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V/STOL TILT ROTOR AIRCRAFT STUDY

VOLUME I

CONCEPTUAL DESIGN OF USEFUL MILITARY AND/OR COMMERCIAL AIRCRAFT

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FOREWORD

This report is one of a series prepared by The Boeing Company, Vertol Division, Philadelphia, Pennsylvania, for the National Aeronautics and Space Administration, Ames Research Center, Moffett Field, California, under Contract NAS2-6598.

The contract was administered by the National Aeronautics and Space Administration with Mr. Gary Churchill as Technical Monitor.

The reports published for the Tilt-Rotor Aircraft Study are:

Volume I - Conceptual Design of Useful Military and/ or Commerical Aircraft (Task I)

Volume II - Preliminary Design of Research Aircraft (Task II)

Volume III - Overall Research Aircraft Project Plan, Schedules and Estimated Cost (Task III)

Volume IV - Wind Tunnel Investigation Plan for a Full-Scale Tilt-Rotor Research Aircraft (Task IV)

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ABSTRACT

This report covers the conceptual designs of four useful tilt-rotor aircraft for the 1975 to 1980 time period conducted under Task I of the V/STOL Tilt-Rotor Aircraft Study, Contract NAS2-6598 with the National Aeronautics and Space Administration. Parametric studies leading to design point selection are described,and the characteristics and capabilities of each configuration are presented. An assessment is made of current technology status,and additional tilt-rotor research programs are recommended to minimize the time, cost, and risk of development of these vehicles.

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CONTENTS

1.0 SUMMARY

This volume covers the conceptual design of useful military and civil tilt-rotor aircraft. Four different applications for the tilt rotor are presented. They are:

a. U. S. Army MAVS - Manned Aerial Vehicle, Surveillance

b. U. S. Air Force SAR - Search and Rescue

c. U. S. Navy - Sea Control Aircraft

d. Civil - Off-Shore Oil Rig Support Aircraft

These missions were selected as those which best satisfied a set of mission acceptability criteria. These criteria, discussed in detail in Section 2.2., may be briefly summarized as requiring:

a. IOC or commerical certification in 1975-1980

b. A favorable environment; i.e., a mission for which there is an acknowledged requirement and for which the tilt rotor's inherent characteristics are well suited.

For each application, a parametric study was made to select a point design best suited to the particular requirements of the individual mission.

a. For the Army aircraft shown in Figure 1-1, the selection studies concentrated on developing an aircraft design of minimum size and weight with broad flight capabilities at low speed required for photo-reconnaissance work and with a dash speed in excess of 300 knots to minimize vulnerability in a hostile environment. The rotor disc loading was constrained to 10 pounds per square foot or less to minimize undesirable groundwash effects.

b. For the USAF SAR aircraft, the emphasis was on mid-point rescue pickup cability. A higher disc loading was permitted than for the Army aircraft but was still constrained to 15 psf or less.

c. The primary interest for the Navy aircraft was in developing a design with extended sortie time capability, an eight-hour sortie goal being achieved.

d. For the civil aircraft, low cost and highest reliability in the design were promoted through an

FIGURE 1-1: U.S.ARMY MAVS -MANNED AERIAL VEHICLE, SURVEII ICE

approach which made maximum use of proven technology and readily available subsystems.

Table 1-1 compares the important geometric, weight, and performance characteristics of the designs.

In addition to the four conceptual designs, it is shown that the tilt rotor has the potential to effectively fill other future missions in the post-1980 time period.

This volume covers the configuration descriptions, materials and structural design, weights, flying qualities characteristics, control systems, noise, aeroelastic stability, and performance of each of the point designs. It concludes by summarizing the current status of tilt-rotor technology and recommending additional research programs. It is adjudged that the technology is now in hand to start on the development of a tilt-rotor flight research aircraft.

TABLE 1-1

COMPARISON OF DESIGN POINT CHARACTERISTICS

	U.S. Army MAVS	U.S.A.F. SAR	U.S. Navy Sea Control	Civil Off-Shore Oil
Rotor Data				
Diameter (FT)	30.0	27.0	30.3	26.0
Disc Loading (PSF)	10.0	14.8	15.0	12.0
Wing				
Area (FT^2)	224	186	229	200
Wing Loading (PSF)	62.9	91.1	94.6	64.0
Weights (Lbs.)				
Weight Empty (Structures, Propulsion, Flt. Controls & Fixed Equip.)	10,851	11,500	10,035	8,846
Fixed Useful Load (Crew & Trapped Liquids)	440	900	760	400
Oper. Weight Empty	11,291	12,400	10,795	9,246
Mission Equipment	1,412	295	2,536	136
Expendable Load	-	-	2,060	2,160
Fuel	1,405†	4,275	6,250	1,268*
Design Gross Weight	14,108	16,970	21,641	12,810
Performance				
Hover Ceiling STD, DGW (FT)	13,500	9,000	3,300	4,700
Airspeed NRP/5000'/ STD (KTS)	300	325	305	282

*Note: Fuel shown corresponds to 109 nautical miles radius; maximum fuel capacity is 2,000 lbs.

†Note: Fuel shown corresponds to 2 hour cruise; maximum full capacity is 3,000 lbs.

2.0 INTRODUCTION

2.1 Background

A wide variety of V/STOL concepts, ranging from jet and fan to propeller and rotor-driven vehicles, has been studied by the aeronautical community, both independently and with government support. These studies have shown that the tilt-rotor aircraft is a promising candidate for military and civil missions.

For military applications, a Boeing study conducted in 1968 of 12 different low disc loading V/STOL concepts applied to the Army Light Tactical Transport Aircraft System (LTTAS) mission showed that the tilt rotor offered the greatest flexibility in terms of speed, range, and altitude. This is illustrated by Figure 2-1. The helicopter, with or without wings, runs out of propulsive force around 200 knots. This can be extended to approximately 250 knots by compounding but at a weight penalty of about 20 percent. In addition, the power required would be from 50 percent to 100 percent greater than that of a 180 to 200-knot helicopter. The tilt rotor, with hover power comparable to that required by a helicopter, offers speeds of 300 to 350 knots.

The tilt-rotor aircraft has the following characteristics:

a. Hover efficiency better than a helicopter (because the rotor blade twist is not compromised by edge-on flight blade loads).

b. Cruise speed in excess of 300 knots and the good cruise and loiter efficiency of the fixed-wing, moderate wing-loading, turboprop aircraft.

c. Low external noise levels in all flight regimes.

d. Good ride quality and low vibration and internal noise levels.

e. Low speed agility of the helicopter.

f. Downwash velocity much lower than jet-lift or fan-lift aircraft, approaching or equaling that of a helicopter.

g. Smooth and continuous transition between hover and cruise.

h. Broad range of flight speeds available with one engine out.

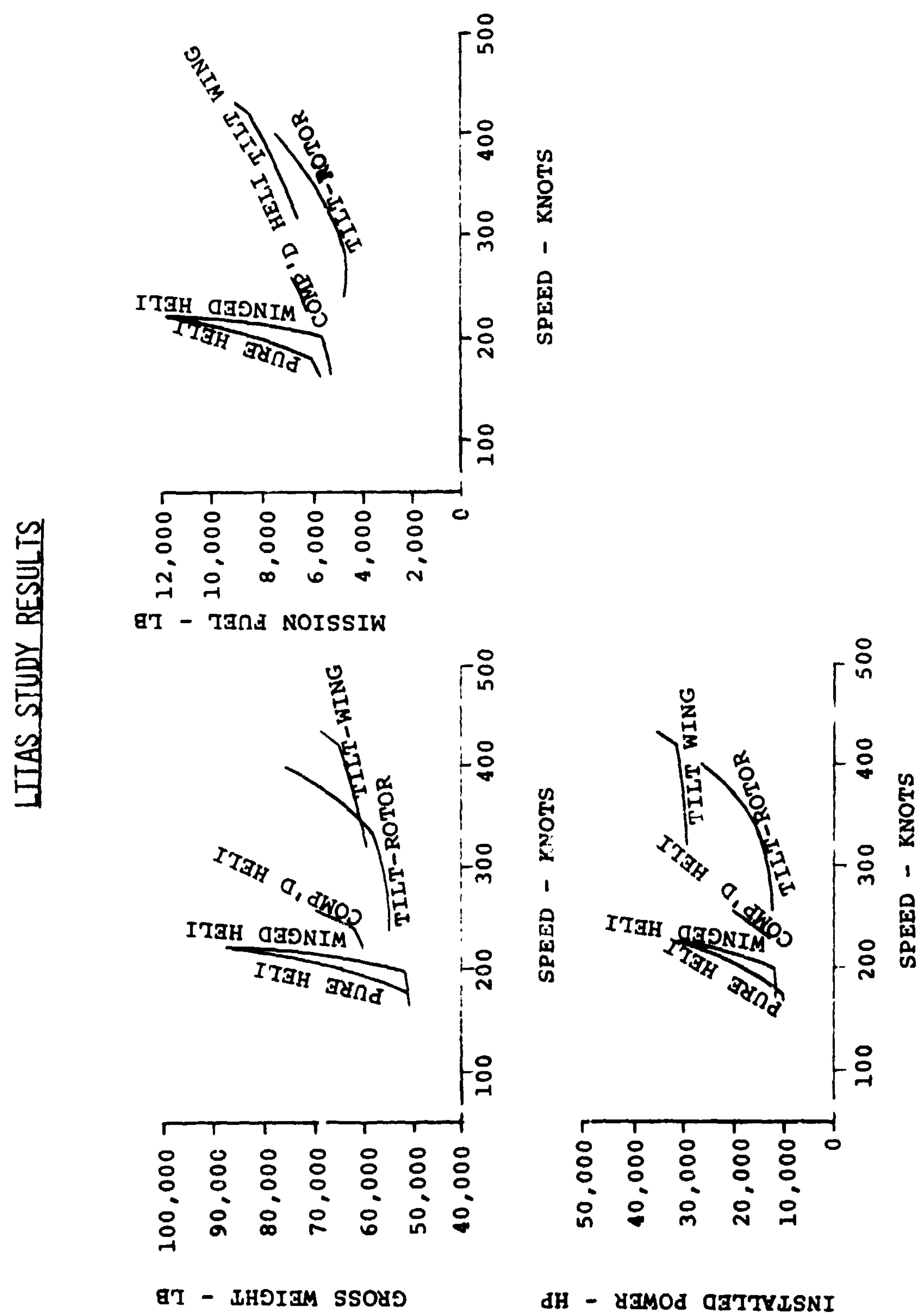


FIGURE 2-1: EFFECT OF DESIGN CRUISE SPEED ON WEIGHT, POWER, AND FUEL

i. Good overload takeoff and landing capability.

j. Ability to autorotate in case of complete power failure.

Since 1966, Boeing has conducted over 3,500 hours of wind tunnel testing using 25 different models. The tilt rotor technological base established to date is sufficient to start building a technology demonstrator aircraft now, leading to useful V/STOL tilt-rotor aircraft for military and/or civil applications in the 1975 to 1980 time period. The purposes of this study, as a key step in this process, are to define a useful operational military and/or civil tilt-rotor conceptual design, design a V/STOL tilt-rotor flight research aircraft, and provide information for planning the next follow-on activity and the overall aircraft program. This volume covers the selection, design, and capabilities of the most promising operational tilt-rotor aircraft for the 1975-1980 time period, together with an assessment of the state of tilt-rotor technology and recommendations for additional programs.

2.2 Criteria For Selecting Applications of Potentially High Payoff

The following criteria were followed in selecting roles and missions to which the tilt-rotor holds the promise of being successfully applied:

a. There must be a real mission existing in the 1975-80 time period; i.e., one for which there is a current or acknowledged need. In addition, either a gap must exist in the current capability of satisfying the mission requirements or there should be a strong motivation to significantly increase the effectiveness of the aircraft serving this role.

b. Either vertical takeoff and landing capability or efficient hovering ability must be an important requirement for this mission.

c. The tilt rotor must provide a significant improvement in system effectiveness over other aircraft concepts. System effectiveness is composed of:

   1. Unique performance capabilities.

   2. Good flying qualities, low internal noise, and low vibration levels for pilot acceptance.

   3. Operational suitability.

4. Low external noise level

5. Economics

6. Maintainability and reliability

7. Safety

8. Survivability

d. Other factors which must exist to maximize the probability of success are:

1. Mission Flexibility - Inherent mission flexibility and growth potential influence successful entry of a new concept through expansion of roles and longevity of use. It would be desirable to find applications which have potentially expanding requirements that could be satisfied by the tilt rotor. For example, a mission which currently has only a minor hover requirement but which potentially could require extended hover capability in the future would be well served by the tilt rotor with its efficient hover characteristics.

2. Technology Risk - There is an optimum level of technological risk to maximize the chances of success for a new program. Clearly, if the risk is very great, the most probable result will be failure. On the other hand, some risk is associated with a program where advances are being made to provide high payoff and a superior system. It is this fact that provides the impetus for the current carefully planned tilt-rotor program in which the technology is demonstrated through flight of a research aircraft prior to development of an operational aircraft. To minimize other technical risks, the only engines considered for the design-point aircraft were those for which some initial development had been completed.

2.3 Applications for First-Generation Tilt-Rotor Aircraft

A large number of potential applications was considered for inclusion in this study. The majority of these missions was eliminated on the basis of violating the criteria previously discussed -- most noticeably with respect to operational timing. However, four of the missions considered were retained for this study and are presented in this volume.

Some of the key missions which were considered and eliminated are:

a. Gunship - The tilt rotor would appear to make an excellent gunship due to its low speed agility coupled with a top speed in excess of 300 knots. These characteristics should enable it to perform the Armed Escort Mission effectively, escorting the fastest future helicopter and capable of being diverted to hit targets of opportunity. While the tilt rotor would be highly effective as a gunship of the future, this application was eliminated as not fitting the time frame of this study.

b. LTTAS - In 1968, Boeing conducted an extensive company-funded study of the Army Light Tactical Transport Aircraft System (LTTAS) mission. Twelve different rotorcraft were studied for this application. These included two pure helicopters, two winged helicopters, three compound helicopters, two advanced rotor concept helicopters, two stowed-rotor aircraft, and the tilt rotor. A major conclusion of the study was that, with respect to twenty-four different evaluation factors, the tilt rotor and winged helicopter ranked first or second in almost every category. In overall ranking, the tilt rotor was first with the winged helicopter second. This evaluation was based on such factors as productivity, noise and vibration, air transportability, reliability, maintainability, safety, fly-away cost and risk -- with cruise speed being a fall-out of the study rather than being a requirement.

   Nevertheless, the most probable timing for procurement of a new LTTAS is 1980-1985, beyond the period specified for this study. Consequently, this mission was eliminated from further consideration in this study.

c. HXM - Studies have been conducted for a number of years to define the basic requirements for a U. S. Marine Corps medium assault helicopter, currently designated HXM. This concept, also referred to as VAMT (Vertical Assault Medium Transport), is basically designed to perform a ship-to-shore assault transport mission although other missions such as crisis control have been considered for it. Design mission radii from 150 to 500 nautical miles have been discussed. System effectiveness is enhanced by reducing the time required to land a complete Marine battalion from the time the first aircraft lands until the last has left the landing zone. Because of its inherently high cruise speed, the tilt-rotor fulfills this requirement very nicely and because of its efficient long-range characteristics, looks particularly attractive as the design radius increases.

However, since the requirements for this mission have not been totally resolved at this time, this mission was not considered further for this tilt-rotor conceptual design study.

d. Commercial Transport - The tilt-rotor's capabilities are well suited to a commercial short haul transport mission. A recent AIAA committee paper[1] suggests that for a successful commercial V/STOL:

> "A machine is required that keep the existing hover and low-speed VTOL capabilities of the helicopter (perhaps with reduced noise) has a cruise speed of 300 to 350 knots and cruise efficiencies giving -75 to -150-mile stage lengths with useful payload and with much reduced noise and maintenance-generating vibration."

The tilt-rotor intrinsically fulfills these requirements and should prove to be an excellent commercial transport. Nevertheless, a necessary prerequisite to successful commercial operation is to gain needed military or utility operating experience. This would require that a commercial transport be a second generation tilt-rotor aircraft and would place initial commercial usage of the tilt-rotor in the 1980 to 1985 time period.

Four of the applications considered did meet all of the acceptance criteria. Therefore, although this study only required the definition of a single operational application for the tilt-rotor, four different applications are presented. The four airplanes whose selection, design, and capabilities are covered in this volume are:

a. U.S. Army MAVS - Manned Aerial Vehicle, Surveillence- A U.S. Army program to develop a tilt-rotor aircraft for surveillance missions.

b. USAF SAR - Search and Rescue - An advanced search and rescue aircraft for the U.S. Air Force.

c. U.S. Navy Sea Control Aircraft - A V/STOL aircraft operating from a new class of ship being developed by

[1]"STOL,VTOL, and V/STOL . . . Where Do They Fit In?,"AIAA Ad Hoc Committee, AIAA 8th Annual Meeting and Technical Display, Washington, D.C., October 1971

B

the U. S. Navy - the Sea Control Ship. The aircraft discussed in this report is a long endurance sensor carrier, also used to deliver weapons against low-resistance targets.

d. Civil Off-Shore Oil Rig Resupply Aircraft - A civilian V/STOL aircraft used to ferry workers to and from the off-shore oil rigs, to provide longer range and greater efficiency than the helicopters currently performing that mission.

2.3.1 U. S. Army MAVS. - The MAVS, or Manned Aerial Vehicle, Surveillance, is a U. S. Army program to develop advanced aerial surveillance capability. The missions include those currently being flown by the OV-1 Mohawk. The tilt-rotor aircraft has VTOL capability which permits landing at forward sites and speeds access time to needed information. In addition, like a helicopter, the tilt rotor can operate from unprepared sites and is not limited by landing field surface conditions. Other unique capabilities of the tilt rotor which provide a significant improvement in system effectiveness relative to other aircraft are:

a. Low Speed Maneuverability - For effective evasion from hostile action, the tilt rotor has excellent maneuverability at low speed. Approach and departure paths are not constrained by terrain features, as is the case with a conventional aircraft.

b. Range of Flight Speeds - The tilt rotor can fly at very low speed if necessary to match sensor characteristics then can smoothly transition to high-speed flight.

c. Low Noise - The perceived noise level of the tilt rotor flying at 200 knots airspeed, 1,000 feet directly above an observer is only 68 PNdB. This compares to 82 PNdB for the OV-1. With each aircraft 2,000 feet from the observer and at an altitude of 1,000 feet, the perceived noise level from the tilt rotor is 53 PNdB compared to 73 PNdB from the OV-1.

d. Survivability - Because it is capable of cruise speeds in excess of 300 knots, the tilt rotor can perform high-speed dashes to evade areas where enemy antiaircraft capability exists. Because of its high-speed capability and good maneuverability, the tilt rotor has good trajectory control.

In addition, the other stated criteria are met. The aircraft size is less than 15,000 pounds in gross weight. In terms of mission flexibility, there are several other Army missions that the tilt rotor could perform well. With a reconfigured fuselage, it will make an excellent utility transport. Because it has VTOL capability, it can operate directly between the main operating base and the front line.

2.3.2 U. S. Air Force SAR. - Current search and rescue operations involve a coordinated effort of different aircraft. HH-3E and HH-53 helicopters are the primary means of recovery while fixed-wing aircraft, notably A-1's, are used to orbit the search area until the helicopters arrive and to provide suppressive fire against enemy troops. The tilt-rotor aircraft can fulfill both functions with the speed, maneuverability, range, and endurance of the fixed-wing aircraft and the hover and vertical landing capability of the helicopter. Consequently, this is a mission that meets the first three criteria very well: the tilt rotor fills the requirements of a well defined mission, the mission inherently requires hovering capability, and the tilt rotor provides a significant improvement in effectiveness over other aircraft.

2.3.3 U. S. Navy Sea Control Aircraft. - The U. S. Navy Sea Control Ship concept involves the development of a relatively small ship (approximately 15,000 tons) from which V/STOL aircraft will operate, carrying weapons and sensors to control the aerospace and hydrospace in the vicinity of a task force or convoy. There are requirements for an aircraft with long endurance to be used as a sensor carrier. It may also be used to attack targets. The tilt rotor's combination of VTOL capability and high speed permits effective use of all the sensors used by fixed-wing aircraft without requiring a long flight deck, catapult, or arresting gear. In comparison to a helicopter, its higher loiter and cruise speed provide increased sweep rate capability in escort or screen applications and its higher cruise speed significantly reduces the time late to datum. It has excellent overload capability to take advantage of wind over deck and limited deck lengths. In addition, the inherent hover efficiency of the tilt rotor makes it quite adaptable to requirements imposed by development of new weapons or sensors which may require recovery, towing, or other low-speed operations.

2.3.4 Civil Off-Shore Oil Rig Support Aircraft. - Helicopters are currently used to support off-shore oil well drilling operations throughout the world with major activities in the Persian Gulf, North Sea, and Gulf of Mexico. Major helicopter operators, including World-Wide Helicopters, Okanagan Helicopters, Bristow Helicopters, and Petroleum Helicopters, provide a contract service to the petroleum companies consisting of support for monitoring of well status, drill rig support, and crew change. The drill rig support mission requires carrying supervisory personnel, geologists, technical people, well logs, and equipment to the drilling site. The crew change mission requires that, once a week, the helicopters transport replacement crews to the oil rigs and return the old crew to shore. A Boeing study conducted in 1970 with the cooperation of Petroleum Helicopters, Inc. indicated that, on the average, operations were conducted 30 to 45 miles further off shore than they had been five years previously.

In the next five years, it is anticipated that the operations will extend another 30 miles, with some operations reaching as far as 200 to 300 miles from shore. The high speed and range of the tilt rotor can provide a step improvement in productivity and a major time saving for such operations.

2.4 Format for This Volume

A conceptual design study was performed to define the characteristics of a first-generation tilt-rotor aircraft for each of the four selected applications.

In each case, a parametric study was made to select a point design aircraft. Section 3 of this volume discusses these trade studies including the detailed definition of the design mission profiles, mission requirements, the ground rules followed for the studies, the parameters considered, and the factors which influenced the design point selection.

Section 4 is a description of each airplane including the following details:

a. Configuration

b. Materials/structural design

c. Weights

d. Noise

e. Flying qualities

f. Control systems

g. Dynamics

h. Performance

Section 5 is an assessment of the status of tilt-rotor technology today and specific recommendations for additional research programs that should be completed before the operational aircraft described in this volume become reality.

3.0 TRADEOFF STUDIES

3.1 U. S. Army MAVS Aircraft

3.1.1 Mission Definition. - The design mission profile for the MAVS aircraft is shown in Figure 3-1. It consists of a two-hour cruise at 200 knots plus other allowances for take-off, reserves, etc. This speed was selected as being representative of photo reconnaissance requirements. The required mission load was 1,412 pounds, consisting of SLAR, photo recon equipment, ECM pods, etc. The aircraft carries a crew of two in a side-by-side configuration.

A vertical climb rate of 500 feet per minute with both engines operating was required at 4,000 feet pressure altitude, 95°F. A normal power cruise speed of at least 300 knots true airspeed was required at 5,000 feet/standard day conditions. An additional performance requirement was that the airplane have sufficient internal fuel capacity for four hours of flight at 200 knots. For this condition, the airplane was permitted to overload.

3.1.2 Study Ground Rules. - In order to satisfy the criteria of minimizing technology risks in areas not directly related to the tilt-rotor aircraft, an engine with high probability of development by 1975-1980 was selected. The engine chosen for this application was a UTTAS engine rated at 1500 shaft horsepower.

The wing, fuselage, and empennage were designed using composite materials, allowing a 15-percent reduction in their weight. The many studies of potential weight savings from the use of composites all indicate possible savings of 25-35 percent so that the 15 percent assumed is considered conservative.

Advanced technology transmissions based on the Army/Boeing Heavy Lift Helicopter program were used. These provide a 15-percent reduction in drive system weight.

The maximum operating speed (V_{MO}) was selected at 350 knots, and the maximum operating Mach number (M_{MO}) was set at 0.569. These values were picked to provide reasonable speed margins relative to the anticipated cruise speed capability of the aircraft.

Engine manufacturer's specific fuel consumption was increased by 5 percent in accordance with MIL-C-5011A.

As discussed in Section 4.5.10, the horizontal and vertical tails were sized on the basis of tail volume coefficients of 1.0 and 0.128, respectively.

1. WARM-UP, TAXI AND TAKEOFF: 5 MIN @ NORMAL RATED POWER, SEA LEVEL
2. CLIMB TO 5000FT. @ NORMAL RATED POWER AND SPEED FOR MAXIMUM RATE OF CLIMB
3. CRUISE OUTBOUND FOR 2 HOURS @ 200 KNOTS TRUE AIRSPEED
4. LAND WITH 10% (INITIAL) FUEL RESERVE

NOTES:

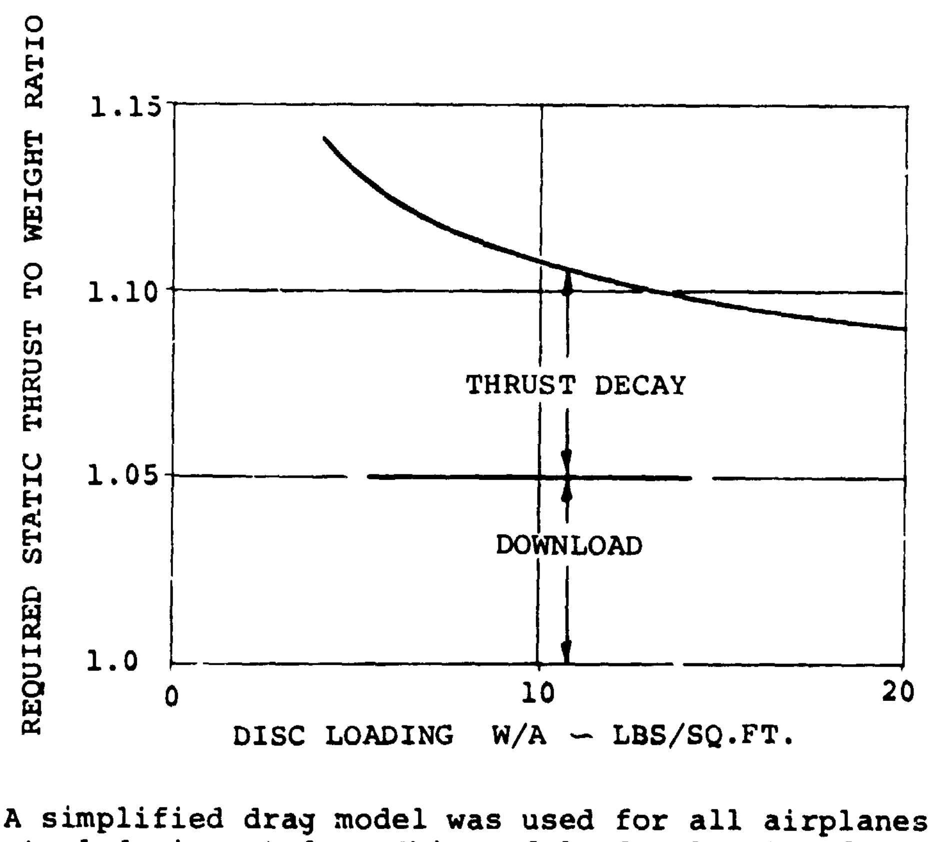
1. MISSION FLOWN @ STANDARD ATMOSPHERE CONDITIONS UNLESS OTHERWISE NOTED
2. SFC INCREASED 5% PER MIL-C-5011A

FIGURE 3-1. MODEL 222-1A ARMY-MAVS TACTICAL AIR OBSERVATION MISSION PROFILE

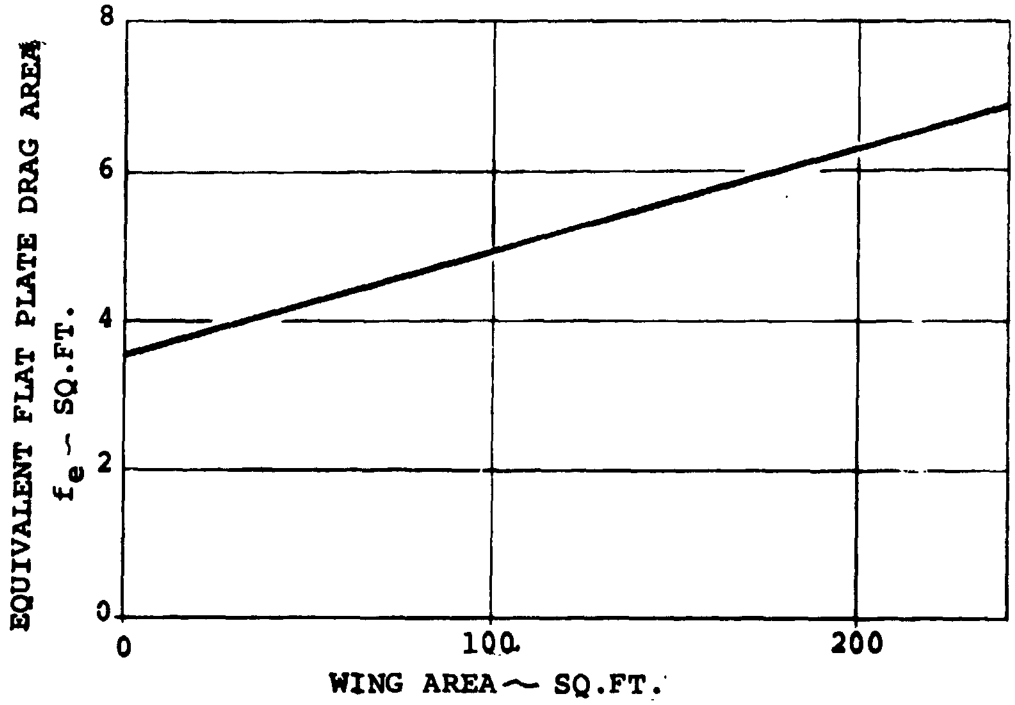
The static thrust-to-weight ratio required to provide 500 fpm vertical rate of climb was assumed to be composed of two parts: (1) a 5-percent download on the wing and fuselage due to the rotor, and (2) the thrust decay at constant power due to the vertical climb. That is,:

$$\text{Static thrust-to-weight ratio} = \frac{T_O}{W} = \frac{1 + \text{download/thrust}}{\text{thrust/static thrust}} = \frac{1.05}{T/T_O}$$

The 5-percent download-to-thrust ratio has been derived from full-scale and model-scale tests as shown in Figure 5-3, Section 5.1. The thrust ratio, T/T_O, was calculated from axial momentum theory. The resultant static thrust-to-weight ratio required, as a function of disc loading, is shown in the sketch below.

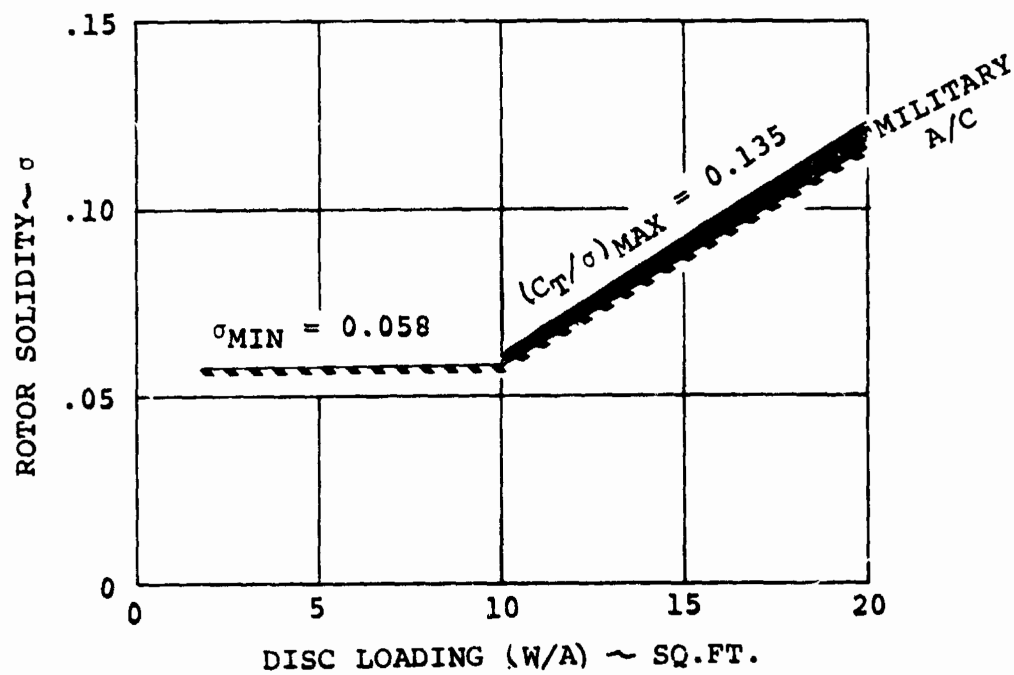


A simplified drag model was used for all airplanes in the conceptual design study. This model, developed and verified as being appropriate by many past detailed design and preliminary design studies at Boeing, represents the drag as being a simple linear function of the wing area. The intercept and slope of the curve shown below was calculated from detailed estimates of the component drag buildup of the aircraft using Boeing Document D8-2194-1, "Drag Estimation of V/STOL Aircraft," Reference 4.



During the parametric studies, the rotor solidity was varied as a function of disc loading as shown by the boundaries in the following sketch. For the military aircraft, a limiting thrust coefficient-to-solidity ratio, C_T/σ, of 0.135 was used. This value may be achieved by aerodynamic and aeroelastic developments such as those being pursued in the Army/Boeing Heavy Lift Helicopter program. This value of C_T/σ provides a minimum permissible value of solidity which is directly proportional to the disc loading, as shown in the sketch, reaching a value of .058 to .062, depending upon ambient conditions, when the disc loading is reduced to 10 psf. For disc loadings less than 10 psf, the minimum permissible solidity was constrained to a value of 0.058 based on practical design and manufacturing considerations of achieving reasonable torsional and flapping stiffness.

All the configuration tradeoffs in the conceptual design studies were done using an advanced computerized aircraft sizing technique called VASCOMP, "V/STOL Aircraft Sizing and Performance Computer Program," (Reference 2). This program, developed by Boeing under a series of NASA contracts, defines design characteristics such as weight breakdown, required propulsive power, and physical dimensions of aircraft which are designed to meet specified mission requirements.



3.1.3 Parametric Trades. - Parametric sizing studies were made with variations in both the design disc loading and the wing size (wing loading). The final design point aircraft was chosen with a disc loading of 10 pounds per square foot and a wing loading of 62.9 pounds per square foot. The aircraft weighs 14,108 pounds and has two 30-foot-diameter rotors.

Figure 3-2 shows the effect of disc loading on the design gross weight, the rotor diameter, and the total aircraft span, rotor tip to rotor tip. A maximum disc loading of 10 psf is shown as a requirement to minimize the downwash environment. It is seen that, within the constraint of this maximum value, the effect of increased disc loading is to reduce the aircraft size and weight. As the disc loading is reduced from 10 psf to 6 psf, the design gross weight increases from 14,100 pounds to 15,550 pounds, the rotor diameter increases from 30 feet to 40.6 feet, and the total aircraft span increases from 67 feet to 89 feet. All aircraft on this figure meet the 500-fpm vertical climb requirement at 4,000 feet/95°F using the UTTAS engine (the maximum disc loading to achieve this is approximately 12 psf for the MAVS aircraft). Based on minimizing size and weight, an aircraft disc loading of 10 psf was chosen.

Figures 3-3 and 3-4 show the effect of wing loading on five different parameters: the design gross weight, cruise

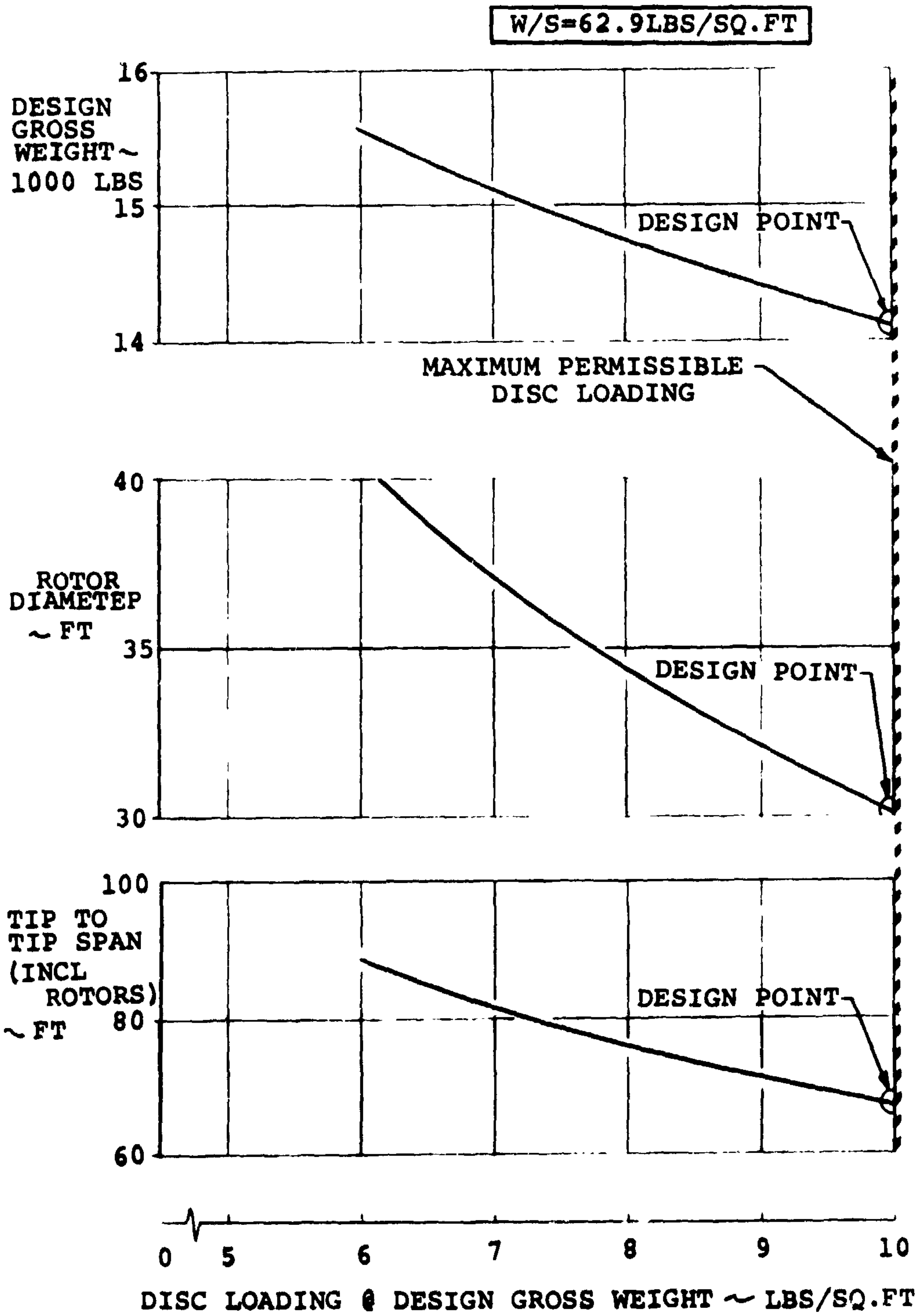


FIGURE 3-2: U.S. ARMY MAVS - PARAMETRIC TRADES

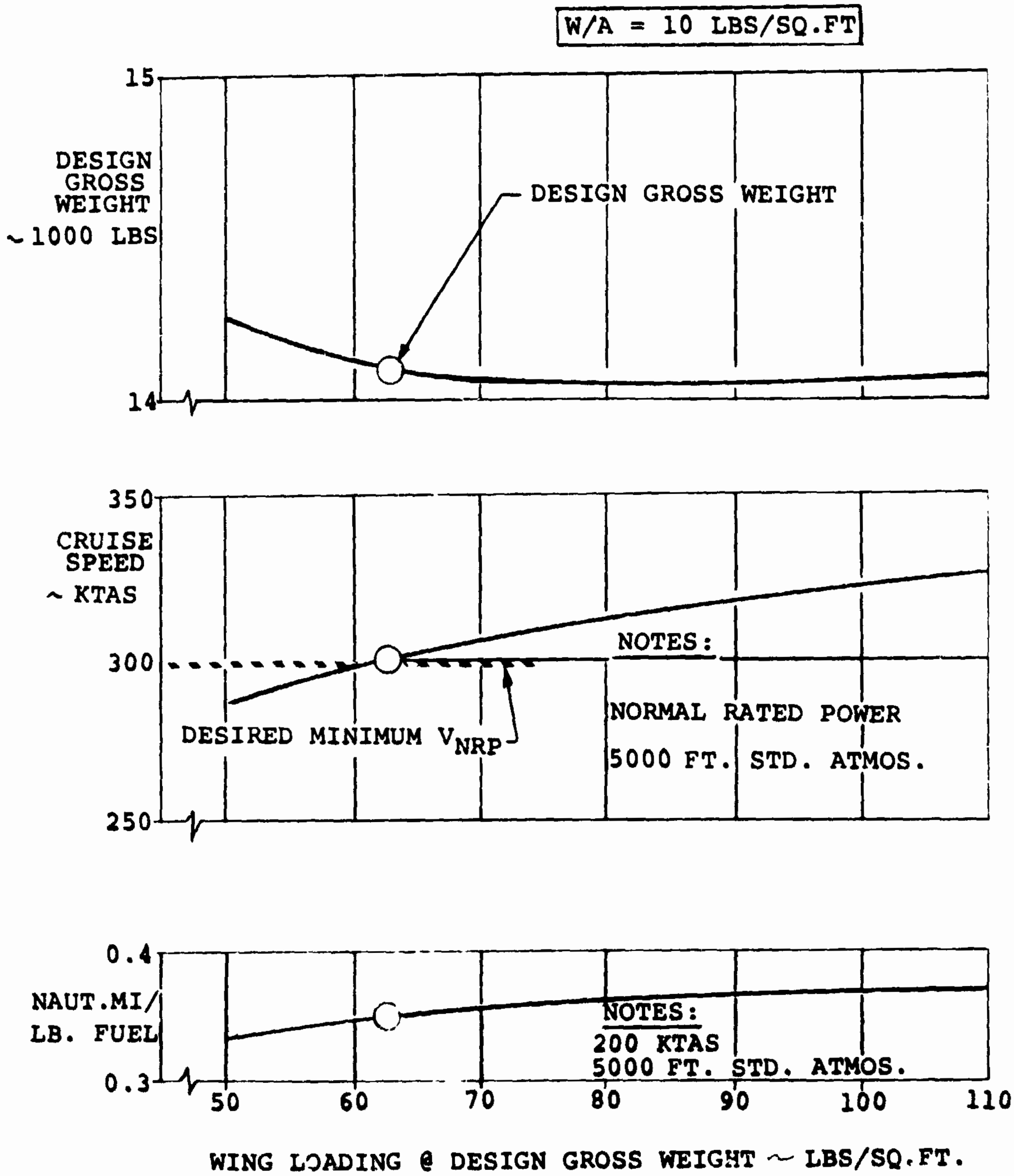


FIGURE 3-3: U.S. ARMY MAVS - WING LOADING SELECTION

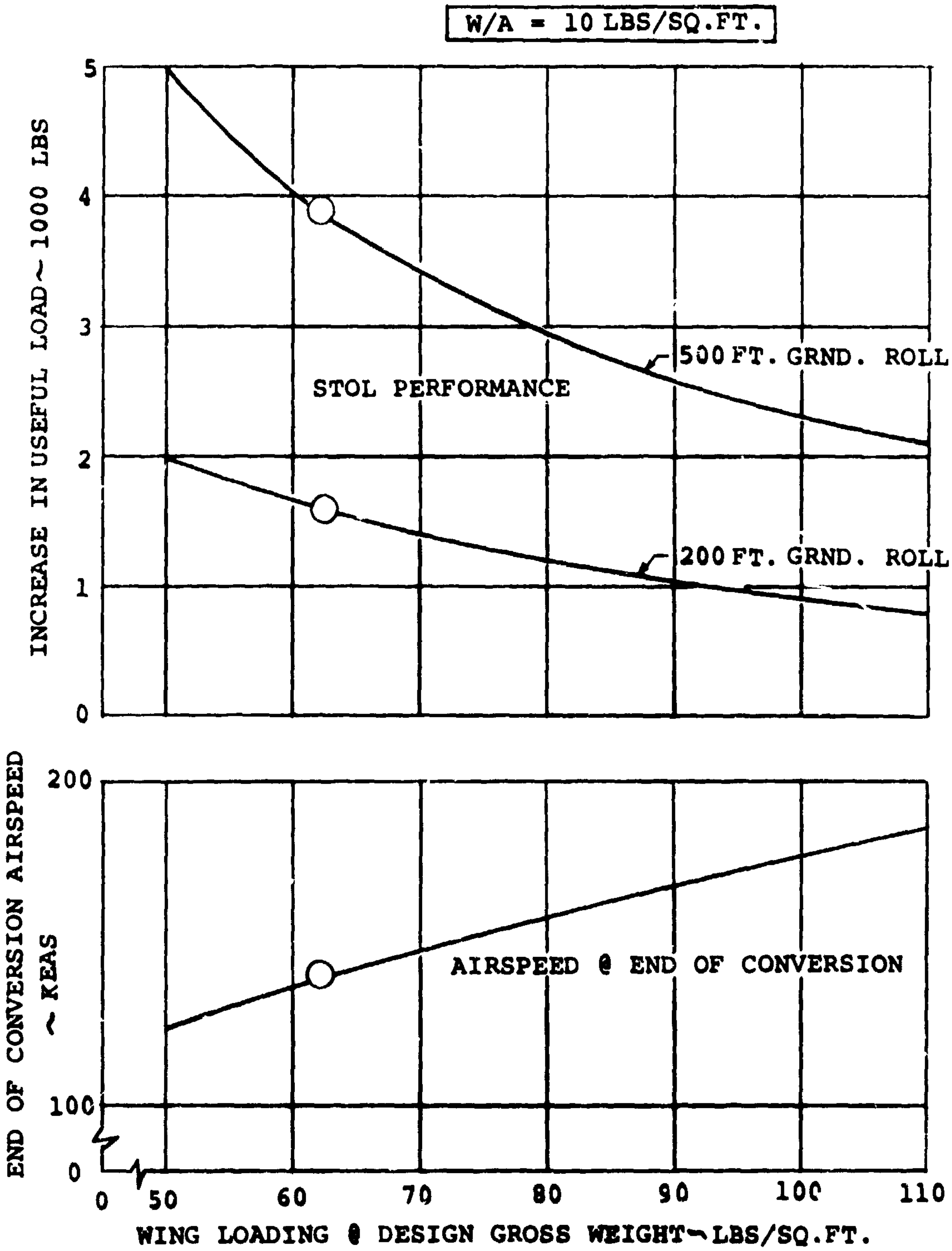


FIGURE 3-4: U.S. ARMY MAVS - WING LOADING SELECTION

speed, the cruise spec range at 200 knots, STOL performance, and the end-of-conversion airspeed. It is seen that the design gross weight is nearly independent of wing loading and the 200-knot cruise performance improves only slightly with increased wing loading. The cruise speed at normal power, 5,000'/STD increases with increased wing loading with a wing loading of 62.8 psf or greater required to obtain the desired level of 300 knots. The end-of-conversion airspeed is seen, on Figure 3-4, to increase with increasing wing loading. For the MAVS operation, where airspeeds between 200 knots and 140 knots are commonly employed during the photo reconnaissance mission, it is advantageous to pick an airplane with a low end-of-conversion speed. In addition, lower wing loading provides greater internal fuel capacity, better STOL performance, and an increase in cruise ceiling. Based on this consideration and on the normal rated power cruise speed shown in Figure 3-3, the design wing loading was picked at 62.9 psf. This provides an end-of-conversion speed of 140 knots and a cruise speed of 300 knots. In addition, as shown by Figure 3-4, this provides increased STOL performance when a short ground roll can be made.

