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CONCEPTUAL DESIGN STUDIES OF 1985 COMMERCIAL VTOL TRANSPORTS THAT UTILIZE ROTORS

VOLUME I

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R. CLARK

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NOVEMBER 1974

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LIST OF SYMBOLS

AEO	All engines operating
C_{T_R}	Rotor thrust coefficient $T/\rho\pi R^2 V_T{}^2$
C_{P_R}	Rotor power coefficient HP x $550/\rho\pi R^2 V_T{}^3$
$C_{T_{PROP}}$	Propeller thrust coefficient
$C_{P_{PROP}}$	Propeller power coefficient
C_{N_β}	Yaw moment coefficient due to sideslip
C_{ℓ_β}	Rolling moment coefficient due to sideslip
C_{Y_β}	Side force coefficient due to sideslip
C_{Ω_p}	Yawing moment coefficient due to roll rate
C_{ℓ_p}	Rolling moment coefficient due to roll rate
C_{Y_p}	Side force coefficient due to roll rate
C_{Ω_r}	Yawing moment coefficient due to yaw rate
C_{ℓ_r}	Rolling moment coefficient due to yaw rate
C_{Y_r}	Side force coefficient due to yaw rate
CG	Center of gravity
DGW (or GW)	Design gross weight
DCP	Differential collective pitch
F/W	Force to weight ratio
Fe (fe)	Equivalent drag area
Ixx	Aircraft inertia (roll)
Iyy	Aircraft inertia (pitch)
Izz	Aircraft inertia (yaw)
IGE	In ground effect
J	Propeller advance ratio $V/\Omega D$
MID WT	Mid-weight

LIST OF SYMBOLS (CONTINUED)

M_α	Moment due to angle of attack
Mq	Pitching moment due to rate of pitch
N_β	Yawing moment due to sideslip
NRP	Normal rated power
n	Normal load factor
OEI	One engine inoperative
OWE	Operating weight empty
OGE	Out of ground effect
PED	Pedal
SAS	Stability augmentation system
SPL	Sound pressure level
t	Time (seconds)
TAS	True airspeed
V_T (V_t)	Tip speed
V_e	Equivalent airspeed
$\bar{V}_H$	Horizontal tail volume ratio
W/S	Wing loading
$\theta, \dot{\theta}, \ddot{\theta}$	Pitch angle, rate, acceleration
$\psi, \dot{\psi}, \ddot{\psi}$	Yaw angle, rate, acceleration
$\phi, \dot{\phi}, \ddot{\phi}$	Roll angle, rate, acceleration
δ_e	Elevator deflection
θ_{75}	Collective pitch
$\Delta\gamma$	Differential cyclic
α_{trim}	Angle of attack to trim
δ_R	Rudder deflection

LIST OF SYMBOLS (CONTINUED)

ω_N	Natural frequency
ζ	Damping

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ABSTRACT

This document presents results of conceptual design studies of commercial rotary wing transport aircraft for the 1985 time period. Two aircraft configurations - a tandem helicopter and a tilt rotor have been designed for a 200 nautical mile short haul mission with an upper limit of 100 passengers. In addition to the baseline aircraft two further designs of each configuration are included to assess the impact of external noise design criteria on the aircraft size, weight and cost.

FOREWORD

This report was prepared by The Boeing Vertol Company for the National Aeronautics and Space Administration, Ames Research Center, under NASA Contract NAS2-8048.

The report contains the results of conceptual design studies of large helicopter and tilt rotor aircraft for the commercial short haul market in 1985.

Mr. D. Giulianetti and Mr. K. H. Edenborough (NASA Ames) were technical monitors for this work.

The Boeing Vertol Program Manager was J. P. Magee, and Project Engineer was R. D. Clark.

SUMMARY

The increasing demand for fast short haul transportation, the increasing congestion at major airports, and the rising cost of fossil fuels are all factors to be considered in assessing the various forms of air transportation to be used in the next decade.

The study reported herein provides preliminary design data for two rotary wing aircraft for the short haul market in the mid 1980's. These aircraft are designed to have vertical takeoff and landing capability to allow operation away from the restrictions of existing airports and traffic patterns, thus relieving congestion.

