Equal as a Basis for Comparison
2 Gross Weights Based on the Requirement To Carry at Least a 4000 lb Payload with 2 hrs Endurance
3 A Fuel Fraction of 11 5% of Gross Weight Is Assumed To Provide a 2 hr Endurance Capability
4 Effects of Vertical Drag and Tip Losses Have Been Neglected To Equalize Rotor Comparisons
5 Some of the Design Options Shown Are Not Feasible in Terms of Power Required or Blade Stall but Are Shown To Illustrate the Comparative Downwash Velocity Versus Rotor Diameter Relationships

load basis to illustrate the empty weight penalty of the tilt rotor design when compared to an equivalent single rotor design. The shaded areas in Table 28 indicate those parameters held equivalent as the basis for comparison.

Consequently, while the tilt rotor design easily meets the high speed flight requirements defined by the PSH Workshop, it fails to meet the size and loading criteria of the users. Sizing the tilt rotor design in terms of weight required to satisfy the user needs will probably result in an unacceptably large aircraft or intolerable downwash velocities.

VII. CONCLUSIONS AND RECOMMENDATIONS

CONCLUSIONS

A previous ORI report in December 1983 found that most of the needs identified by the July 1980 User's Workshop can be classified as design needs, since either the required technology is already in hand or the need is actually a manufacturing design option. A comparison of the remaining technology needs with NASA Aeronautics Program activities focusing on rotorcraft technology indicates that, taken individually, the NASA program is addressing nearly all of the needs requiring further advances in the current technology base. However, some of the stated needs have an impact upon other needs, requiring either tradeoffs or new thrusts to advance the state-of-the-art in rotorcraft technology in four areas. These areas are:

- o Vehicle Design
- o Propulsion
- o Auxiliary Systems
- o Monitoring and Diagnostic Systems

Vehicle Design

1. The desired vehicle characteristics for a public service helicopter (PSH) are similar to the Army's program objectives for development of the LHX utility helicopter in several respects, e.g., weight, size, and speed boundaries.

2. The major conflicts in the PSH users' needs involve the combination of required load capacity, hover and speed capabilities, and the specified limitation of a 20 foot diameter rotor.

3. To meet the most stringent load carrying requirements specified by the Workshop working groups, a need to lift a 4,000 lb. payload with 2 hours fuel for environmental control missions, the aircraft design must be sized to the maximum gross weight limit of 10,000 lb. This results in an aircraft at least 2.3 times larger than the next stated most severe loading requirement of 6 passengers and 4 hours fuel.

4. As a compromise, nearly all of the loading requirements can be accommodated by an aircraft sized at an 8,000 lb. Design Gross Weight but structurally capable of operating at a 10,000 lb. maximum gross weight. An alternative design sized at 4,300 lb. Design Gross Weight could meet the loading requirements for 6 passengers and 4 hours of fuel.

5. The 20 ft. limitation on rotor diameter is not considered feasible at the specified loading requirements due to downwash velocity, power required and maneuver loading problems. These analyses suggest that the minimum feasible rotor diameter is 30 ft., while the optimum design region is in the 35 to 45 foot range.

6. Downwash velocity represents the most significant design problem. This factor is a function of aircraft weight and rotor diameter only and is not amenable to reduction or improvement within known design constraints. Current alternative rotorcraft designs (i.e., Tilt Rotor, ABC and other coaxial rotor types) offer no solution to this problem and tend only to aggravate the condition.

7. Aerodynamic breakdown (i.e., blade stall and compressibility) of the small rotor at the required loadings is also a significant problem, but is probably conducive to solution. Alternative rotorcraft designs could offer some potential solutions to this problem.

8. The single engine hover criteria specified by the Workshop results in unusually high installed power requirements with correspondingly low power loadings on the order of 2.5 to 3.0 lb./shp. Although current engine technology offers engine designs capable of providing the required power without severely degrading the empty weight fraction, the resulting fuel consumption in routine flight operations is adversely affected due to the high specific fuel consumption at the reduced partial power settings, unless one engine is shut down during routine flight operations.

9. The requirement for a 200 knot continuous cruise speed presents no basic design conflicts and appears to be feasible with a conventional single rotor helicopter design; however, the 300 knot dash speed requirement is probably not possible with the conventional single rotor helicopter design operating at typical rotor tip speeds in the range of 615 to 725 ft./sec due to the effects of compressibility at these speeds. The combination of the 300 knot forward flight speed and the typical range of rotor tip speeds results in advancing blade tip Mach numbers of 1.0 to 1.1, indicating that portions of the rotor are operating supersonically. Limiting the advancing blade tip Mach number to 0.95, which represents the current technological limit, results in maximum forward flight speeds of 200 to 260 knots for the conventional single rotor helicopter operating at tip speeds of 725 to 615 ft./sec, respectively. Through further investigation of advanced concepts, the 300 knot speed objective may be achievable with a single rotor helicopter design by considering the following options:

- Designing the rotor for sustained operation at supersonic advancing blade tip Mach numbers on the order of Mach 1.0 to 1.1 to resolve rotor blade stall problems at high speeds. (Higher levels of noise and required power may be associated with this option.)