3.2 U. S. Air Force SAR Aircraft

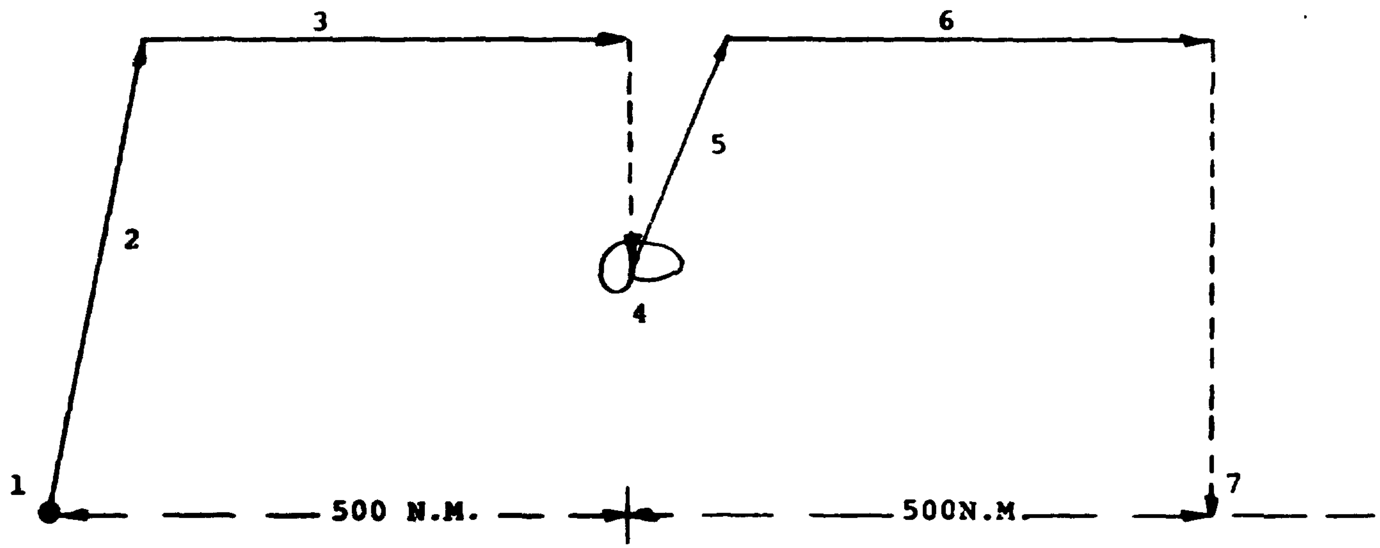
3.2.1 Mission Definition. - The mission profile to which the Air Force Search and Rescue Aircraft was designed is shown in Figure 3-5. It consists of climbing to optimum altitude, cruising 500 nautical miles at speed for normal rated power, hovering for one-half hour, pick-up of 3 men weighing 200 pounds each, and return. Appropriate reserves are added. The optimum cruise altitude for this aircraft (minimum fuel) was 20,000 feet.

The mission load consists of 150 pounds of rescue equipment (litters, forest penetrator, rescue sling, flares and gun, life raft, etc.), a 5.56-mm gun and ammunition, plus special rescue electronics including airborne equipment to locate the rescuee.

The aircraft carries a four-man crew consisting of two flight officers, a crew chief, and a paramedic.

The airplane was required to have a cruise speed of 300 knots TAS or greater at an altitude of 10,000 feet at mid-point hover gross weight.

An important requirement that dictated engine power and rotor diameter was that the aircraft be capable of hovering at mid-point after pickup of seven rescuees, consisting of the normal complement of three plus an additional group of four people representing the crew of a downed sister ship. This requirement did not influence the required mission fuel --



1. WARM UP, TAXI AND TAKEOFF: 3 MIN. @ NORMAL RATED POWER, SEA LEVEL
2. CLIMB TO OPTIMUM ALTITUDE @ MILITARY POWER AND SPEED FOR MAXIMUM RATE OF CLIMB
3. CRUISE OUTBOUND @ NORMAL RATED POWER SPEED
4. HOVER 1/2 HR., EFFECT RESCUE OF 3 PEOPLE (600 LBS) @ 5000'/95°F, MILITARY POWER
5. CLIMB TO OPTIMUM ALTITUDE @ MILITARY POWER AND SPEED FOR MAXIMUM RATE OF CLIMB
6. CRUISE INBOUND @ NORMAL RATED POWER SPEED
7. LAND WITH 10% (INITIAL) FUEL RESERVE

NOTES:

1. MISSION FLOWN @ STANDARD ATMOSPHERE CONDITIONS UNLESS OTHERWISE NOTED.
2. SFC INCREASED 5% PER MIL-C-5011A

FIGURE 3-5. USAF-SAR - HI-HI MISSION PROFILE the additional fuel required coming either from the reserves or from in-flight refueling.

3.2.2 Study Ground Rules. - The Lycoming PLT-27 turboshaft engine, rated at 1950 horsepower, was chosen in order to meet the mid-point hover requirement with a reasonable rotor diameter.

As was the case with the U. S. Army MAVS aircraft, the wing, fuselage, and empennage made use of composite materials for a component weight saving of 15 percent. In addition, the drive system was an advanced technology design based on Heavy Lift Helicopter experience. A 15-percent reduction in drive system weight was assumed.

A 5-percent increase in specific fuel consumption was applied per MIL-C-5011A.

The maximum operating speed (V_{MO}) and maximum operating Mach number (M_{MO}) were selected to be 350 KEAS and 0.569, respectively.

The mid-point hover requirement was interpreted as dictating a thrust-to-weight ratio of 1.10. This provides a 5-percent margin for download and an additional 5 percent for maneuver, control, etc.

The maximum desirable disc loading was 15 psf.

3.2.3 Parametric Trades. - Figure 3-6 shows the effect of rotor diameter on the following parameters:

a. Design Disc Loading - Disc loading at design gross weight (initial takeoff weight).

b. Normal power cruise speed at 10,000 feet, standard day conditions

c. Thrust-to-weight ratio available at mid-point after picking up 4 rescuees in addition to the normal 3 people.

d. Design gross weight.

It is seen that a minimum rotor diameter of 26.48 feet is required to meet the mid-point hover criterion. A more critical requirement on diameter is the 15-psf-or-less criterion, which dictates a rotor diameter of at least 26.75 feet. For the range of diameters studied, it is seen that all aircraft exceed the 300-knot cruise speed requirement.

The design point aircraft was selected with a diameter of

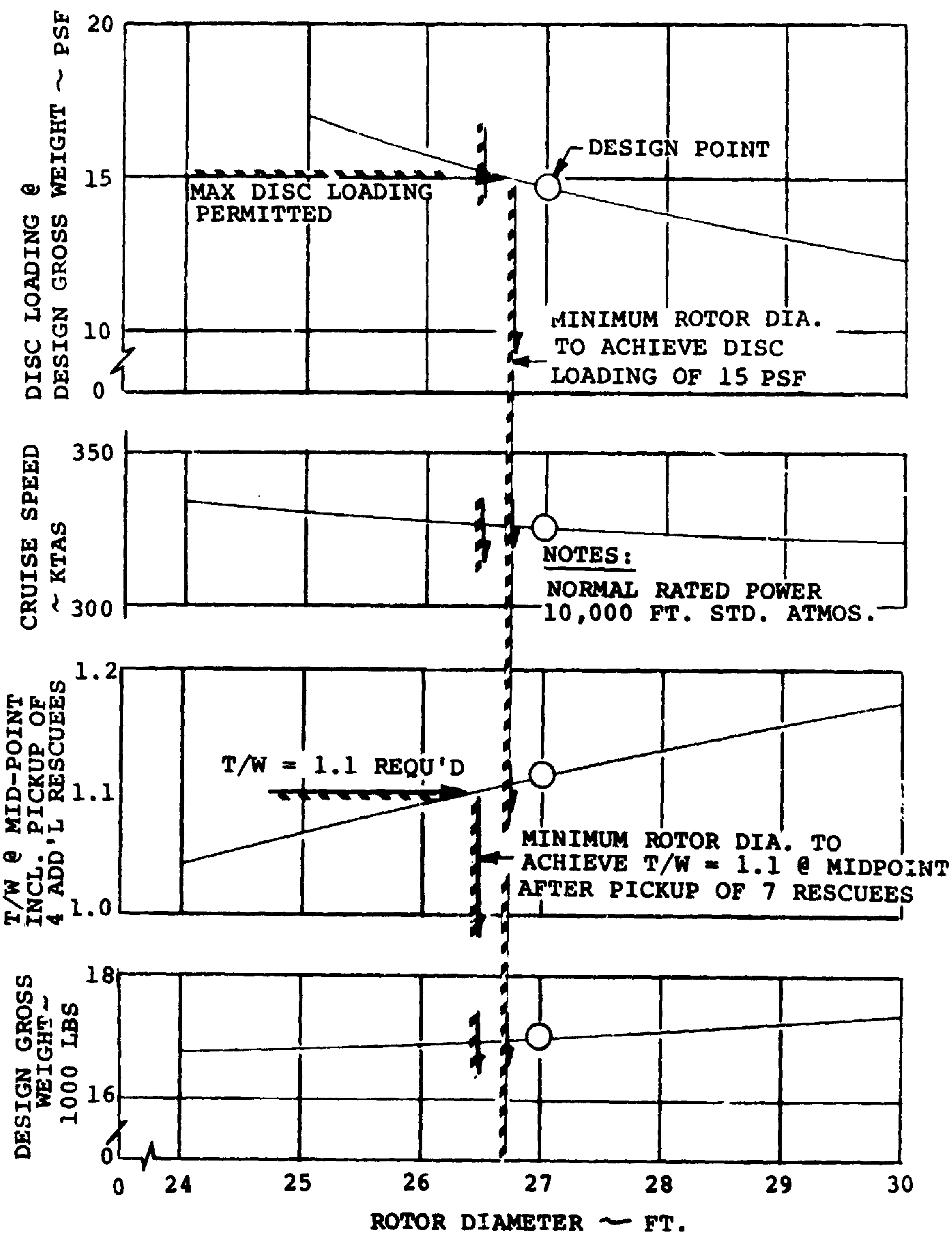


FIGURE 3-6: USAF SAR - PARAMETRIC TRADES

27 feet, resulting in a design gross weight of 16,970 pounds. Disc loading is 14.8 psf.

3.3 U. S. Navy Sea Control Aircraft

3.3.1 Mission Definition. - The Navy Sea Control Aircraft was designed to the mission profile shown in Figure 3-7. This represents an antisubmarine warfare mission. It consists of a 150-nautical-mile cruise at the speed for 99 percent of best range to a mid-point where the airplane loiters, engages in combat, and returns home. The allowance for taxi, takeoff, and reserves that were used is indicated on Figure 3-7. The mid-point loiter consisted of two parts: an extended loiter at 5,000 feet altitude and a 15-minute loiter at 500 feet altitude to simulate the combat. The loiter time at 5,000 feet altitude was a variable in the study with a minimum requirement of 3 hours and a desired goal of 6.7 hours (to give a total sortie time of 8 hours).

The mission load consists of 2,400 pounds of mission electronics, 1,060 pounds representing 2 MK46 torpedoes, and 1,000 pounds of expendable stores. In addition, the airplane carries 136 pounds of rescue and survival equipment and a crew of four. The aircraft was required to have a vertical rate of climb of 500 fpm at sea level, 90°F.

3.3.2 Study Ground Rules. - The Lycoming PLT-27 turboshaft engine, rated at 1950 horsepower, was chosen for this aircraft. It will be seen that this was the smallest engine which would provide 8-hour sortie capability. Included in the tradeoff study was the effect of derating the engines varying amounts, dependent upon mid-point loiter requirements, to provide an additional margin of reliability to both engines and transmissions.

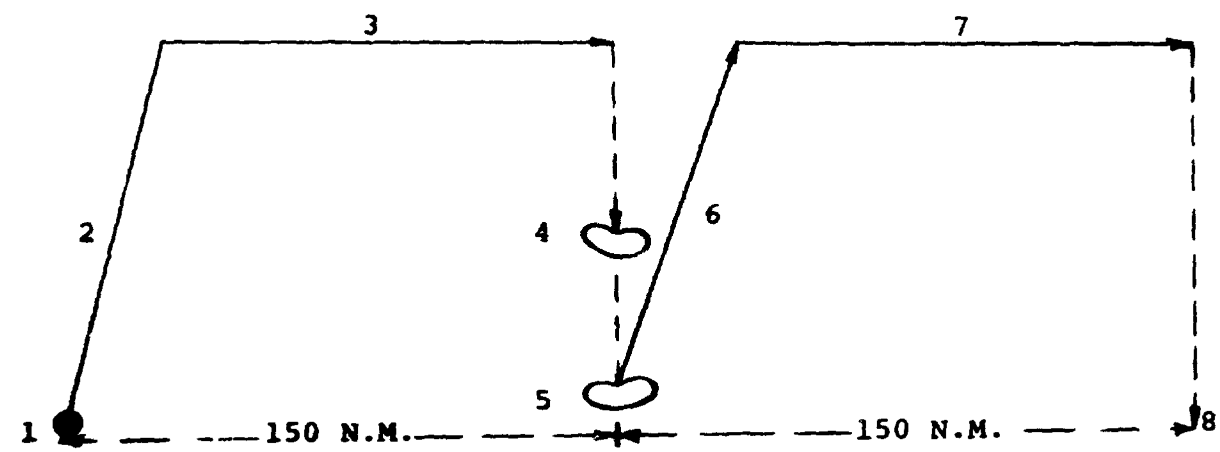
Approximately 80 percent of the secondary structure and 70 percent of the primary structure was assumed to be fabricated of composite materials. This results in a decrease of airframe maintenance of approximately 50 percent as a consequence of increased fatigue resistance, fracture toughness, corrosion resistance, and ease of repair of the composite structures. A weight savings of 15 percent was credited to use of composites.

The following limitations were imposed on the configurations studied:

a. Maximum rotor diameter of the order of 30 feet.

b. Maximum disc loading of 15 psf.

d. Minimum loiter time (including the 15 minutes for combat) of three hours.



1. WARM-UP, TAXI AND TAKEOFF: 2 MIN. @ NORMAL RATED POWER, SEA LEVEL, 90°F
2. CLIMB TO 10,000 FT @ MILITARY POWER AND SPEED FOR MAX. RATE OF CLIMB
3. CRUISE OUTBOUND @ SPEED FOR 99% BEST RANGE
4. LOITER @ 5000 FT
5. LOITER @ 500 FT FOR 15 MINUTES, FOR COMBAT
6. CLIMB TO 10,000 FT @ MILITARY POWER AND SPEED FOR MAX. RATE OF CLIMB
7. CRUISE INBOUND @ SPEED FOR 99% BEST RANGE
8. LAND WITH 10% (INITIAL) FUEL RESERVE

NOTES:

1. MISSION FLOWN @ STANDARD ATMOSPHERE CONDITIONS UNLESS OTHERWISE NOTED
2. SFC INCREASED 5% PER MIL-C-5011A

FIGURE 3-7. MODEL 222-1N NAVY-SEA CONTROL ASW MISSION PROFILE

The first two of these limitations were chosen to take advantage of existing technology. Although a three-hour minimum loiter time was imposed, it was desired to obtain 6.7 hours at mid-point in order to achieve an 8-hour sortie.

3.3.3 Parametric Trades. - Figure 3-8 is a design chart which shows the characteristics of a family of tilt-rotor aircraft designed for the ASW mission. The design gross weight is displayed on this figure as a function of rotor diameter and the desired mid-point loiter time. Curves of disc loading and required horsepower are superimposed on this chart. All aircraft on this figure are capable of a vertical climb of 500 feet per minute at sea level, 90°F, at the design gross weight. The boundaries shown by the cross-hatch represent the limits on rotor diameter, disc loading, and minimum loiter time. Two interesting design choices on this figure are designated by the letters "A" and "B."

Configuration "A" is the lightest aircraft which falls within the permissible envelope, weighing 17,720 pounds. It is capable of three hours of loiter at mid-point (4.3 hours total mission time). The PLT-27 engines can be derated to 1420 horsepower at sea level, standard day conditions, and still provide sufficient power for 500 foot-per-minute vertical rate of climb at sea level, 90°F. Rotor diameter is 27.4 feet.

Configuration "B" provides the greatest loiter capability of any aircraft within the permissible envelope, with 6.7 hours at mid-point and a total mission time of 8.0 hours. The aircraft weighs 21,640 pounds and has 30.3-foot-diameter rotors. This configuration was chosen as the design point.

3.4 Civil Off-Shore Oil Rig Support

3.4.1 Mission Definition - The primary mission for the off-shore oil rig support aircraft is shown in Figure 3-9. The design profile includes a cruise at 20,000 feet altitude to the oil rig with a vertical landing at sea level, 95°F. The airplane carries sufficient fuel to return to the shore plus one-half-hour of endurance. The Boeing study of operations at Petroleum Helicopters, Inc. (PHI) indicates that, for the crew exchange mission, the average flights are of 50 to 125 statue miles range. The passenger load is less than 6 passengers 10 percent of the time, 6 to 9 passengers 25 percent of the time, and 10 passengers 65 percent of the time. As a result, 12 passengers were chosen for the design payload and 125 statute miles (109 nautical miles) was chosen for the design radius.

The engine power was required to be sufficient to give a hover thrust-to-weight ratio of 1.1 (5 percent download plus 5 percent additional margin) at initial takeoff weight.

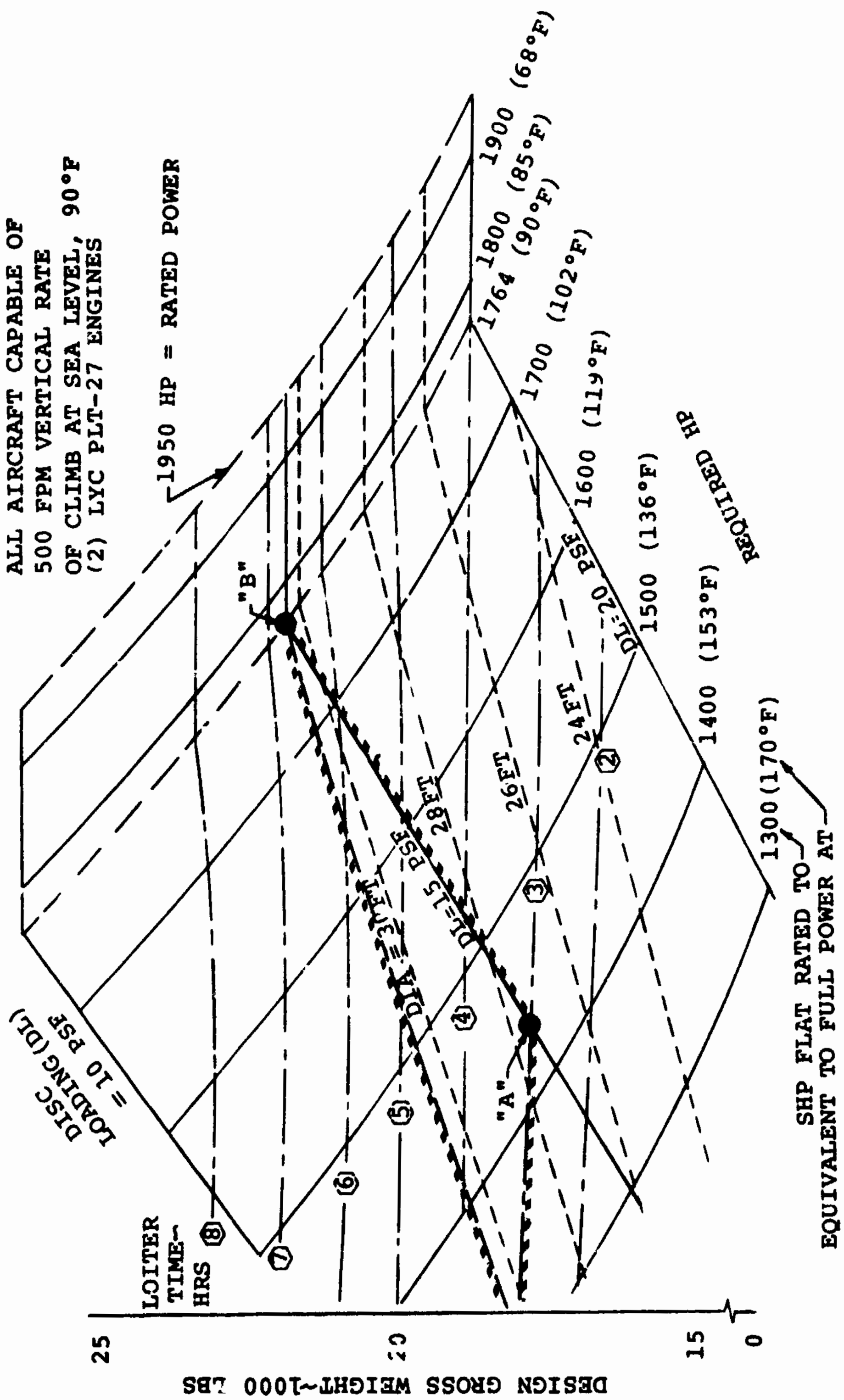
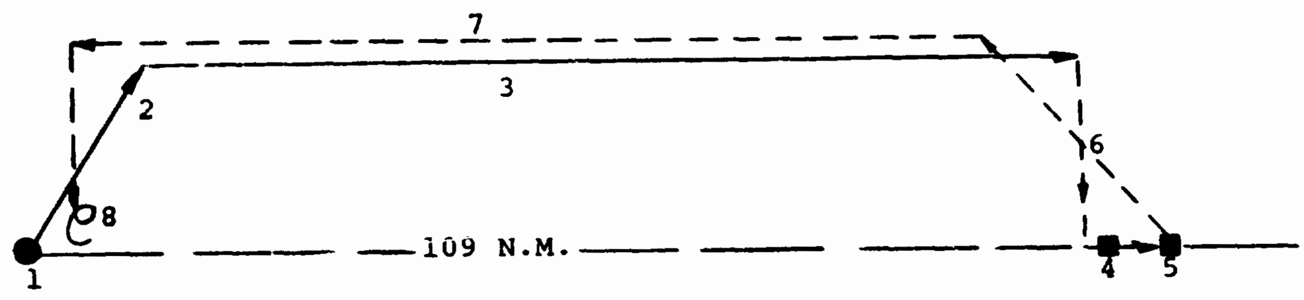


FIGURE 3-8: MATRIX OF U.S. NAVY ASW TILT ROTOR OPERATIONAL AIRCRAFT



1. WARM-UP, TAXI AND TAKEOFF: 1 MIN @ MAXIMUM POWER, SEA LEVEL 95°F
2. CLIMB TO 20,000 FT @ MILITARY POWER AND SPEED FOR MAX. RATE OF CLIMB
3. CRUISE OUTBOUND @ NORMAL RATED POWER
4. LANDING: 1 MIN @ MAXIMUM POWER, SEA LEVEL, 95°F
5. TAKEOFF: 1 MIN @ MAXIMUM POWER, SEA LEVEL, 95°F
6. CLIMB TO 20,000 FT @ MILITARY POWER AND SPEED FOR MAX. RATE OF CLIMB
7. CRUISE INBOUND @ BEST RANGE SPEED
8. LAND WITH 1/2 HOUR FUEL RESERVE @ MAXIMUM ENDURANCE SPEED

NOTES:

1. MISSION FLOWN @ STANDARD ATMOSPHERE CONDITIONS UNLESS OTHERWISE NOTED
2. SFC INCREASED 5% PER MIL-C-5011A

FIGURE 3-9. MODEL 222-1C CIVIL-OFFSHORE OIL OFFSHORE OIL MISSION PROFILE

The airplane carries a crew of two.

3.4.2 Study Ground Rules. - The primary requirements imposed for the selection of a design point configuration for the off-shore oil mission were that it make maximum use of proven technology and readily available subsystems.

The structure was fabricated of metal, rather than the composite materials used in the three military aircraft. The transmissions were a standard design concept, rather than the advanced technology transmissions chosen for the other configurations. The rotor solidity was based on a maximum C_T/σ of 0.09 rather than the value of 0.135 used for the military aircraft.

The rotor diameter was fixed at 26 feet in order to take maximum advantage of the Boeing analysis, design, and test experience derived from the research aircraft activities.

Only available proven engines were considered, specifically the Pratt & Whitney PT6C-40, the Lycoming T53-L-13B, and the General Electric T58-8F.

3.4.3 Parametric Trades. - All the engines meet the payload-radius requirement, the PT6 just meeting it, and the T53 and T58 exceeding the minimum requirement. With the candidate engines each capable of providing the required performance, the prime consideration for engine selection for a civil aircraft becomes engine reliability.

In considering engine reliability, a considerable amount of engine data is available for all three engines, each of which has significant service experience. The table below reflects the relative reliability levels of the basic engine families at this time expressed as the frequency of unscheduled removals per 1,000 engine operating hours due to engine causes. Also shown is the relative reliability of the candidate derivatives of these engines appropriate to the 1975 to 1980 time period, projected to that time.

	T53		T58		PT6	
Current values based on service and experience	-11	-13A	-8B	-10	-20	-27
	.31	.65	1.51	1.04	.13	.17
Projected values of derivative engines	-13B		-8F		-40	
	.35		1.20		.20	

The reliability predictions for the candidate engines are based on the service experience of the basic family, the specific configuration changes from the basic family, and the time at which the candidate engines would be incorporated into the proposed aircraft.

Based on these relative reliability levels coupled with unusually high TBO levels of 3,000 to 4,000 hours, the PT6 was selected. Another factor considered is the extensive network of logistics support and maintenance training facilities developed throughout the world on the PT6 based on its wide commercial applications.

4.0 AIRCRAFT DESCRIPTIONS

4.1 Aircraft Configurations

4.1.1 Introduction. - The discussion on configuration is divided into two sections. The first section (4.1.2) concerns features that are common to the four configurations depicted. The second section (4.1.3) spotlights the features that are unique to a particular configuration.

4.1.2 Configuration Approach. - The fuselage configuration for any given aircraft is primarily dictated by the mission requirements and the empennage configuration by stability and control requirements. For the four configurations presented where critical Mach number considerations are not particularly demanding, the wing size and geometry have been chosen for the most efficient and simple structural arrangement and nacelle attachment, consistent with the required relationship between the nacelle tilt pivot and wing for correct center of gravity location in hover and cruise flight. Some of the configuration features common to all four aircraft are:

a. High wing configuration selected to provide adequate nacelle-to-ground clearances.

b. Rotor center-to-center distance designed to give a 12-inch clearance between fuselage and rotor tip in cruise attitude.

c. Rotor-to-wing clearance in the cruise attitude designed to give 12-inch minimum clearance.

d. Fail operative nacelle tilt actuation system configured around a unique ball screw design.

e. Cross-shaft to provide power transfer for one engine out operation.

f. Nacelle tilt axis positioned on wing to give minimum cyclic trim requirement.

g. Cabin and aft compartment (where applicable) are pressurized.

h. Hingeless rotor requiring minimum maintenance.

i. Wing download reduction devices.

4.1.3 Specific Configurations.

4.1.3.1 U. S. Army MAVS (Model 222-1A). - Figure 4-1 is a 3-view drawing of the U. S. Army MAVS aircraft, including inboard profiles of the aircraft and the nacelle. The features of this aircraft include:

a. Bubble canopy designed for optimum visibility for both crewmen

b. North American LW-3 zero-zero escape system installed for both crewmen

c. Control and display console (radar or IR) is installed on the right, in front of observer

d. Easily removable integrated slar antenna enclosed in PRD 49 (radar transparent) bay

e. An automatic data annotation system for complete identification of all sensor imagery

f. Inertial navigation system

g. Infrared or radar displays

h. Horizon-horizon vertical panoramic camera system

i. Radiological monitoring system

j. Aural recorder for transcribing the crew's descriptions of visual observations

4.1.3.2 U. S. Air Force SAR (Model 222-1F). - Figure 4-2 is a 3-view drawing of the Air Force SAR configuration. Its features are:

a. Chin turret with 360-degree fire suppression coverage

b. Nose radar

c. Downward looking night TV scanners

d. Pressurized cockpit and cabin compartments

e. Engine placement in conjunction with cross-shaft provides minimum vulnerability

4.1.3.3 U. S. Navy Sea Control Aircraft (Model 222-1N). - Figure 4-3 is a drawing of the Navy Sea Control Aircraft. It includes the following features:

a. Removable heated fairing for torpedoes

b. Interchangeable weapon pylons

c. Aircraft has small spotting area in folded configuration

d. Powered folding system

e. APS 115 radar with 360-degree scanning coverage

f. Crew stations equipped with teardrop domes for maximum visibility

In addition to carrying two MK46 torpedoes and a total of 2,400 pounds of mission electronics, this aircraft has provisions for 1,000 pounds of additional mission equipment. These may be comprised of combinations of the following:

a. Twelve sonobuoy launch tubes - rechargeable from magazine - 30 sonobuoy capacity

b. Ten Marine smoke marker launch tubes - rechargeable from magazine - 20 marker capacity

c. MAD ASQ-81 equipment housed in tail cone

d. Dipped sonar system

e. TRRAPS and ADD installed at aircraft CG position

4.1.3.4 Civil Off-Shore Oil Rig Support Aircraft (Model 222-1C). - The aircraft chosen for the civil off-shore oil rig support mission is shown in Figure 4-4. It includes:

a. Space for 12 passengers and 2 crew plus baggage

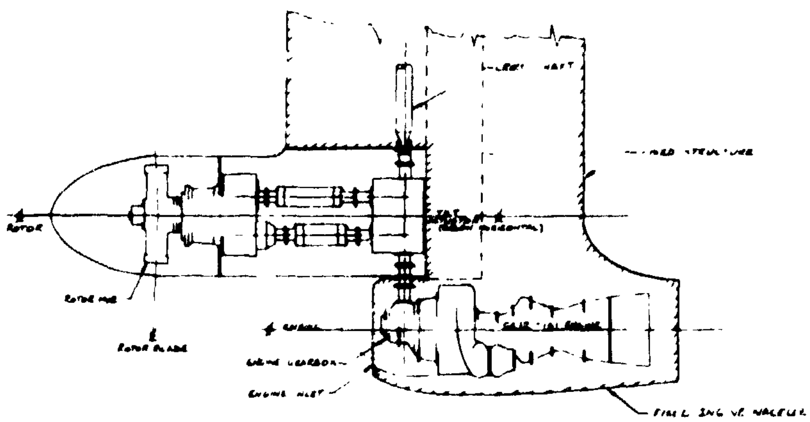
b. Nose radar for zero-zero visibility operation

c. Pressurized cockpit and cabin compartments

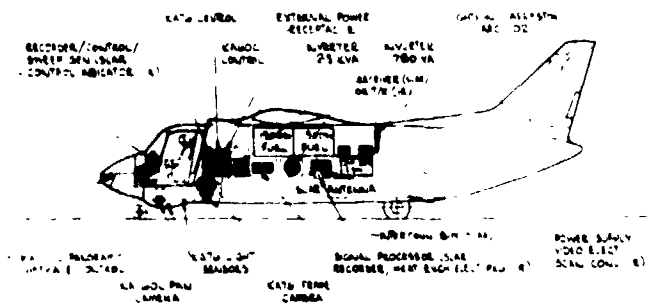
4.1.4 Summary of Characteristics. - Table 4-1 presents a summary of characteristics for each of the four aircraft.

4.2 Materials/Structural Design

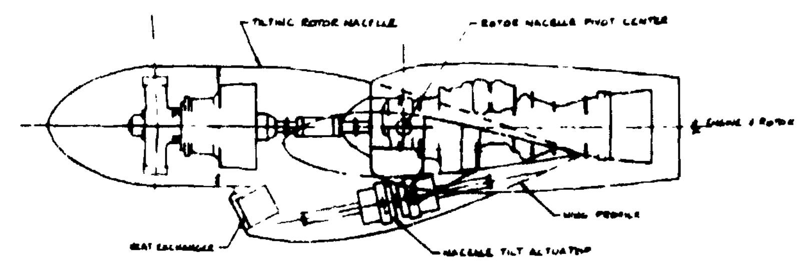
Extensive use of advanced technology materials, e.g., graphite/boron and PRD-49 epoxy composites, and titanium alloys, has been applied to the SAR, Sea Control, and MAVS aircraft primary wing, empennage, and fuselage structure. Preliminary studies indicate that substantial savings in structural weight can be achieved from the use of advanced technology composites by suitably tailoring the strength and stiffness properties to desired values. However, it should be noted that a considerable test and development effort will be required, particularly to develop component allowables and design and analysis



NACELLE PLAN
(SCALE 1:10)



INBOARD PROFILE



NACELLE ELEVATION
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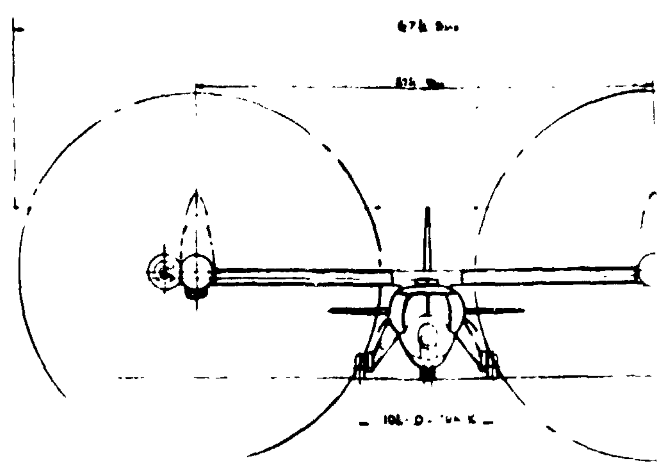

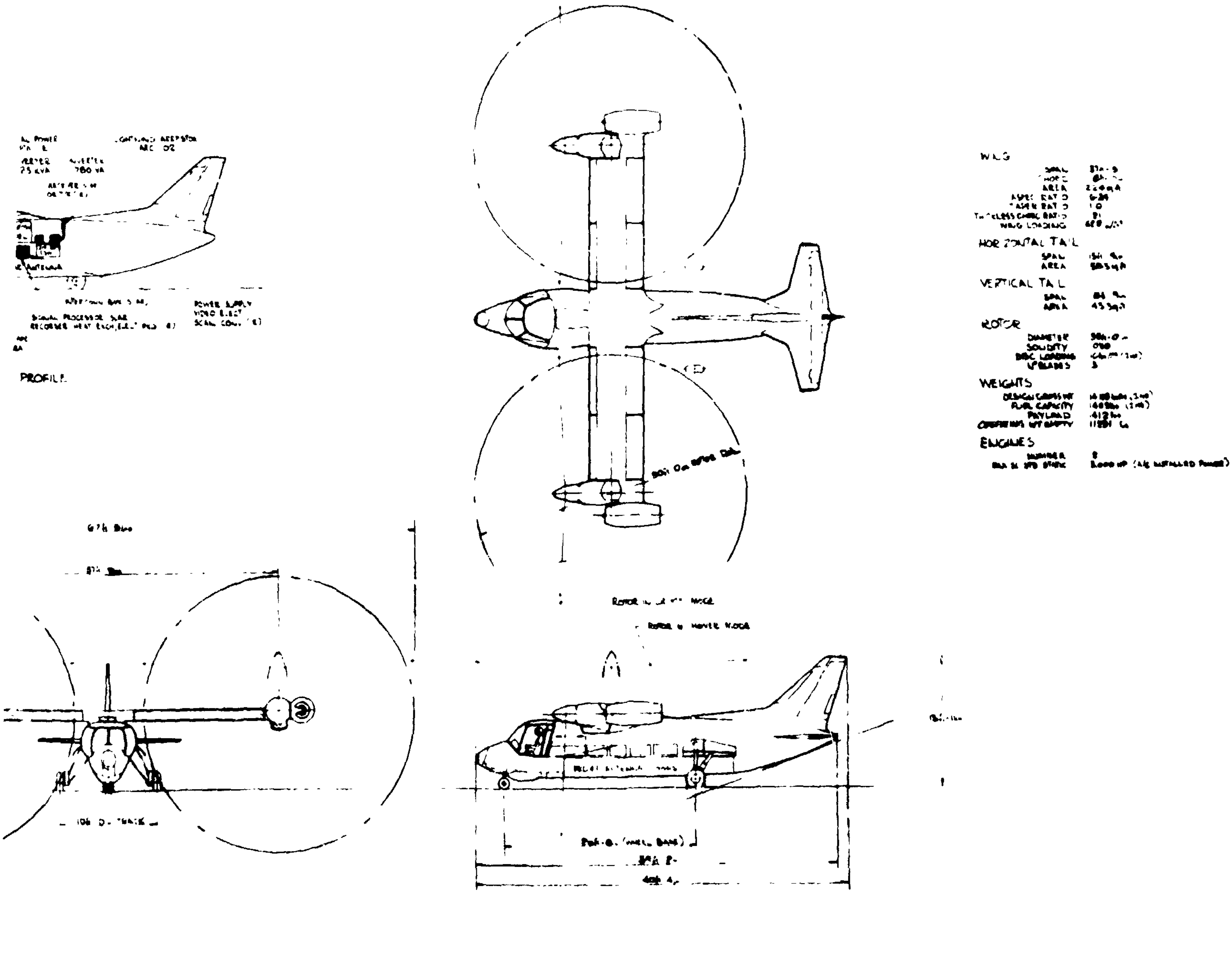

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FIGURE 4-1: MODEL 222-1A - TILT ROTOR SURVEILLANCE AIRCRAFT

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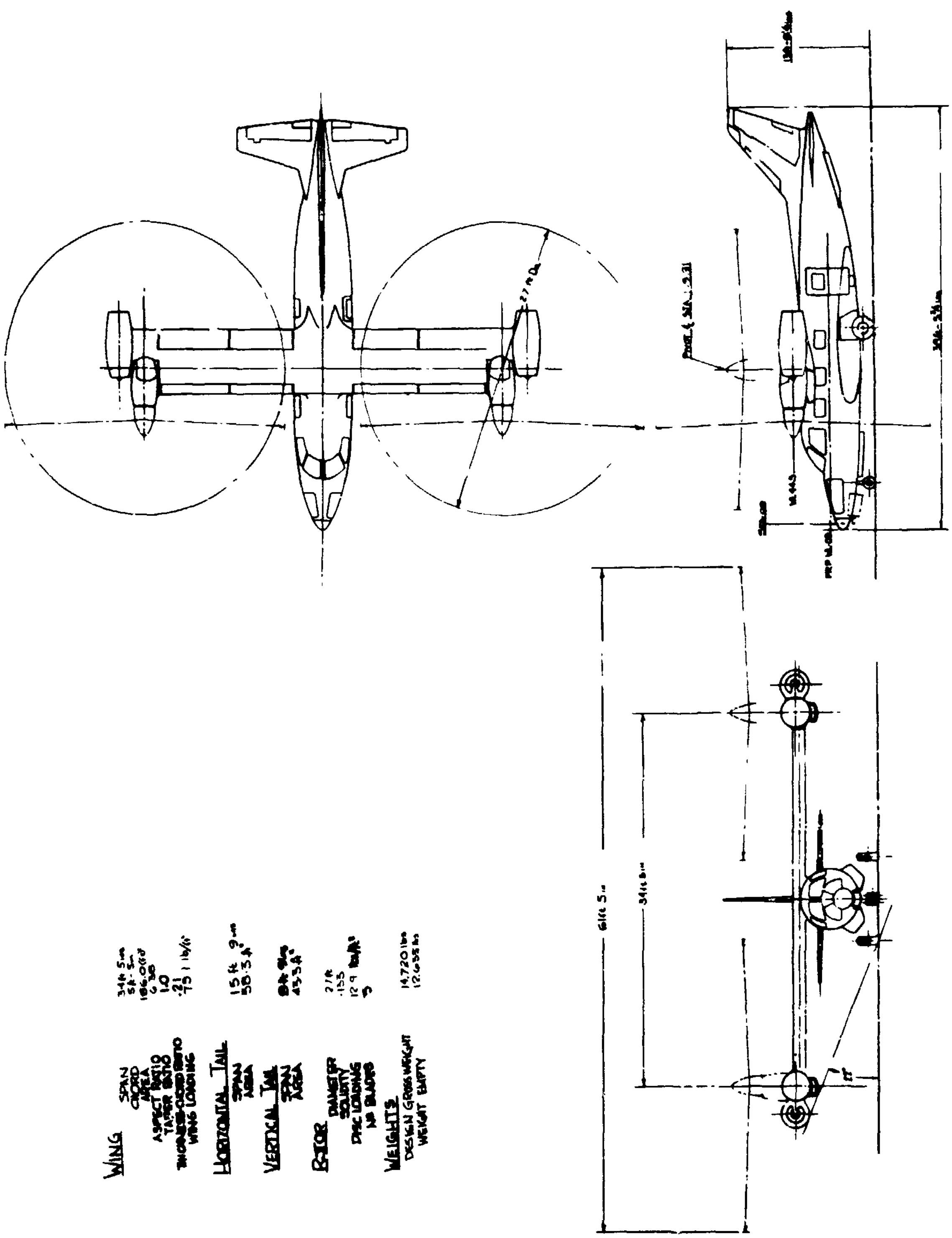


FIGURE 4-2: MODEL 222-1F - AIR FORCE SAR

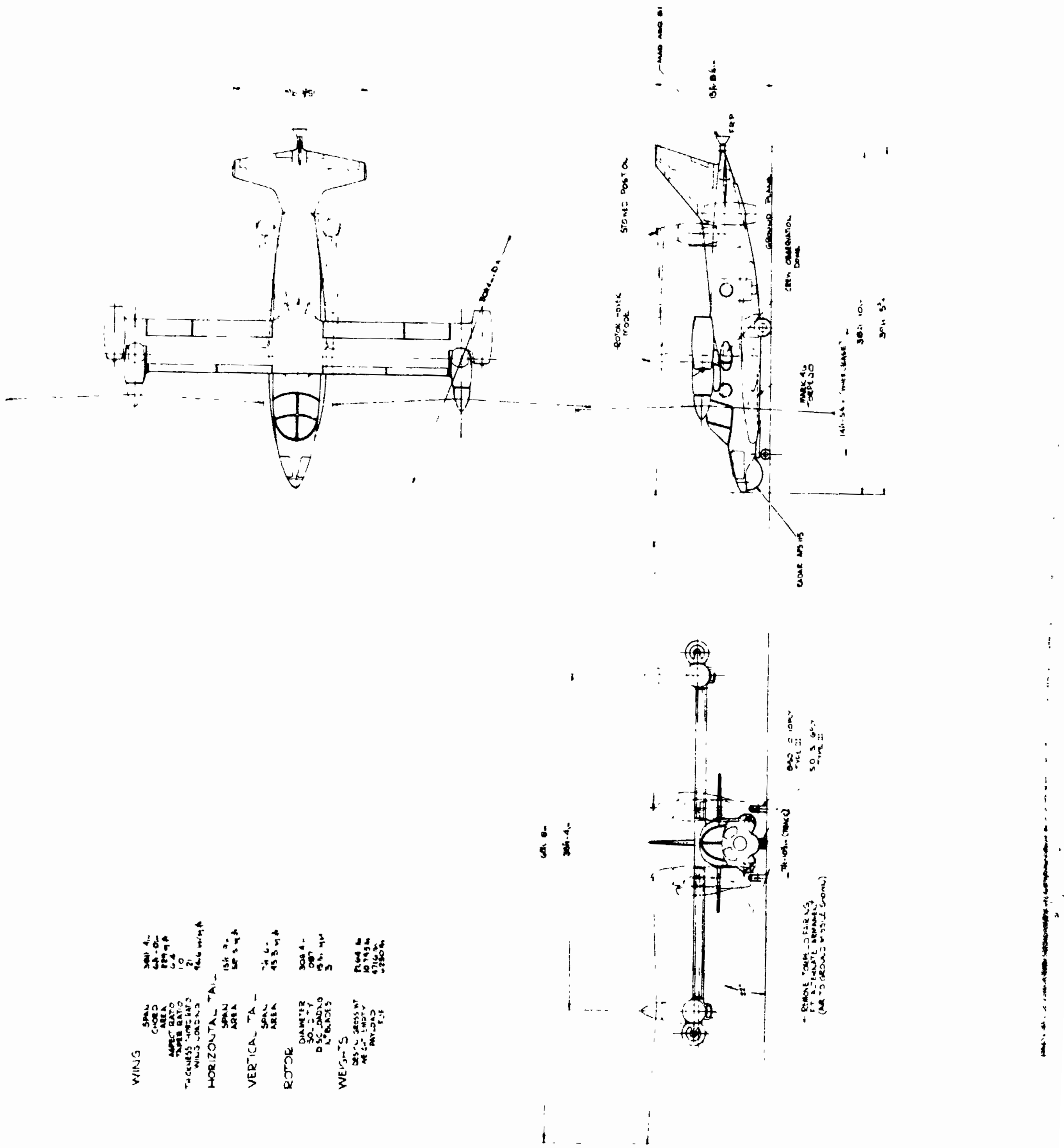


FIGURE 4-3: MODEL 222-1N - NAVY SEA CONTROL

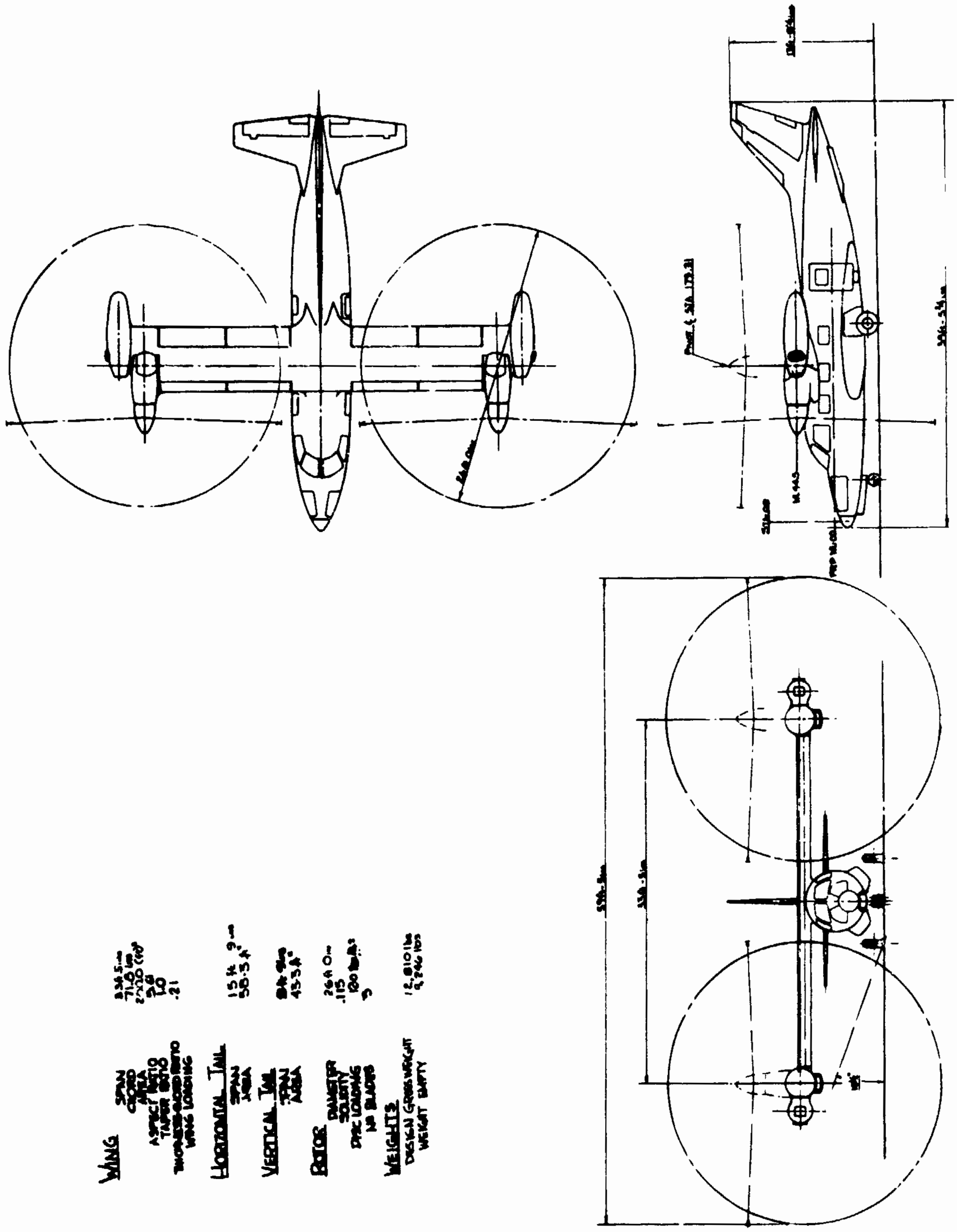


FIGURE 4-4: MODEL 222-1C - CIVIL-OFFSHORE OIL RIG SUPPLY

TABLE 4-1
DESIGN POINT VEHICLES

VEHICLE USER DESIGNATION	MAVS ARMY M222-1A	SAR USAF M222-1F	SEA CONTROL NAVY M222-1N	OFFSHORE OIL CIVIL M222-1C
Power Plant	UTTAS	LYC PLT-27	LYC PLT-27	P&W PT6A-40
Rated Power (SHP)	1500	1950	1950	1150
Aircraft Weights (LBS)				
Design	14,108	16,970	21,641	12,810
Design Ambient Condition (FT/°F)	4000/95	SL/95	SL/90	SL/95
VTOL @ SL/STD	19,675	21 700	22,100	14,400
Operating Weight Empty	11,291	12,400	10,795	9,246
Rotor Data				
C_T/σ//Solidity	.135//.058	.135//.133	.135//.087	.09//.115
Diameter (FT)	30.0	27.0	30.3	26.0
Disc Loading (PSF)				
Design Weight	10.0	14.8	15.0	12.
VTOL Weight SL/STD	13.9	19.0	15.3	13.6
Rotor Tip Speed (Hover/Cruise)				
FEET/SEC	750/525	750/525	750/525	750/525
REV/MIN	478/334	530/371	473/331	551/386
Wing Data				
Area (FT^2)	224.	186.	229.	200.
Span (FT)	37.8	34.4	37.8	33.4
Aspect Ratio	6.24	6.38	6.23	5.61
Thickness (%)	21	21	21	21
Taper Ratio	1	1	1	1
Sweepback (DEG)	0	0	0	0
Wing Loading (PSF)				
Design Weight	62.9	91.1	94.6	64
VTOL Weight	87.8	116.7	96.5	72.0
Fuel Capacity (LBS)	3000	4275	6250	2000
Tail Data				
Vertical				
Area (FT^2)	50.8	43.3	56.9	43.3
Thickness (%)	8	8	8	8
Aspect Ratio	5.72	5.72	5.72	5.72
Horizontal				
Area (FT^2)	60.6	58.3	67.6	58.3
Thickness (%)	10.	10.	10	10.
Aspect Ratio	4.61	4.61	4.61	4.61
Fuselage Data				
Length (FT)	39.17	38.5	38.84	39.06
Width (FT)	5.62	5.62	5.62	5.62
Height (FT)	5.62	5.62	5.62	5.62

methodology for joints and fittings before the full potential offered by the specific properties of these composites can be realized.