The two configurations studied were a tandem rotor helicopter and a tilt rotor aircraft. Each configuration was designed to carry 100 passengers and luggage over a 200 nautical mile range.

The design point tandem helicopter has a takeoff design gross weight of 30,470 Kg(67,175 pounds). The tilt rotor aircraft takeoff design gross weight is 33,905 Kg (74,749 pounds). These weights are reflected in the aircraft "fly-away" or initial costs and result in the helicopter initial cost of $4.17 million and the tilt rotor initial cost of $5.15 million. However, the tilt rotor shows advantages which result from its high cruise speed capability.

The tilt rotor design cruise speed is 349 knots at 14,000 feet altitude. The tandem helicopter design cruise speed is 165 knots at 5,000 feet altitude. This marked difference in cruise speed produces faster block times and trip times for the operator and short haul traveller, and combined with lower fuel requirements, results in lower direct operating costs for the tilt rotor aircraft.

At 230 statute miles the tilt rotor has a direct operating cost of 2.19¢ per seat mile (1974 dollars) compared with the tandem helicopter at 3.21¢ per seat mile.

The design point tilt rotor has a lower fuel consumption than the tandem rotor helicopter and can operate up to 47.5 passenger miles per gallon at 100% load factor compared with the helicopter at 28.8 passenger miles per gallon.

External noise is an important consideration if short haul VTOL aircraft are to operate close to areas of high population density. The 500 foot side line noise level for the design point helicopter at takeoff is 92.3 PNdB compared with the design point tilt rotor at 98.2 PNdB.

This noise difference is negated when the operational environment is studied. A 95 PNdB noise level is observed over a larger total area (.58 sq. mi.) for the helicopter than for the tilt rotor (.24 sq. mi.) when both takeoff and landing is considered.

The effect of imposing external noise constraints on the designs has been investigated by sizing both configurations to be 5 PNdB more noisy and 5 PNdB less noisy than the baseline configuration designs.

For the tandem rotor helicopter, decreasing the 500' sideline noise level at takeoff by 5 PNdB increased the aircraft gross weight to 33,669 Kg (74,227 pounds), increased the aircraft initial cost to $4.76 million and the direct operating cost at 230 statute miles to 3.34¢ per seat mile.

If the external noise level at takeoff is allowed to increase by 5 PNdB the aircraft gross weight reduces to 29,866 Kg (65,843 pounds), and the aircraft initial cost reduces to $3.98 million. The direct operating cost at 230 statute miles did not decrease, but increased to 3.5¢ per seat mile.

For the tilt rotor configuration a reduction in external noise of 5 PNdB requires an increase in takeoff design gross weight to 36,143 Kg (79,682 pounds) and a resulting increase in initial cost to $5.6 million. The direct operating cost increases to 2.36¢ per seat mile.

A 5 PNdB increase in external noise reduces the tilt rotor takeoff weight to 33,210 Kg (73,217 pounds) and the initial cost to $5.03 million. The direct operating cost of the aircraft is slightly higher than the baseline tilt rotor at 2.20¢ per seat mile.

The helicopter is thus the slower, more expensive in terms of direct operating cost, less expensive in terms of initial cost and less noisy of the two aircraft at 500 feet sideline although it effects a larger area than the tilt rotor.

The tilt rotor is faster, cheaper in terms of direct operating cost, more expensive in terms of initial cost and more noisy on takeoff at 500 feet sideline distance. Its perceived noise contours encompass a smaller area than the helicopter case.

Details of the designs are presented in this document. The report also includes an evaluation of the technical risk associated with large rotary wing aircraft and component development programs are proposed which minimize such risks. In the case of the tilt rotor this component development activity includes a flight test program. This is envisioned as an intermediate gross weight vehicle program which would use existing airframe components (e.g., CH-47 fuselage), but would embody full size dynamic components and composite material rotors. A test program of progressively more severe operating conditions and increasing gross weight will permit system development to commercially acceptable levels of payload. An additional attraction of this approach is that the intermediate sized aircraft of initially low disc loading, comes close to being a prototype of a vehicle which would be suitable for a number of military missions (LTTAS, etc.). Thus, this test bed vehicle would have a range of utilization spanning both military and civil activities. The following table is a summary of the aircraft designs used in this study.