- Incorporating significant forward tilt of the main rotor shaft axis, on the order of 15 to 30 degrees, to offset the effects of blade stall and compressibility in high speed flight. (Provisions for variable shaft tilt over a range of 0^o to 20^o may be required to preserve hover performance.)

- Operating the rotor at reduced tip speeds in the range of 550 to 575 ft./sec at conventional shaft inclinations of 0^o to 10^o of forward tilt. (Provisions for slowing the rotor to the reduced tip speeds in high speed flight may be necessary to preserve hover performance.)

- Integrating combinations of the above options in various degrees and, if required, some form of auxiliary lift system (e.g., a wing) and auxiliary forward propulsion system to provide an optimum rotorcraft design capable of attaining 300 knots forward speed without excessively degrading other design parameters.

10. Current design alternatives to the conventional single rotor helicopter, such as Advancing Blade Concept coaxial rotor systems and tilt rotor designs, offer higher forward speed advantages by reducing or totally eliminating the problems of compressibility and blade stall, which present a significant limitation for the conventional single rotor helicopter at high forward speeds. This advantage must be weighed against the potential penalties associated with these designs; i.e., increased aircraft size, increased gross weight and downwash velocity, and relatively poor empty weight fractions.

Propulsion

1. Coordinated NASA and Army technology programs are addressing advanced designs, such as the Convertible Fan Shaft Engine, and building upon an established technology base in rotorcraft propulsion.

2. Further research may be required to develop particle separators which provide for fine particle separation, such as volcanic ash in the atmosphere.

Auxiliary Systems

1. There is an apparent need to adapt military developments in night vision systems for use in PSH operations.

2. Specialized user needs to identify, lock-on, and stop automobiles appear to be related to functional needs which are outside the scope of the NASA Aeronautics program.

Monitoring and Diagnostic Systems

1. Further systems integration work appears to be required to adapt head-up displays for use in PSH operations.

RECOMMENDATIONS

The following recommendations are made for NASA consideration:

1. The scope of the NASA Rotorcraft Technology Program should be expanded to include:

   a. Research in supersonic rotors for use on high speed helicopters.

   b. Investigations of advanced concepts, such as a variable shaft tilt, single rotor design and potential combinations of auxiliary lift and propulsion systems, to provide an optimum design configuration for forward speeds up to 300 knots without excessively degrading other design parameters.

   c. Development of optimization methodology to tradeoff and integrate supersonic rotor capabilities, slowed rotor concepts, and variable shaft tilt options to achieve maximum rotor performance at minimum power and noise conditions.

2. PSH users should be provided with the results of the needs evaluation and provided a forum for revising their stated needs.

3. A Public Service Helicopter Technology and Systems Plan should be developed to meet the revised technology needs of the PSH users.

APPENDIX A

DEFINITION OF WEIGHT AND LOADING TERMS

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APPENDIX A

DEFINITION OF WEIGHT AND LOADING TERMS

Empty Weight. The dry weight of the aircraft without crew, mission specific equipment or payload.

Operational Empty Weight. The Empty Weight plus crew, mission specific equipment and trapped oils and fluids, but no payload or consumable fuel (mission fuel plus reserve fuel).

Useful Load. The sum of the payload weight and consumable fuel weight, to include reserve fuel, at take-off or launch.

Gross Weight. The sum of the Operational Empty Weight plus Useful Load.

Minimum Mission Weight. The minimum permissible gross weight of the aircraft at any time during a mission comprised of the Operational Empty Weight plus minimum reserve fuel, no mission fuel and no payload. Only occurs at landing with no payload.

Minimum Gross Weight. The minimum permissible gross weight of the aircraft at take-off or launch comprised of the Operational Empty Weight plus minimum mission fuel, no payload.

Design Gross Weight. The normal gross weight of the aircraft at take-off or launch comprised of the Operational Empty Weight plus Normal Useful Load. It is also the maximum gross weight for which a single engine Hover-Out-of-Ground-Effect capability exists from Sea Level/STD. to 10,000 ft./STD.

Alternate Design Gross Weight. The maximum gross weight of the aircraft at take-off or launch comprised of the Operational Empty Weight plus Maximum Useful Load (fuel and payload). While a single engine Hover-Out-of-Ground-Effect capability may not exist at the maximum gross weight, straight and level forward flight at endurance speed on one engine is possible. The maximum gross weight represents an overload condition and jettison provisions for excess fuel carried externally and/or excess payload carried modularly or via sling should be incorporated to reduce the gross weight from the Alternate Design Gross Weight condition to the Design Gross Weight in order to achieve a single engine Hover-Out-of-Ground-Effect capability in the event of an engine failure in hover during take-off.

Minimum Mission Fuel. The minimum permissible fuel load at take-off comprised of sufficient fuel for warm-up, take-off, a minimum 30-minute mission, landing and reserve.