Boeing-Vertol is at present engaged in an extensive test program to determine basic properties of advanced composite materials. Further, in the advanced technology components development phase for the HLH program, composite fittings and joints are being designed and tested under both static and fatigue load conditions. Although these tests are aimed at determining the optimum designs for specific load transfer conditions, the data from the test results could be used to establish appropriate design methodology. Design and analysis methods so established will reduce the amount of test and development effort required for the optimum utilization of advanced composites in tilt-rotor aircraft. It is proposed to use conventional materials, e.g., titanium, steel, and aluminum alloys for most dynamic components (except blades) as well as in areas where relevant experience or test data is not available or where other considerations preclude use of composites.

Application of advanced technology composite materials requires, as indicated above, a considerable amount of test and development prior to undertaking detail design. Material selection for the civil aircraft program, therefore, has been limited to the use of conventional materials for reasons of cost as well as a higher degree of conservatism required in a civil program which necessitates a low risk design relying heavily on past experience.

The structural design of the several tilt-rotor aircraft will conform to the appropriate requirements laid down by the relevant military and civil agencies. The structure will be optimized to meet the strength and stiffness criteria at minimum weight utilizing finite element structural analysis computer programs currently available at Boeing-Vertol such as NASTRAN, ASTR, S06, S-47, etc. It is also envisioned that some new computer programs for analysis and optimum design with composite materials will be available in the near future to complement existing composite programs which are basically useful in laminate analysis.

4.3 Weights

Summary weight statements and mass moments of inertia for each of the four Task I study aircraft are included in Tables 4-2 through 4-5. A weights comparison chart comparing all four aircraft is presented in Table 4-6. The weights were developed around the aircraft geometry, design parameters, materials, and structural designs discussed in Sections 4.1 and 4.2 of this report.

TABLE 4-2
SUMMARY WEIGHT STATEMENT
MODEL 222-1A ARMY-MAVS

ENG. H.P. EA. ROTOR DIA.	1500 30'					
ROTOR GROUP	877					
WING GROUP	870					
TAIL GROUP	200					
BODY GROUP	1125					
BASIC						
SECONDARY						
SECOND.-DOORS, ETC.						
ALIGHTING GEAR	616					
FLIGHT CONTROLS	1045					
ENGINE SECTION	400					
PROPULSION GROUP	2271)					
ENGINES(S)			Mass Moments of Inertia			
AIR INDUCTION			About Aircraft C.G.			
EXHAUST SYSTEM			VTOL - 14108#			
COOLING SYSTEM						
LUBRICATING SYSTEM	1022		Ixx(Roll) 52191 SLUG FT²			
FUEL SYSTEM						
ENGINE CONTROLS			Iyy(Pitch) 13200 SLUG FT²			
STARTING SYSTEM						
PROPELLER INST.	-		Izz (Yaw) 57200 SLUG FT²			
*DRIVE SYSTEM	1249					
AUX. POWER PLANT	-					
INSTR. AND NAV.	123					
HYDR. AND PNEU.	132					
ELECTRICAL GROUP	816					
ELECTRONICS GROUP	1164					
ARMAMENT GROUP	162					
FURN. & EQUIP. GROUP	465	3447				
PERSON. ACCOM.						
MISC. EQUIPMENT						
FURNISHINGS						
EMERG. EQUIPMENT						
AIR COND. & DE-ICING	372		Fixed Equip. 3447# is Identical			
PHOTOGRAPHIC	105		to that carried in the			
AUXILIARY GEAR	19		Mohawk OV-1D Surveillance			
			Aircraft			
MFG. VARIATION	89					
WEIGHT EMPTY	10851					
FIXED USEFUL LOAD						
CREW (2)	400	° Slar Equip.				750
TRAPPED LIQUIDS	40	° Photographic Equip.				219
ENGINE OIL		° ECM Wing Pods (2)				227
		° Oxygen Installation				51
FUEL	1405	° Misc				165
Mission Equip.	1412					
PASSENGERS/TROOPS			(Mohawk Equip.)			
GROSS WEIGHT	14108					

TABLE 4-3

SUMMARY WEIGHT STATEMENT

MODEL 222-1F USAF-SAR

ENG. H.P. EA. ROTOR DIA.	1950 27.0'					
ROTOR GROUP	1145					
WING GROUP	757					
TAIL GROUP	200					
BODY GROUP	1186					
BASIC						
SECONDARY						
SECOND.-DOORS, ETC.						
ALIGHTING GEAR	620					
FLIGHT CONTROLS	1267					
ENGINE SECTION	450					
PROPULSION GROUP	(2575)					
ENGINES(S)	620		Mass Moments of Inertia			
AIR INDUCTION			About Aircraft C.G.			
EXHAUST SYSTEM			VTOL - 16970			
COOLING SYSTEM	200					
LUBRICATING SYSTEM			Ixx (Roll) 75750 SLUG FT^2			
FUEL SYSTEM	428					
ENGINE CONTROLS			Iyy (Pitch) 17000 SLUG FT^2			
STARTING SYSTEM						
PROPELLER INST.			Izz (Yaw) 80454 SLUG FT^2			
*DRIVE SYSTEM	1327					
AUX. POWER PLANT	-					
INSTR. AND NAV.	135					
HYDR. AND PNEU.	130					
ELECTRICAL GROUP	800					
ELECTRONICS GROUP	1500					
ARMAMENT GROUP	175	3300				
FURN. & EQUIP. GROUP	350					
PERSON. ACCOM.			Fixed equip. 3300# was estimated			
MISC. EQUIPMENT			from discussions with MAC rescue			
FURNISHINGS			equp. personnel at Scott AFB			
EMERG. EQUIPMENT						
AIR COND. & DE-ICING	100					
PHOTOGRAPHIC						
AUXILIARY GEAR	110		• Forest Penetrator		(1)	22
			• Folding Litters		(2)	36
			• Rescue Litter		(1)	18
MFG. VARIATION			• Troop Seat Instl.		(2)	12
WEIGHT EMPTY	11500		• Life Raft		(1)	30
FIXED USEFUL LOAD			• Rescue Sling		(1)	4
CREW (4)	860		• Flares & Gun		(1)	4
TRAPPED LIQUIDS	40		• Water & Container		(1)	10
ENGINE OIL			• Misc		-	14
Mission Equip.	150					
FUEL	4275					
CARGO						
PASSENGERS/TROOPS						
Gun & Ammo	145 -	5.56 MM	gun 35#,	3000 M's	Ammo 110#	
GROSS WEIGHT	16970					

TABLE 4-4

SUMMARY WEIGHT STATEMENT

MODEL 222-1N NAVY SEA CONTROL

ENG. H.P. EA. ROTOR DIA.	1950 30.3'					
ROTOR GROUP	1322					
WING GROUP	1075					
TAIL GROUP	200					
BODY GROUP	1140					
BASIC						
SECONDARY						
SECOND.-DOORS, ETC.						
ALIGHTING GEAR	650					
FLIGHT CONTROLS	1320					
ENGINE SECTION	450					
PROPULSION GROUP	(2878)					
ENGINES(S)	620		Mass Moments of Inertia			
AIR INDUCTION			about Aircraft C.G.			
EXHAUST SYSTEM			VTOL - 21641#			
COOLING SYSTEM	200					
LUBRICATING SYSTEM			Ixx (Roll) 76750 SLUG FT²			
FUEL SYSTEM	481					
ENGINE CONTROLS			Iyy (Pitch) 17461 SLUG FT²			
STARTING SYSTEM						
PROPELLER INST			Izz (Yaw) 81454 SLUG FT²			
*DRIVE SYSTEM	1577					
AUX. POWER PLANT						
INSTR. AND NAV.	100					
HYDR. AND PNEU.	125					
ELECTRICAL GROUP	350					
ELECTRONICS GROUP						
ARMAMENT GROUP						
FURN. & EQUIP. GROUP	340	1000#				
PERSON. ACCOM.						
MISC. EQUIPMENT			Fixed equip. 1000# is identical			
FURNISHINGS			to that carried on the BO-105 LAMPS			
EMERG. EQUIPMENT			aircraft			
AIR COND. & DE-ICING	75					
PHOTOGRAPHIC						
AUXILIARY GEAR	10					
			° Forest Penetrator		(1)	22
			° Folding Litters		(2)	36
MFG. VARIATION			° Rescue Litter		(1)	18
WEIGHT EMPTY	10035		° Troop Seat Instl.		(2)	12
FIXED USEFUL LOAD			° Life Raft		(1)	30
CREW (4)	720		° Rescue Sling		(1)	4
TRAPPED LIQUIDS	40		° Flares & Gun		(1)	4
ENGINE OIL			° Water + Cont.		(1)	10
	136					
FUEL	6250					
	1000	- Expendable Stores				
	1060	- Mark 46 Torpedoes (2)				
Total Electronics	2400					
GROSS WEIGHT	21641					

TABLE 4-5

SUMMARY WEIGHT STATEMENT

M222-1C CIVIL OFFSHORE OIL

ENG.H.P. EA. ROTOR DIA.	1150 26.0'					
ROTOR GROUP	1100					
WING GROUP	840					
TAIL GROUP	213					
BODY GROUP	1211					
BASIC						
SECONDARY						
SECOND.-DOORS, ETC.						
ALIGHTING GEAR	590					
FLIGHT CONTROLS	1143					
ENGINE SECTION	400					
PROPULSION GROUP	(2149)					
ENGINES(S)	642		Mass Moments of Inertia			
AIR INDUCTION			about Aircraft C.G.			
EXHAUST SYSTEM			VTOL - 12810#			
COOLING SYSTEM	200					
LUBRICATING SYSTEM			Ixx (Roll) 50130 SLUG FT^2			
FUEL SYSTEM	200					
ENGINE CONTROLS			Iyy (Pitch) 13230 SLUG FT^2			
STARTING SYSTEM						
PROPELLER INST.			Izz (Yaw) 56659 SLUG FT^2			
*DRIVE SYSTEM	1107					
AUX. POWER PLANT	-					
INSTR. AND NAV.	108					
HYDR. AND PNEU.	-					
ELECTRICAL GROUP	305					
ELECTRONICS GROUP	230					
ARMAMENT GROUP						
FURN. & EQUIP. GROUP	439	1200				
PERSON. ACCOM.						
MISC. EQUIPMENT						
FURNISHINGS						
EMERG. EQUIPMENT						
AIR COND. & DE-ICING	108					
PHOTOGRAPHIC			Fixed equip. 1200# resulted from			
AUXILIARY GEAR	10		studies done at Boeing-Vertol for			
			a demonstrator aircraft			
MFG. VARIATION						
WEIGHT EMPTY	8846					
FIXED USEFUL LOAD			Rescue litter			
CREW (2)	360		Life Rafts			
TRAPPED LIQUIDS	40		Rescue Sling			
ENGINE OIL			Flares & Gun			
Mission Equip.	136		Water & Container			
FUEL	1268					
CARGO						
PASSENGERS (12)	2160					
GROSS WEIGHT	12810					

TABLE 4-6
SUMMARY WEIGHT STATEMENT[illegible]
TILT ROTOR CONFIGURATION MODEL 2[illegible]

	ARMY 222-1A		NAVY 222-1N	USAF 221-1F		CIVIL 222-1C
ROTOR GROUP	877		1322	1145		1100
WING GROUP	870		1075	757		840
TAIL GROUP	200		200	200		213
BODY GROUP	1125		1140	1186		1211
BASIC						
SECONDARY						
SECOND.-DOORS, ETC.						
ALIGHTING GEAR	616		650	620		590
FLIGHT CONTROLS	1045		1320	1267		114[illegible]
ENGINE SECTION	400		450	450		
PROPULSION GROUP	(2271)		(2878)	(2575)		(2149)
ENGINES(S)			620	620		642
AIR INDUCTION						
EXHAUST SYSTEM						
COOLING SYSTEM	1022		200	200		200
LUBRICATING SYSTEM						
FUEL SYSTEM			481	428		200
ENGINE CONTROLS						
STARTING SYSTEM						
PROPELLER INST.						
*DRIVE SYSTEM	1249		1577	1327		1107
AUX. POWER PLANT	–		–	–		–
INSTR. AND NAV.	12[illegible]		100	135		108
HYDR. AND PNEU.	132		125	130		–
ELECTRICAL GROUP	816		350	800		305
ELECTRONICS GROUP	1164		In useful load	1500		230
ARMAMENT GROUP	162	3447	–	175	1200	–
FURN. & EQUIP. GROUP	465		340	350		439
PERSON. ACCOM.						
MISC. EQUIPMENT		1000			3300	
FURNISHINGS						
EMERG. EQUIPMENT						
AIR COND. & DE-ICING	372		75	100		108
PHOTOGRAPHIC	105					
AUXILIARY GEAR	19		10	110		10
MFG. VARIATION	89					
WEIGHT EMPTY	10851		10035	11500		8846
FIXED USEFUL LOAD						
CREW	(2)400		(4)720	(4) 860		(2)360
TRAPPED LIQUIDS	40		40	40		40
ENGINE OIL						
Mission Equip			136	150		136
FUEL	1405		6250	4275		1268
CARGO	1412		1000			
PASSENGERS			1060			2160
(See Detail Sheets)			2400	145		
GROSS WEIGHT	14108		21641	16970		12810

Weights were determined using VASCOMP (V/STOL Aircraft Sizing and Performance Computer Program), Reference 2. The weights segment of the program, Reference 3, contains detailed statistical weight trend equations which compute the weights of the structure, flight controls, and propulsion groups. Fixed equipment (auxiliary power plant through auxiliary gear groups on weight statements), fixed useful load and payload are weight input values. Examples of the trends are included in the weights section of Volume II.

VASCOMP computes the weights of the aircraft wing, flight controls, engine section, engine installation, fuel system, rotor/prop assembly, and drive system. The weights of fuselage, empennage, engines, and landing gear were inputs to the program.

All configuration weights, with the exception of the civil aircraft, utilize advance composite materials in the structure (wing, fuselage, engine section) and rotor assembly. Advanced technology has been considered in the drive system (higher Hertz stress levels in the gearing) for all configurations except the civil aircraft. Weight savings of between 15 to 20 percent of the individual groups are realized through the use of the advanced materials and advanced technology.

The civil aircraft assumes current technology and utilizes advanced composite material only in the rotor/propeller assembly.

The Navy Sea Control aircraft includes automatic rotor blade folding and wing folding. The weight penalties associated with these features are 200 and 170 pounds, respectively.

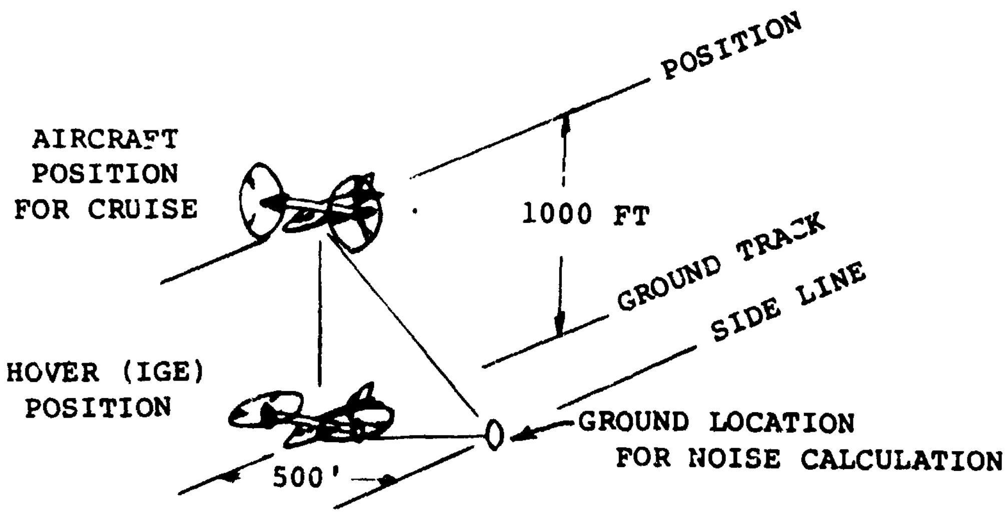
The weights of the fixed equipment, mission, and rescue equipment were determined for each aircraft on an independent basis. The approach is described on the summary weight sheets.

4.4 Noise

The tilt-rotor aircraft is one of the quietest configurations to be developed for VTOL flight. The absence of high noise devices such as direct lift engines, turbojets, and anti-torque propellers leaves the shaft-driven turbine and the rotor as the primary noise sources. In this respect, the hover and low-speed noise are very similar to that of a non-overlapped tandem-rotor helicopter. The capability to design and build a low noise configuration was demosntrated by the Boeing-Vertol Model 347, which displays low external noise level and an absence of impulsive noise components generally referred to as rotor "slap" or "bang." Noise level of the tilt-rotor aircraft is projected to be less than that of the Model 347. These advantages are also manifest in the low aural detection signature and resulting warning times displayed by the tilt-rotor aircraft which has been compared with similar characteristics of the OV-1 Mohawk. With both aircraft at 200 knots true airspeed at 1,000 feet altitude, an observer located 500 feet from the flight path would first hear each approaching aircraft at the distances noted below, under low ambient noise levels, typical winter atmospheric conditions, and a ground cover of trees.

	Tilt Rotor	OV-1 Mohawk
Aural Detection Range (ft)	14,600	43,900
Warning Time (sec)	43	130

The noise levels of both aircraft have been calculated from theoretical estimating procedures (References 6 and 7). For each aircraft, only the noise of the rotors



was considered to contribute to the far field acoustic radiation. Powerplant noise and other sources were assumed to be acoustically treated to the extent that they did not enter into the aircraft noise signature.

Perceived noise levels (PNL) were calculated for each aircraft in hover and cruise. For hover, the aircraft was operating in ground effect and the observer location was 500 feet from the rotor centerline. For cruise, the aircraft was at an altitude of 1,000 feet and the observer was at a 500-foot-sideline distance from the flight path.

Table 4-7 presents the PNL for each configuration along with the input variables which, in large part, determine far field sound levels. Note that in hover, thrust and perceived noise generally are ranked together. In cruise, the perceived noise tends to be ranked with thrust also, although a reduction in blade chord and thickness for the MAVS aircraft has increased the fundamental frequency of vortex noise and thus the perceived noise for that particular configuration. All tilt-rotor aircraft are well within the noise guidelines of 95 PNdB at a 500-foot distance.

TABLE 4-7

	U.S. ARMY MAVS	U.S. AIR FORCE SAR	U.S. NAVY SEA CONTROL	CIVIL OFF-SHORE OIL RIG SUPPORT
HOVER				
Thrust/Rotor Pounds	7760	8100	11903	7046
Rotor rpm	479	530	473	551
Perceived Noise PNdB	90	88	93	87
CRUISE AT NORMAL POWER				
Airspeed Knots	300	325	310	283
Thrust/Rotor Pounds	1140	1330	1350	995
Perceived Noise PNdB	69	66	67	64

4.5 Stability and Control

4.5.1 Introduction. - All of the aircraft configurations of this study are similar, and the NASA/Army Flight Research Aircraft (described in Volume II) and the Navy Sea Control Aircraft represent the extremes of gross weights, wing loadings, and inertias. Therefore, if the characteristics of these air-aircraft are acceptable, those of the other aircraft are anticipated to be acceptable. It is to be noted that although the NASA/Army demonstrator aircraft, the Task II vehicle, is not called out specifically as one of the four design-point vehicles discussed in this volume, its configuration and dimensions are the same as those of the Civil Off-Shore Oil aircraft and the design weight is comparable -- 12,000 pounds compared to 12,810 pounds.

4.5.2 Flying Qualities Criteria. - The requirements of Military Specification MIL-F-83300 have been used as the primary criteria for tilt-rotor aircraft flying qualities in the hover and transition regimes through conversion speed, V_{CON}.

AGARD Report-577 criteria were reviewed and the requirements are substantially in agreement with those of MIL-F-83300. The AGARD report provides expanded detail regarding specific criteria and better definition of requirements concerning control configurations. Guidance provided in AGARD R-577 will be utilized in the design stages.

The criteria of MIL-F-8785B (ASG) are applicable for speeds above V_{CON}.

4.5.2.1 Control Criteria - Low Speed. - Maneuver response requirements for the tilt-rotor aircraft in the hover and transition flight regimes were determined based on a review of the following applicable data:

a. MIL-F-83300, Flying Qualities of Piloted V/STOL Aircraft, dated 31 December 1970.

b. NASA TN D-5594, Airworthiness Considerations for STOL Aircraft, dated January 1970.

c. Boeing data gathered in support of tilt wing/tilt rotor controllability studies.

The data suggests that the minimum angular acceleration responses in hover to full, single axis control deflections should be of the following magnitudes:

Axis	Angular Acceleration (rad/sec^2)
Pitch	0.6
Roll	1.0
Yaw	0.5

A reasonable reduction in these levels can be tolerated through the transitional regime as the rotor controls are phased out and the aerodynamic control surfaces (which are always working) become effective. The recommended minimum angular accelerations about each axis through transition are shown in Figure 4-5 (see Volume II for rationale). These minima are slightly greater than the recommendations in NASA TN D-5594 for the STOL speed range (near end of transition).

4.5.3 Control Configuration. - Control of the tilt-rotor aircraft described herein is accomplished by utilization of rotor longitudinal cyclic, differential cyclic, and differential collective control for hover and operation of conventional aircraft control surfaces for primary control in cruise. The

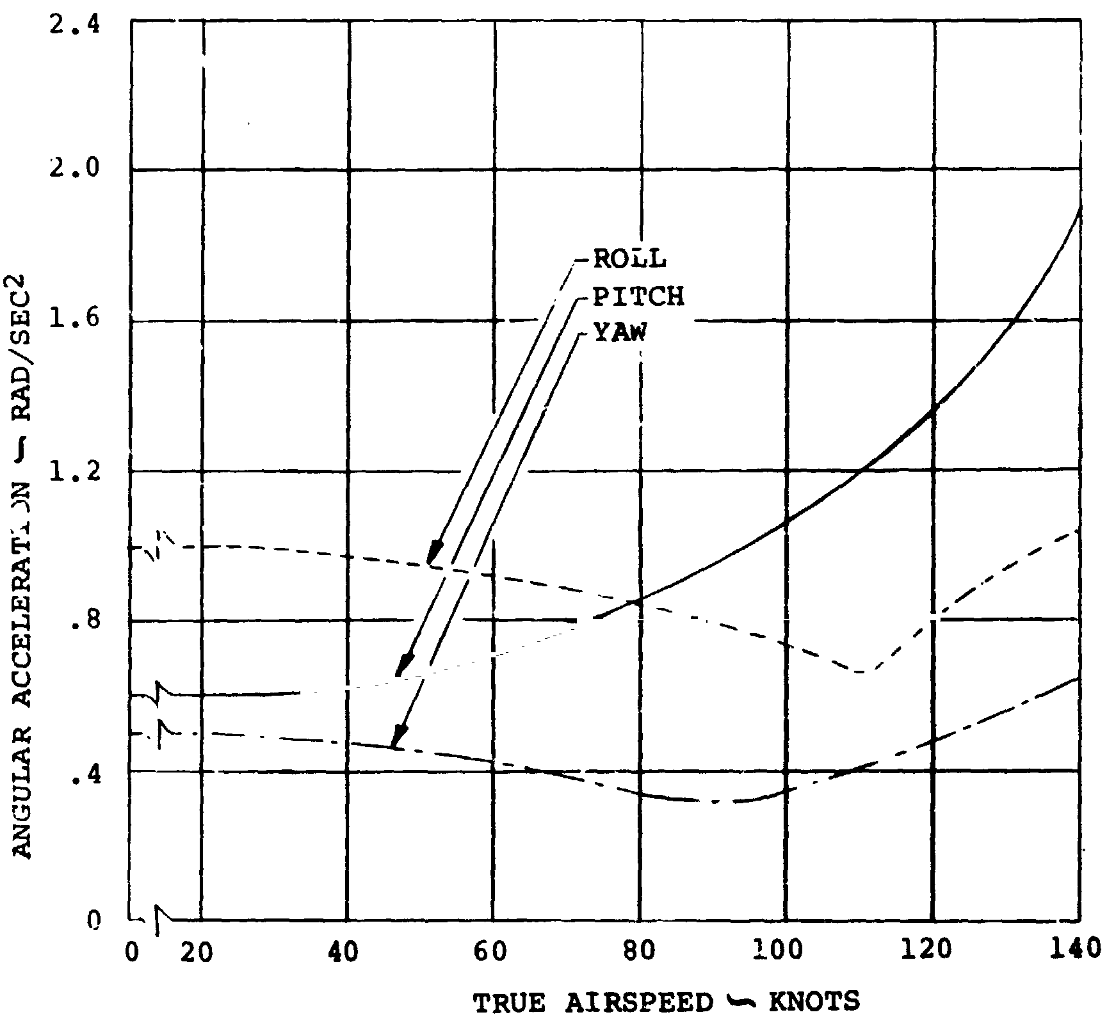


FIGURE 4-5: MINIMUM RECOMMENDED CONTROL POWER

E

aircraft surfaces are operated at all times, but their effectiveness is very low at low transition speeds and near zero in hover. Control in transition is accomplished by using a mixture of rotor and aircraft controls which are automatically proportioned as a function of nacelle incidence.

The cyclic control inputs will be phased so as to result in maximum inplane forces oriented along the rotor disc X-axis. This is desirable to provide maximum yawing moment on the aircraft where differential cyclic is commanded through rudder pedal application at low speed. This results in a small reduction in the pitching moment attainable per degree of longitudinal cyclic control.

The rudder and elevator controls are conventional. Roll control in cruise is accomplished by use of the outboard semispan of the flap operating downwards in conjunction with use of a spoiler operating upwards on the opposite wing. This permits use of a more effective single-slotted, full-span flap for low-speed loiter in the cruise configuration and permits better aerodynamic-tailoring, i.e., minimizing of the resulting yawing moment due to roll.

Control requirements for maneuver and control scheduling are discussed in the following paragraphs.

4.5.4 Cyclic Control Required in Hover.

4.5.4.1 Maneuver. - The simultaneous and differential longitudinal cyclic control required to meet the hovering pitch and yaw angular acceleration requirements is shown in Figure 4-6. The alleviation of required differential cyclic associated with wing flexibility and nacelle tilt is also indicated. Control requirements are indicated for both the NASA Research Aircraft, 12,000 pounds, and the Navy Sea Control Aircraft, 21,641 pounds. It is to be noted that the heavier aircraft requires slightly less cyclic for both pitch and yaw control, both requirements being easily attainable. The reasons for the trend are: (1) as the disc size increases, the longitudinal control power per degree cyclic increases more rapidly than the increase in pitch inertia, and (2) the increase in inplane force along the rotor disc X-axis as a function of the increased thrust, required at the heavier weights in hover, when multiplied by the moment arm from the nacelle to the center of gravity of the aircraft results in yawing moment increasing more rapidly than the yaw inertia.

4.5.4.2 Trim. - The amount of cyclic required to trim the aircraft longitudinally in hover with the center of gravity offset from the pivot is relatively small, amounting to approximately 0.10 degree cyclic per percent MAC cg offset for the 12,000-pound aircraft and 0.11 degree per percent MAC cg

NOTES:
1. CYCLIC PHASED FOR MAX IN-PLANE X-FORCE
2. THRUST VECTORING: 1.0° TILT PER 1.0° CYCLIC

RESEARCH AIRCRAFT:
- ROTOR ON RIGID WING
- ROTOR ON FLEXIBLE WING
- ROTOR WITH THRUST VECTORING, RIGID WING
- ROTOR WITH THRUST VECTORING, FLEXIBLE WING

SEA CONTROL AIRCRAFT:
- ROTOR ON RIGID WING
- ROTOR W/THRUST VECTORING, RIGID WING

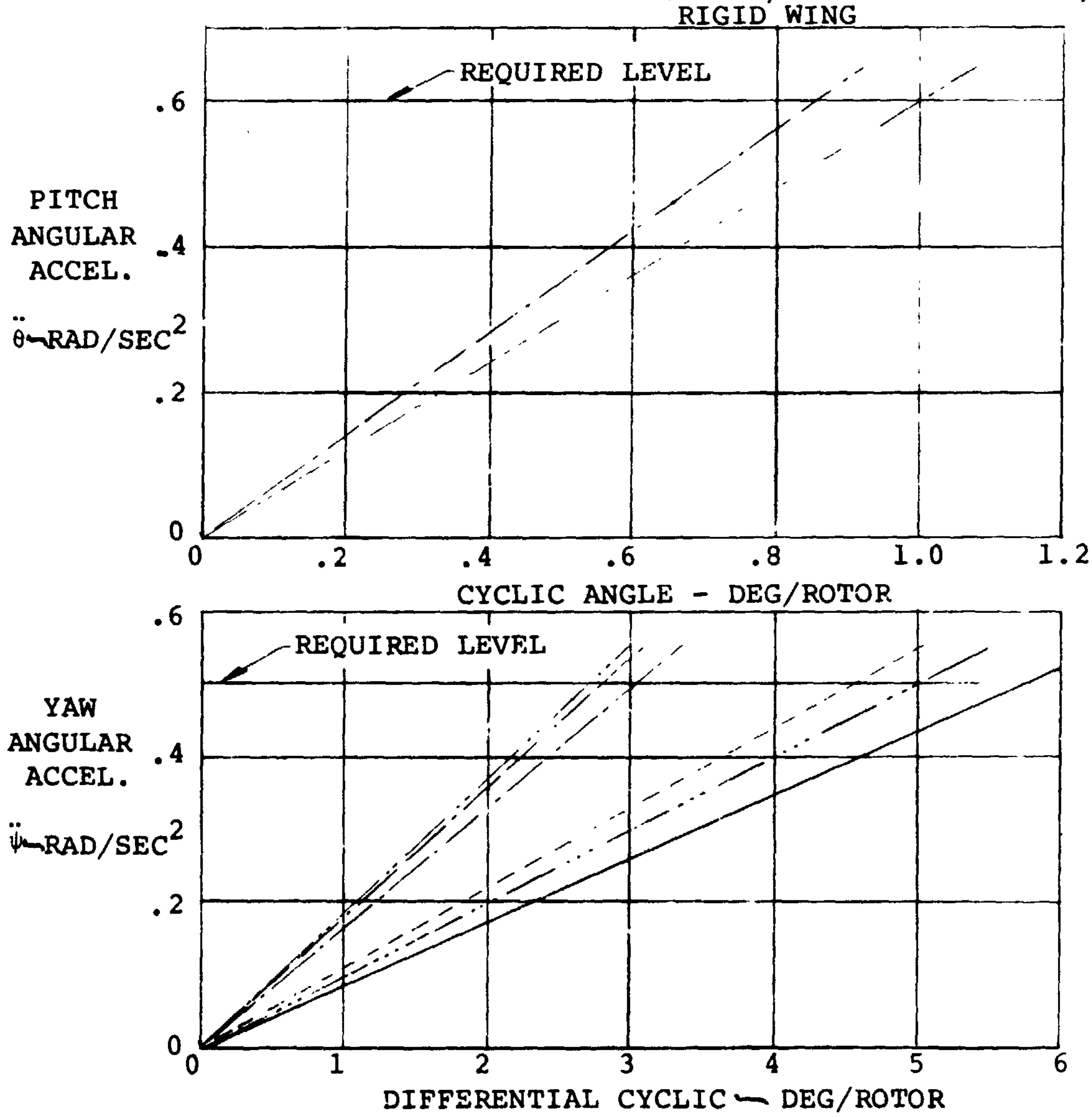


FIGURE 4-6: HOVER CONTROLLABILITY

offset for the 21,641-pound aircraft. The pivot is located such that for the condition of nacelle vertical, hover mode, the aircraft center of gravity will normally be within approximately ± 5 percent MAC of the pivot.

4.5.5 Control Scheduling.

4.5.5.1 Hover. - During operation in the hover mode, pitch control will be accomplished by application of rotor longitudinal cyclic control in response to stick command. Roll control results from application of differential collective pitch in response to lateral stick command. Yaw is accomplished by application of differential cyclic control and differential nacelle tilt in response to rudder pedal input. The differential nacelle tilt results from structural deflection of the nacelle/pivot mechanism and yields approximately one degree nacelle tilt per degree cyclic command in hover.

4.5.5.2 Transition.

a. General - The controls are "scheduled" in transition to phase out the rotor cyclic and collective response and differential nacelle tilt as nacelle incidence is decreased, with increasing speed, and as the aerodynamic control surfaces, which are always working, become more effective. The controls are also scheduled so as to minimize response coupling of the aircraft about the roll and yaw axes, i.e., to result in minimum yaw response for a pure roll input and vice versa.

b. Pitch Control - Pitch control is accomplished by combined rotor longitudinal cyclic and elevator control surface deflection in response to longitudinal stick command with the cyclic control "phased out" as nacelle incidence is decreased.

c. Yaw Control - Yaw controls are scheduled such that at low speeds, differential cyclic and differential nacelle tilt are the prime contributors to yaw. As speed increases and nacelle incidence is decreased, the amount of differential cyclic and nacelle tilt per inch of rudder pedal travel is decreased, and differential collective pitch is phased in to yeild uncoupled yaw responses to rudder pedal deflection. As nacelle incidence is decreased to zero for cruise flight, the rotor differential cyclic and collective pitch inputs and differential nacelle tilt are phased out completely and rudder deflection alone provides adequate yaw response to pedal input.

d. Roll Control - Roll control is accomplished at low transition speeds by differential collective pitch inputs to the rotors. As nacelle incidence is decreased, differential collective response to lateral stick inputs is reduced, and differential cyclic and differential nacelle tilt are phased in to minimize yaw coupling. As nacelle incidence is reduced further, the cyclic, collective, and nacelle tilt controls are phased out and roll results from only the aircraft aerodynamic controls, outboard flaperon deflection down on one wing, and spoiler deflection upward on the opposite wing.

Typical control phasing is illustrated by Figures 4-7 and 4-8. The magnitudes of the control inputs illustrated represent the amount of control required to meet the transition control criteria specified in Figure 4-5 during operation of the 12,000-pound NASA flight research aircraft in accelerating transition at constant altitude with near-level fuselage attitude. These control travels provide control moments in excess of the minima of Figure 4-5 during unaccelerated transition, i.e., constant speed.

4.5.6 Longitudinal Trim in Transition. - There are various "possible" ways to operate a tilt-rotor aircraft in and through the transition flight regime. This is true because of the number of variables associated with control of the aircraft such as thrust, nacelle incidence, tail incidence/elevator deflection, flap deflection, and fuselage attitude. It is desirable, however, because of the many variables and the relatively short time required to accomplish an accelerating or decelerating transition to automate as many of the variables as possible to decrease the pilot's workload. Scheduling, or automation, of the various parameters can be made a function of velocity, dynamic pressure, power lever position, nacelle incidence, or other variables. It is desirable in order to simplify mechanization and improve system reliability to reduce the scheduling to functions of a minimum number of variables of parameters. Therefore, investigations to date have been directed toward scheduling of flaps, cyclic, tail surface trim, etc. with nacelle incidence and leaving control of nacelle incidence to the pilot. The following ground rules were established to define the nominal transition schedules for transition.

a. Hub moments shall be essentially zero.

b. There shall be a smooth variation of the trim parameters between hover and end-transition speed.

c. Flap deflection will be scheduled to minimize thrust required in transition consistent with maintaining comfortable (i.e., small nose up or down) fuselage attitude.

NOTES:

1. AIRCRAFT SURFACE CONTROLS OPERATIVE AT ALL TIMES AND DIFFERENTIAL NACELLE TILT OF 1.0 DEGREE PER DEGREE CYCLIC ASSUMED.
2. CONTROL FOR MAX. ROLL COMMAND.

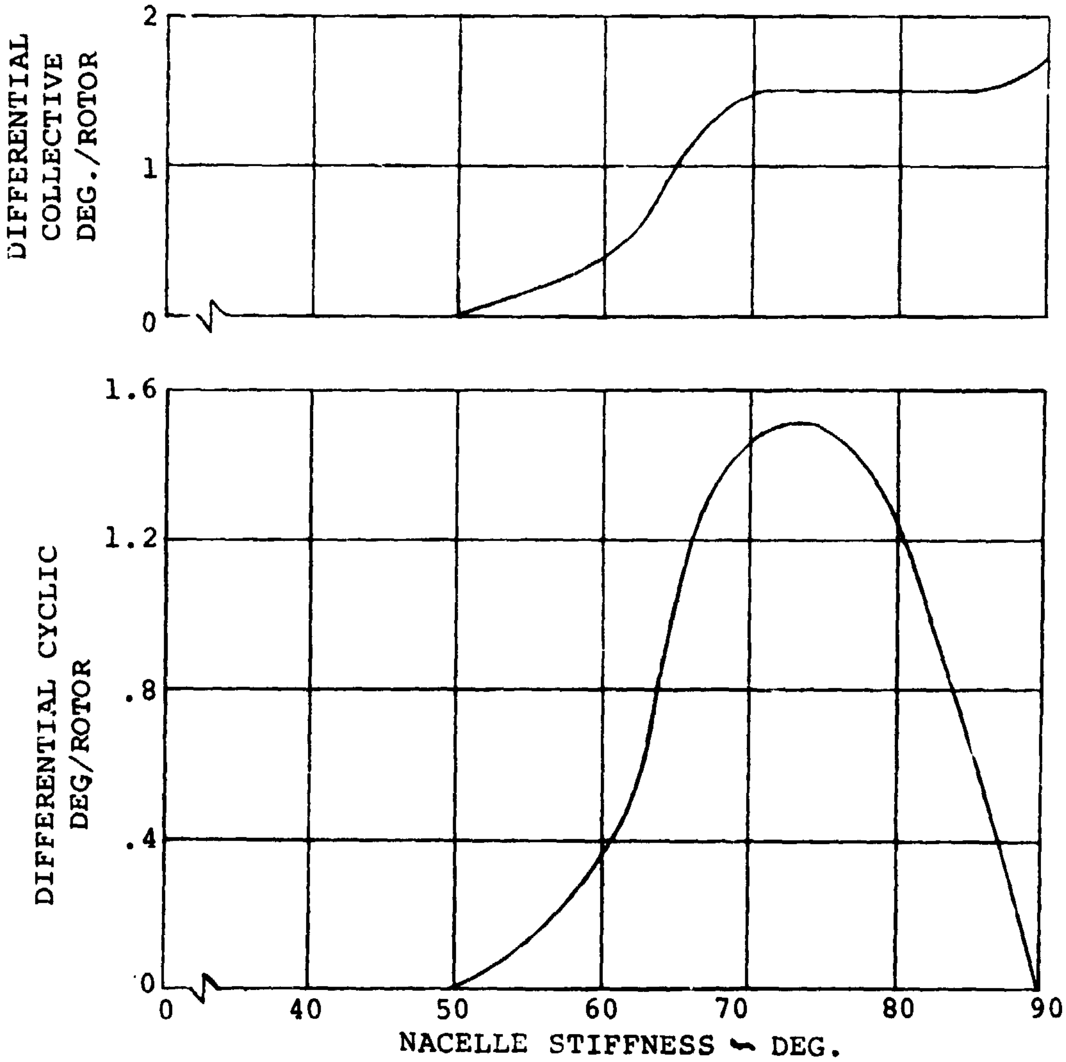


FIGURE 4-7: ROLL CONTROL PHASING TO MINIMIZE INITIAL YAW COUPLING

NOTES:
1. AIRCRAFT SURFACE CONTROLS OPERATIVE AT ALL TIMES AND DIFFERENTIAL NACELLE TILT OF 1.0 DEGREE PER DEGREE CYCLIC IS ASSUMED.
2. DIFFERENTIAL COLLECTIVE SHOWN PER DEGREE YAW CYCLIC.
3. DIFFERENTIAL CYCLIC SHOWN FOR MAX. YAW COMMAND.

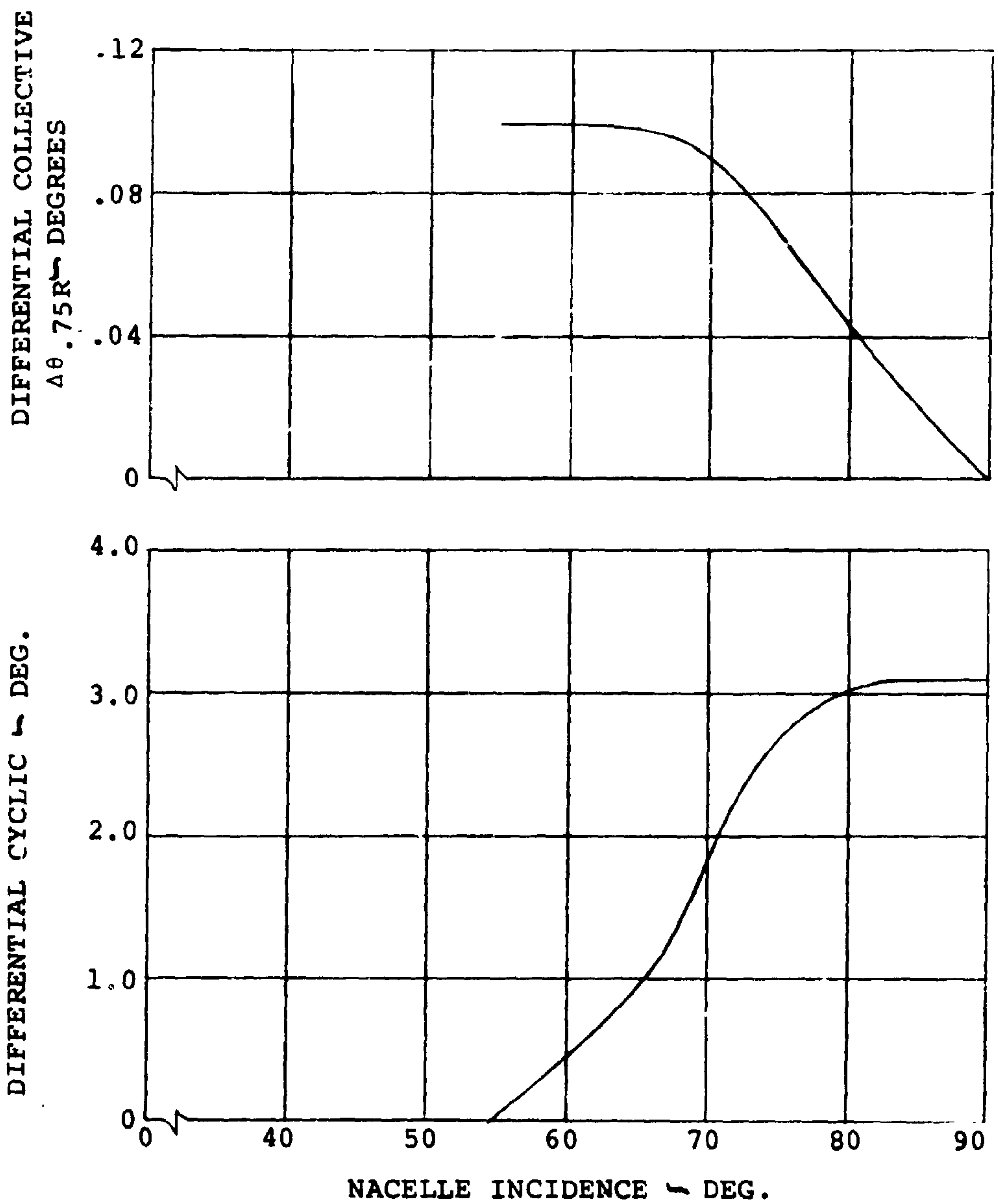


FIGURE 4-8: YAW CONTROL PHASING TO MINIMIZE INITIAL ROLL COUPLING

d. The trim conditions at the end of transition shall be coincident with the trim required in the aircraft configurations, i.e., cruise configuration.

The resultant variations for the trim parameters with speed are shown in Figures 4-9 and 4-10 for the research aircraft and Navy Sea Control configurations, respectively. It is evident, from the trends illustrated, that reasonable schedules of nacelle incidence with airspeed and tail incidence and flap deflection with nacelle incidence can be achieved. For the 12,000-pound-weight research aircraft, hub moments of less than ±800 ft-lb are achieved through transition. The maximum cyclic control angles required to trim are 3.9 degrees of longitudinal cyclic at 120 KTAS and 1.5 degrees of lateral cyclic at 110 KTAS. For the 21,641-pound-weight Sea Control aircraft, hub moments are between +3,360 and -4,140 ft-lb, and the maximum cyclic requirements are 4.38 degrees longitudinally at 120 KTAS and 1.54 degrees laterally at 110 KTAS. A smooth variation of fuselage attitude, of acceptable magnitude, from hover to end-transition speed is achieved, and wing and tail angles of attack are well within limits. These transitions were calculated using the same tail incidence versus nacelle incidence and nacelle incidence versus velocity schedules. Minor variation of the tail incidence versus nacelle incidence and nacelle incidence versus velocity schedules would permit transition with zero hub moment over nearly the entire transition speed range.

The effect of the minor differences in configuration and differences in weight of the two aircraft can easily be compensated for by minor changes in scheduling of the transition control parameters.

4.5.7 <u>Cruise Trim</u>. - Trim conditions in the airplane configuration cruise mode were calculated for the research aircraft and Sea Control aircraft. Typical examples of the cruise trim requirements are illustrated in Figures 4-11 and 4-12 for the 12,000 and 21,641-pound aircraft, respectively. The longitudinal and lateral cyclic control required to zero the rotor hub moments have been calculated as a function of velocity for each of the two aircraft. A feedback system is being developed to automatically apply cyclic control to maintain hub moments near zero, and the cyclic control indicated on these figures is that which will be applied by the feedback system. This system is described in more detail in Volume II.