SUMMARY OF AIRCRAFT DESIGNS

	BASELINE TANDEM HELICOPTER	+5 PNdB TANDEM HELICOPTER	-5 PNdB TANDEM HELICOPTER	BASELINE VTOL TILT ROTOR	+5 PNdB VTOL TILT ROTOR	-5 PNdB VTOL TILT ROTOR
GROSS WEIGHT - Kg (LBS)	30,470 (67,175)	29,866 (65,843)	33,669 (74,227)	33,905 (73,217)	33,211 (73,217)	36,143 (99,682)
EMPTY WEIGHT - Kg (LBS)	18,226 (40,181)	17,305 (38,152)	21,107 (46,533)	22,710 (50,068)	22,116 (48,757)	24,820 (54,718)
CRUISE SPEED - KTS TAS	165	141	181	349	340	355
CRUISE ALTITUDE - m (FT)	1,524 (5,000)	1,524 (5,000)	1,524 (5,000)	4,267 (14,000)	4,267 '14,000)	4,267 (14,000)
BLOCK TIME - HRS	1.337	1.53	1.24	.742	.76	.73
DOC - ¢/SEAT MILE	3.21	3.50	3.34	2.19	2.2	2.36
500' SIDELINE PERCEIVED NOISE - PNdB	92.3	97.2	87.1	98.2	103.2	93.4
95 PNdB AREA - TAKE-OFF - Sq Km (Sq. Miles)	0.18 (.07)	0.49 (.19)	.03 (.01)	0.23 (.09)	0.49 (.19)	0.08).03)
95 PNdB AREA - LANDING Sq Km (Sq. Miles)	1.39 (.535)	2.28 (.88)	.76 (.295)	.39 (.15)	.75 (.29)	.18 (.07)
BLOCK FUEL - Kg(LBS)	2,310 (5,093)	2,536 (5,590)	2,541 (5,603)	1,431 (3,157)	1,403 (3,094)	1,618 (3,567)
ROTOR DIA. - m (FT)	21(68.9)	20.8(68.2)	22.1(72.5)	17.16(56.3)	17.0(55.7)	17.74(58.2)
DISC LOADING - Kg/m^2 (LBS/FT^2)	43.94 (9.0)	43.94 (9.0)	43.94 (9.0)	73.2 (15)	73.2 (15)	73.2 (15)
WING LOADING - Kg/m^2(LBS/FT^2)	----	-----	-----	488(100)	488(100)	488(100)
HOVER TIPSPEED - m/s (ft/s)	221(725)	247(810)	195(640)	236(775)	279(915)	195(640)
CRUISE TIPSPEED m/s (ft/sec)	221(725)	247(810)	195(640)	165(543)	195(641)	137(448)
INS.POWER-Watts(HP)	10.79×10^6 (14,472)	10.27×10^6 (13,770)	12.88×10^6 (17,277)	12.36×10^6 (16,480)	11.98×10^6 (16,072)	14.52×10^6 (19,476

It is concluded that no insurmountable technology barrier is identified which is associated with size in either the tandem helicopter or tilt rotor configurations. The amount of design and development work required to bring a tandem helicopter into service will be smaller than those associated with a tilt rotor since substantial helicopter development at these gross weights has already been accomplished.

1.0 INTRODUCTION

This report documents the results of conceptual engineering design studies of two VTOL transport configurations for the 1985 time frame. These studies were performed by the Boeing Vertol Company for NASA-Ames Research Center, under NASA Contract NAS2-8048.

The studies required the definition of a tandem rotor helicopter and a tilt rotor aircraft for a short naul commercial transport mission. The aircraft have been sized for 100 passengers, the maximum number of passengers permitted by the study groundrules, and a 200 nautical mile design mission.