Normal Mission Fuel. The normal internal fuel load at take-off comprised of sufficient fuel for warm-up, take-off, approximately 2 hours of mission endurance, landing and reserve.

Maximum Mission Fuel. The maximum fuel load of the aircraft comprised of normal full internal fuel plus jettisonable external auxiliary fuel sufficient for warm-up, take-off, approximately 4 hours of mission endurance, landing and reserve.

Minimum Useful Load. The combination of fuel and payload possible at the Minimum Gross Weight comprised of Minimum Mission Fuel only, no payload.

Normal Useful Load. The combination of fuel and payload possible at the Design Gross Weight comprised of either Minimum Mission Fuel and Maximum Payload, Normal Mission Fuel and Normal Payload, or Maximum Mission Fuel and Minimum Payload.

Maximum Useful Load. The combination of fuel and payload possible at the Alternate Design Gross Weight or overload condition comprised of either Minimum Mission Fuel and Maximum Payload, Normal Mission Fuel and Normal Payload, or Maximum Mission Fuel and Minimum Payload.

Minimum Payload. The payload possible with Maximum Mission Fuel (approximately 4 hr. mission endurance) for any Gross Weight.

Normal Payload. The payload possible with Normal Mission Fuel (approximately 2 hr. mission endurance) for any Gross Weight.

Maximum Payload. The payload possible with Minimum Mission Fuel (approximately 30-minute minimum mission) for any Gross Weight.

APPENDIX B

LIST OF ABBREVIATIONS AND ACRONYMS

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APPENDIX B

LIST OF ABBREVIATIONS AND ACRONYMS

ABC -- Advancing Blade Concept

ARTI -- Advanced Rotorcraft Technology Integration

DARPA -- Defense Advanced Research Projects Agency

DOD -- Department of Defense

EMS -- Emergency Medical Service

ft. -- feet

ft./sec. -- feet per second

g -- force of gravity

GW -- Gross Weight

HIGE -- Hover-In-Ground-Effect
(less than one rotor diameter above ground)

HOGE -- Hover-Out-of-Ground-Effect
(more than one rotor diameter above ground)

hp -- horsepower

Int. -- Internal

ISA -- International Standard Atmosphere

kn -- knot, unit of speed of one nautical mile per hour

lb -- pounds

LHX -- Light Helicopter Experimental: U.S. Army Develcpment Program

LHX-UTIL -- Utility version of the LHX

LHX-SCAT -- Scout and attack versions of the LHX

Mach -- Mach number, which is a dimensionless number representing the ratio of the speed of an object to the speed of sound in the surrounding medium, as air, through which the object is moving.

NASA -- National Aeronautics and Space Administration

PSH -- Public Service Helicopter

rpm -- revolutions per minute

SAR -- Search And Rescue

sec. -- second

SFC -- Specific Fuel Consumption

shp -- shaft horsepower

SL -- Sea Level

STD -- Standard Day, atmospheric standards of temperature and pressure at Sea Level of 59°F, 14.7 lb./sq. inch and air density of 0.002377 slug/cu. ft. (Standard temperature change with altitude is a decrease of 3.57° per 1,000 ft.)

VTOL -- Vertical Take-Off and Landing

Wt. -- Weight

C_T -- Rotor Thrust Coefficient ($T/\rho A V_t^2$): Non-dimensional

σ -- Rotor Solidity (Blade Area/Disk Area): Non-dimensional

μ -- Advance Ratio (V_{ff}/V_t): Non-dimensional

V_{ff} -- Forward Flight Velocity: In ft./sec.

V_t -- Rotor Tip Speed: In ft./sec.

C_T/σ -- Rotor Lift Coefficient: Non-dimensional

$(C_T/\sigma)_{max}$ -- Maximum Rotor Lift Coefficient: Non-dimensional

a -- Speed of Sound (1,116 ft./sec at SL/STD): In ft./sec.

M_t -- Rotor Tip Mach Number in Hover (V_t/a): Non-dimensional

M_{90} -- Advancing Blade Tip Mach Number in Forward Flight ($(V_{ff} + V_t)/a$): Non-dimensional

T -- Rotor Gross Thrust: In lb.

T_O -- Rotor Gross Thrust at 1.0 g: In lb.

g -- Maneuver Load Factor: Non-dimensional

A -- Rotor Disk Area

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15. Supplementary Notes

Project Monitor, John F. Ward

16. Abstract

This report examines the design requirements for a family or type of Public Service Helicopter (PSH) to satisfy the needs of municipal and state governments in the following mission areas:

Emergency Medical Services—Airborne Rescue Squad
Law Enforcement
Search and Rescue
Environmental Control (Fire Fighting, Pollution, Resource Management)

The report compares both design and performance requirements as specified by the PSH user's group against current technological capabilities, RTOPS and US Army LHX design requirements. The study explores various design trade-offs and options available to the aircraft designer/manufacturer in order to meet the several criteria specified by the PSH user's group. In addition, the report includes a brief assessment of the feasibility of employing certain advanced rotorcraft designs to meet the stringent combination of operational capabilities desired by the Public Service Helicopter Users.

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