The aircraft is indicated to be statically stable, with reference to trim versus velocity, through the cruise range at the nominal gross weight. The heavy weight aircraft indicates a mild trim instability between 150 and 200 KTAS as evidenced by the reversal in the trim tail incidence angle variation with speed. This instability can be corrected by several means such as programmed trim versus velocity.

NOTES:
1. GW = 12,000 LBS
2. CG = 28% NACELLE DN
FS 214.2
3. V_{TIP} = 750 FPS

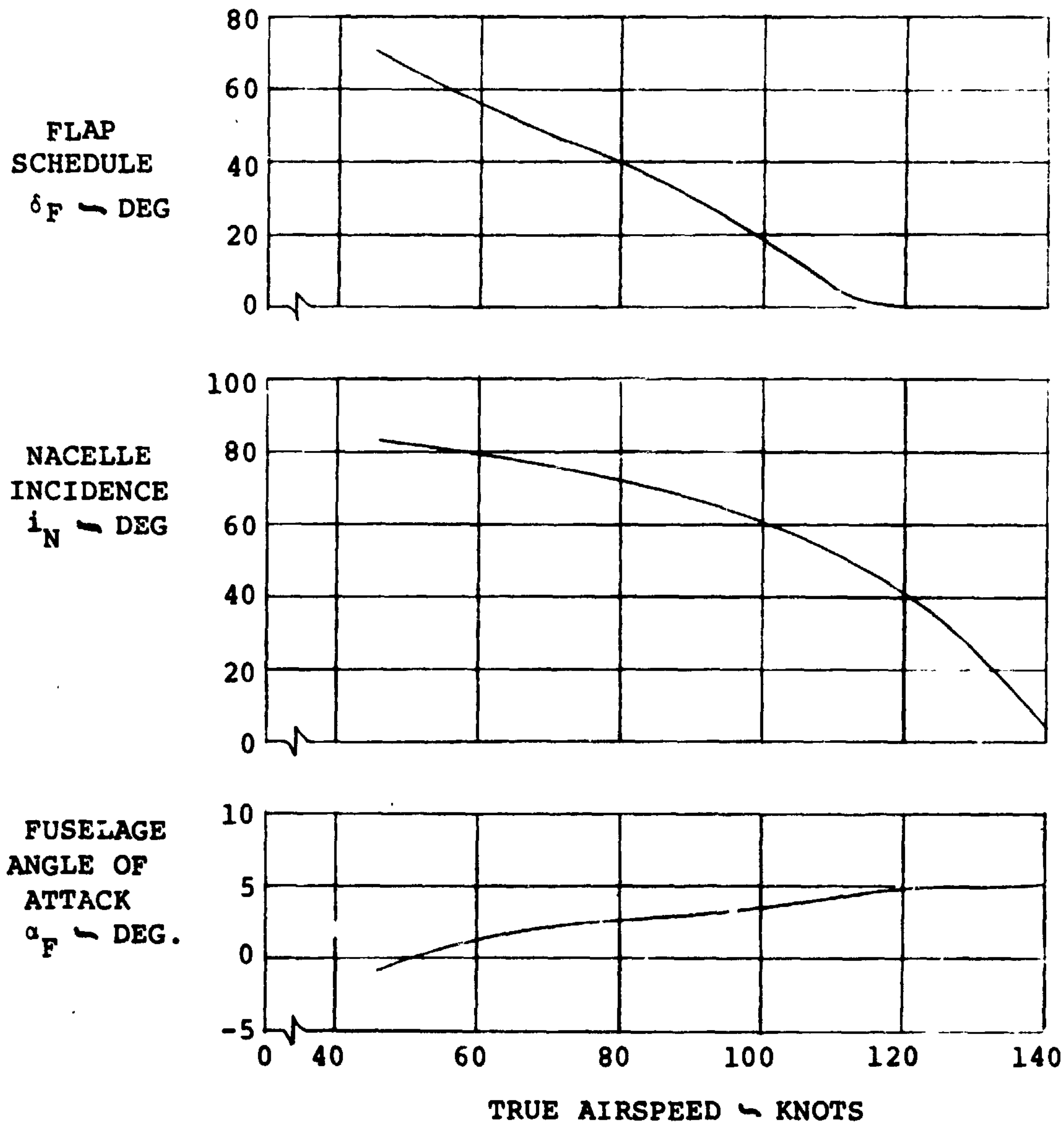


FIGURE 4-9: TRIM IN TRANSITION

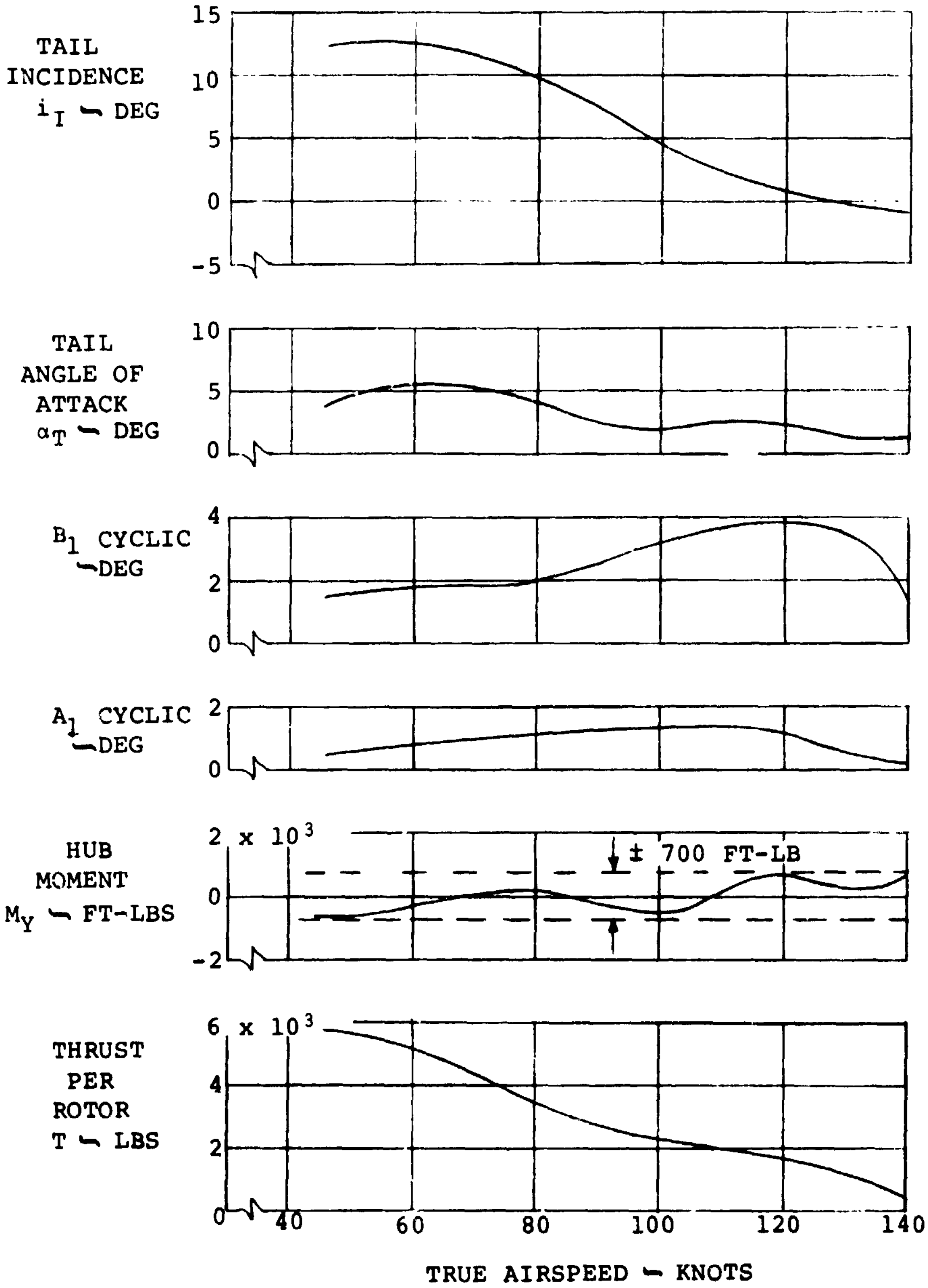


FIGURE 4-9 CONCLUDED

NOTES:

1. GW = 21,641 LBS
2. CG = 28% NACELLE DN
   FS 214.2
3. V_{TIP} = 750 FPS

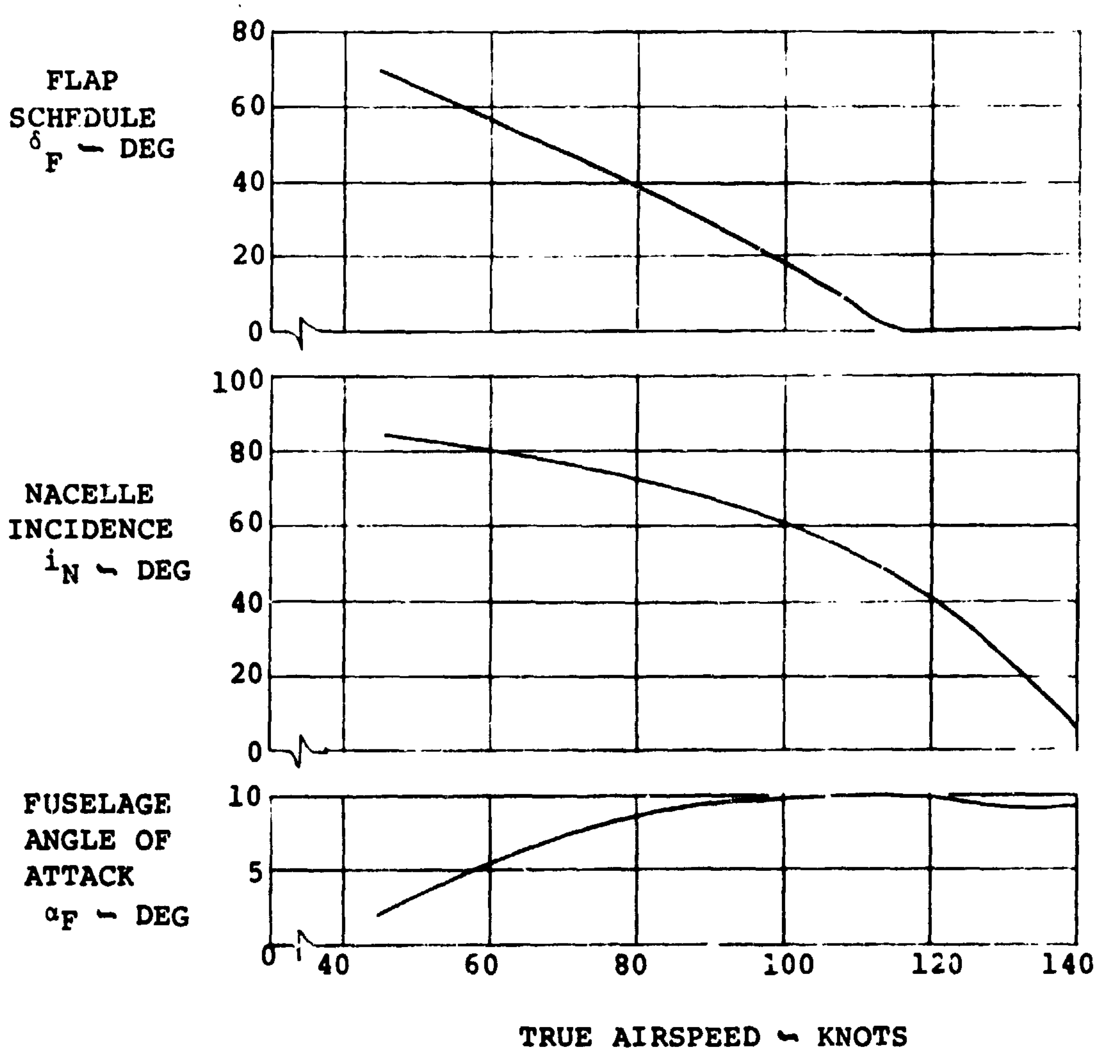


FIGURE 4-10: TRIM IN TRANSITION

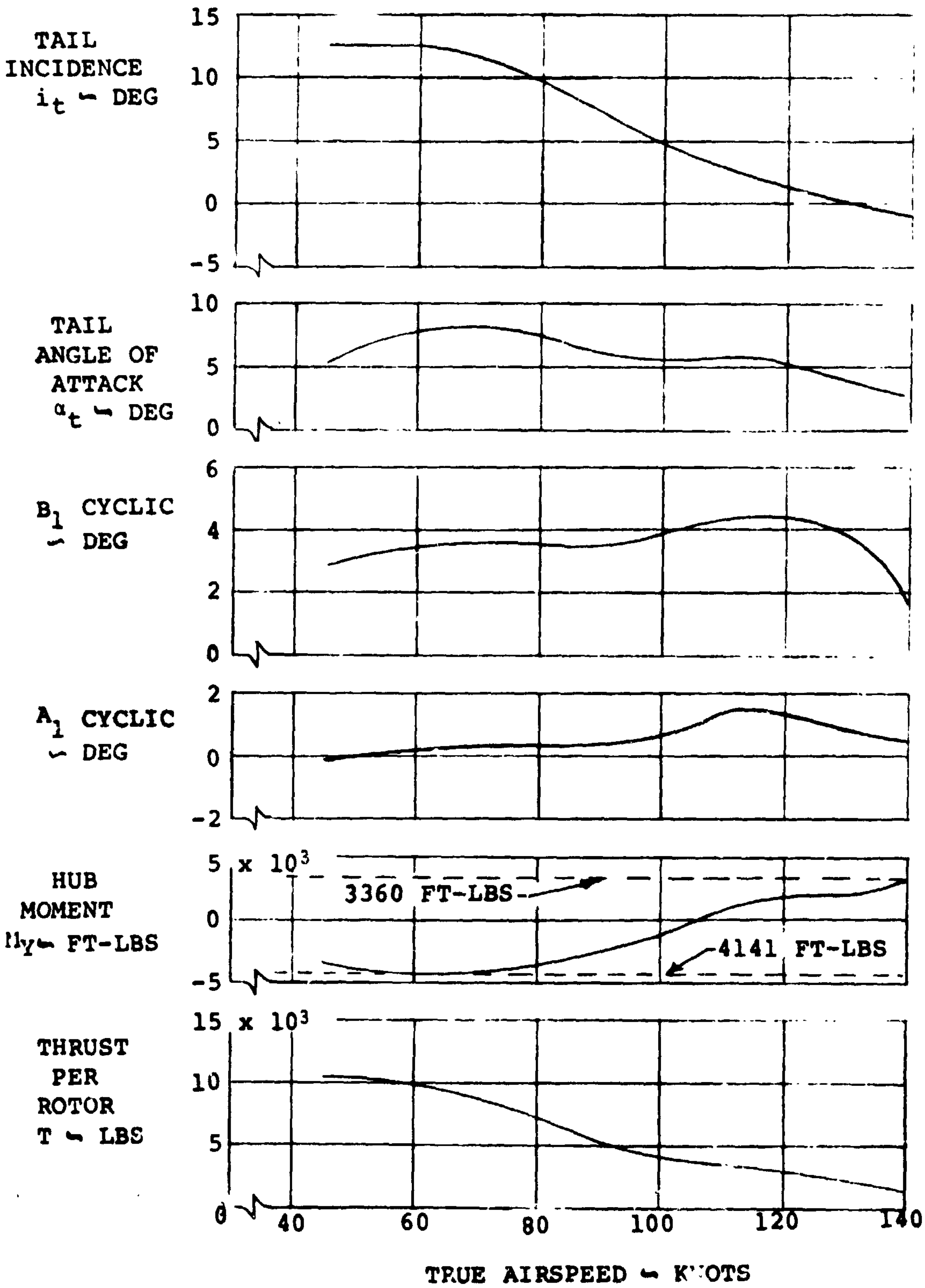


FIGURE 4-10 CONCLUDED

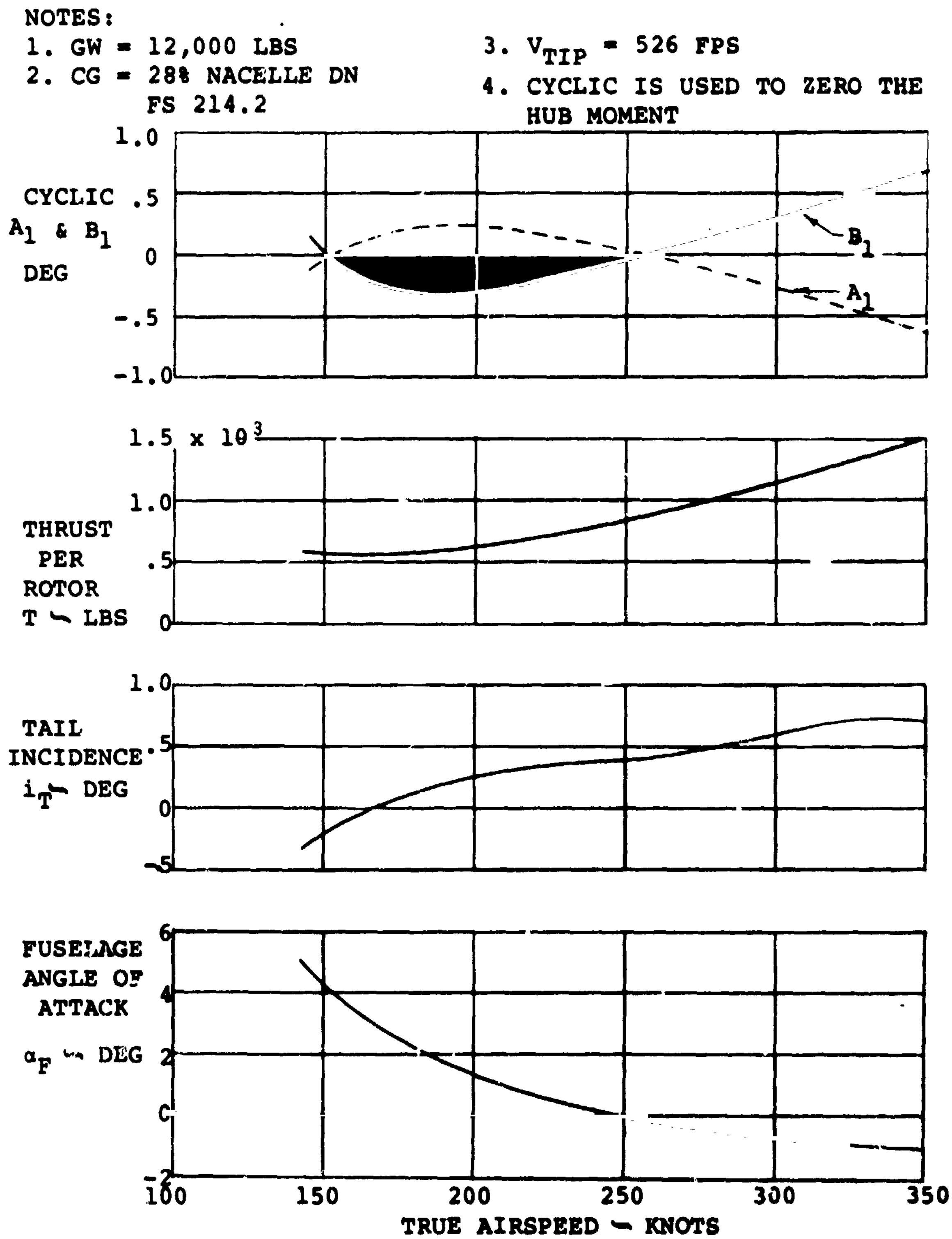


FIGURE 4-11: CRUISE TRIM

NOTES:

1. GW = 21,641 LBS
2. CG = 28% NACELLE DN FS 214.2
3. V_{TIP} = 526 FPS
4. CYCLIC IS USED TO ZERO THE HUB MOMENT

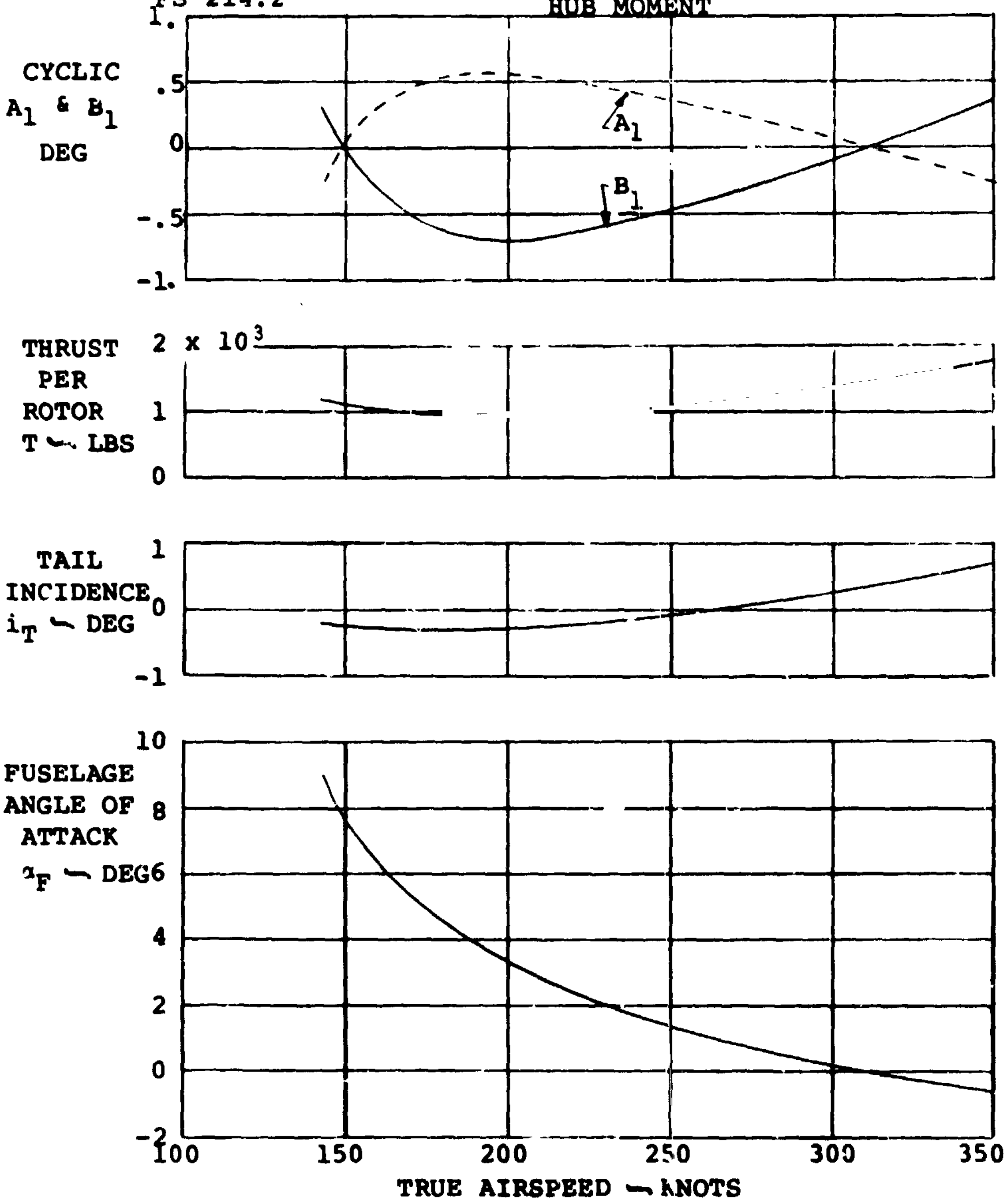


FIGURE 4-12 CRUISE TRIM

4.5.8 Longitudinal Maneuver Control in Cruise. - Data are presented in Figures 4-13 through 4-16 illustrating the longitudinal maneuver characteristics of the aircraft during cruise flight. Variation of tail incidence and angle of attack is illustrated for maneuver normal load factors from -1.0g to +3.0g as a function of velocity. The tail incidence variation per g may be converted to equivalent elevator deflection per g by multiplying by 2.0. The effects of weight and configuration differences between the nominal and maximum weight aircraft with the center of gravity approximately at the forward limit may be determined by comparing Figures 4-13 and 4-14. These figures assume trim thrust for each maneuver condition at the start of the pullup. Comparison of Figures 4-14 and 4-15 indicate the difference in control required for the maximum weight aircraft with thrust maintained at the value for trim level flight as compared to thrust required to maintain velocity initially at the maneuver load factor. Comparison of Figures 4-13 and 4-16 indicate the difference in maneuver requirements of the lightweight aircraft with thrust required to maintain velocity initially at the maneuver load factor for pullup compared to the characteristics with thrust-as-required during steady state turns at the same load factors.

4.5.9 Longitudinal Dynamics in Cruise. - Preliminary estimates of longitudinal short period and phugoid dynamics are presented in Figure 4-17 to indicate the effects of aircraft velocity and size. The short period frequency and damping are good with regard to military specification compliance. The short period natural frequencies of the Sea Control aircraft are slightly lower than for the lighter weight aircraft. Phugoid damping is indicated to be somewhat low at low speeds as anticipated. The assumed damping level of the phugoid, as shown, is pessimistic since the thrust variation with speed at constant power setting and density variation with altitude were neglected. If necessary, stability augmentation can be used to improve the phugoid damping. No stability augmentation effects were included in the calculations. A stability augmentation system will be incorporated into the aircraft and will operate in conjunction with the rotor-aircraft load alleviation system mentioned earlier. Augmentation of the phugoid mode damping can be accomplished by utilization of available pickoffs from the SAS/load-alleviation system if necessary.

4.5.10 Tail Sizing. - Adequacy of the tail area of both the horizontal and vertical tails on the Model 222 aircraft configurations is dependent on the rotor characteristics. Correlations of rotor characteristics predictions with wind tunnel test data obtained to date from tests of flexible rotors, i.e., with both lead-lag and flapping flexibility, indicate that the more conventional methods used to calculate rotor forces and moments are inadequate to predict the

F

NOTES:
1. GW = 12,000 LBS.
2. CG = .15$\overline{C}$
3. δ_F = 0°
4. NACELLE INCIDENCE = 0°
5. LONGITUDINAL ACCELERATION = 0.g
6. CYCLIC CONTROL = 0°

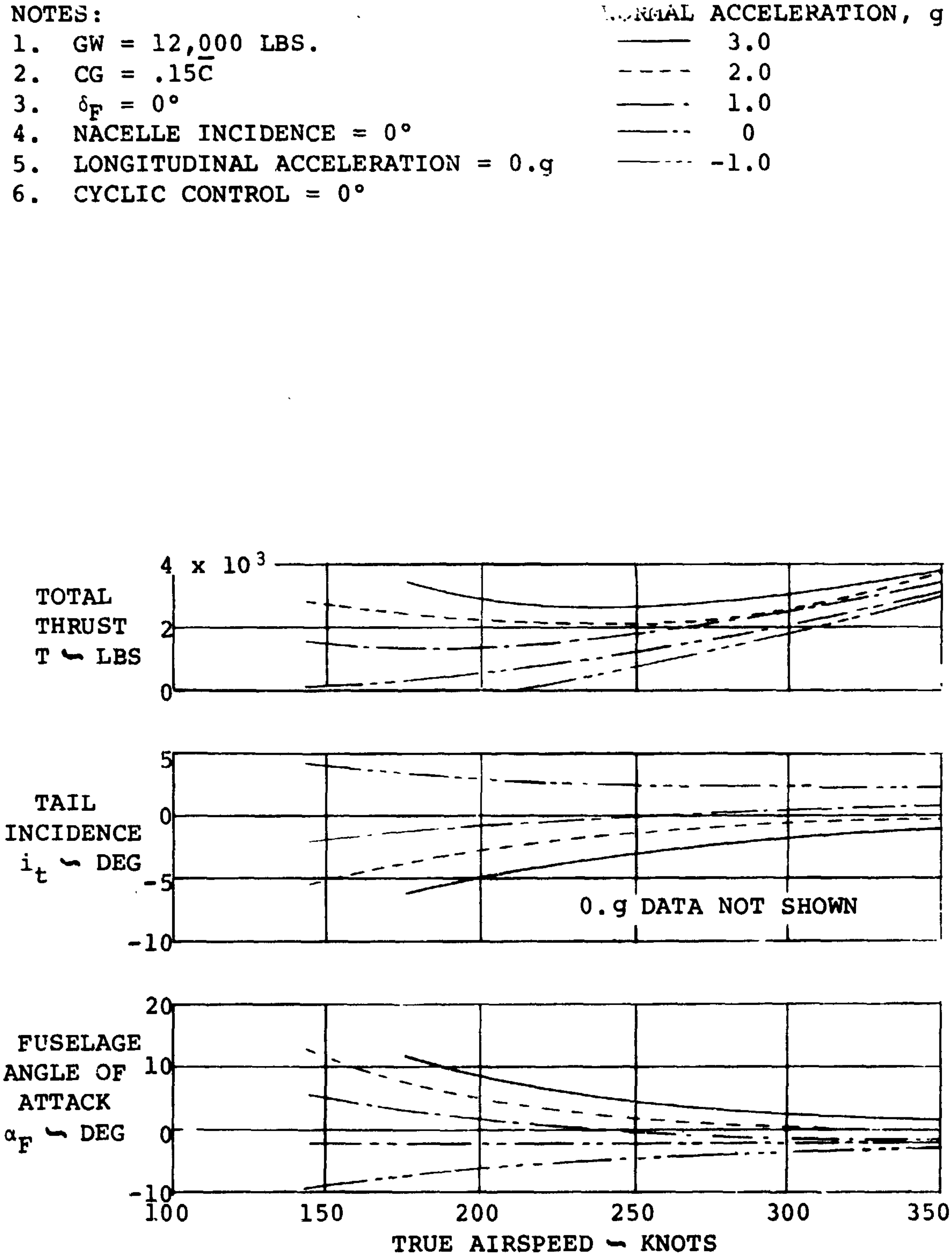


FIGURE 4-13: NORMAL ACCELERATION - LONGITUDINAL PULL-UP

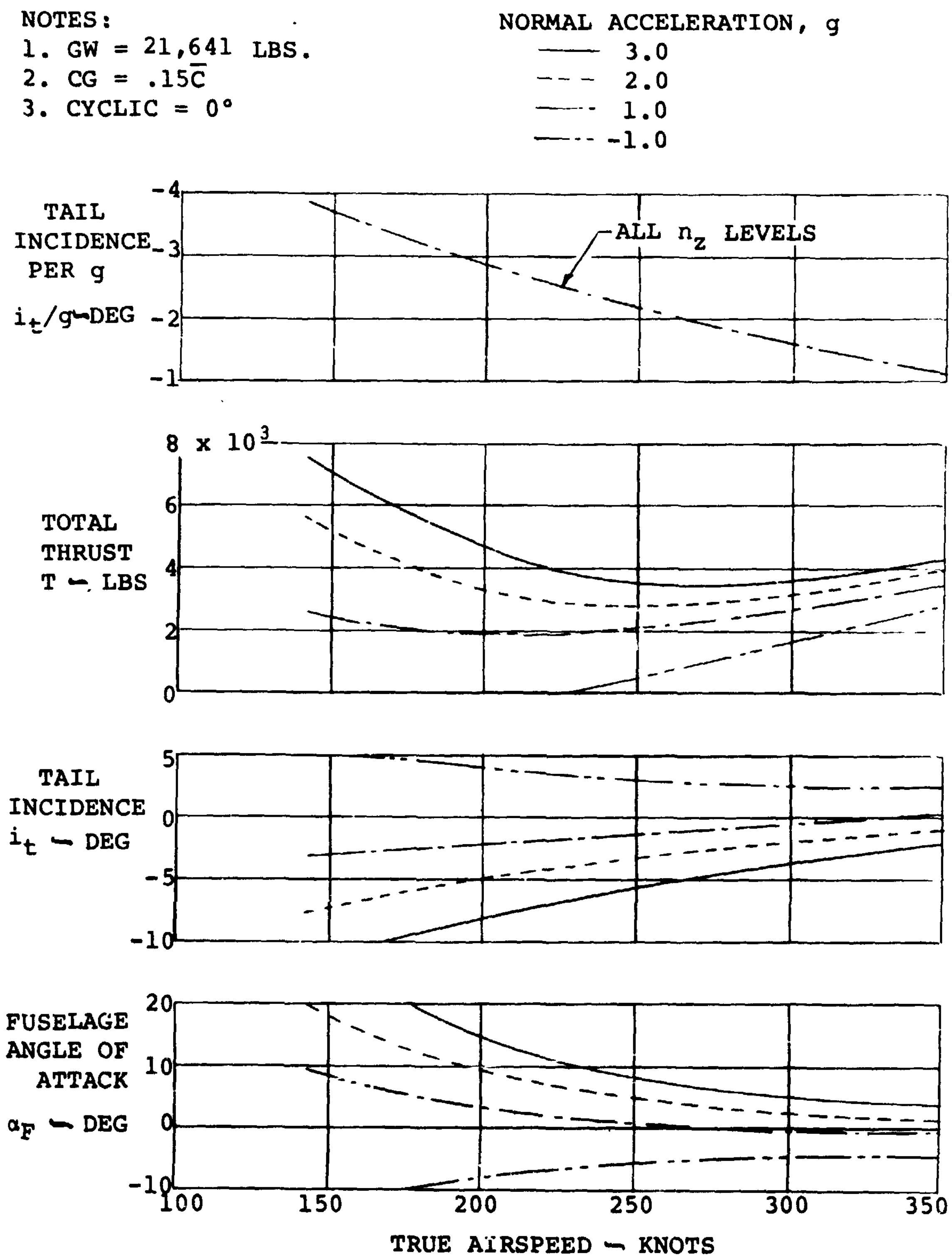


FIGURE 4-14: NORMAL ACCELERATION-LONGITUDINAL PULL-UP

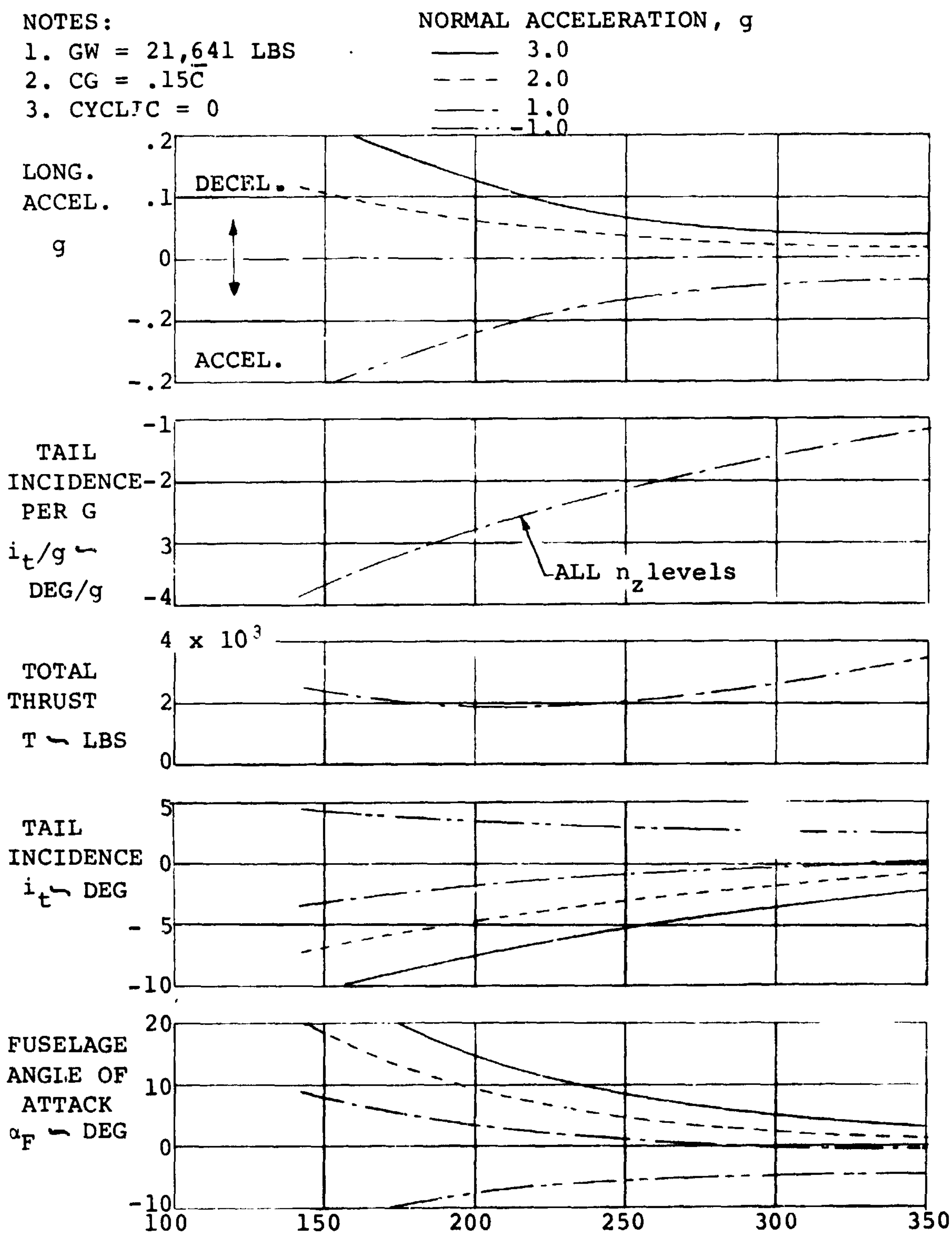


FIGURE 4-15: LONGITUDINAL PULL-UP AT CONSTANT (LEVEL FLIGHT) THRUST

NOTES:
1. GW = 12,000 LBS.
2. CG = .15 $\bar{C}$
3. δ_F = 0°
4. NACELLE INCIDENCE = 0°
5. LONGITUDINAL ACCELERATION = 0 g
6. CYCLIC CONTROL = 0

NORMAL ACCELERATION:
——— 3.0
- - - 2.0
—— · 1.0

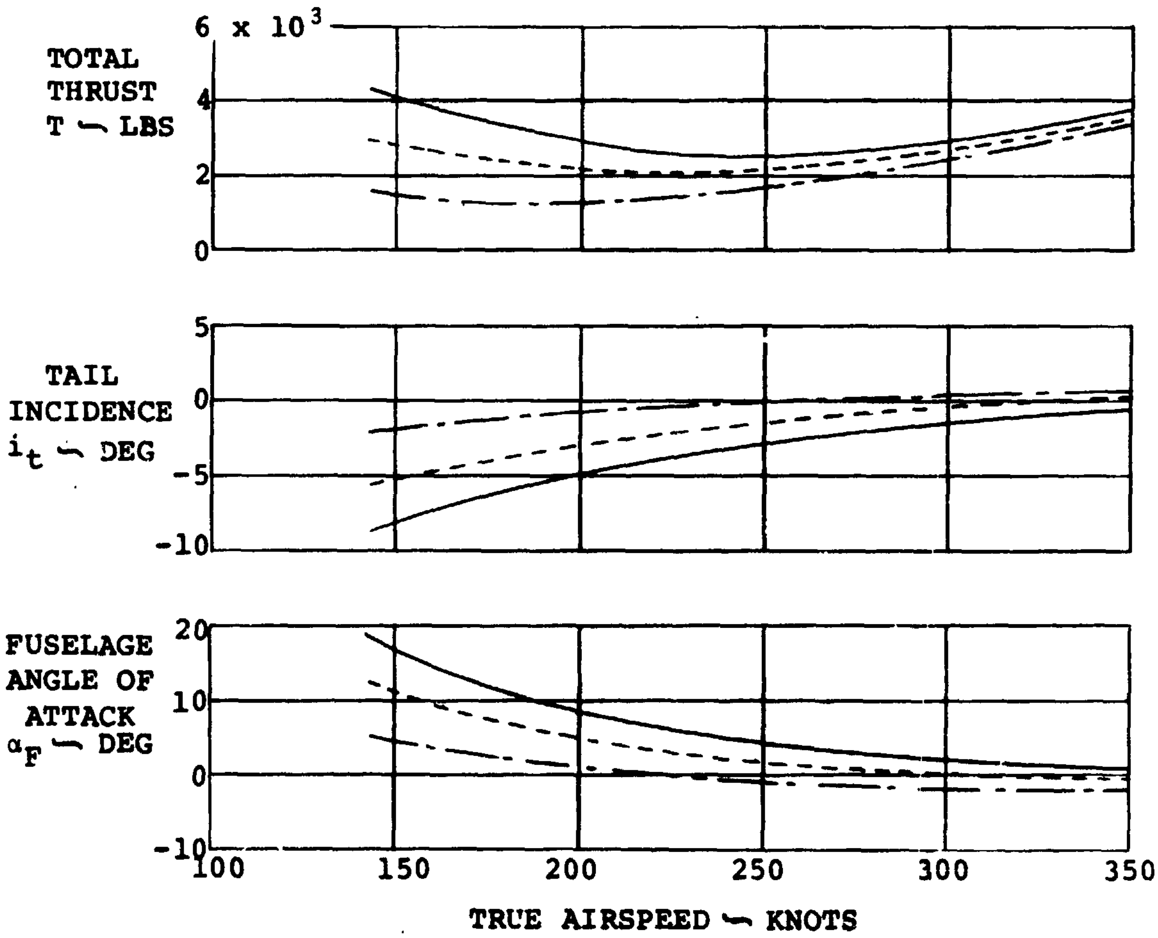


FIGURE 4-16: ACCELERATED TRIM-LEVEL FLIGHT TURNS

NOTE:
LEVELS REFER TO MIL-F-8785B
REQUIREMENT IN FLIGHT CATEGORY B
FLIGHT PHASE CRUISE (CR)

——— RESEARCH AIRCRAFT: GW = 12,000 LBS, CG @ .28$\bar{C}$
--- SEA CONTROL AIRCRAFT: GW = 21,641 LBS, CG @ .28$\bar{C}$

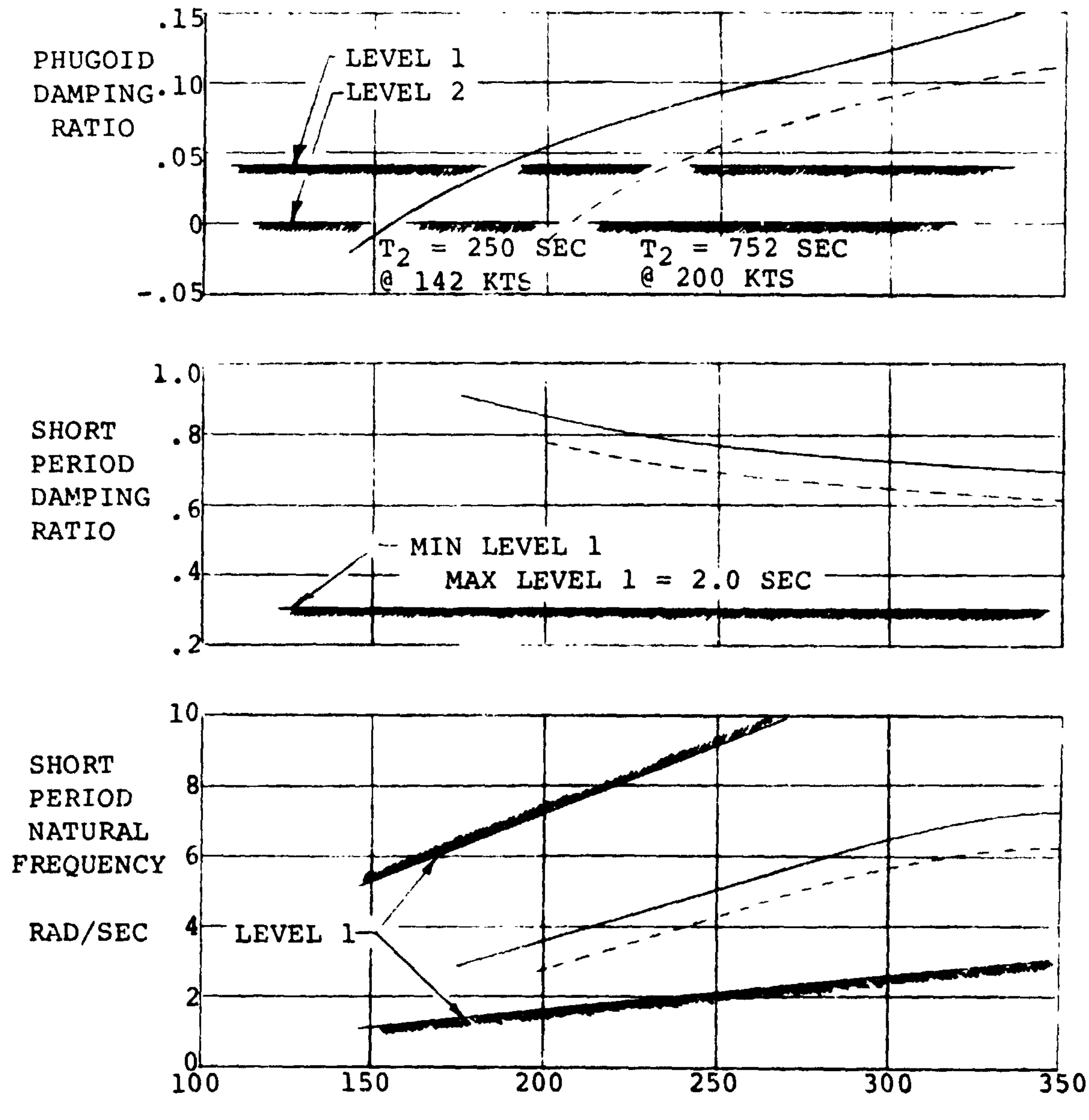


FIGURE 4-17: LONGITUDINAL STABILITY

characteristics of a soft inplane rotor because they ignore inplane flexibility effects. Figure 4-18 illustrates a comparison of rotor force and moment coefficients obtained from tunnel test data with the predicted levels of the force and moment coefficients obtained, using digital programs for: (1) rigid rotors, (2) flexible in flapping only, and (3) flexible in flapping and lead-lag both. The wind tunnel data were obtained from tests of a 1/9-scale dynamically-similar folding tilt-rotor/semi-span-wing model having frequency characteristics at the intermediate test rpm's near those of the Model 222 rotors. The frequency ratios were varied over a relatively wide range during the test by varying the test rpm of the rotor. It is noteworthy that the trends of the coefficients are matched very well, utilizing the Boeing-developed digital program methods which include the lead-lag in addition to flapping frequency effects. Note also that the sign of the pitching moment coefficient changed from + to - in the intermediate rpm range; i.e., the hub moment changed from a destabilizing to a stabilizing contribution, and that the normal force coefficient is of decreased magnitude as compared to the predicted level for the rigid rotor or rotor free to flap only. The normal force times its moment arm is more powerful in its contribution to stability than is the hub moment for the Model 222, and the rotor total contribution to satibility is, therefore, still destabilizing. However, the destabilizing influence of the Model 222 rotor is much smaller in both pitching and yawing of the aircraft than would be true if the rotor were rigid, or nearly so, inplane.

Proper selection of the frequencies as compared to design operational rpm of the rotor permits full advantage to be taken of the anticipated effects of the inplane frequency contribution to stability at both transition and cruise speeds because of the change in rpm between transition and cruise mode. Thus, the tail sizes of the Model 222, horizontal and vertical, are substantially smaller, approaching 50 percent, than would be required if the inplane frequency effects were ignored. Preliminary analysis indicates that a vertical tail volume coefficient $\ell_T S_T / S_w b$ of 0.128 and a horizontal tail volume coefficient $\ell_T S_T / S_w C_w$ of 1.0 provide adequate directional and longitudinal static and dynamic stability and control characteristics in cruise. These values have been used for the operational airplanes in this study.

4.6 Control Systems

The pilot's controls consist of a conventional stick and rudder pedals plus a power control. The power control lever sets power and rotor collective pitch. The governing system modifies the collective pitch to hold constant rpm. The rotor tilt position is selected by a switch on the power control lever. Rudder pedals are connected through a power cylinder

NOTES:

1. FORCE = $1/2\rho\pi R^2V^2\ \alpha$(COEFF.)
2. MOMENT = $\rho\pi R^3V^2\ \alpha$(COEFF.)
3. $dC_L/d\alpha$ = 5.7
4. TUNNEL VELOCITY ≈ 85 FPS
5. SYMBOLS DENOTE TEST PTS
6. COEFFICIENTS: PER RADIAN

——— FLAP & LEAD LAG
- - - RIGID BLADES
—— - FLAP ONLY

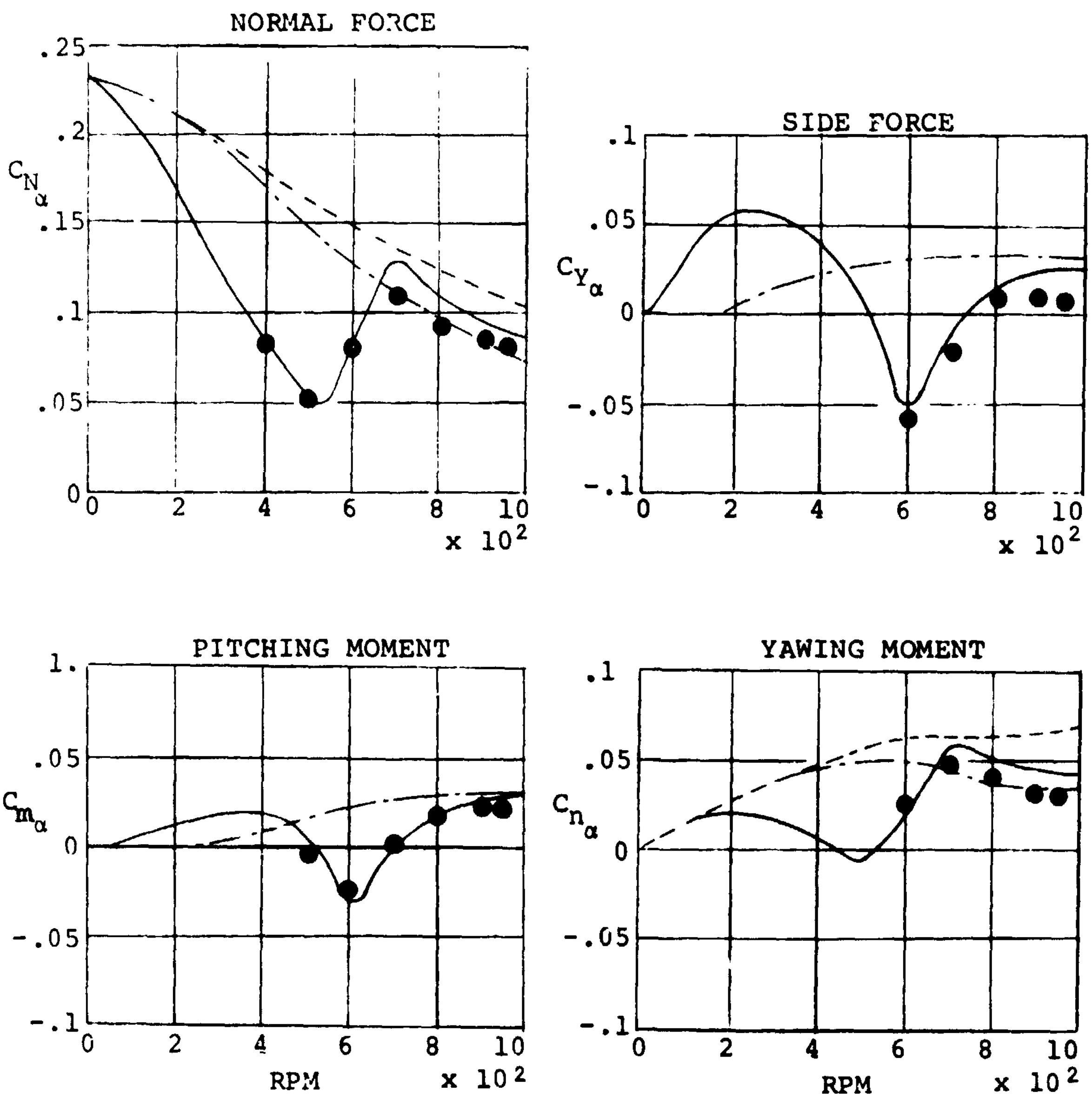


FIGURE 4-18: 1/9 SCALE CONVERSION MODEL DERIVATIVE VARIATION WITH RPM

to the conventional aircraft rudder and to the differential longitudinal rotor cyclic pitch system.

In hover, full yaw cyclic is obtained by full pedal displacement. As the transition (i.e.,rotor tilt) progresses toward aircraft flight, a mechanical gain changer reduces the amount of cyclic that is obtained by the pedal displacement as the rudder effectiveness increases with forward airspeed.

Longitudinal motion of the stick moves the elevator and produces cyclic pitch in the rotors. As in the case of the pedals, this cyclic pitch is phased out as the aircraft transitions from hover to cruise.

Lateral motion of the stick moves the flaperons and provides differential collective pitch in the rotors during hover. This differential collective pitch is also phased out as the aircraft transitions from hover to cruise.

During transition, no additional tasks are performed by the pilot since the setting of flaps, longitudinal trim, etc., is pre-programmed as a function of nacelle tilt position. This transition programming automatically places the flaps in a 70-degree position for hover and opens the wing leading-edge download reduction devices. These leading-edge devices are closed in transition at approximately 40 knots, and the flap angle is reduced in accordance with a transition schedule. Flap deflection will be reduced to zero as nacelle incidence goes to zero.

Sets of dual hydraulic power cylinders are provided at each rotor swashplate to prevent feedback of rotor control loads into the control system. Cockpit power cylinders are provided to ensure light control feel and prevent feedback of aircraft control surface loads to the stick. A variable control force feel system is incorporated to provide good control forces for all flight regimes. Control feel changes will be programmed with nacelle incidence and with dynamic pressure.

A stability augmentation system (SAS) is installed to provide the desired damping and attitude characteristics during hover, transition, and aircraft flight. This augmentation system also provides feedback inputs to the rotor collective and cyclic controls to improve the ride qualities and reduce the rotor loads during flight in gusty air. A conventional longitudinal trim wheel is provided to permit the pilot to compensate for center of gravity variations. This wheel biases the longitudinal stick position.

4.7 Aeroelastic Stability

The tilt rotor class of aircraft has a number of potential instability mechanisms as a result of prop/rotor size and flexibility. These include whirl flutter and ground and air resonance, along with nonclassical effects related to the internal properties of the blade and steady deflection effects such as pitch/flap/lag coupling. Because of the similarity existing throughout the class of tilt rotor aircraft covered in this study, the flight research aircraft has been analyzed in detail since it is considered that the results obtained for a typical configuration will apply to all of the aircraft.

4.7.1 Prediction Capability. - A high level of confidence in our ability to predict whirl flutter and ground and air resonance behavior has been demonstrated on a series of dynamically similar models. For example, the testing for divergence and whirl flutter boundaries on the 1/9-scale model, Reference 5, as shown in Figure 4-19, has shown the accuracy of the predictions. The onset of air resonance has also been predicted accurately for a 1/9.2-scale model, Figure 4-20. A recently developed capability is the prediction of blade pitch-lag-flap flutter. This has been experienced on a number of model blades most notably during a NASA/Boeing test of a 13-foot dynamic model at ONERA. The instabilities experienced on the test were caused by coupling between blade torsion and blade flapping and lagging conditions when the blade was deflected due to steady thrust and torque conditions. The onset of instability is now predicted using the most recent methodology as shown in Figure 4-21.

4.7.2 Stability Boundaries. - Stability boundaries for a typical tilt/prop-rotor aircraft are shown in Figure 4-22. These are for the cruise configuration (i.e., nacelles down) which is identified to be the most critical regime of operation. Whirl flutter occurs at high advance ratio, high rpm conditions. Two modes of symmetric flutter are present and one mode involving antisymmetric motion of the airframe and rotors. These boundaries are well clear of any operational conditions. The use of a soft inplane hingeless rotor provides high speed capability without whirl flutter and without adding weight to the wing which was designed from strength considerations. Typically, an articulated or gimballed rotor requires a substantial increase in wing torsional stiffness. At high advance ratios, any tendencies to air resonance mechanical instability are suppressed by aerodynamic damping in the blade lead-lag modes. At low advance ratios, the blade collective pitch settings are smaller and lead-lag damping reduced, with the result that a region of air resonance exists centered around 160 percent of normal cruise rpm. This is outside the operational cruise range which extends down only as far as an advance ratio of .45. The system is still free from instability

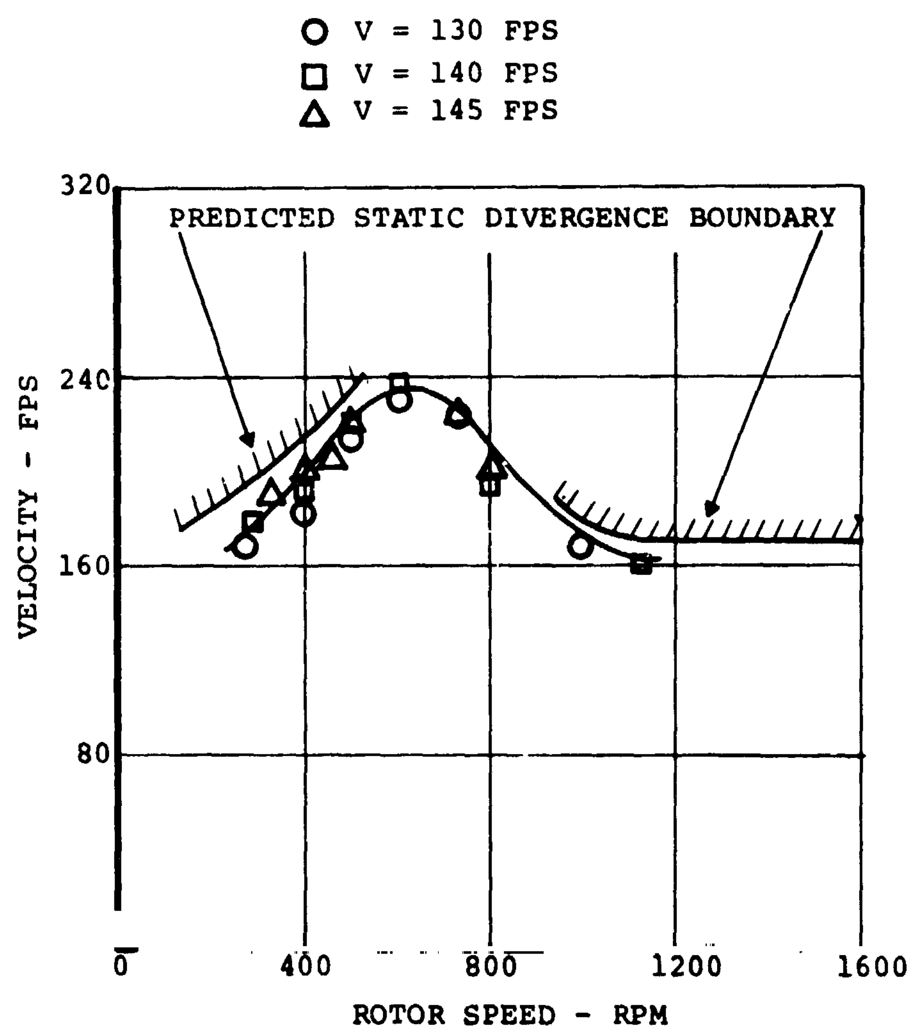


FIGURE 4-19: STATIC DIVERGENCE BOUNDARY EXTRACTED FROM TEST DATA - RUN 126 - REDUCED TORSION STIFFNESS SPAR - 1/9 SCALE CONVERSION MODEL

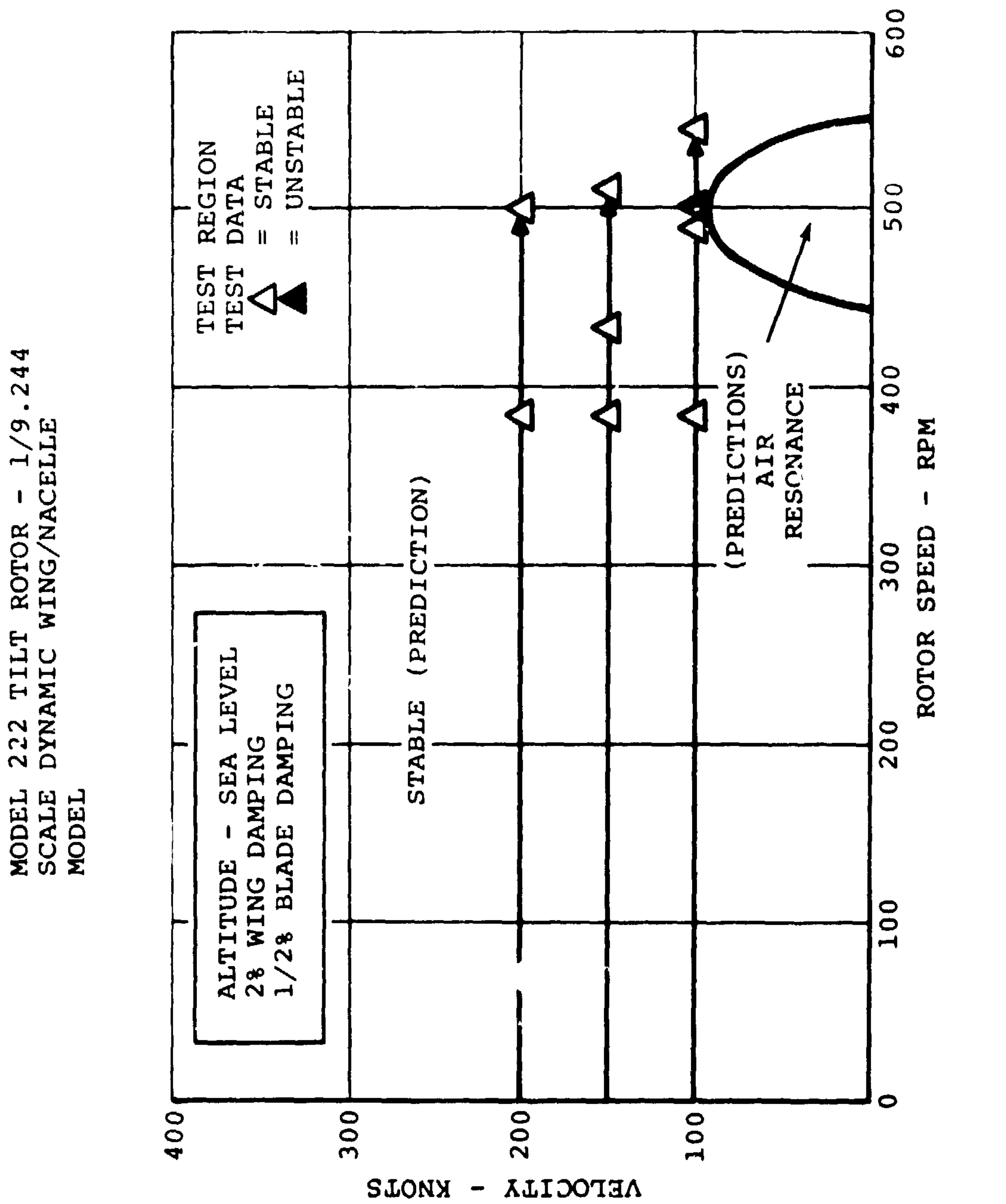


FIGURE 4-20: CRUISE AEROELASTIC STABILITY PROFILE, CORRELATION BETWEEN ANALYSIS AND TEST DATA

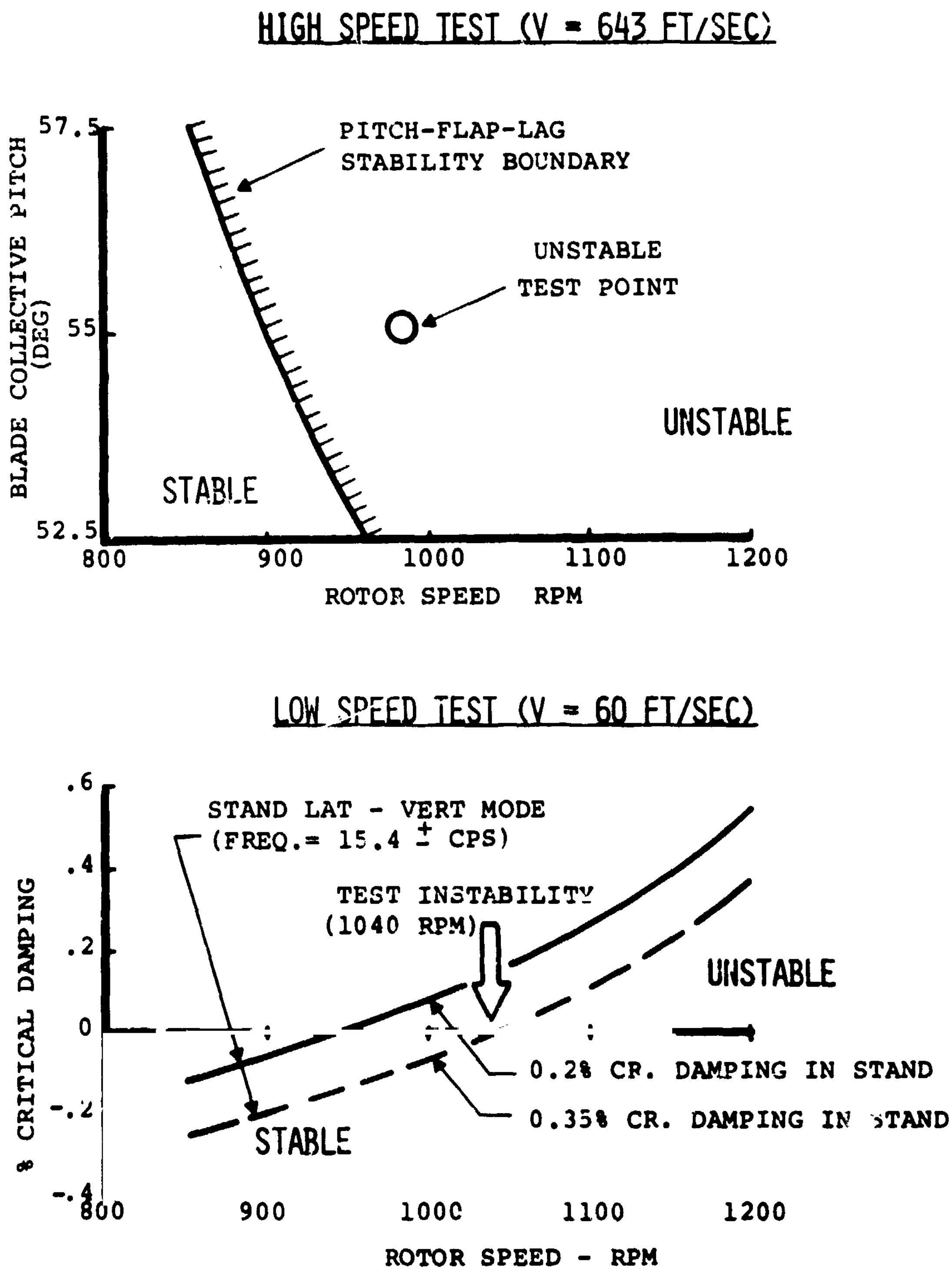


FIGURE 4-21:PREDICTED PITCH-LAG-FLAP MODAL DAMPING VS. RPM (ONERA TEST OF JUNE 1970)

MODEL 222 TILT ROTOR DEMONSTRATOR AIRCRAFT

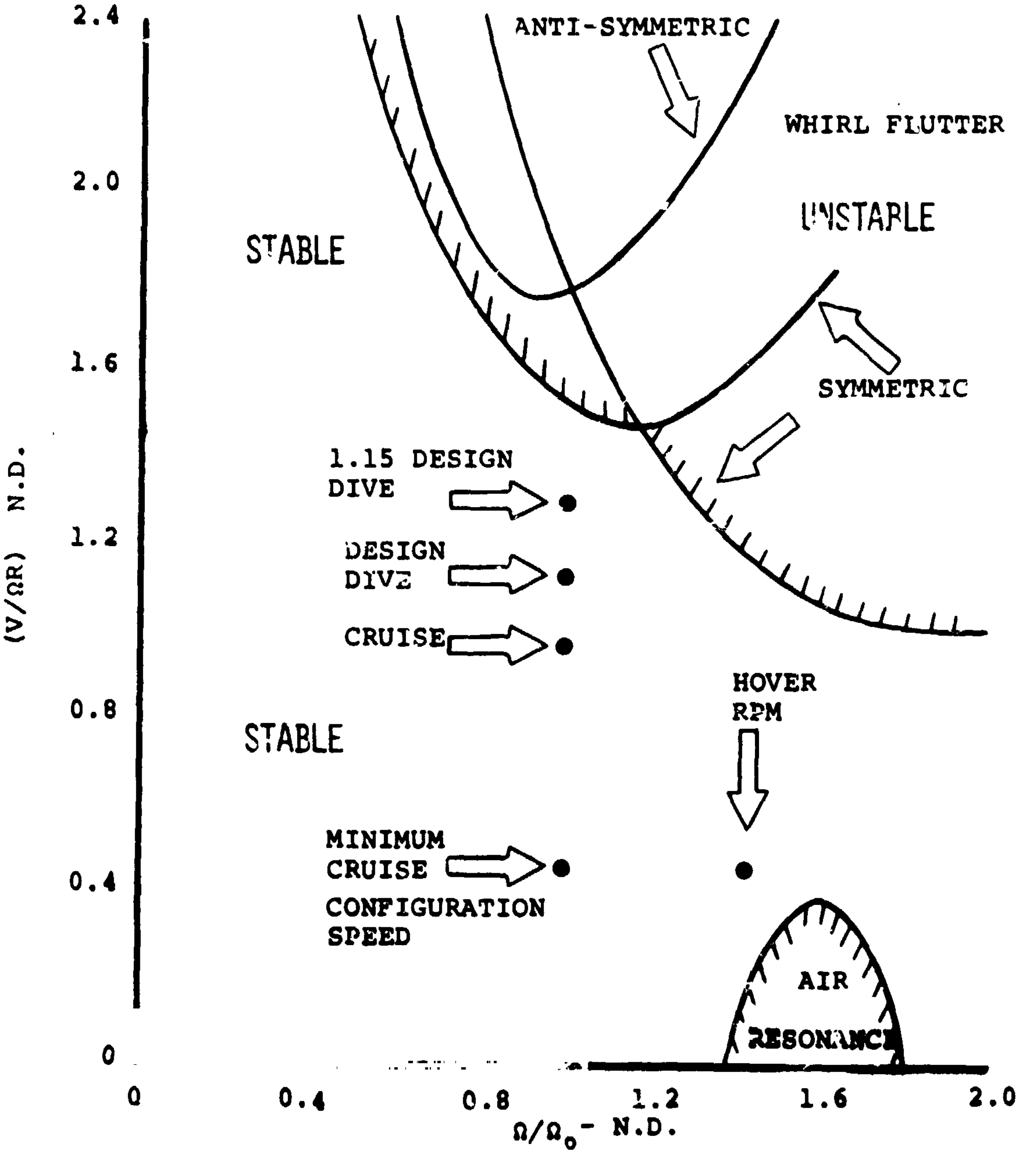


FIGURE 4-22: AEROELASTIC STABILITY BOUNDARY CRUISE MODE

even at hover rpm at the lowest cruise configuration advance ratio.

4.8 Performance

The performance of each of the four useful tilt-rotor aircraft is shown in this section. The following data is presented:

a. Hover ceiling

b. Flight envelope

c. STOL performance

d. Payload-radius-loiter performance

4.8.1 U. S. Army MAVS.

4.8.1.1 Hover Ceiling. - Figure 4-23 presents the out-of-ground effect hover capability of the Army version of the tilt rotor. This data is based on the capability to hover at a T/W ratio sufficient to provide a 500 ft/min vertical rate of climb. The climb margin is consistent with current Army design criteria which requires that new aircraft possess this inherent capability, specifically at 4,000 feet, 95°F, at military rated power. Superimposed on the curve is the design gross weight which substantiates the fact that the point design vehicle exceeds the criteria by approximately 500 pounds in gross weight or conversely the aircraft's climb capability exceeds the 500-fpm requirement.

4.8.1.2 Flight Envelope. - Figure 4-24 presents the normal rated power level flight speed envelope for cruise rotor rpm (70 percent of hover rpm). The data is presented for standard day conditions; rotors down; flaps up. This shows a normal rated power speed range between 290-300 knots up to approximately 20,000 feet. The military rated power speed at 10,000 feet is 314 knots. In addition, these data show a ceiling capability of 29,500 feet.

4.8.1.3 STOL Performance. - Figure 4-25 presents the short takeoff performance capability of the Army MAVS design at sea level for standard day conditions in terms of takeoff gross weight as a function of ground roll distance. Two levels of performance are presented; one which reflects maximum capability and assumes that the aircraft lifts off at the end of rotation at 10 degrees maximum wing angle of attack at lift-to-weight ratio = 1.0, and the other, a normal takeoff capability, assumes that in STOL mode, the lift-off speed corresponds to lift-to-weight ratio = 1.1 at a limiting wing angle of attack of 10 degrees. The first set of assumptions

NOTES:
1. T/W = 1.1
2. MILITARY POWER
3. (2) UTTAS 1500 SHP ENGINES
4. HOVER TIP SPEED = 750 FT/SEC
5. OUT-OF-GROUND EFFECT

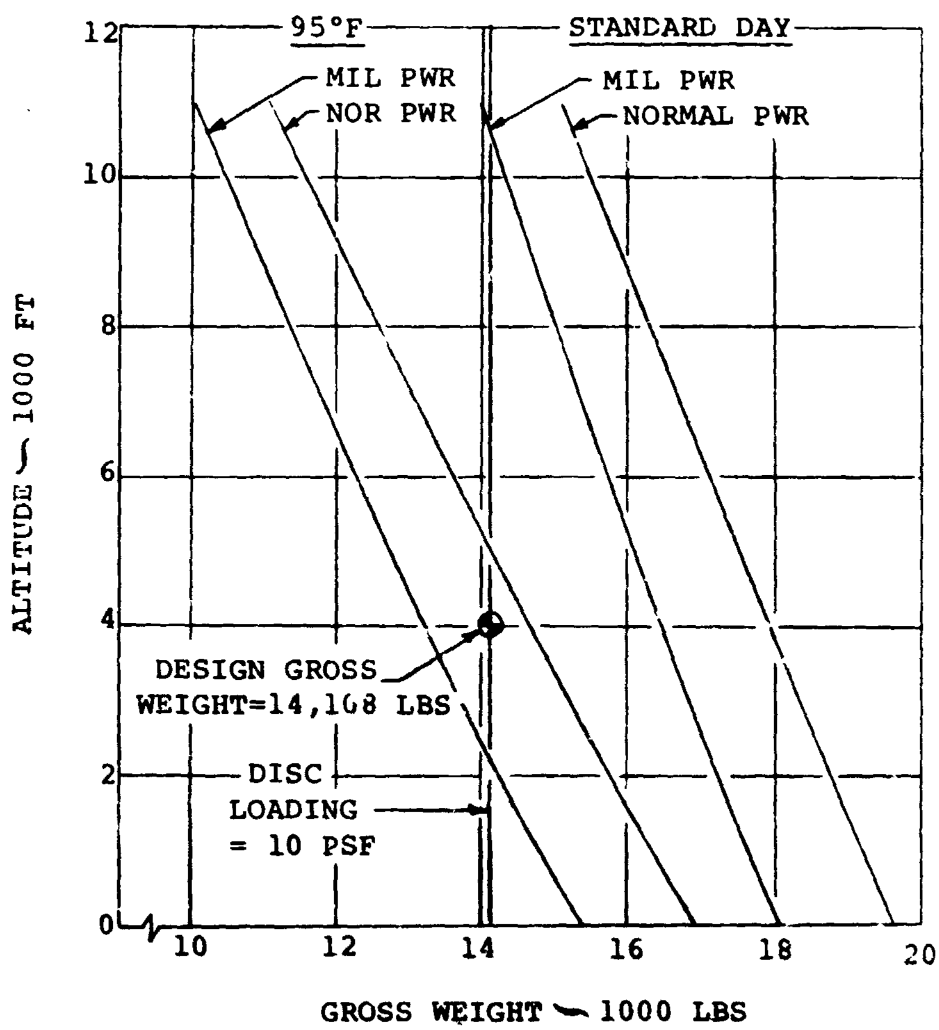


FIGURE 4-23: MODEL 222-1A-ARMY MAVS OGE HOVER CAPABILITY

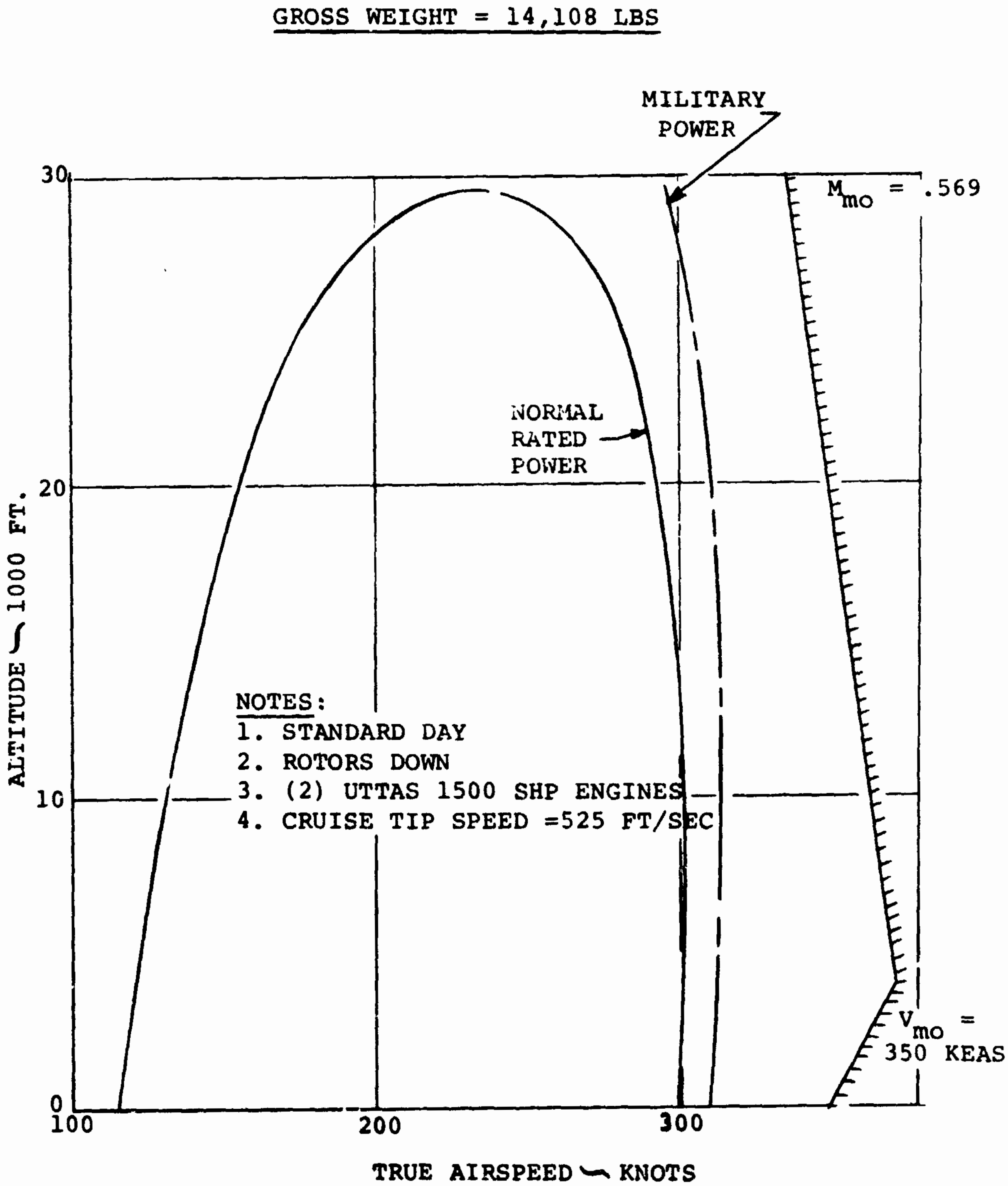


FIGURE 4-24: MODEL 222-1A ARMY - MAVS
SPEED ALTITUDE CAPABILITY

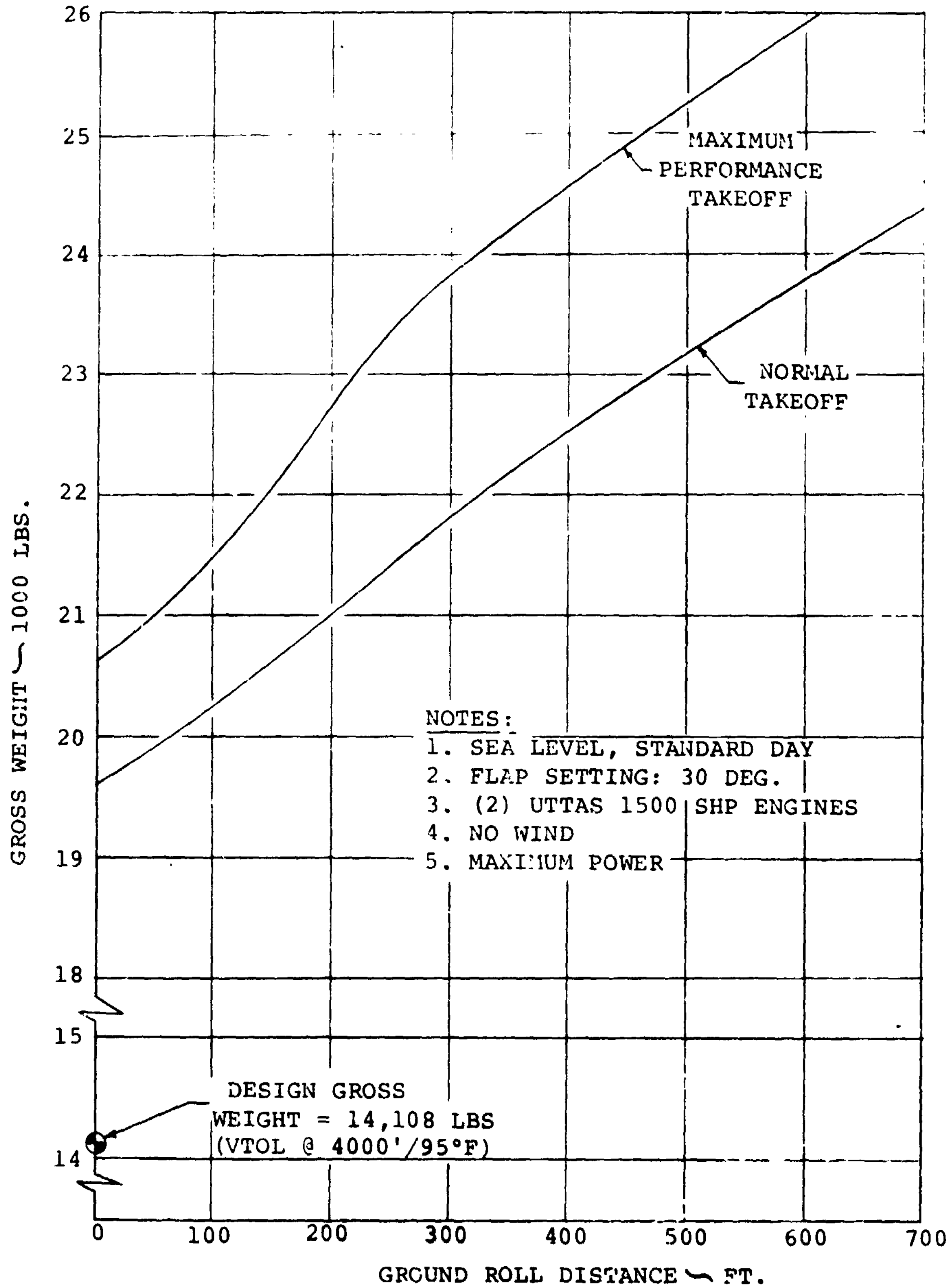


FIGURE 4-25: MODEL 222-1A ARMY-MAVS
SHORT TAKEOFF PERFORMANCE - GROSS WEIGHT

results in a lower lift-off speed and distance and is referred to as maximum performance takeoff. In the limit, this permits vertical takeoff with thrust-to-weight ratio of 1.05 to offset the 5-percent download. The second assumption is referred to as takeoff with normal load factor margin of 10 percent. In the zero speed limit, this has been faired to permit vertical takeoff at thrust-to-weight ratio of 1.1. Superimposed on the plot is the design gross weight which, as can be seen, is significantly less than the VTOL weight limitation. This is because the aircraft has a vertical climb rate at 4,000 feet/95°F in excess of 500 fpm. Figure 4-26 shows the corresponding variation of mission load with takeoff distance.

4.8.1.4 Payload-Radius-Loiter Performance. - Figure 4-27 presents the mission performance capability for the generalized surveillance/observation mission shown in Figure 4-29. Takeoff is at design gross weight. For a typical Army mission radius of 100 nautical miles, the data shows that the aircraft is capable of carrying a payload of approximately 1,540 pounds with an attendant loiter capability of one hour. The maximum radius (without employing auxiliary fuel) and with one hour of loiter at mid-point is 122 nautical miles with 1,412 pounds of mission with no mid-point loiter.

A basic requirement for the MAVS aircraft is that it be capable of a four-hour cruise on internal fuel at an overload weight not to exceed the maximum alternate gross weight (minimum load factor). The total internal fuel capacity of this aircraft is 3,000 pounds. The maximum alternate gross weight is 19,751 pounds. It is seen on Figure 4-28 that the four-hour cruise requirement is met with a takeoff gross weight of 15,513 pounds and using 2,810 pounds of internal fuel.

4.8.2 U. S. Air Force SAR.

4.8.2.1 Hover Ceiling. - Figure 4-30 presents the out-of-ground effect hover capability of the USAF version of the tilt rotor. The data is based on the capability to hover at a thrust-to-weight ratio of 1.1 which accounts for 5 percent to download and an additional margin of 5 percent for maneuver and control. As shown, the aircraft is capable of hovering at a gross weight of 21,750 pounds at sea level, standard day conditions. At the design gross weight, the hot day (95°F) hover ceiling is approximately 4,000 feet.

4.8.2.2 Flight Envelope. - Figure 4-31 presents the normal rated power level flight speed envelope for the design gross weight. It is noted that for a standard day cruise mode configuration, the normal rated power speed capability exceeds 320 knots up to approximately 12,000 feet and 300 knots up to 19,000 feet. The associated military rated power speed is 347 knots.

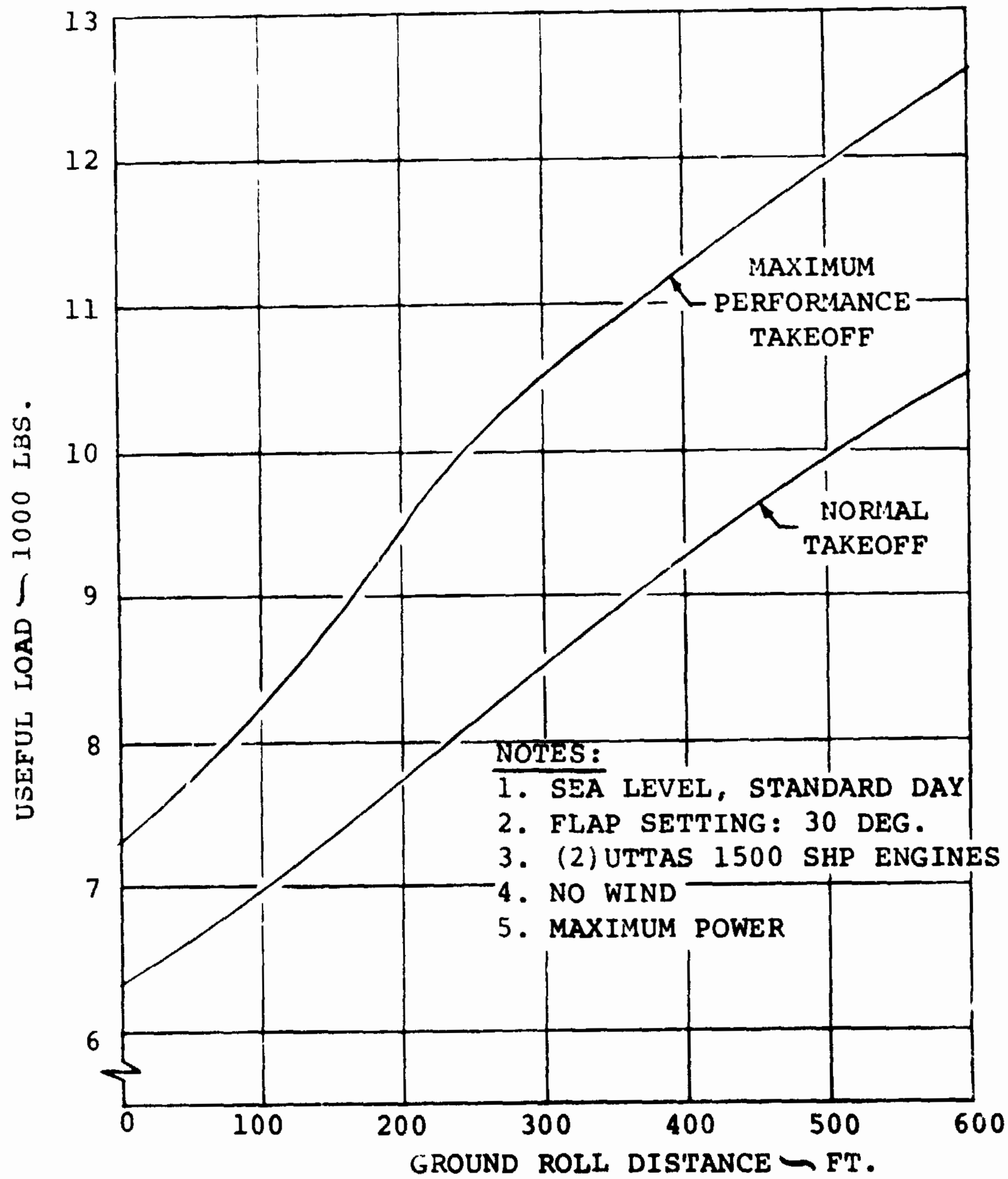


FIGURE 4-26: MODEL 222-1A ARMY-MAVS
SHORT TAKEOFF PERFORMANCE - USEFUL LOAD

TAKEOFF GROSS WEIGHT = 14,108 LBS

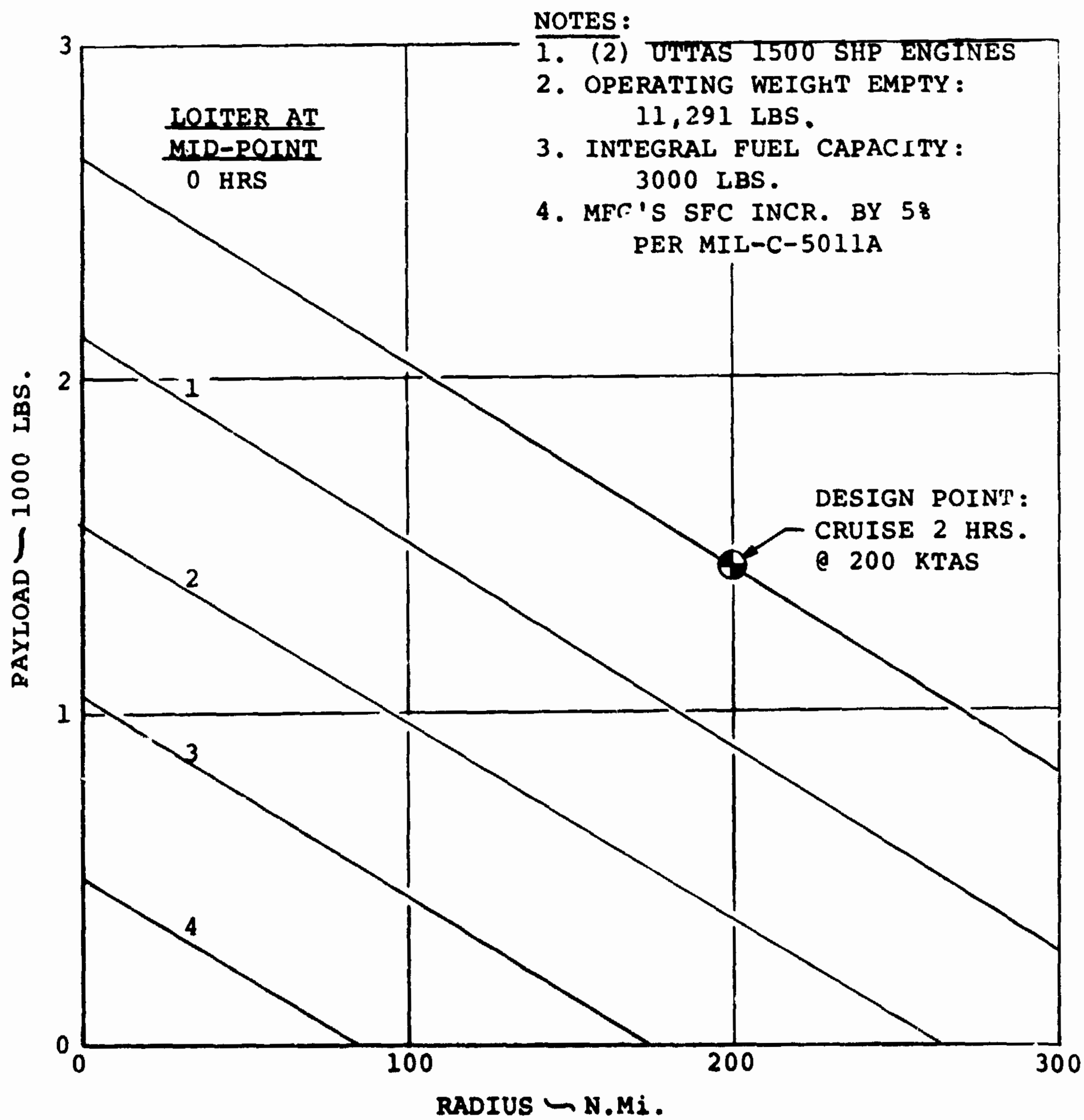


FIGURE 4-27: MODEL 222-1A ARMY-MAVS
MISSION CAPABILITY

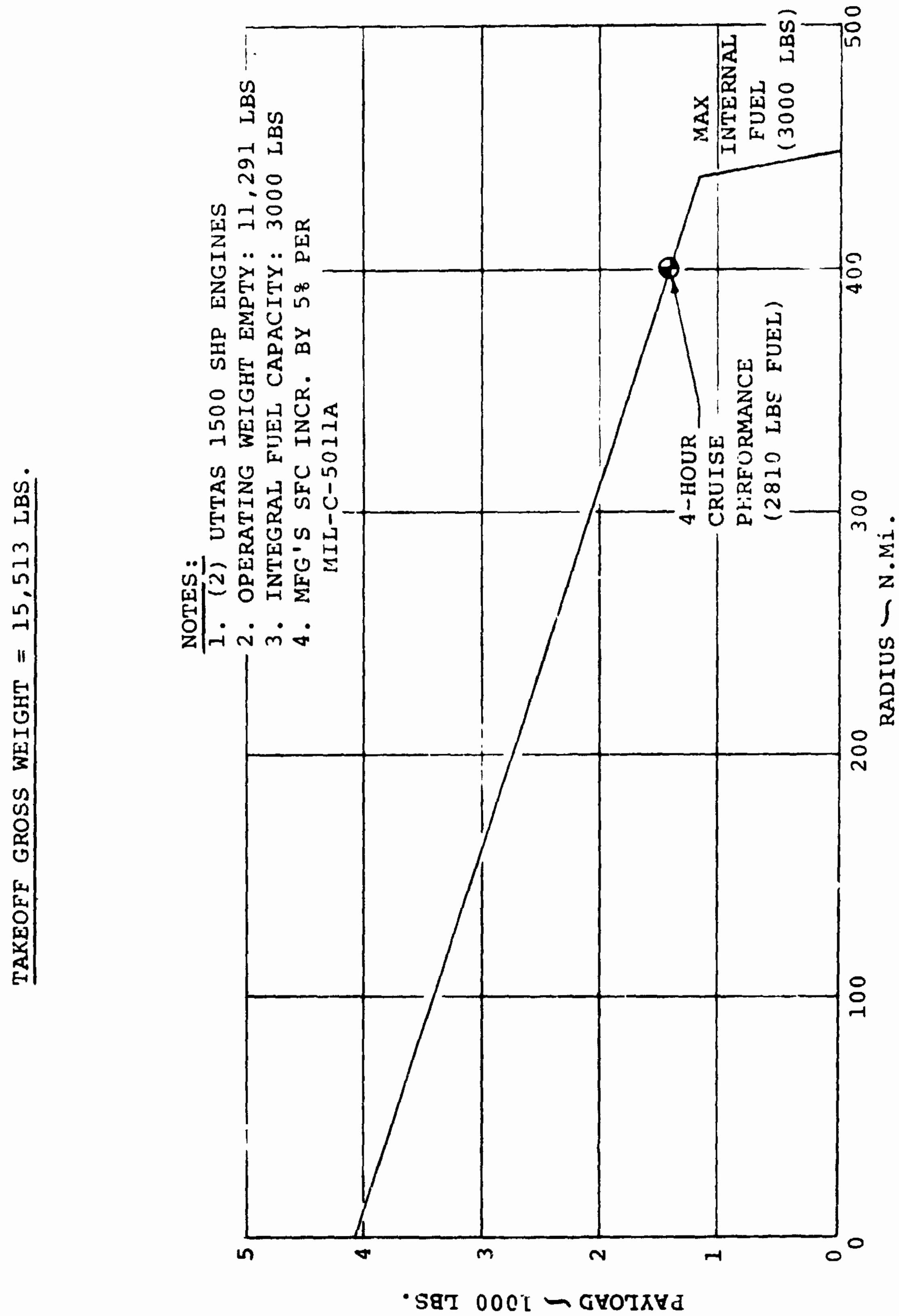
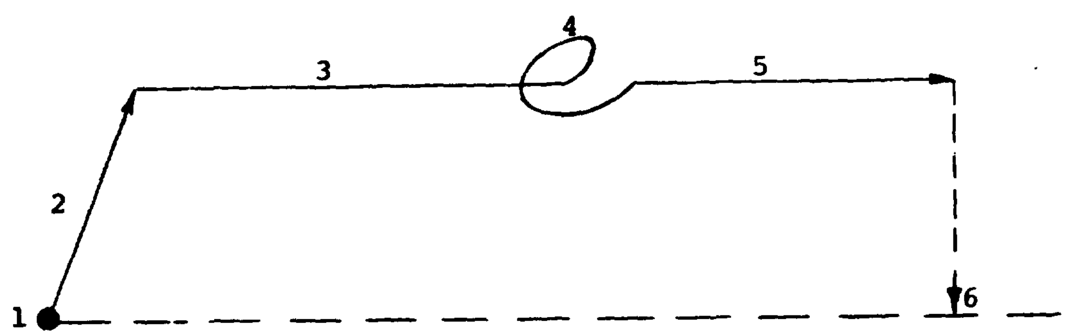


FIGURE 4-28: MODEL 222-1A ARMY-MAVS
MISSION CAPABILITY



1. WARM-UP, TAXI AND TAKEOFF: 5 MIN @ NORMAL RATED POWER, SEA LEVEL
2. CLIMB TO 5000 FT @ NORMAL RATED POWER AND SPEED FOR MAXIMUM RATE OF CLIMB
3. CRUISE OUTBOUND AT 200 KNOTS TRUE AIRSPEED
4. LOITER FOR X HRS @ 5000 FT
5. CRUISE INBOUND @ 200 KNOTS TRUE AIRSPEED
6. LAND WITH 10% (INITIAL) FUEL RESERVE

NOTES:
1. MISSION FLOWN @ STANDARD ATMOSPHERE CONDITIONS UNLESS OTHERWISE NOTED
2. SFC INCREASED 5% PER MIL-C-5011A

FIGURE 4-29: MODEL 222-1A ARMY-MAVS TACTICAL AIR OBSERVATION MISSION PROFILE

NOTES:
1. T/W = 1.1
2. MILITARY POWER
3. (2) LYC. PLT-27 ENGINES
4. HOVER TIP SPEED = 750 FT/SEC
5. OUT-OF-GROUND EFFECT

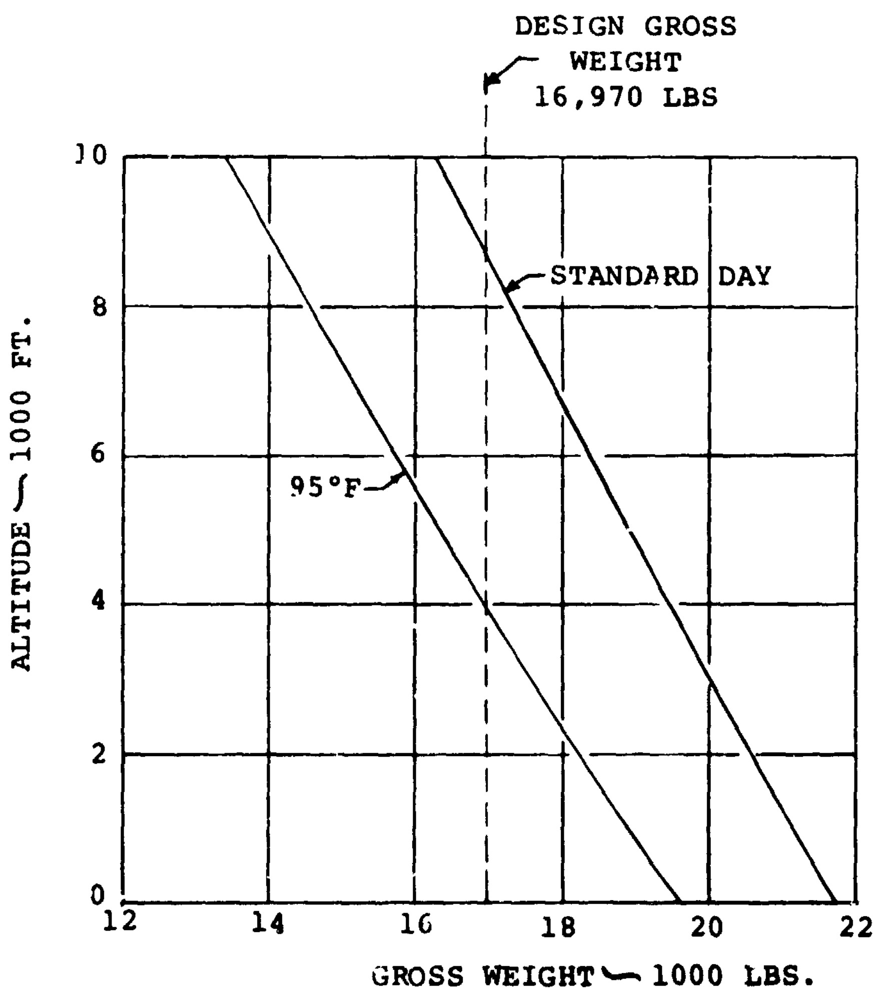


FIGURE 4-30: MODEL 222-1F USAF-SAR OGE HOVER CAPABILITY

GROSS WEIGHT = 16,970 LBS.

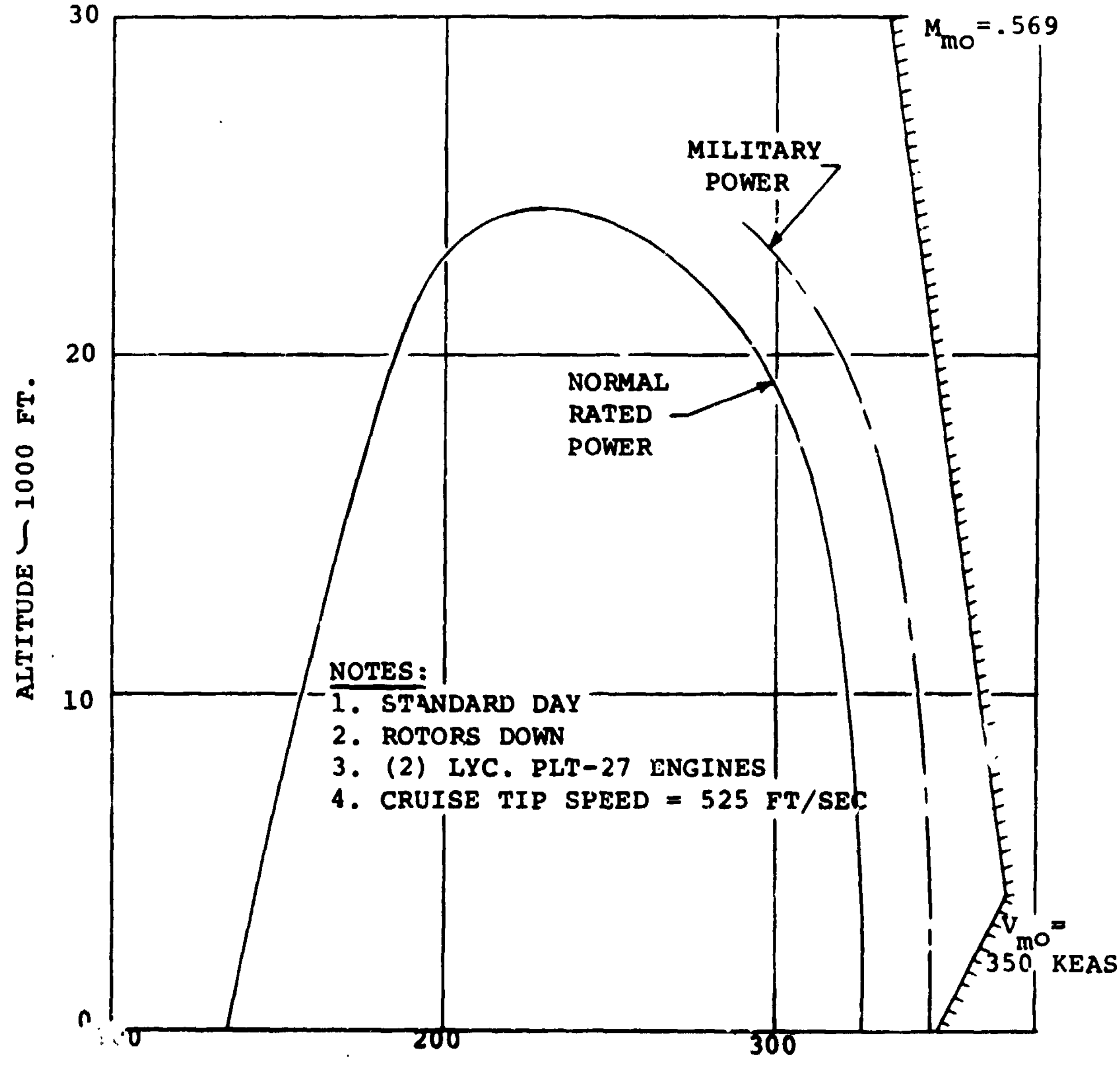


FIGURE 4-31: MODEL 222-1F USAF-SAR SPEED ALTITUDE CAPABILITY

4.8.2.3 STOL Performance. - Figure 4-32 presents the STOL performance of the USAF version of the tilt rotor. In addition to standard day performance, hot day (95°F) capability is also shown to demonstrate the performance capability of the vehicle operating in a more severe rescue environment. It is noted that even at this condition .e aircraft takeoff capability is substantially in excess of the design gross weight of 16,970 pounds. As noted, two sets of data are presented, one depicting maximum performance and the other showing normal takeoff performance.

Figure 4-33 presents useful load as a function of takeoff distance. As presented, the data does not reflect a specific mission profile as there is no accountability for fuel requirements.

4.8.2.4 Payload - Radius Performance. - The load carrying capability of the USAF SAR design point aircraft is shown in Figure 4-34. The useful load shown on this figure includes certain items of fixed equipment and fixed useful load as indicated on the vertical scale. This has been done to provide flexibility in selecting the desired combination of mission equipment and crew. This performance is shown for the high-low-low-high mission of Figure 4-35. Curves are shown for a mission in which no refueling is permitted and for a mission for which the aircraft is refueled on the return leg. In addition, the effect of flying the low leg at best range speed instead of normal rated power is shown. The initial takeoff is VTOL at sea level, 95°F. This figure shows that the SAR aircraft can pick up 3 rescuees at a radius of 500 nautical miles including a mid-point hover at 6,000 feet/95°F. An additional 4 rescuees can be picked up at 500-nautical mile radius with refueling on the return leg if the rescue altitude were 5,000 feet/95°F.

4.8.3 U. S. Navy Sea Control Aircraft.

4.8.3.1 Hover Ceiling. - Figure 4-36 presents out-of-ground effect hover capability for the proposed Navy Sea Control tilt-rotor aircraft. In addition to the standard day and 95°F day performance, the capability under tropical atmosphere conditions is also presented to reflect a more realistic Navy environment. The data reflects a T/W ratio of 1.1 to account for download and additional hover margin.

Figure 4-37 presents hover capability at sea level as a function of ambient temperature for two levels of thrust/weight ratio. The thrust/weight ratio of 1.05 reflects accountability for download with no additional margin on power.

4.8.3.2 Flight Envelope. - Figure 4-38 presents the normal power level flight capability for the Sea Control tilt

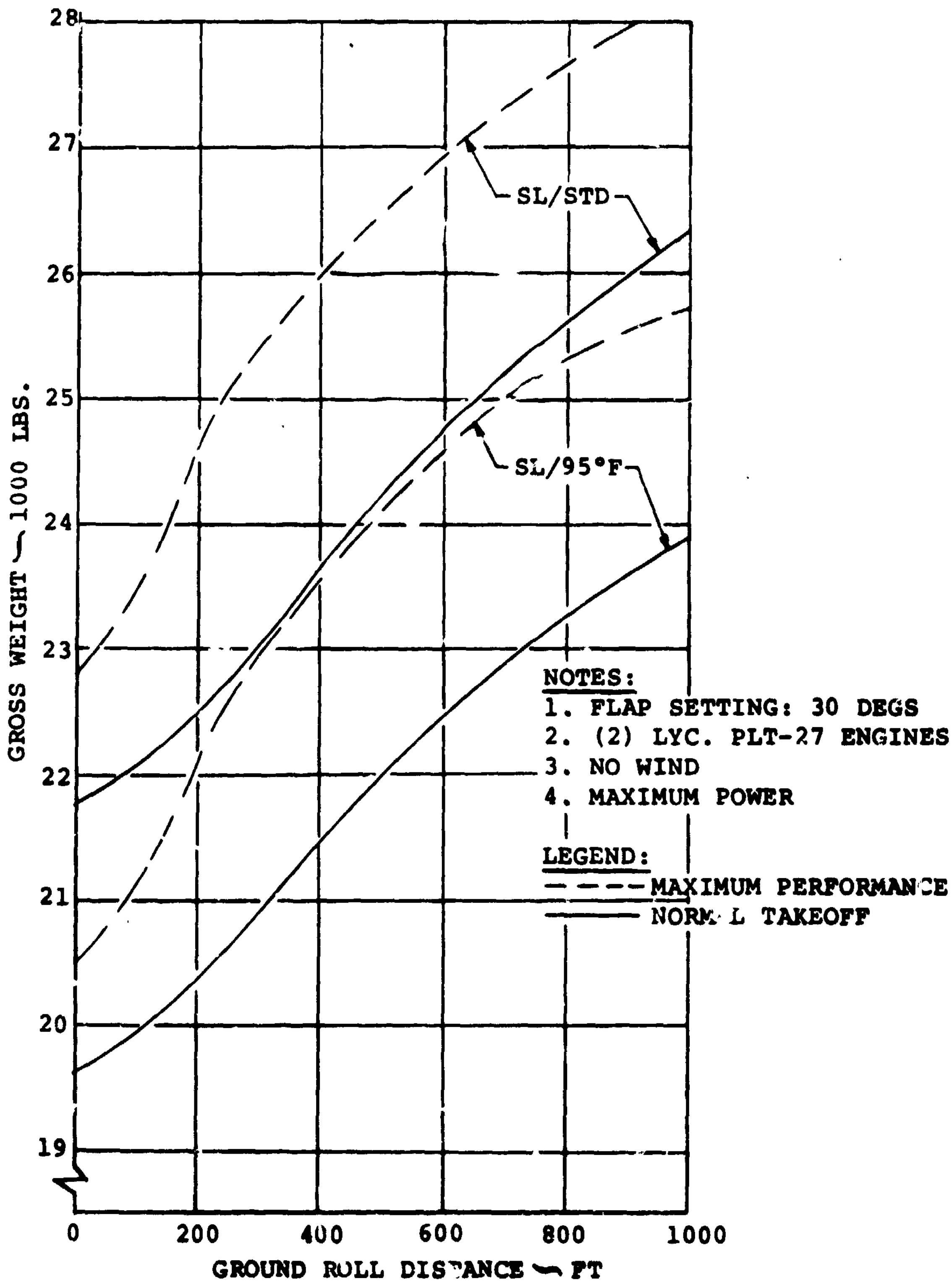


FIGURE 4-32: MODEL 222-1F USAF SAR - SHORT TAKEOFF PERFORMANCE - GROSS WEIGHT

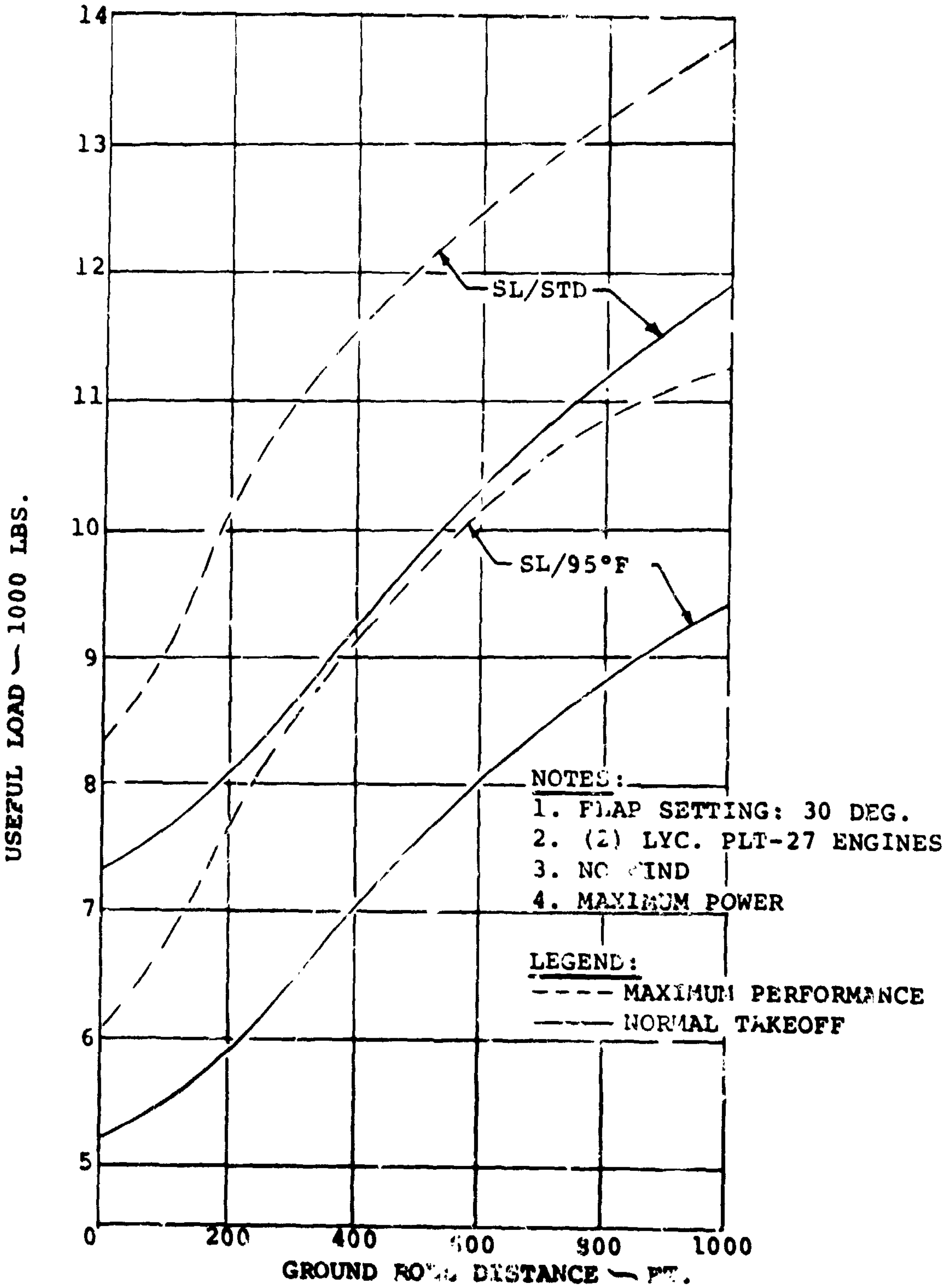


FIGURE 4-33: MODEL 222-1F USAF SAR
SHORT TAKEOFF PERFORMANCE - USEFUL LOAD

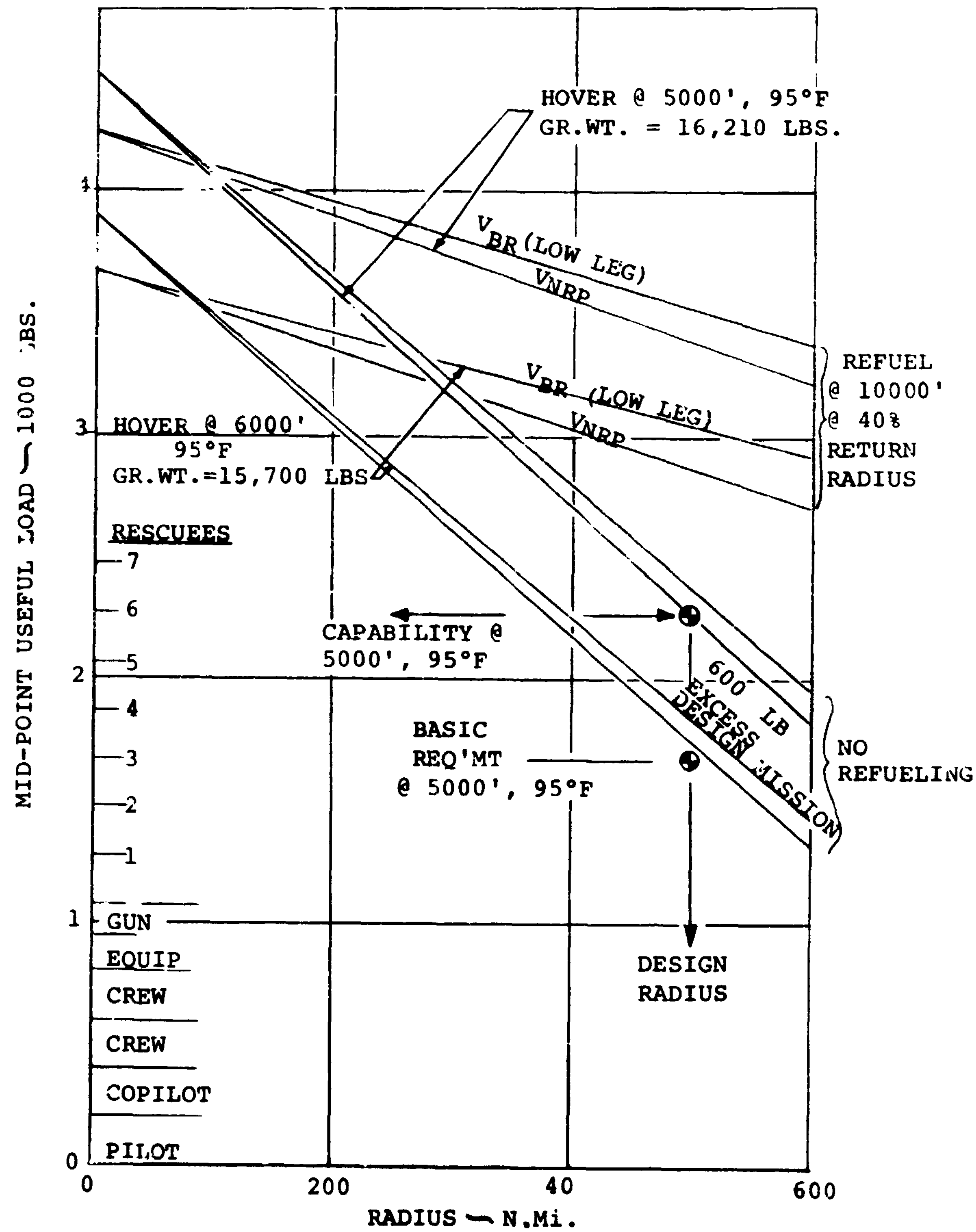
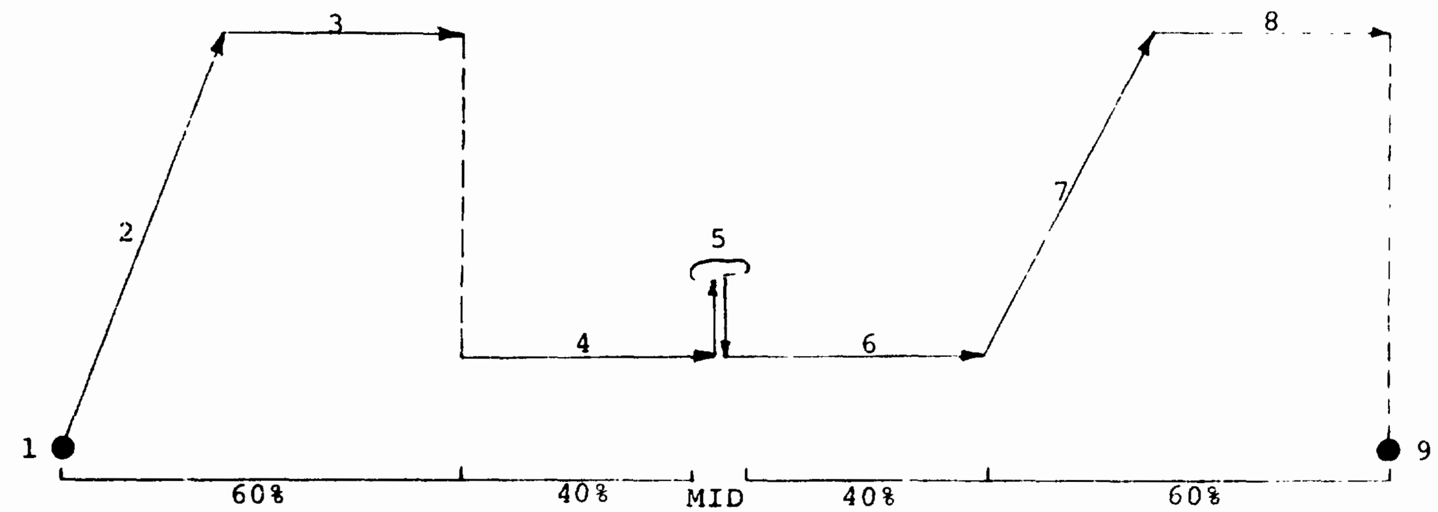


FIGURE 4-34: MODEL 222-1F USAF-SAR
MISSION LOAD CAPABILITY

C2



1. WARM UP, TAXI AND TAKEOFF: 3 MIN. @ NORMAL RATED POWER, SEA LEVEL, 95°F
2. CLIMB TO 20,000 FT @ MILITARY POWER AND SPEED FOR MAXIMUM RATE OF CLIMB
3. CRUISE OUTBOUND @ NORMAL RATED POWER
4. CRUISE @ 3000 FT, @ NORMAL RATED POWER, 95°F
5. HOVER 1/2 HR., EFFECT RESCUE OF 3 PEOPLE (600 LBS) @ 5000'/95°F
6. CRUISE @ 3000 FT @ NORMAL RATED POWER, 95°F
7. CLIMB TO 20,000 FT @ MILITARY POWER AND SPEED FOR MAXIMUM RATE OF CLIMB
8. CRUISE INBOUND @ NORMAL RATED POWER
9. LAND WITH 10% (INITIAL) FUEL RESERVE

NOTES:
1. MISSION FLOWN @ STANDARD ATMOSPHERE CONDITIONS UNLESS OTHERWISE NOTED.
2. SFC INCREASED 5% PER MIL-C-5011A.

FIGURE 4-35: USAF-SAR
HI-LO-LO-HI MISSION PROFILE

NOTES:
1. T/W = 1.1
2. MILITARY POWER
3. (2) LYC. PLT-27 ENGINES
4. HOVER TIP SPEED = 750 FT/SEC
5. OUT-OF-GROUND EFFECT

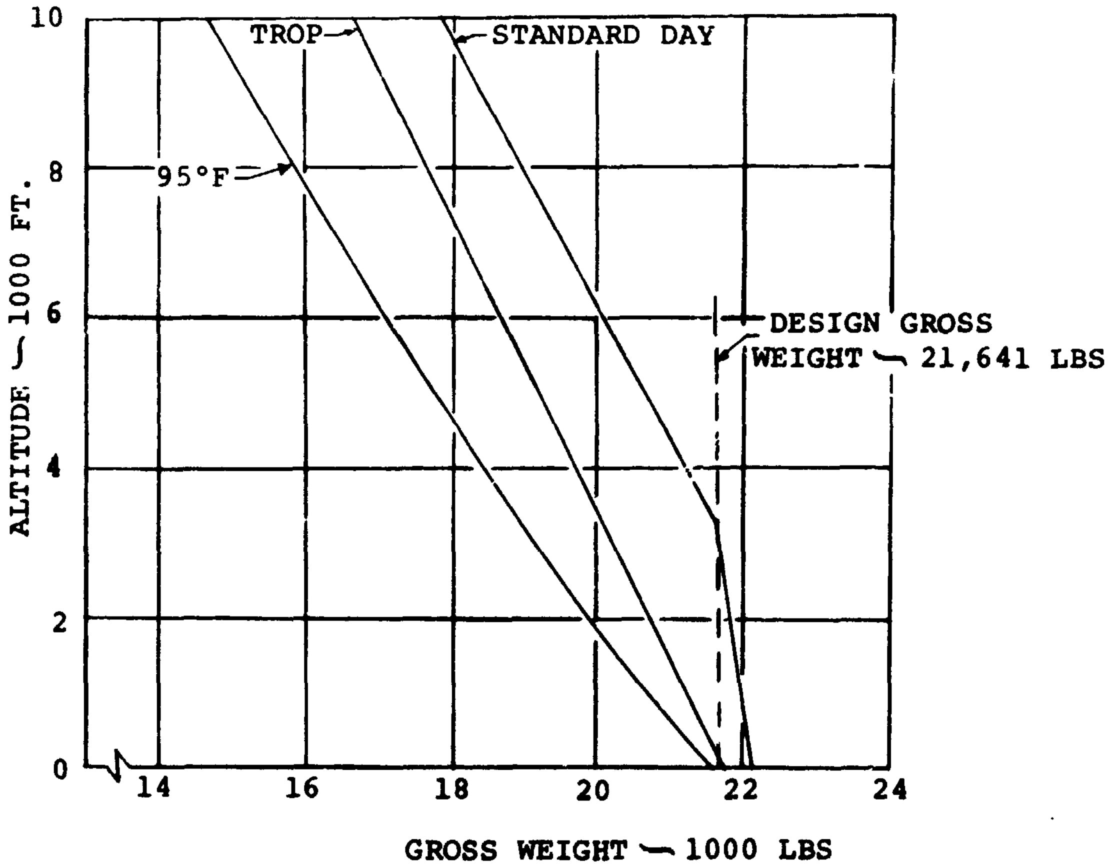


FIGURE 4-36: MODEL 222-1N NAVY-SEA CONTROL OGE HOVER ALTITUDE CAPABILITY

SEA LEVEL

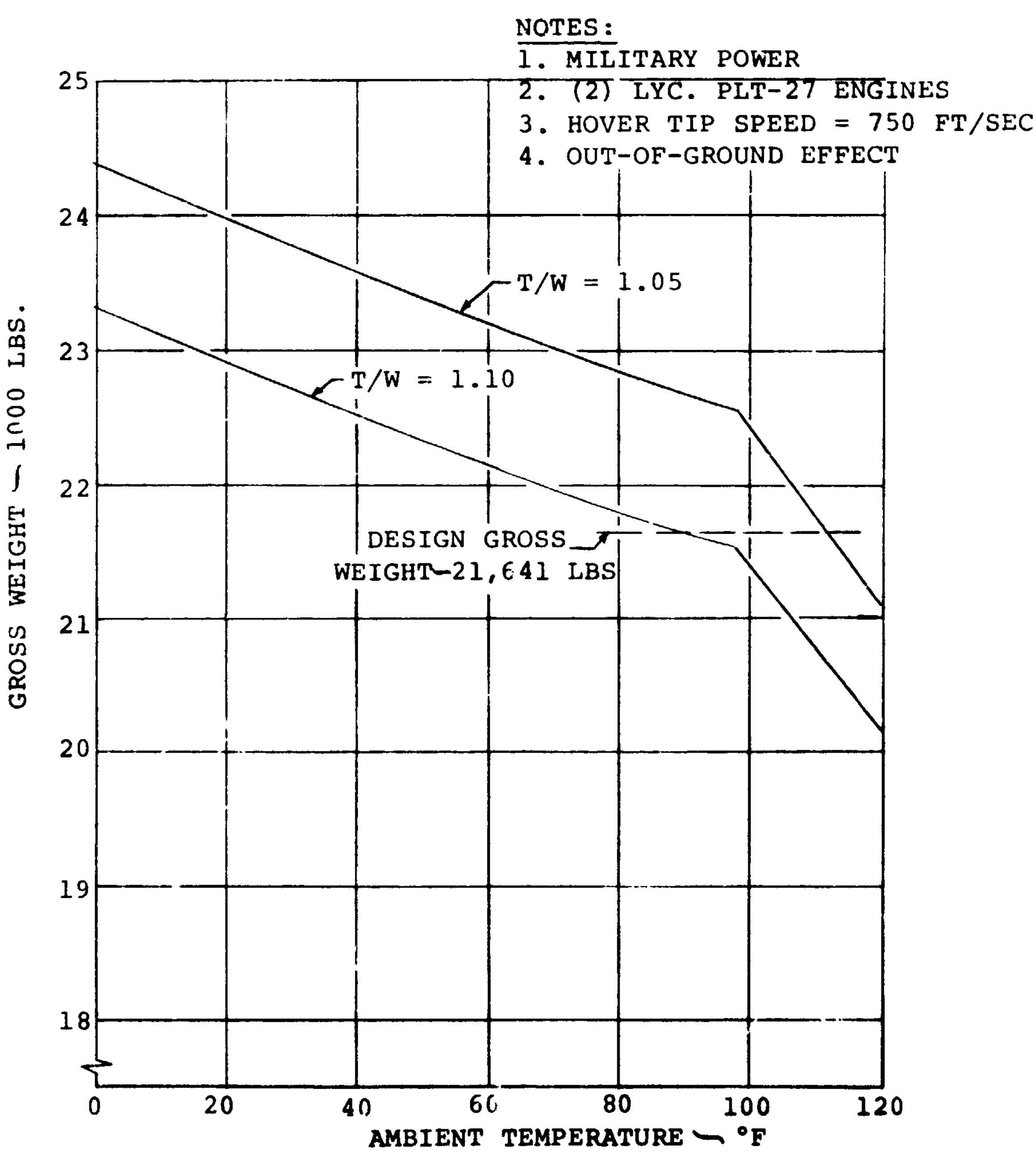


FIGURE 4-37: MODEL 222-1N NAVY - SEA CONTROL OGE HOVER CAPABILITY- SEA LEVEL

GROSS WEIGHT = 21,641 LBS

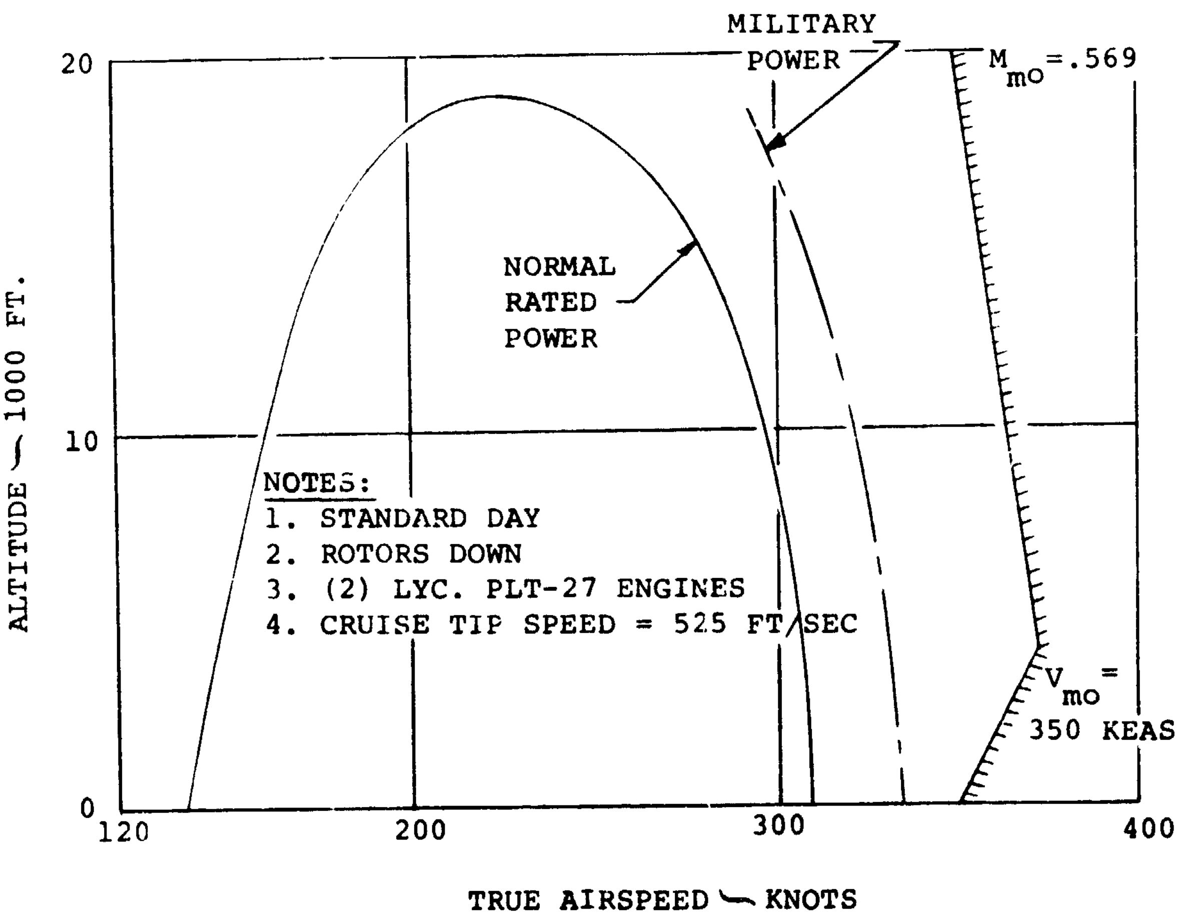


FIGURE 4-38: MODEL 222-1N NAVY - SEA CONTROL SPEED ALTITUDE CAPABILITY

rotor aircraft at the design gross weight. As noted, the speed capability exceeds 300 knots up to altitudes of 8,000 feet. The ceiling capability is of the order of 20,000 feet.

4.8.3.3 STOL Performance. - Figures 4-39 and 4-40 show the effect of ground roll distance on takeoff gross weight and useful load respectively for takeoffs at sea level, 90°F and with zero wind over deck.

Figure 4-41 shows the effect of wind over deck and takeoff ground roll on the useful load capability. This indicates that the 4,460-pound mission load of expendable stores, electronics, and torpedoes can be increased by 2,000 pounds (44 percent of design mission load) with 50 feet of ground roll and 30 knots of wind over deck.

4.8.3.4 Payload-Radius-Loiter Performance. - Figure 4-42 presents the ASW mission capability for the Navy Sea Control configuration of the tilt rotor. Performance is presented as a function of radius for a spectrum of mid-point loiter times. Dependent upon specific requirements at a given time, a flexible range of mission capability is available to the user. The design point superimposed upon the curve is based on a total mission time of 8 hours with 6.7 hours of loiter time. Figure 4-43 depicts the mission profile used in estimating the ASW mission performance.

4.8.4 Civil Off-Shore Oil Rig Support Aircraft.

4.8.4.1 Hover Ceiling. - Figure 4-44 presents the estimated OGE hover altitude capability of the civil version of the tilt-rotor vehicle for three ambient conditions. The data includes accountability for control power margin and download effects. In addition to the typical ambient conditions of ISA and 95°F, data is presented for a tropical atmosphere which is more representative of the environment of commercial operations.

Figure 4-45 presents the hover capability at sea level as a function of temperature. Data is presented for conditions of T/W = 1.05 and also for T/W = 1.1 which represents additional control margin. As noted, at the design gross weight, the aircraft is capable of hover operations at temperatures of 95°F or 106°F, depending upon which operating criteria is applied.

4.8.4.2 Flight Envelope. - Figure 4-46 presents the design gross weight, normal rated power, level flight speed capability for a standard atmosphere. As shown, the operating ceiling is in excess of 20,000 feet with an attendant speed capability of approximately 280 knots up to 10,000 feet.

4.8.4.3 STOL Performance. - Figure 4-47 presents sea

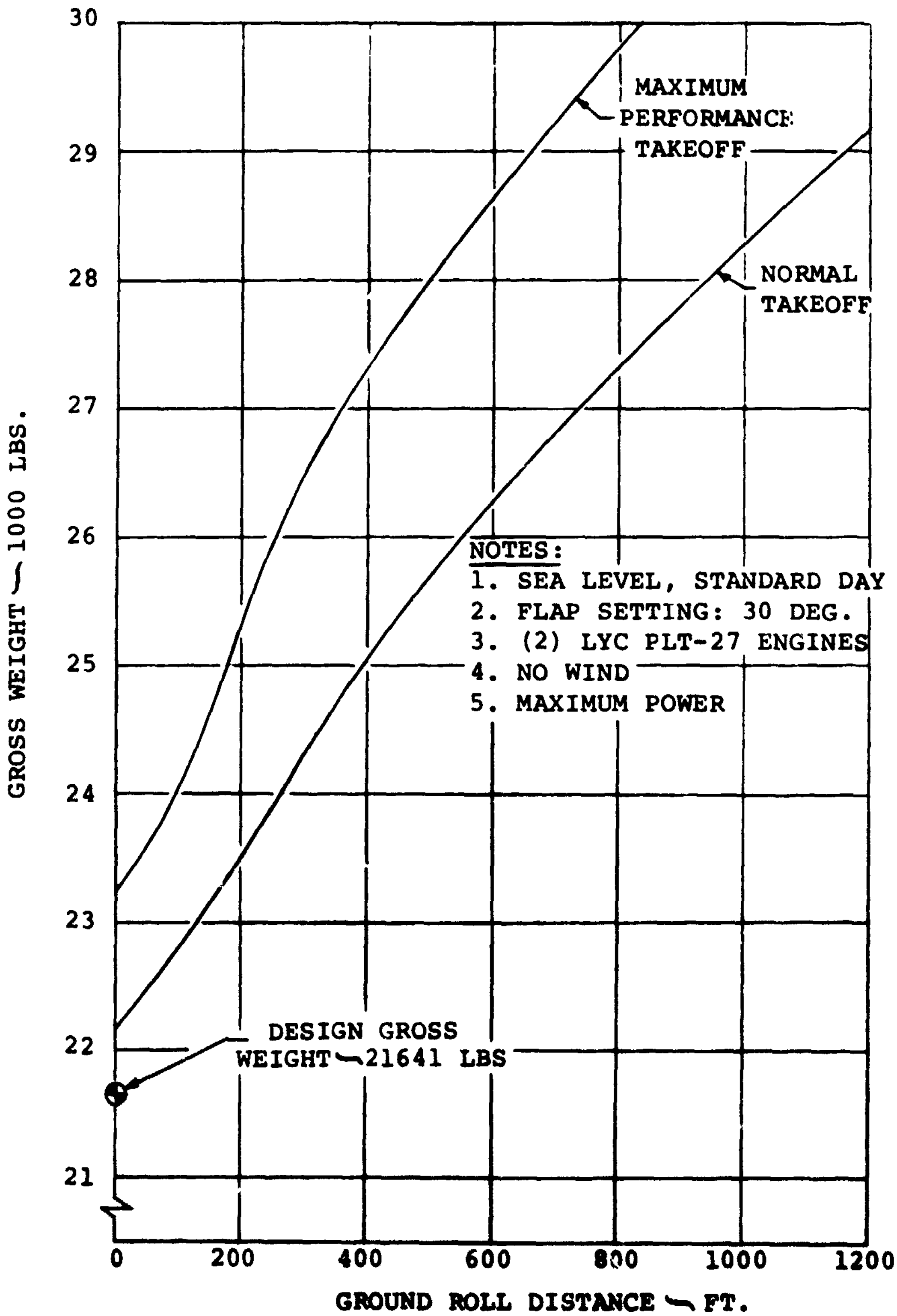


FIGURE 4-39: MODEL 222-1N NAVY - SEA CONTROL SHORT TAKEOFF PERFORMANCE - GROSS WEIGHT

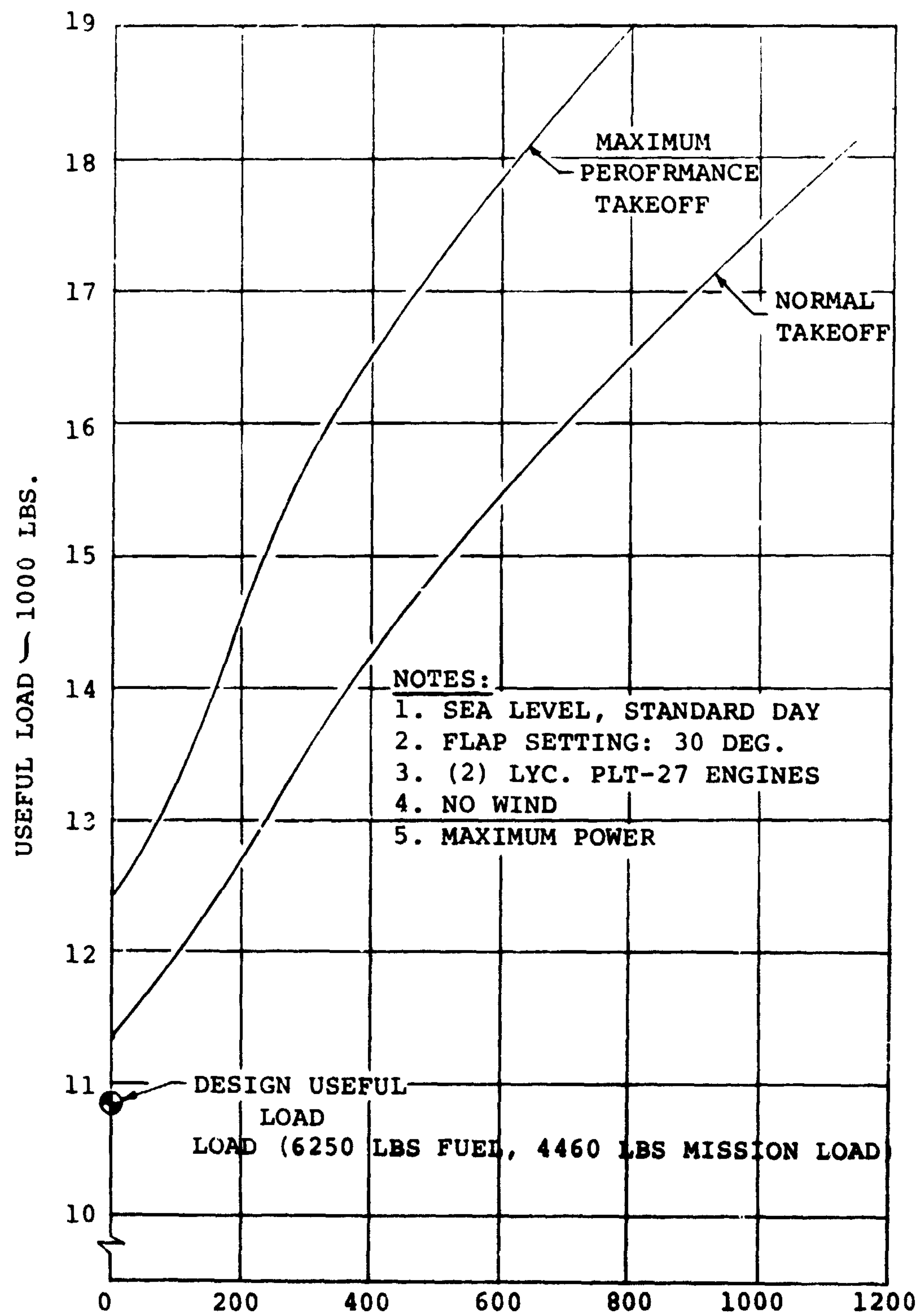


FIGURE 4-40: **MODEL 222-1N NAVY-SEA CONTROL SHORT TAKEOFF PERFORMANCE - USEFUL LOAD**

NOTES:
1. SEA LEVEL, STANDARD DAY
2. FLAP SETTING: 30 DEG.
3. (2) LYC. PLT-27 ENGINES
4. MAXIMUM POWER

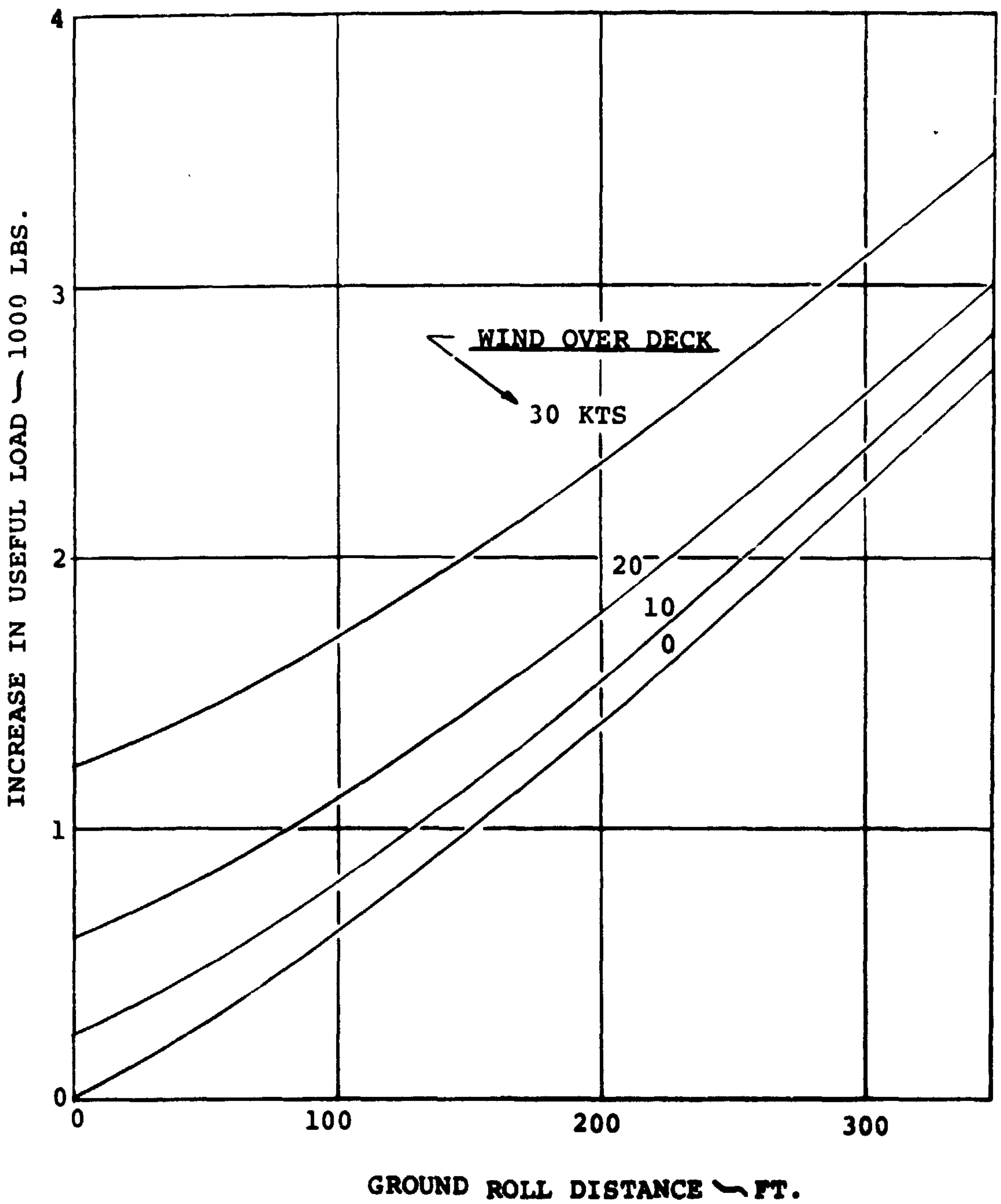


FIGURE 4-41: MODEL 222-1N NAVY - SEA CONTROL
EFFECT OF WIND ON STOL PERFORMANCE

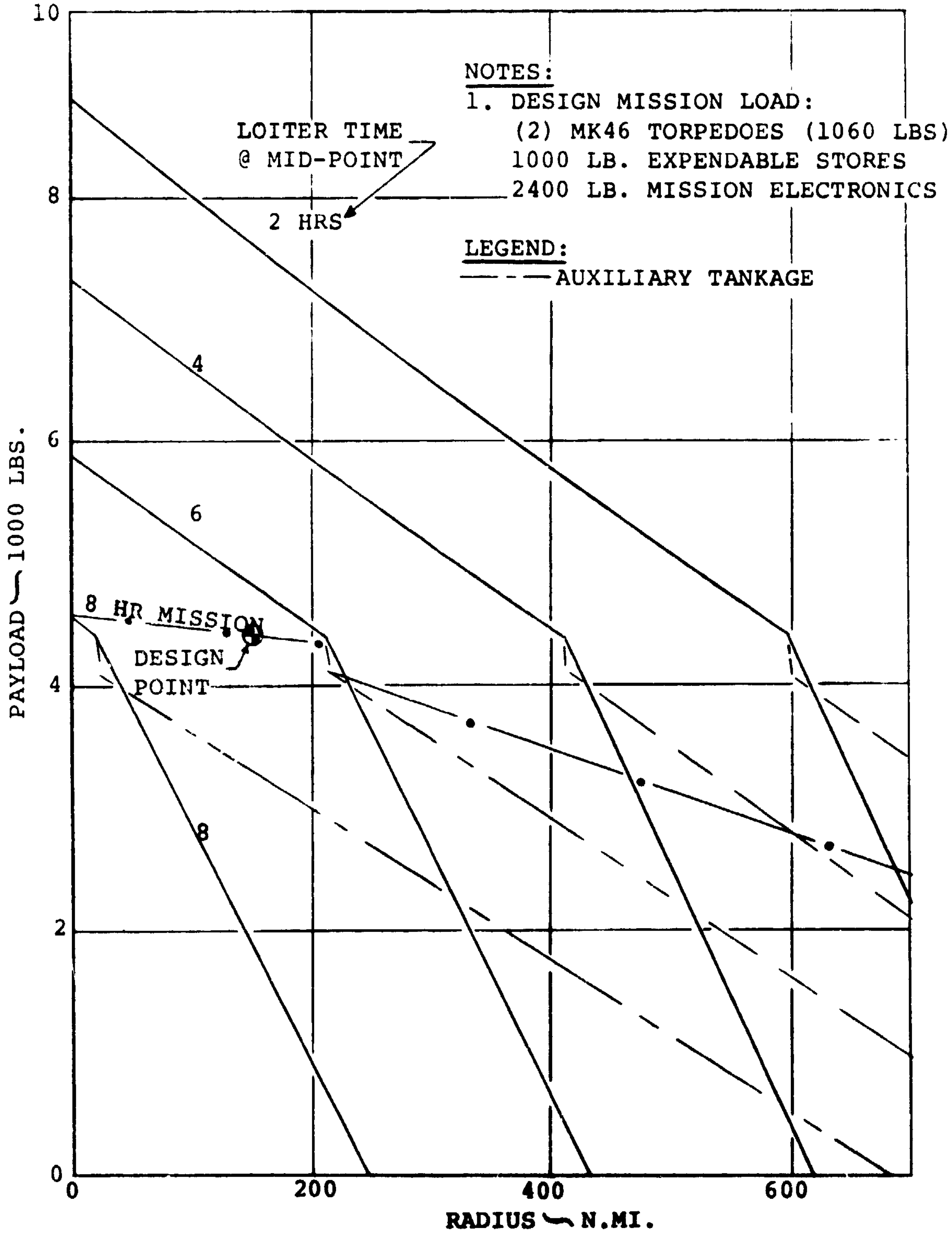
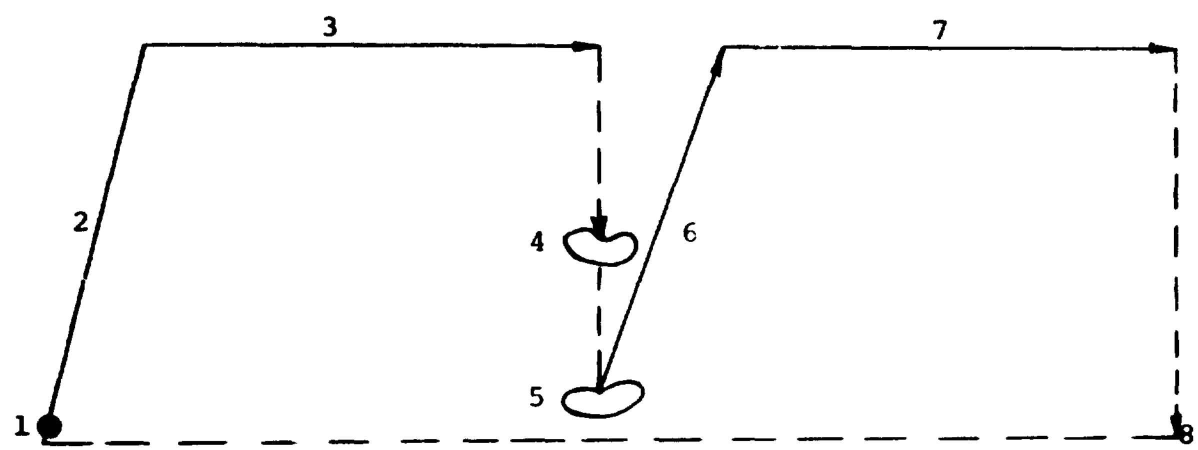


FIGURE 4-42: MODEL 222-1N NAVY - SEA CONTROL ASW MISSION CAPABILITY



1. WARM-UP, TAXI AND TAKEOFF: 2 MIN. @ NORMAL RATED POWER, SEA LEVEL, 90°F
2. CLIMB TO 10,000 FT @ MILITARY POWER AND SPEED FOR MAX. RATE OF CLIMB
3. CRUISE OUTBOUND @ SPEED FOR 99% BEST RANGE
4. LOITER @ 5000 FT.
5. LOITER @ 500 FT. FOR 15 MINUTES, FOR COMBAT
6. CLIMB TO 10,000 FT. @ MILITARY POWER AND SPEED FOR MAX. RATE OF CLIMB
7. CRUISE INBOUND @ SPEED FOR 99% BEST RANGE
8. LAND WITH 10% (INITIAL) FUEL RESERVE

NOTES:
1. MISSION FLOWN @ STANDARD ATMOSPHERE CONDITIONS UNLESS OTHERWISE NOTED
2. SFC INCREASED 5% PER MIL-C-5011A

FIGURE 4-43: MODEL [illegible]-1N NAVY-SEA CONTROL ASW MISSION PROFILE

NOTES:
1. T/W = 1.1
2. MILITARY POWER
3. (2) P&W PT6C-40 ENGINES
4. HOVER TIP SPEED = 750 FT/SEC
5. OUT-OF-GROUND EFFECT

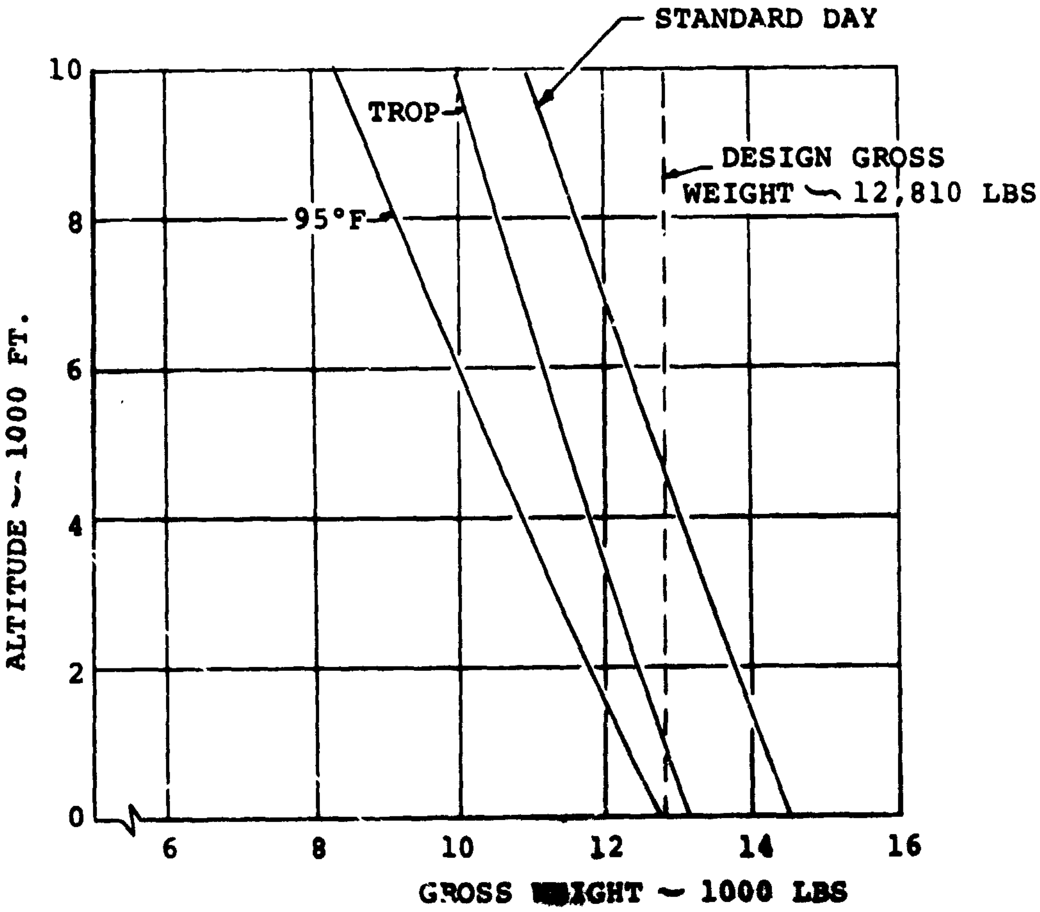


FIGURE 4-44: MODEL 222-1C CIVIL-OFFSHORE OIL OGE HOVER ALTITUDE CAPABILITY

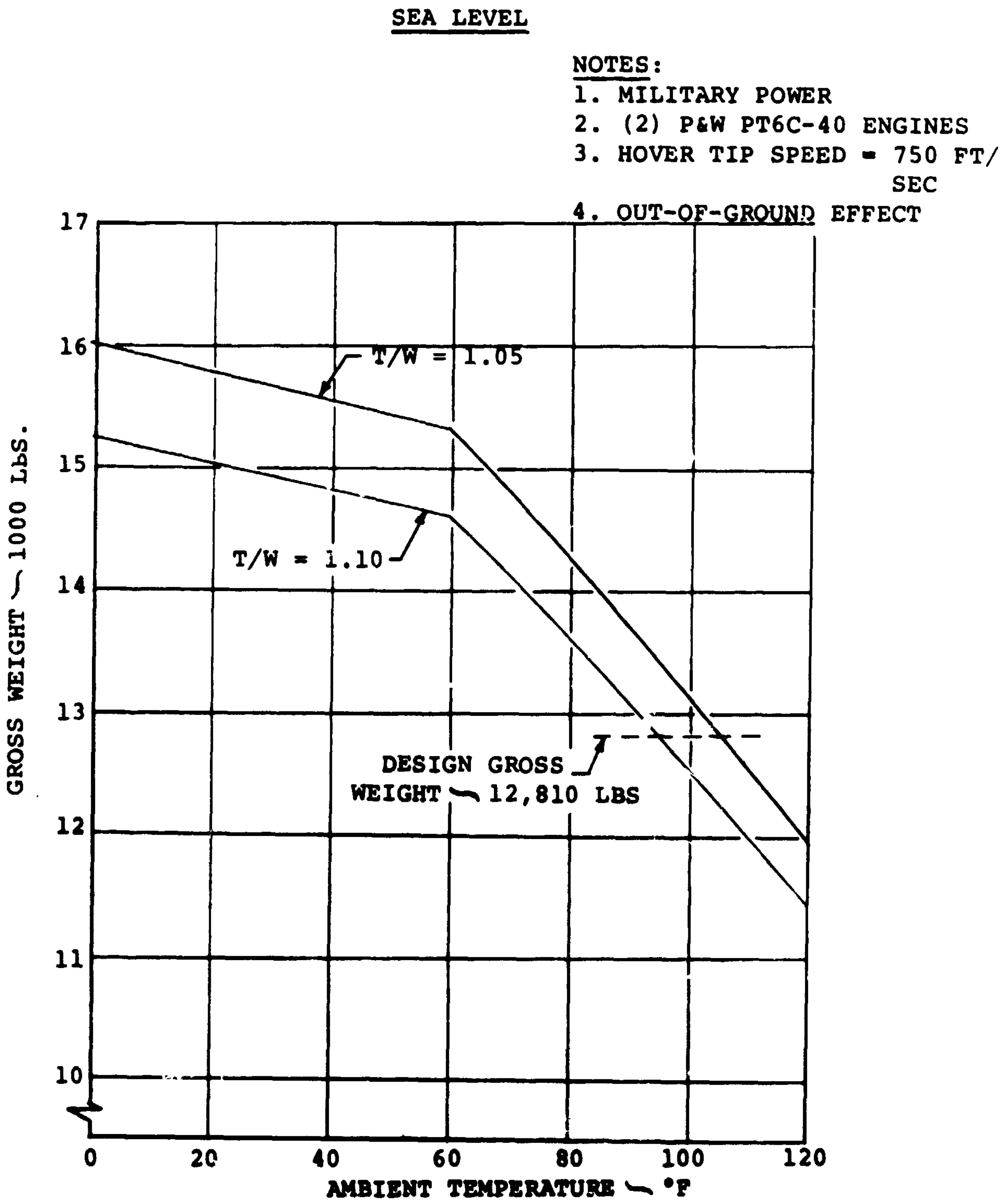


FIGURE 4-45: MODEL 222-1C CIVIL-OFFSHORE OIL OGE HOVER CAPABILITY-SEA LEVEL

GROSS WEIGHT ~ 12,810 LBS.

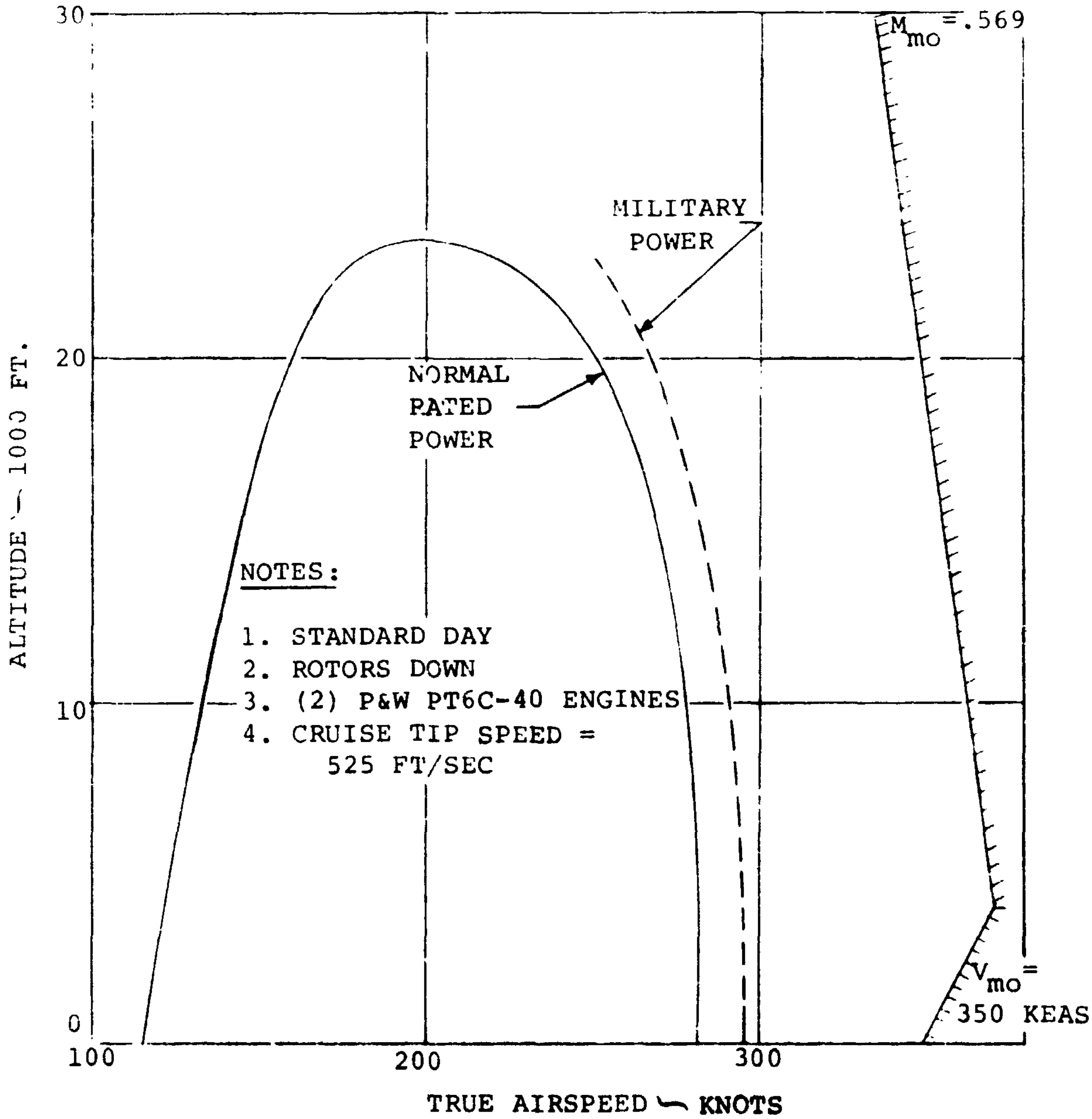


FIGURE 4-46: MODEL 222-1C CIVIL-OFFSHORE OIL SPEED ALTITUDE CAPABILITY

B

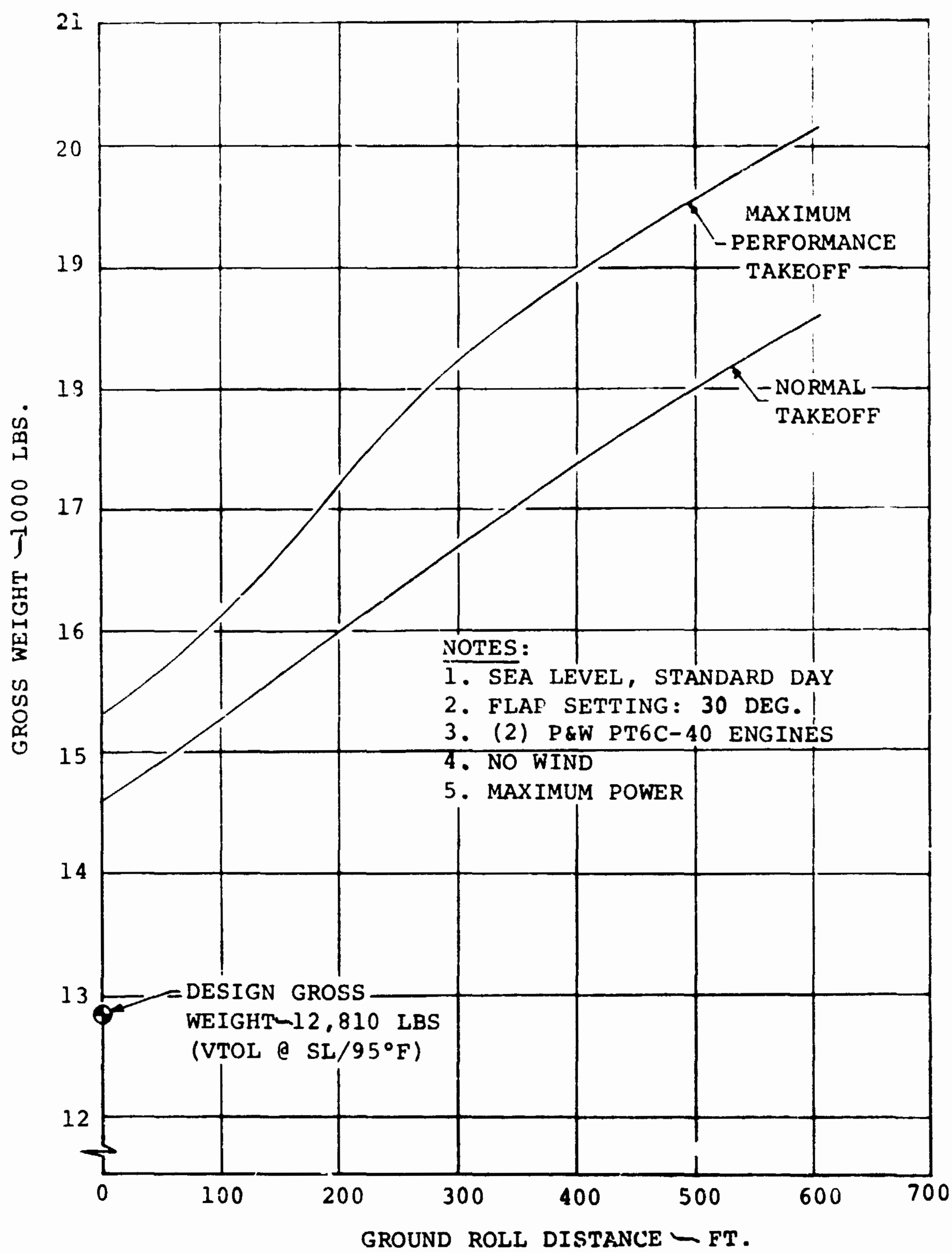


FIGURE 4-47: MODEL 222-1C CIVIL-OFFSHORE OIL SHORT TAKEOFF PERFORMANCE - GROSS WEIGHT

level standard day takeoff capability for maximum performance and normal performance takeoffs. Superimposed on the plot is the design point gross weight. Figure 4-48 depicts the useful load capability which is available as a function of takeoff distance.

4.8.4.4 Payload-Radius Performance. - Figure 4-49 presents the mission capability for the civil off-shore oil configuration of the tilt rotor. The data is based on hovering OGE at sea level, 95°F. The design point condition depicted matches the requirement to transport 12 passengers 125 statute miles (109 nautical miles). This requires less fuel than the full tank capacity of 2,000 pounds. For extended range, payload can be traded for fuel. Figure 4-50 depicts the mission profile used to estimate the performance presented in Figure 4-49 for a typical off-shore oil operation.

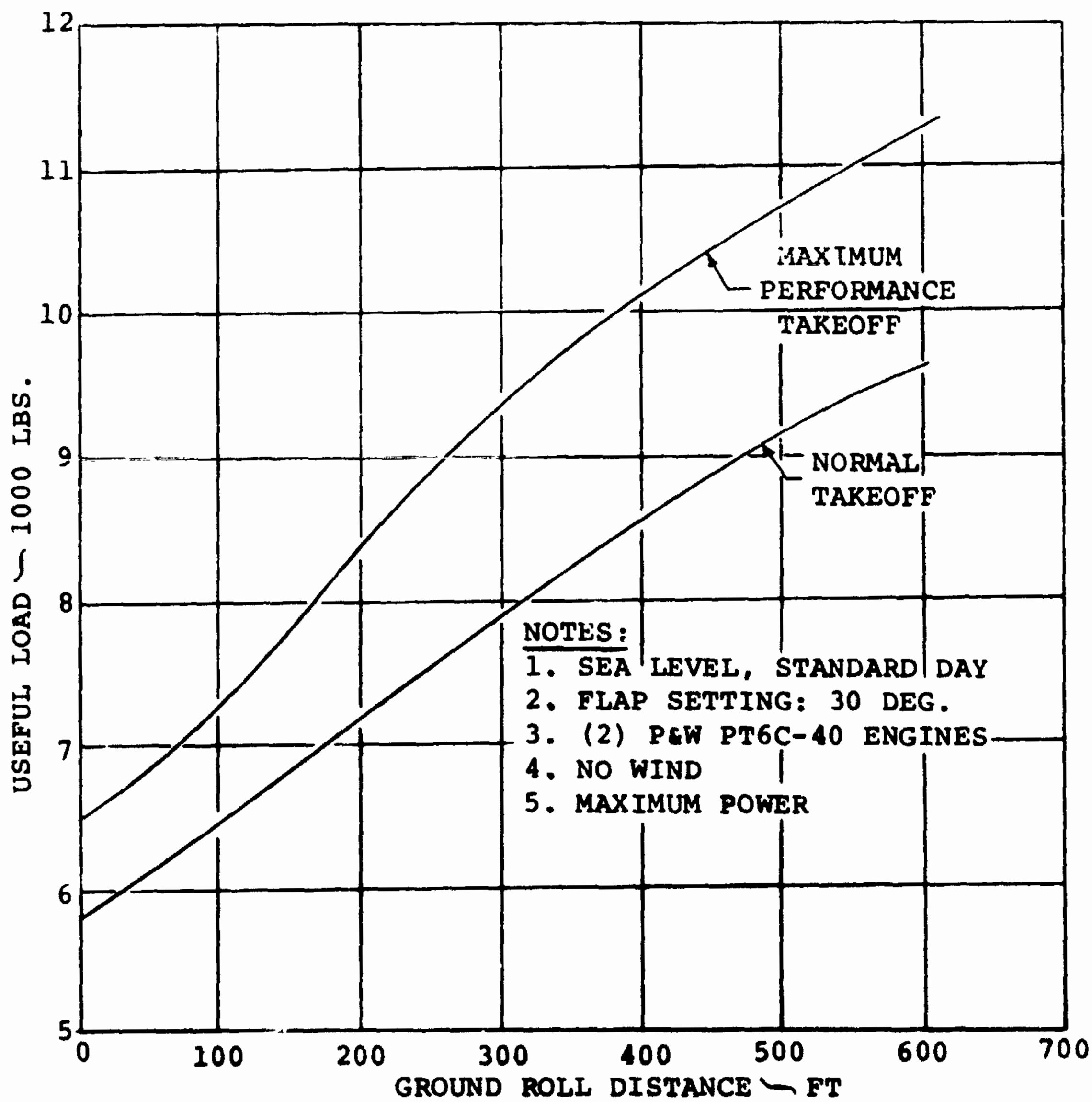


FIGURE 4-48: MODEL 222-1C CIVIL-OFFSHORE OIL SHORT TAKEOFF PERFORMANCE - USEFUL LOAD

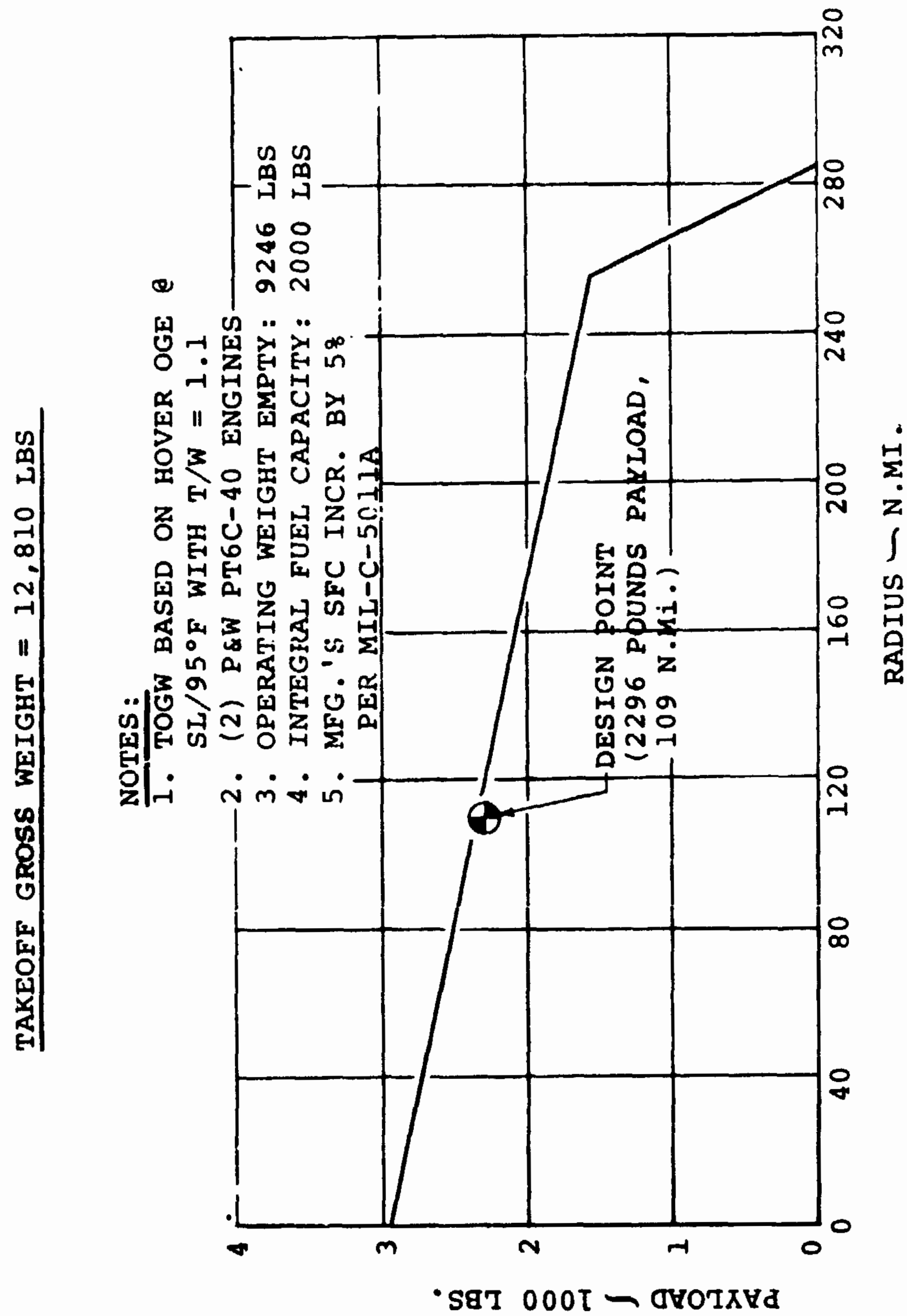
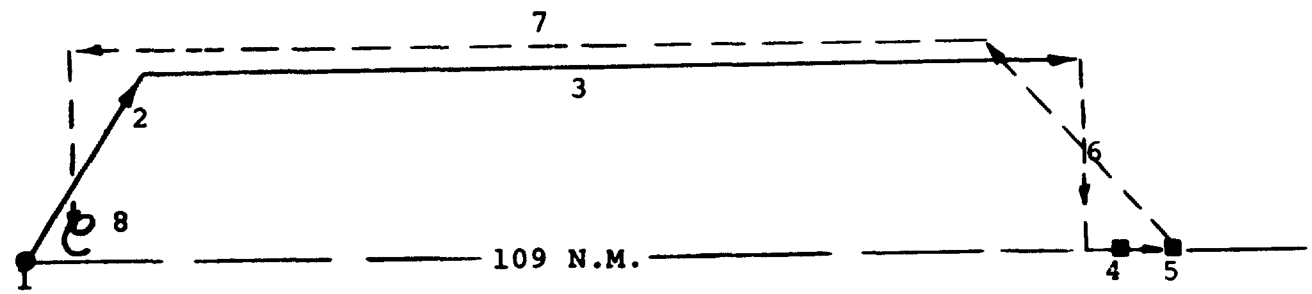


FIGURE 4-49: MODEL 222-1C CIVIL-OFFSHORE OIL MISSION CAPABILITY



1. WARM-UP, TAXI AND TAKEOFF: 1 MIN @ MAXIMUM POWER, SEA LEVEL 95°F
2. CLIMB TO 20,000 FT @ MILITARY POWER AND SPEED FOR MAX. RATE OF CLIMB
3. CRUISE OUTBOUND @ NORMAL RATED POWER
4. LANDING: 1 MIN @ MAXIMUM POWER, SEA LEVEL 95°F
5. TAKEOFF: 1 MIN @ MAXIMUM POWER, SEA LEVEL 95°F
6. CLIMB TO 20,000 FT @ MILITARY POWER AND SPEED FOR MAX. RATE OF CLIMB
7. CRUISE INBOUND @ BEST RANGE SPEED
8. LAND WITH 1/2 HOUR FUEL RESERVE @ MAXIMUM ENDURANCE SPEED

NOTES:

1. MISSION FLOWN @ STANDARD ATMOSPHERE CONDITIONS UNLESS OTHERWISE NOTED
2. SFC INCREASED 5% PER MIL-C-5011A

FIGURE 4-50: MODEL 222-1C CIVIL-OFFSHORE OIL OFFSHORE OIL MISSION PROFILE

5.0 TECHNICAL ASSESSMENT AND RECOMMENDATIONS FOR ADDITIONAL RESEARCH

5.1 Technology Status

During the past six years, Boeing has carried out an intensive program for development of tilt-rotor technology. The philosophy of this program has been the concurrent development of analytical methodology and wind tunnel model investigation so that the analytical methods could be used to plan logical and productive wind tunnel programs and the wind tunnel tests could be used to validate and improve the analytical methodology. During the course of this program, over 3,500 hours of testing has been carried out on 25 models. One important feature of the model test program was the development of the technology for designing, building, and running dynamically-scaled models because of the importance of dynamic phenomena on the tilt-rotor configuration. Over 1,000 hours of the model test program were devoted to the testing of 9 dynamically-scaled models. The following paragraphs give a brief summary of some of the more important technical areas which have been explored in the model test program.

a. Rotor Performance - Rotor performance in both hover and cruise modes has been investigated by full-scale tipspeed tests on 5-foot and 13-foot-diameter models under both NASA and Boeing sponsorship. Correlation with predictions in the hover mode is good, as shown in Figure 5-1, and in the cruise mode is also good, as shown in Figure 5-2, except that at Mach numbers of 0.6 and above, considerable differences have been found between data from different tests. It is expected that this will be resolved by a NASA-sponsored program for additional testing on the 13-foot models.

b. Download - Download reduction devices, consisting of large deflection trailing edge flaps and leading edge umbrellas, have been developed by both model and full-scale testing. These reduced the download on the wing from about 13 percent for an unflapped wing to less than 5 percent with flaps and umbrellas operative, as shown in Figure 5-3.

c. Aeroelastic Stability - Aeroelastic stability boundaries have been explored on both semi-span and full-span dynamic models. A semi-span windmilling model using a 5.5-foot-diameter rotor is shown in Figure 5-4 and the correlation of whirl flutter boundaries with predictions is shown in Figure 5-5. Another model, which is dynamically-scaled from the 26-foot-diameter flightworthy rotor now under construction for NASA, is shown

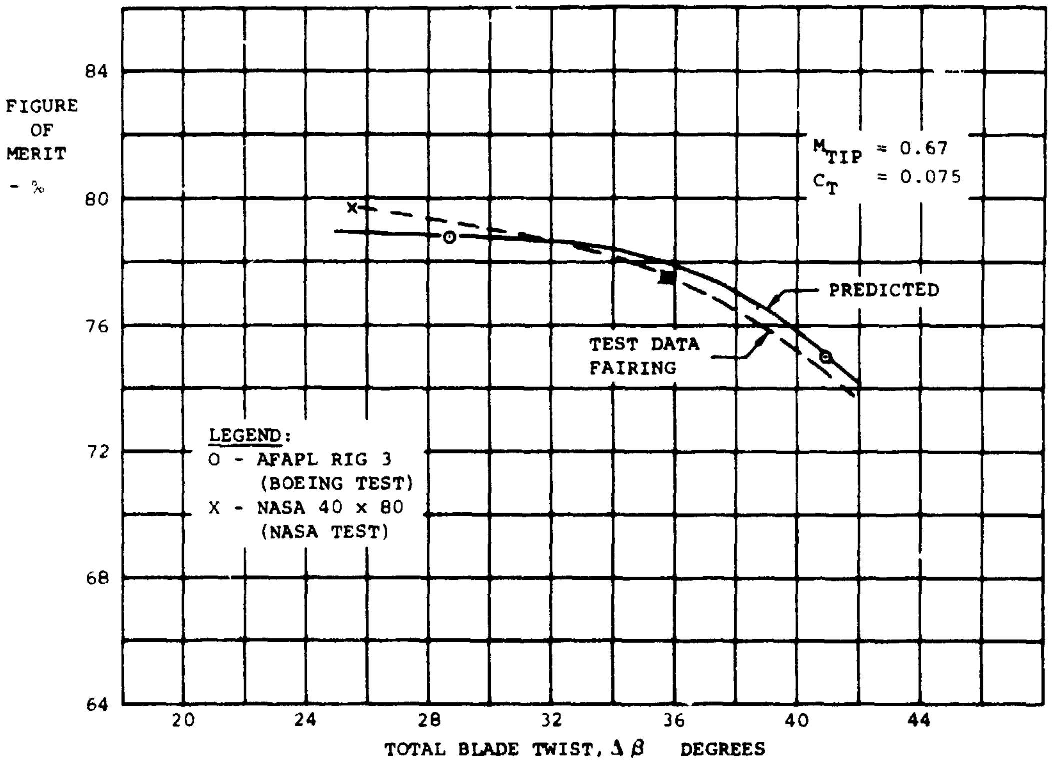


FIGURE 5-1: HOVER PERFORMANCE CORRELATION

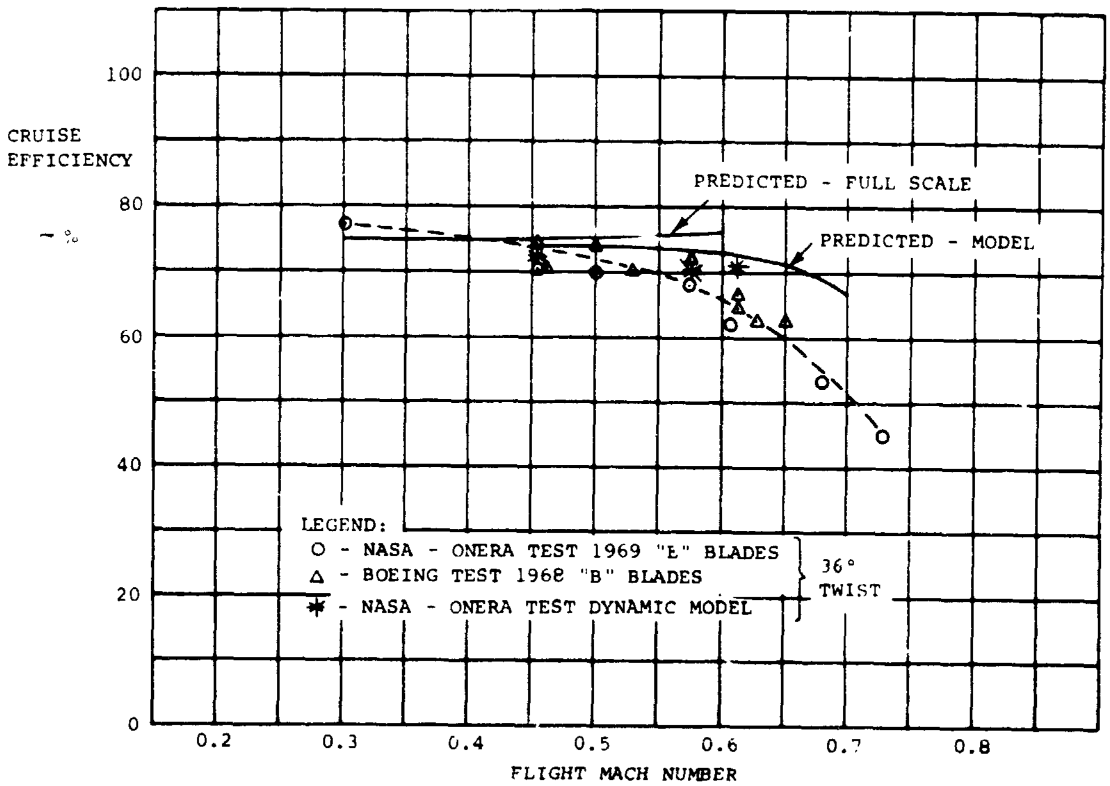


FIGURE 5-2: CRUISE PERFORMANCE CORRELATION

HOVER DOWNLOAD

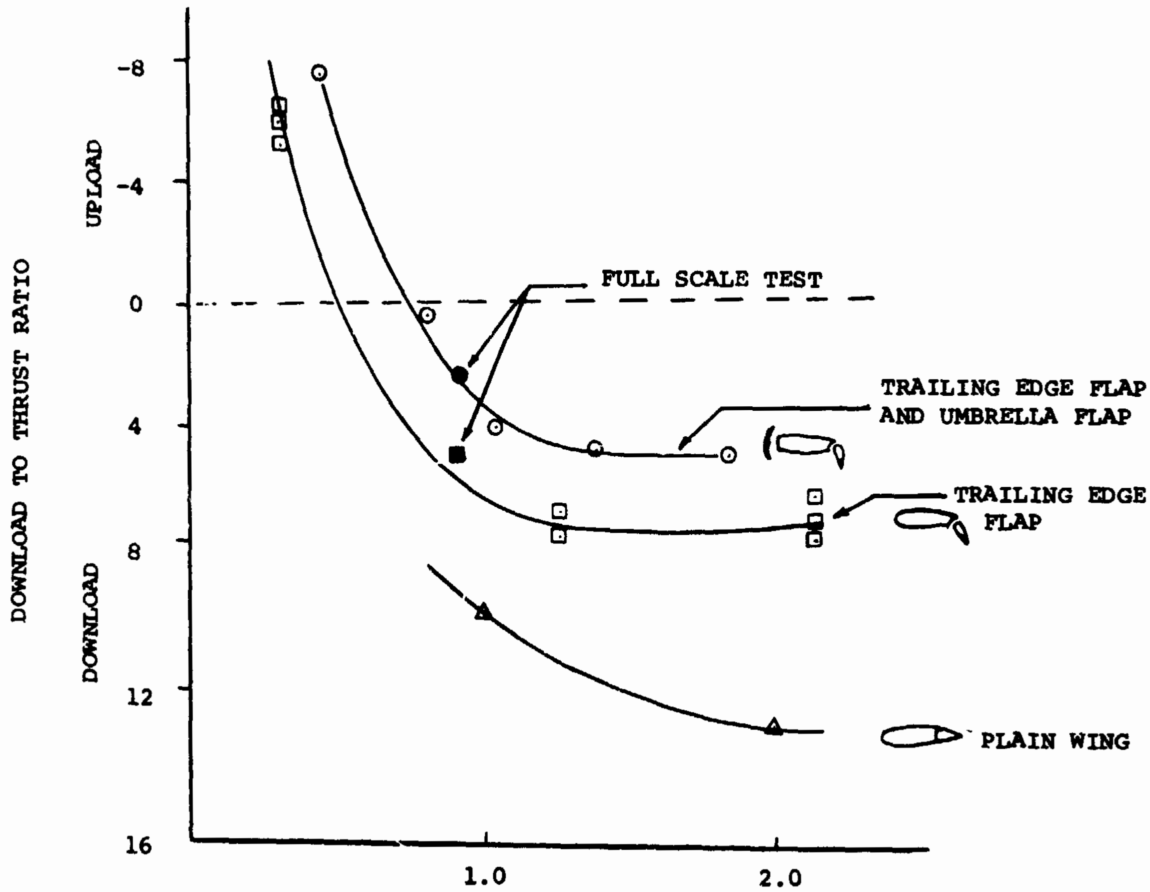


FIGURE 5-3: ROTOR HEIGHT/DIAMETER RATIO

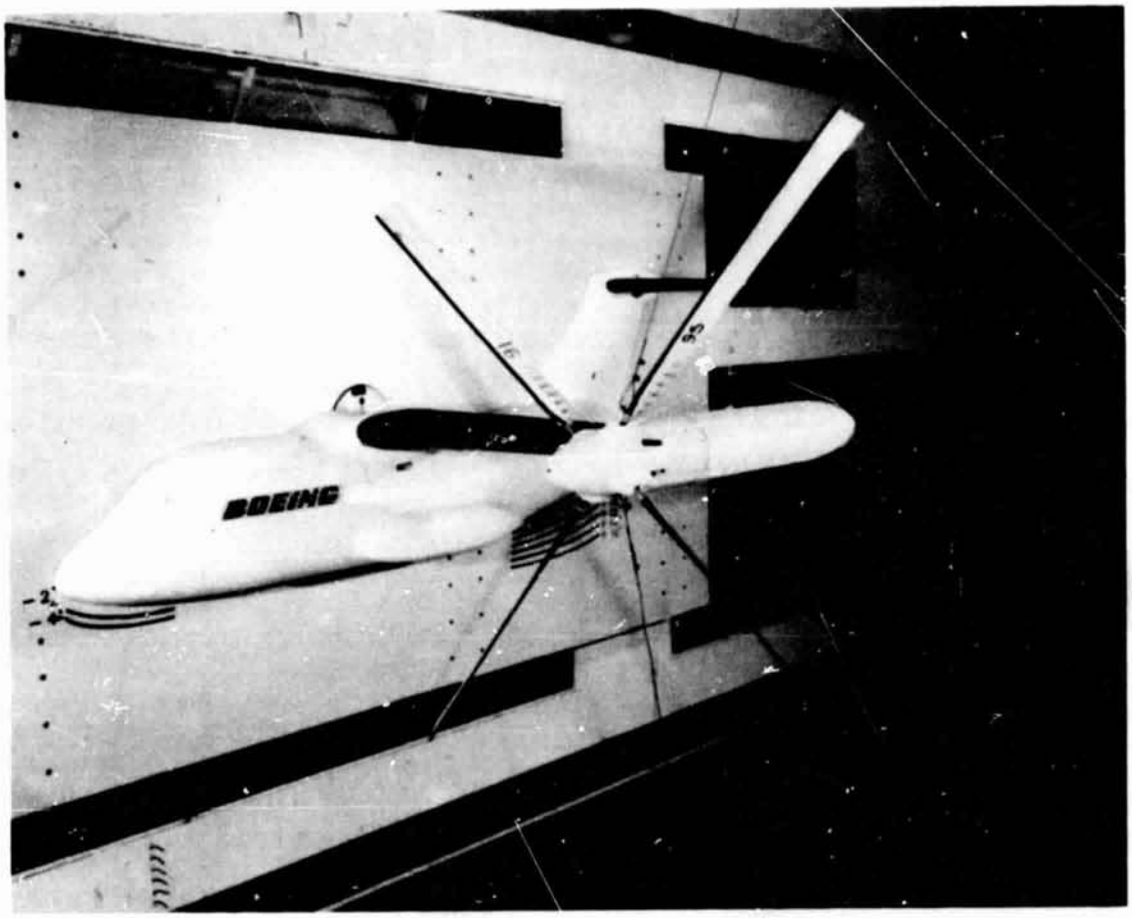


FIGURE 5-4: SEMI-SPAN WINDMILLING MODEL WITH 5.5' DIAMETER ROTOR

1/9 SCALE CONVERSION MODEL

1/4 STIFF WING SPAR IN TORSION
WINDMILLING CONDITION

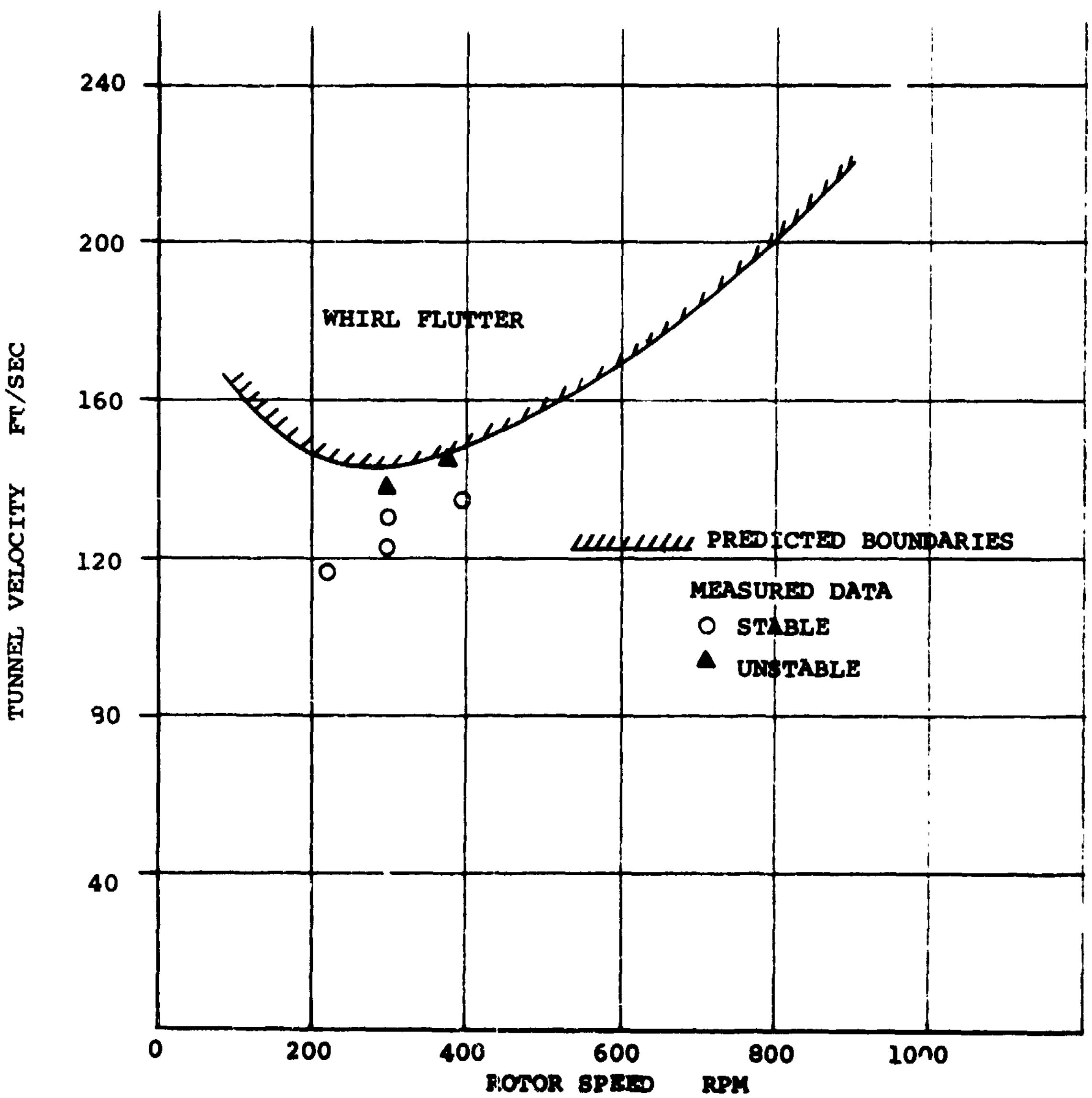


FIGURE 5-5: WHIRL FLUTTER CORRELATION

in Figure 5-6. A correlation of the damping in the air resonance mode obtained on this model is shown in Figure 5-7.

d. Flying Qualities - A full-span dynamically-scaled model of the tilt rotor, gimbal-mounted on a monkey pole to provide 4 degrees of rigid body freedom in addition to its elastic motions, was tested in the Boeing-Vertol V/STOL tunnel. This model, shown in Figure 5-8, was used to explore both the aeroelastic characteristics and the flying qualities of the aircraft, since it could be disturbed and time histories obtained of resulting rigid body and flexible motions.

   Additional flying qualities data, particularly on rotor derivatives, was obtained from the semi-span dynamic models discussed previously. The rotor derivatives obtained by test correlate well with the theory developed which predicts the effects of both flapwise and lagwise flexibility of the blades. Correlation of predicted and measured pitching moment derivatives is shown in Figure 5-9.

e. Blade Loads - Variation of blade loads throughout the various regimes of flight has been explored on all the models tested. New analyses allowing for the high blade twist and skewed flow encountered in tilt rotors provide improved correlation with test data as shown in Figure 5-10.

   A brief description of some of the major computer programs available at Boeing for the analysis of aerodynamic flying qualities, designs, and blade loads is given in Table 5-1.

5.2 Areas for Additional Research

Based on the technology development summarized in the preceding paragraphs, Boeing considers that the technology is now in hand to start on the development of the research aircraft. However, there are certain areas in which additional research would be desirable. These areas can be divided into four categories:

1. Work which would minimize development time and cost in the wind tunnel and flight test of the research aircraft.

2. Areas which required exploration or substantiation on the research aircraft, either in the wind tunnel or in flight.

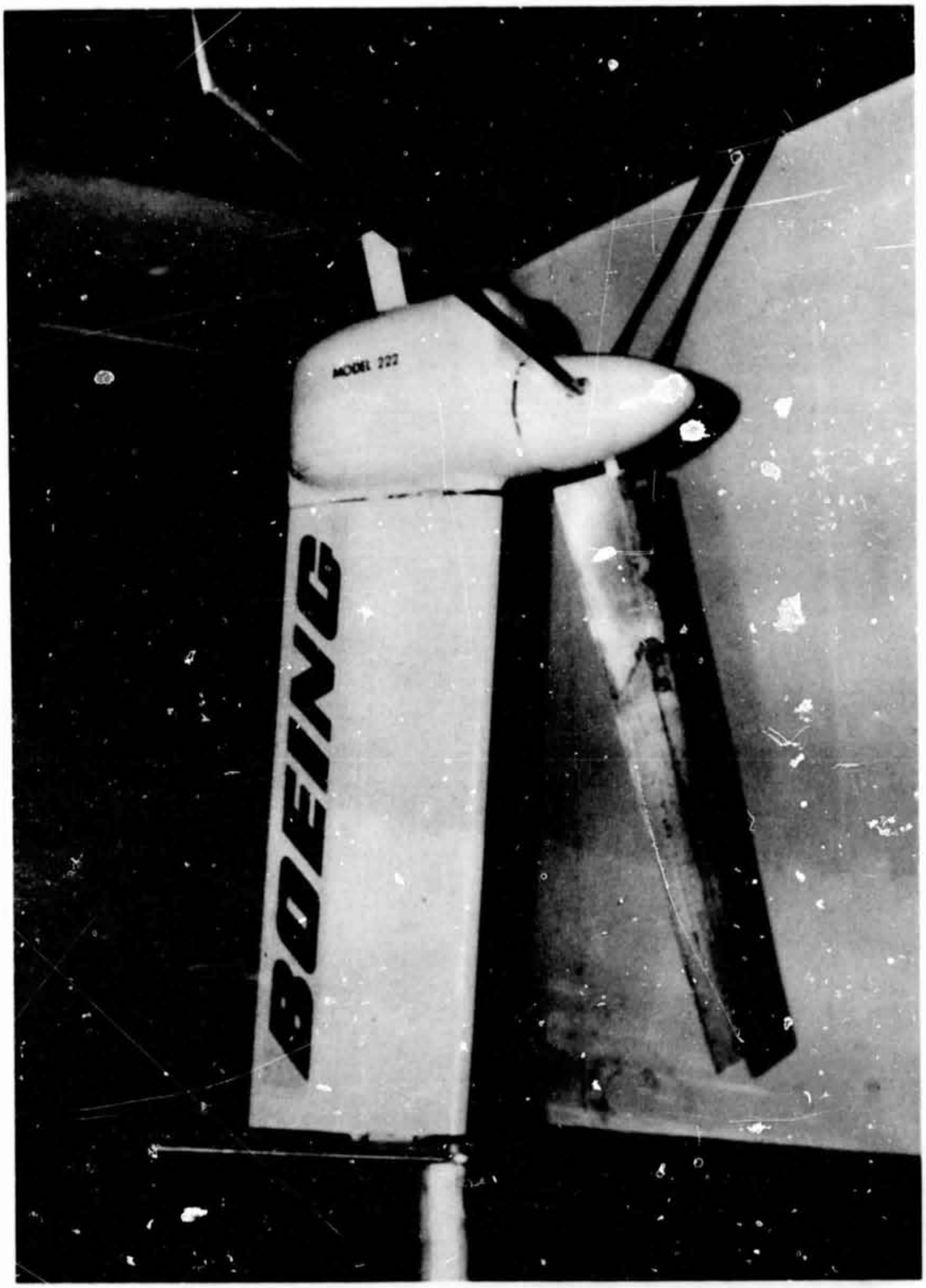


FIGURE 5-6: 1/9.244 DYNAMICALLY SCALED MODEL 222

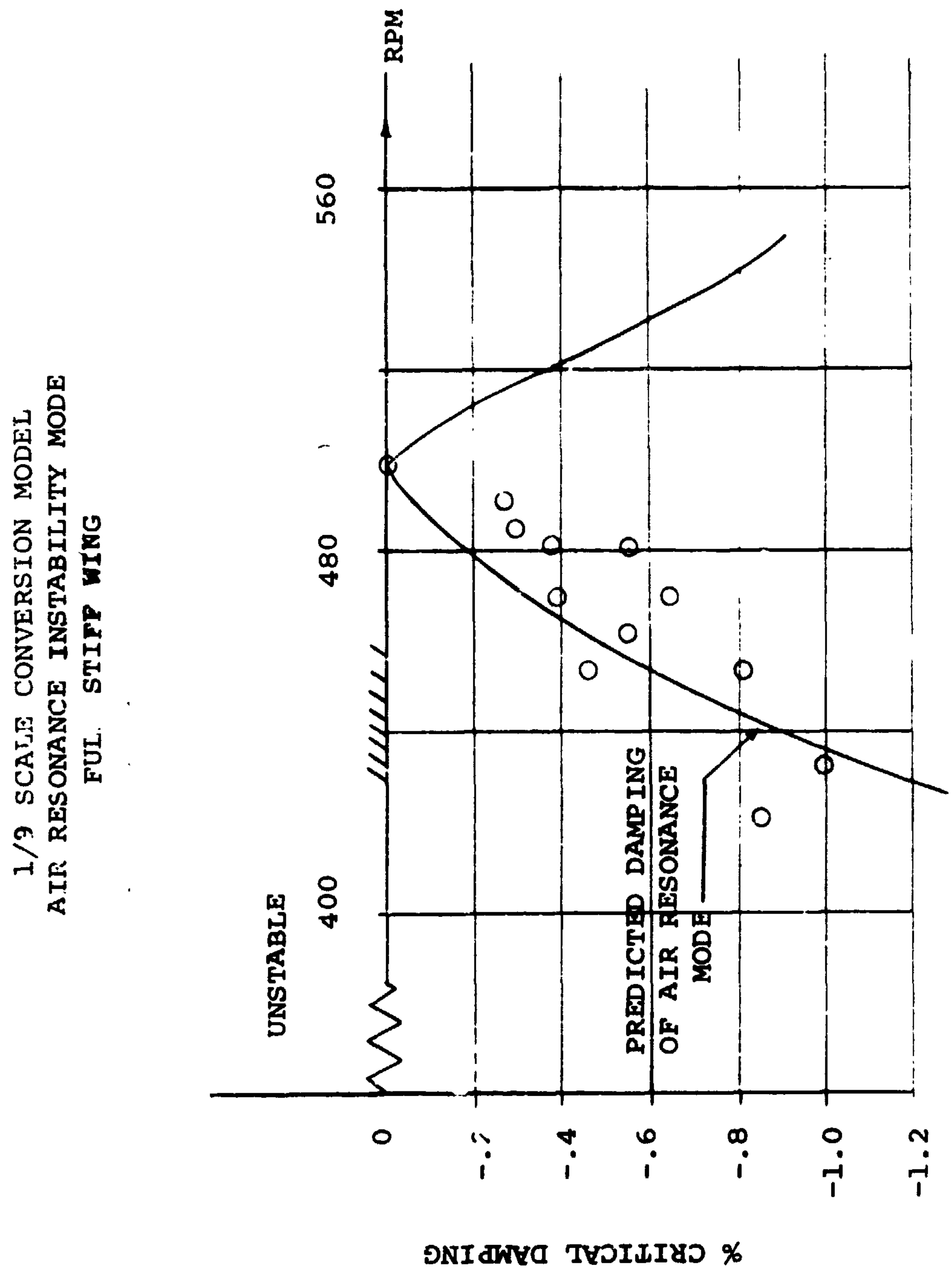


FIGURE 5-7: AIR RESONANCE CORRELATION

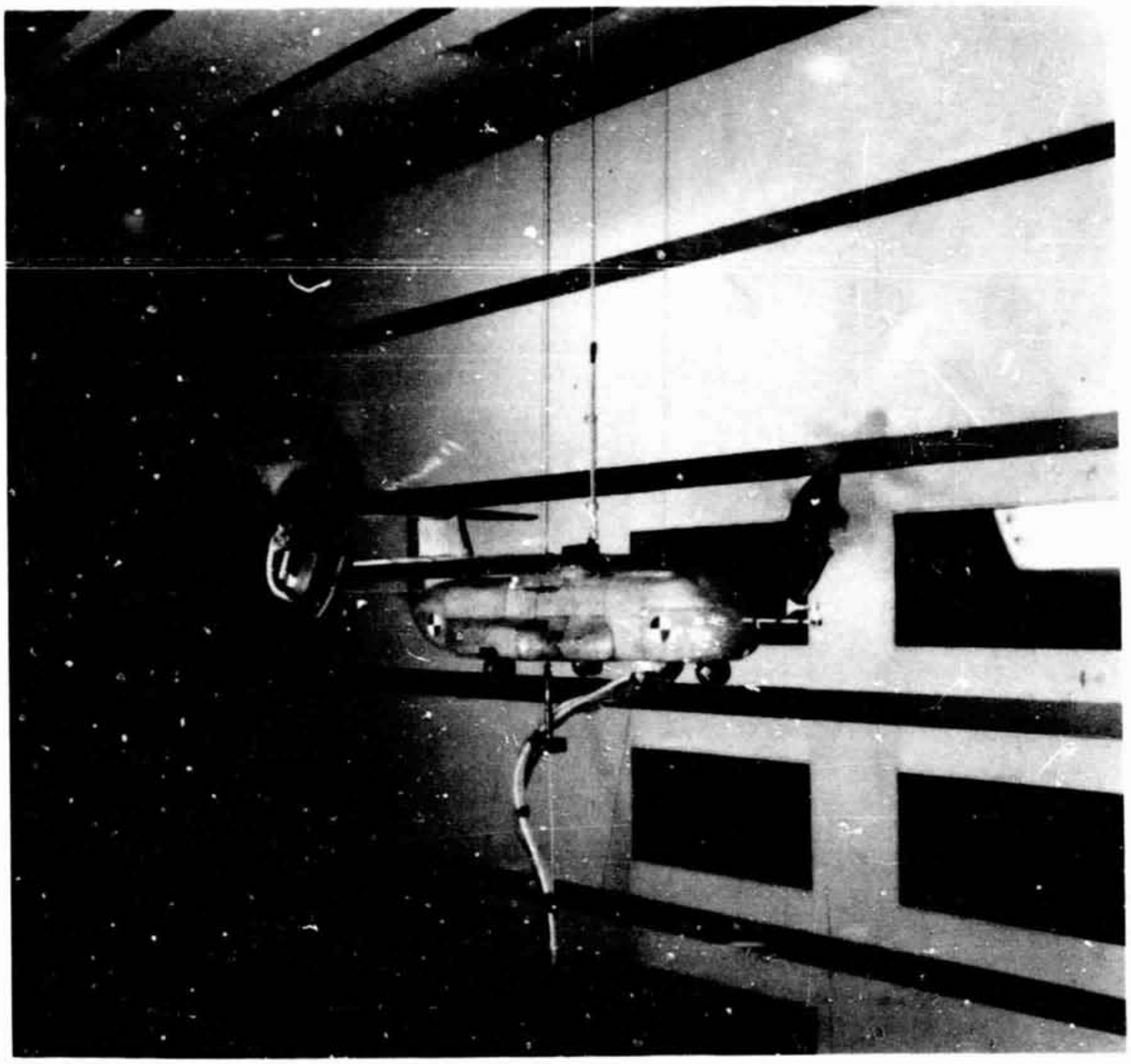


FIGURE 5-8: FULL SPAN DYNAMICALLY SCALED MODEL OF THE TILT ROTOR

NOTE: 1/9 SCALE
ROTOR PITCHING MOMENT DERIVATIVE
VARIATION WITH ROTOR RPM

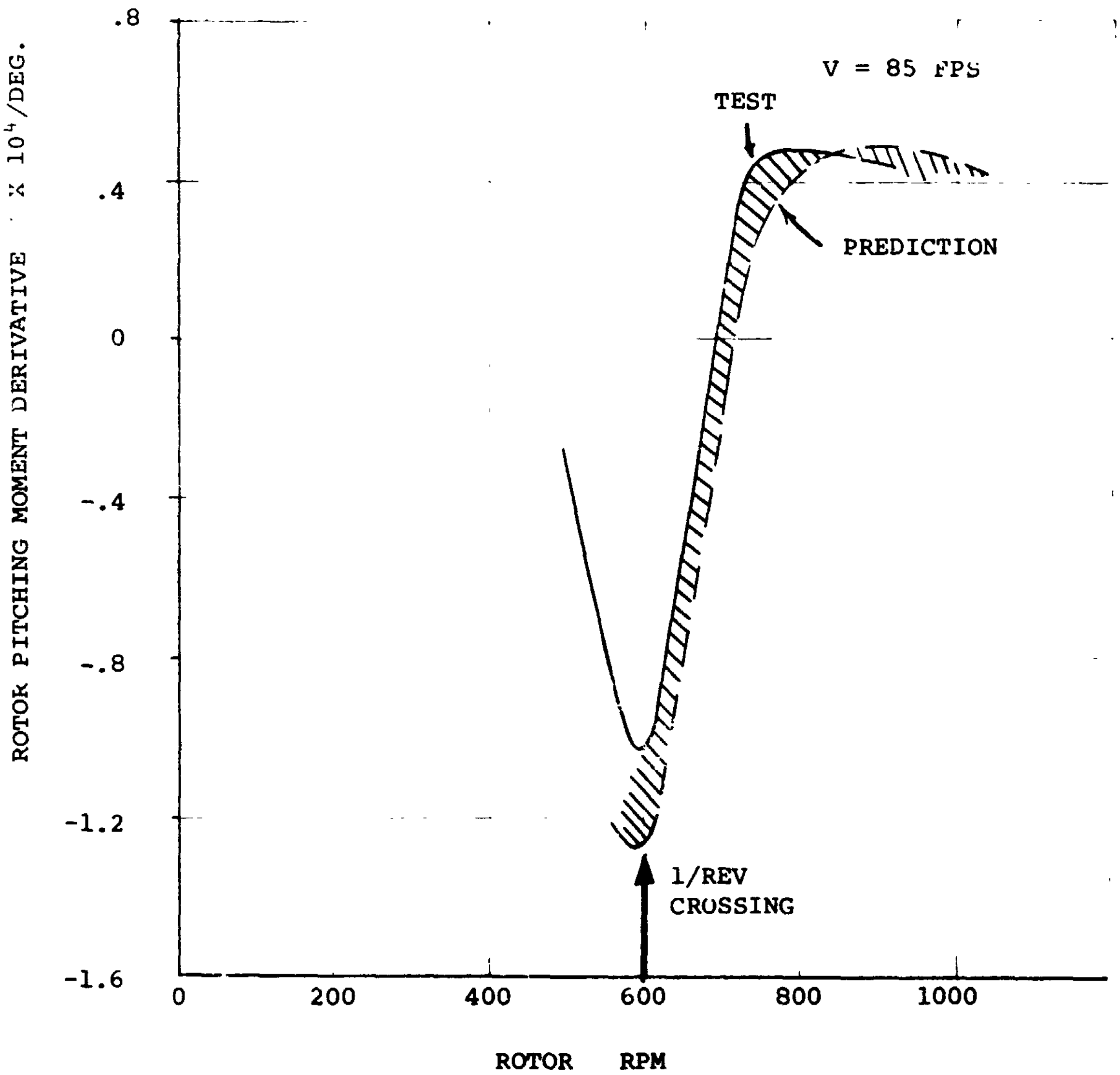


FIGURE 5-9: ROTOR DERIVATIVE CORRELATION

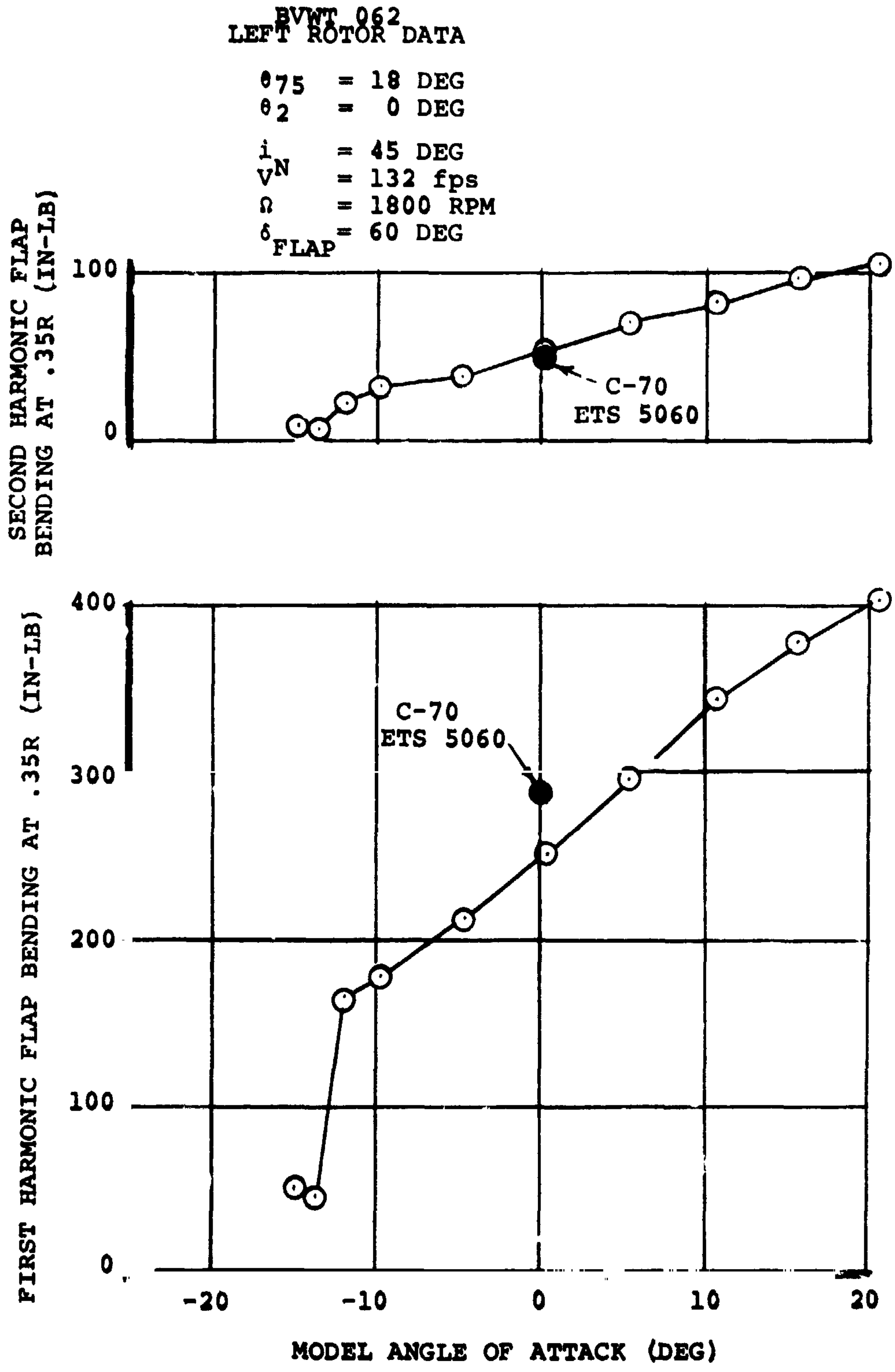


FIGURE 5-10: MODEL 160 PERFORMANCE MODEL PREDICTED AND MEASURED FIRST AND SECOND HARMONIC FLAP BENDING MOMENTS FOR MODEL ANGLE OF ATTACK VARIATION IN TRANSITION

TABLE 5-1
SUMMARY OF COMPUTER PROGRAMS

PROGRAM NO.	DESCRIPTION	COMMENT
B-92	Program B92, Analysis of Propeller and Rotor Performance in Static and Axial Flight by an Explicit Vortex Influence Technique, is documented in Boeing Report R-372. This analysis establishes a radial distribution of induced velocity based on a prescribed wake contraction schedule to calculate isolated rotor induced and total power coefficients as specified thrust or thrust coefficients. The radial airload distribution is also defined.	
B-93	VASCOMP II - The program may be used for the sizing of aircraft for which the type of aircraft and mission profile are specified. Alternatively, the program may be used for mission calculations for aircraft for which sizing details (gross weight, fuel available, engine power and fuel consumption, etc.) are known. As a combination of these two capabilities, the program may be used to first size an aircraft for a given mission and then calculate the off-design-point performance for other missions. The program contains size trends equations which reflect the variation of aircraft dimensions with gross weight, detailed statistical weights - trends equations, a routine for sizing of engines to match	

TABLE 5-1
SUMMARY OF COMPUTER PROGRAMS

PROGRAM NO.	DESCRIPTION	COMMENT
B-93 (Cont.)	airframe requirements, and a comprehensive library of engine cyclic data, and a propeller performance capability.	
B-94	"V/STOL Aircraft Takeoff and Landing Computer Program", Program B-94, will calculate take-off and landing performance independently or together for a wide variety of VTOL/STOL/OTOL aircraft. The program uses a two degree-of-freedom point mass analysis. Fourth-order Runge-Kutta integration is used for the take-off routine. Constraints on take-off speed margins at lift off and during climb out may be varied.	
B-87	Propeller Performance Program The program was developed by Curtiss-Wright Corporation to predict propeller performance in the axial flow state. Blade section, twist, chord, thickness, and design lift coefficient distribution are input. The program utilizes Theodorsen's strip analysis to calculate the induced losses and propeller efficiency.	

TABLE 5-1
SUMMARY OF COMPUTER PROGRAMS

PROGRAM NO.	DESCRIPTION	COMMENT
S.W. 142	Tilt Rotor Flight Path Simulation This hybrid computer program can be used to simulate the longitudinal motion of the tilt rotor airplane. Although developed to study autorotation maneuvers following partial or complete power failure, the program can be used to simulate any longitudinal maneuver such as transition from hover to forward flight. A digital program calculates initial equilibrium conditions of the aircraft that are then used as initial conditions in a hybrid computer simulation of the airplane motion. The airframe is represented by a rigid body with three degrees of freedom; the rotors are rigid and have a rotational degree of freedom. The aerodynamics of the aircraft are represented in tabular form with separate inputs for the airframe, rotor and interference terms.	
B-67	Rotor airloads and performance program in hover and axial flight uses an empirically defined family of wake shapes in hover to define the non-uniform inflow required to compute angle of attack of blade sections.	

TABLE 5-1
SUMMARY OF COMPUTER PROGRAMS

PROGRAM NO.	DESCRIPTION	COMMENT
B-95	This program combines a vortex distortion analysis with a rotor airloads and performance analysis. The vortex distortion analysis computes the induced velocities throughout the wake and allows the wake to drift until the required wake shape is achieved by an iterative procedure. This wake geometry is used to compute the non-uniform inflow required for airloads and performance.	
C-26	Six degrees-of-freedom/aeroelastic stability analysis includes representation of wing bending (nacelle offset vertical translation and yaw) and rotor disc coning (β_0) since β_0 couples in the non-symmetrical problem. Representation of structural damping can be added. Includes blade feathering feedback effects.	Increments wing-nacelle pitch and yaw natural frequencies to determine stability boundary. Hub motion as a function of time is also obtained.
C-27	Nine degrees-of-freedom - treats hingeless rotor blade as two coupled flap-lag elastic modes of each blade. Wing-nacelle representation in pitch and yaw is same as C-26. Wing bending is equivalent flapping about pivot point which is coincident with equivalent yaw pivot point.	Increments advance ratio, nacelle pitch, yaw, and wing vertical bending natural frequencies to determine stability boundaries. Time history of hub whirl is in work.

TABLE 5-1
SUMMARY OF COMPUTER PROGRAMS

PROGRAM NO.	DESCRIPTION	COMMENT
C-28	Calculates the roots and vectors of a complex matrix. The program is set up to read in inertia, damping, and stiffness matrices by cards and to read in airload matrices by cards or from C-24 tape. For flutter problems the program calculates the flutter frequency, damping coefficient, flutter velocity and eigenvector associated with each root. For vibration problems the program calculates the frequency, damping rate, and eigenvector associated with each root. Uses QR algorithm method.	Provides separate output formats for flutter and vibration problems. Prints complex roots of matrix and either the flutter data needed to make V-f and V-g plots or the vibration data, along with the complex eigenvectors.
C-39	Prime method for aeroelastic stability analysis. Treats hingeless rotor system with blades initially deformed under centrifugal and aerodynamic forces. Three general blade vibratory modes and general airframe representation. Aerodynamic theory includes unsteady effects, non-axial flow and non-uniform inflow. Emphasis on effects of precone, inertial coupling due to initial deflections and feedback coupling.	Very general mathematical model applicable to configurations ranging from helicopter, 1 or 2 rotor, through tilt wing, plus low disc loading tilt rotor and conventional fixed wing designs. Evaluates eigenvalues plus vectors of equations for specified ranges of cases.

TABLE 5-1
SUMMARY OF COMPUTER PROGRAMS

PROGRAM NO.	DESCRIPTION	COMMENT
C-39.01	Evaluates hub force and moment derivatives for shaft angles ranging from cruise to hover conditions. Dynamic derivatives suitable for transient analysis and static derivatives suitable for trim analyses are computed. The dynamic derivatives are the partial differentials of hub forces and moments with respect to hub positions, rates and accelerations and include inertial and gyroscopic effects as well as aerodynamic effects. For the static derivatives a constant shaft angle to the relative wind is assumed and the resulting blade motion computed. The effects of blade aerodynamic and inertia and gyroscopic forces are combined to give the hub derivatives due to constant shaft angle and constant rate of change of shaft angle.	Program based on C-39 and has the same level of aero-dynamic and kinematic sophistication.
C-40	Accepts up to 20 airframe degrees-of-freedom including six rigid body modes, also up to four prop/rotors. The blades have two coupled flap-lag flexure modes. Axial flow is assumed and strip theory is assumed for wing and empennage oscillatory air forces.	Has been used for tilt rotor ground and air resonance studies, and for whirl flutter studies on fixed wing aircraft with conventional propellers.

TABLE 5-1
SUMMARY OF COMPUTER PROGRAMS

PROGRAM NO.	DESCRIPTION	COMMENT
C-41	The hub force and moment derivatives are computed for steady and transient shaft angle conditions. The blade representation is the same as C-27, from which this program was derived.	Accepts a range of parametric conditions, e.g., speed, RPM, blade natural frequency and computes and prints out derivative matrices.
C-46	Airframe natural frequency and mode shapes. The complex aircraft structure is considered as simple structural elements (beam, axial, and skin) meeting at structural nodes. These stiffness parameters together with the mass distribution provide the program input. The IBM 360 generates a dynamic matrix from the stiffness and mass properties which is solved for natural frequencies and modes.	The solution provides for 2750 structural elements, connecting a maximum of 600 structural nodes. The computer performs a double precision natural frequency and mode solution for 139 degrees-of-freedom.
D-88	Prediction of aeroelastic rotor loads required for vibration prediction and response characteristics of the airframe using finite element methodology. Includes effects due to: a. hub motion b. compressible non-linear unsteady aerodynamics c. non-uniform downwash	

TABLE 5-1
SUMMARY OF COMPUTER PROGRAMS

PROGRAM NO.	DESCRIPTION	COMMENT
B-96	The damped forced response of the aircraft structure is computed utilizing the natural modes computed in the D-46 program. Excitation loads required for the response are obtainable experimentally (model and/or flight testing) or analytically from the L-22 program which computes rotor hub shaking forces and moment.	The solution provides for 30 modes each having 139 degrees-of-freedom and structural damping variation between modes.
L-21	Computes the coupled flap and chord bending natural frequencies and mode shapes of a highly twisted blade for both pinned and cantilevered root-end boundary conditions.	Utilizes the lumped parameter method of analysis employing finite difference equations to relate the dynamic aeroelastic quantities of adjacent blade station, whose maximum number is 40.
L-22	Computes the coupled flap and chord bending forces response of a highly twisted blade in hover, transition and cruise flight conditions. The steady and vibratory forced response includes blade bending deflections and moments and rotor hub forces.	Utilizes the lumped parameter method of analysis employing finite difference equations to relate the dynamic aeroelastic quantities of adjacent blade station, whose maximum number is 40.

TABLE 5-1
SUMMARY OF COMPUTER PROGRAMS

PROGRAM NO.	DESCRIPTION	COMMENT
L-32	Computes the steady-state vibratory and transient vibratory flap bending response of a helicopter or propeller blade to control and gust inputs, and to a prescribed vortex of arbitrary shape and duration.	Utilizes the generalized coordinate theory employing the uncoupled flap bending natural modes derived from Program L-01; their maximum number being eight.
Y-08	Computes the blade loading and non-uniform induced velocities for a propeller or rotor.	Includes a simulation of a distorted, contracted wake; the analyses permit variation of the cyclic pitch through the flight.
Y-71	A theory and calculation method for predicting the static, vibration, and dynamic stability characteristics of a propeller or rotor system mounted on a flexible wing. The equations of motion of a blade are derived using the lumped mass approach. Aerodynamic forces are expressed on the basis of quasi-steady flow, using strip theory. The equations are of the form of a set of linear higher order differential equations with periodically varying coefficients. An extension of the "Floquet Method" is used in solving the equations.	

TABLE 5-1
SUMMARY OF COMPUTER PROGRAMS

PROGRAM NO.	DESCRIPTION	COMMENT
WATFOR Divergence Program	Accepts static derivative data for prop/rotors and combines with wing aerodynamics and structural influence coefficients to form and solve static divergence equations. Rotor derivatives may be theoretical or empirical.	Program examines equations for degree of static stability over a range of air speeds. The mode of twist due to the static loading is calculated, since this may differ significantly from the fundamental vibratory torsion mode. This procedure therefore provides a more reliable estimate of static divergence speed than the estimates obtained from flutter calculations.
Tilt Rot 5	Wing on rotor interference program - uses a lifting line representation of the wing and computes incremental induced velocities and inflow angle in the plane of the rotor.	WATFOR Program

TABLE 5-1
SUMMARY OF COMPUTER PROGRAMS

PROGRAM NO.	DESCRIPTION	COMMENT
G-24	Tilt Rotor Trajectory Performance and Noise Prediction G-24 is a mathematical model which describes the kinematic performance and acoustical characteristics of tilt-rotor aircraft. Vehicle accelerations are enforced as inequality constraints. The noise generated by the tilt-rotor aircraft is assumed to be produced entirely by the rotor. The technique used to predict the rotational noise of the rotor is a modification of analytical method described in Studies of Helicopter Rotor Noise by M. V. Lowson and J. B. Ollerhead, USAAVLABS TR68-60, January 1969. This modification accounts for the changes in inflow the rotor encounters when tilting. The broad-band noise generated by the rotor is predicted by an empirical method developed by R. Schlegel, R. King and H. Mull in Helicopter Rotor Noise Generation and Propagation, USAAVLABS TR66-4, October 1966 and modified by M. V. Lowson and J. B. Ollerhead in their AIAA Paper Problems of Helicopter Noise Estimation and Reduction, No. 69-195, February 1969. The 1/3 octave band sound pressure level spectrum resulting from the prediction methods is used to calculate the perceived noise level.	

TABLE 5-1
SUMMARY OF COMPUTER PROGRAMS

PROGRAM NO.	DESCRIPTION	COMMENT
C-60	Predicts prop/rotor blade loads and hub loads, using compressible, non-linear unsteady aerodynamics and non-uniform downwash. The analysis is limited to steady state flight at constant rotor tip speeds for blades with small built-in twist. Coupling of deflections with airloads, flap deflections and torsional deflections are neglected.	Not a true prop/rotor analysis since it can only approximate high twist rotors.
C-70	Predicts prop/rotor blade loads and hub loads, using compressible, non-linear unsteady aerodynamics and non-uniform downwash. Large blade twist, shear center offset from the pitch axis, large shaft tilts and full flap-lag-pitch coupling are considered. The analysis is limited to steady state flight at constant rotor tip speeds.	
C-48	A thrust management and cyclic pitch feedback control capability which includes all the aircraft features of C-40 and in addition thrust management and cyclic pitch control representation.	Is used for concept feasibility and sensitivity trend studies of feedback system parameters.

3. Areas where tilt-rotor technology should be further refined or expanded to increase confidence or reduce effort in developing Task I airplanes from the research aircraft.

4. General areas of advanced technology, not specifically related to tilt rotors, which have been assumed to be incorporated into the Task I aircraft.

Only categories 1 and 3 are addressed in this section. Category 2 is covered under Task II, while category 4 is considered beyond the scope of this study except for a general listing of areas.

Category 1 - Desirable Before Research Aircraft

a. Feedback Control System - As discussed under Task II (Volume II), major improvements in flying qualities, blade load reduction, and increase in damping of aeroelastic modes can be obtained by use of feedback control systems. Development of analytical methods and wind tunnel testing is required. Boeing's current limited analytical capability is being expanded to predict the capabilities of such systems. Wind tunnel data can be obtained by additional testing of Boeing's existing 1/9-scale dynamic model of the Model 222 rotor, the 1/4.6-scale airplane model which Boeing is building during the first half of 1972, and from additional testing of the full-scale 26-foot-diameter rotor.

b. Ground Effect - Ability to predict ground effect is poor, even on fixed-wing aircraft. Some data on the effect of ground proximity on performance is available from tests on a 1/10-scale performance model of the Boeing Model 160. Limited testing on the 1/10-dynamically-scaled full-span model of the Boeing Model 160 indicated a slight tendency to skittishness while hovering in ground effect, which was readily correctable by an attitude SAS. The effect of ground proximity on both performance and flying qualities needs further investigation by additional model testing. This could be done on Boeing 1/4.6-scale model of the Model 222.

c. Transient Rotor Loads - A very limited capability exists for the prediction of transient rotor blade loads during maneuvers and gusts. The methodology needs improvement and validation by test. Data on transients following control application can be obtained also on an oscillating model and in simulated gusts. Such data could be obtained on Boeing's

1/9-dynamically-scaled 222 model in the Princeton tunnel. Maneuver data can also be obtained from Boeing's 1/4.6-scale full-span dynamic model.

d. Free-Free Aeroelastic Tests - Basic analytical methodology for coupled airframe/rotor dynamics has been well validated by model tests and will be further substantiated by the 26-foot rotor tests. Almost all testing, however, has been on semi-span models which cannot simulate antisymmetric or free-free modes. Additional testing is desirable on a full-span dynamically scaled model. This could be performed on Boeing's 1/4.6-scale 222 model.

Category 3 - Desirable Before Proceeding with Task I Airplanes

a. Stall Flutter - Correlation between test data and current stall flutter criteria is poor. Current criteria may be extremely conservative resulting in excessive blade weight and reduced performance in hover and cruise. Additional parametric testing is required to improve the understanding of the effects of airfoil characteristics, flight conditions, blade torsional properties, and control stiffness and damping. While certain advanced sections appear to be clear of stall flutter in hover, this phenomenon may rapidly reappear as forward flight speed is increased and rotor tilt decreased during transition. Tests of rotors using advanced airfoils are required.

b. Autorotation and Descent - A limited amount of data on low power and autorotation performance was obtained from Boeing's 1971 tests on a "rigid" model. This is sufficient to permit the research aircraft to start exploring the area. However, quantitative data exploring parametric variations of airspeed, shaft angle, collective pitch, wing interference effects, etc. can be more accurately and cheaply obtained by model tests. Additional model testing is recommended to provide the volume of parametric data needed to substantiate and improve the prediction methods. This could be performed on Boeing's 1/4.6-scale 222 model.

c. Individual Blade Instabilities - Current analytical methods can predict aeroelastic stability characteristics of individual blades and of coupled rotor/airframe modes. The coupled mode analyses, such as whirl flutter and air/ground resonance, are well substantiated by test data. For individual blade instabilities, such as coupled pitch/flap/lag motion, however, available test data to validate the analysis is extremely limited. For the research aircraft, model tests and the full-scale 26-foot-diameter rotor tests will substantiate freedom from these instabilities. For the new rotors which will be used on the Task I vehicles, however, it would be desirable to obtain sufficient test data to substantiate the analytical prediction of stability boundaries. Since the mechanism of these instabilities is specifically associated with the deflected shape of the blade under aerodynamic and inertia forces, parametric variations can be obtained on a single blade design by varying the operating conditions of thrust, rpm, advance ratio, and shaft angle. Boeing's 1/9-scale and 1/4.6-scale dynamic models would be suitable for this purpose.

Category 4 - Non-Tilt Rotor Technology

Advanced technology not specific to tilt rotors which has been assumed to be incorporated into the Task I airplane includes extensive use of advanced composites in the airframe, advanced transmission technology in gears, bearings and lubrication systems, advanced airfoils for wings as well as rotors, and advanced technology turbine engines. All of these technologies are under development now by NASA, military, or industrial efforts and should be in hand in the time-scale of operational aircraft developments. These areas should be monitored over the period between the present time and the initiation of operational aircraft designs to assure an adequate technology base is indeed developed.

REFERENCES

1. STOL, VTOL, and V/STOL -- Where Do They Fit In?, AIAA AdHoc Committee, AIAA 8th Annual Meeting and Technical Display, Washington, D. C., October 1971

2. Schoen, A. H., User's Manual for VASCOMP II - The V/STOL Aircraft Sizing and Performance Computer Program, Boeing Document D8-0375, Volume VI, Vertol Division, The Boeing Company, Philadelphia, Pa., October 1971.

3. Wisniewski, John S., Weight Trends Data for VASCOMP - The V/STOL Aircraft Sizing and Performance Computer Program, Boeing Document D8-0375, Volume V, Vertol Division, The Boeing Company, Philadelphia, Pa., April 1967.

4. Gabriel, E. A., Drag Estimation of V/STOL Aircraft, Boeing Document D8-2194-1, Vertol Division, The Boeing Company, Philadelphia, Pa., April 1969.

5. Sandford, Robin W., et al, Design Studies and Model Tests of the Stowed Tilt Rotor Concept, Volume VIII. Summary of Structural Design Criteria and Aerodynamic Prediction Techniques, Air Force Flight Dynamics Laboratory Report AFFDL-TR-76-62, Volume III, October 1971.

6. Lowson, M. V., Ollerhead, J. B., Studies of Helicopter Rotor Noise, USAAVLABS TR68-60, January 1969.

7. Schlegel, R., King, R., Mull, H., Helicopter Rotor Noise Generation and Propagation, USAAVALBS TR66-4, October 1966.