The objectives in performing these studies were twofold. The first objective was to provide design data for the two rotary wing configurations. The data is required as input information for a larger VTOL transportation systems study to be performed by NASA. The second objective was to identify the size and performance of rotary wing commercial transport aircraft in the short haul environment at a time when increasing fuel costs, environmental issues, and the efficient use of existing and new terminal facilities becomes increasingly important in the selection of future vehicles. As the reliability levels of rotorcraft rise and vibration decreases through continued research, the rotary wing machine can offer a flexible, viable alternative to other forms of short haul transportation.

Details of the design point aircraft defined by this study (a tandem rotor helicopter and a tilt rotor) are discussed in Section 2.0 of this volume.

In addition, derivative aircraft have been designed to varying levels of external noise. For each configuration two additional aircraft were defined having noise characteristics ±5 PNdB in relation to the basic design point aircraft. These aircraft are described in Section 3.0 of this report.

Comparisons of the two configurations and of the effects of noise criteria are drawn in Section 4.0.

The broader and less easily quantified topics which fall under the general heading of risk are discussed in Section 5. This includes such issues as the technical risks associated with component size and economic visability. These naturally tend to conflict. Technical risk must be assumed to increase, the further one proceeds beyond the level of past experience, while the probability of good economic performance improves up to the sizes which have been considered. In Section 5 it has been concluded that the technical and engineering risks associated with the 100 passenger size helicopter and tilt rotor are acceptable, provided that a decision to build is accompanied by an orderly and comprehensive program of component test and development.

Throughout the study it has been assumed that levels of comfort and reliability, at least as good as current jet transports, will be required to gain passenger and operator acceptance. This will require special efforts to ensure the fullest use of vibration reducing equipment, and in the case of the tilt rotor, the application of advanced active control system techniques in order to attain acceptable ride qualities characteristics.

2.0 DESIGN POINT AIRCRAFT

This section describes the two baseline design point aircraft - a tandem rotor helicopter and a tilt rotor aircraft. These vehicles have been selected and refined from initial vehicle trend studies. Each configuration carries one hundred (100) passengers and has a 200 nautical mile design mission. The design selections are based on minimum operating cost and are constrained by the NASA design guidelines (see Section 4, Volume II.)

For each configuration the design layout, weights data, vehicle performance, stability and control, noise and cost data are presented. The comprehensive background technology data which support the summary information presented in this section are contained in Volume II.

A design identification numbering system has been adopted to allow ease of discussion in comparing designs - for example, TH-100 (92.3) and TR-100 (98.2). The initial letters indicate the configuration: TH - tandem helicopter; TR - tilt rotor. The -100 number indicates 100 passenger designs and the number in parentheses (92.3) is the PNdB value at 500 feet side line in hover to distinguish between the vehicles designed to various noise criteria as discussed in Section 3.0.

2.1 DESIGN POINT TANDEM ROTOR HELICOPTER - TH-100 (92.3)

The tandem rotor helicopter configuration was selected over other pure helicopter types for this study because of the inherently lower risk of large helicopter development for this type. The primary risks in the development of these aircraft are related to rotor size, transmission and rotor gearbox torque capability as discussed in Section 5.0. The individual components are generally smaller and more within the manufacturing state-of-the-art in a tandem design than for a single rotor machine. Other advantages of the tandem configuration include ease of handling large CG excursions and the ability to locate the engines away from the passenger cabin. This latter capability keeps engine noise, fumes and carbon deposition away from passenger areas. In addition, Boeing experience with tandem rotor helicopters ranging in size from 5,000 pounds to 120,000 pounds gross weight provides a high degree of confidence in prediction and design techniques.

2.1.1 Design TH-100 (92.3) - Configuration and Layout

The tandem rotor design point aircraft is shown in Figure 2.1. The major aircraft dimensions and pertinent data are shown in Table 2.1 and a threeview is shown in Figure 2.2.

This vehicle weighs 30,470 Kg (67,175 pounds) design takeoff gross weight and has an installed shaft horsepower of 3.597×10^6 Watts (14,472 HP) at sea level standard day. The two 68.9 foot rotors are four-bladed articulated rotors with a solidity ratio of 0.099. The selection of rotor solidity has been made to provide freedom from stall flutter loads over the entire maneuver envelope. The rotor overlap has been held to zero to eliminate rotor "bang" due to one rotor

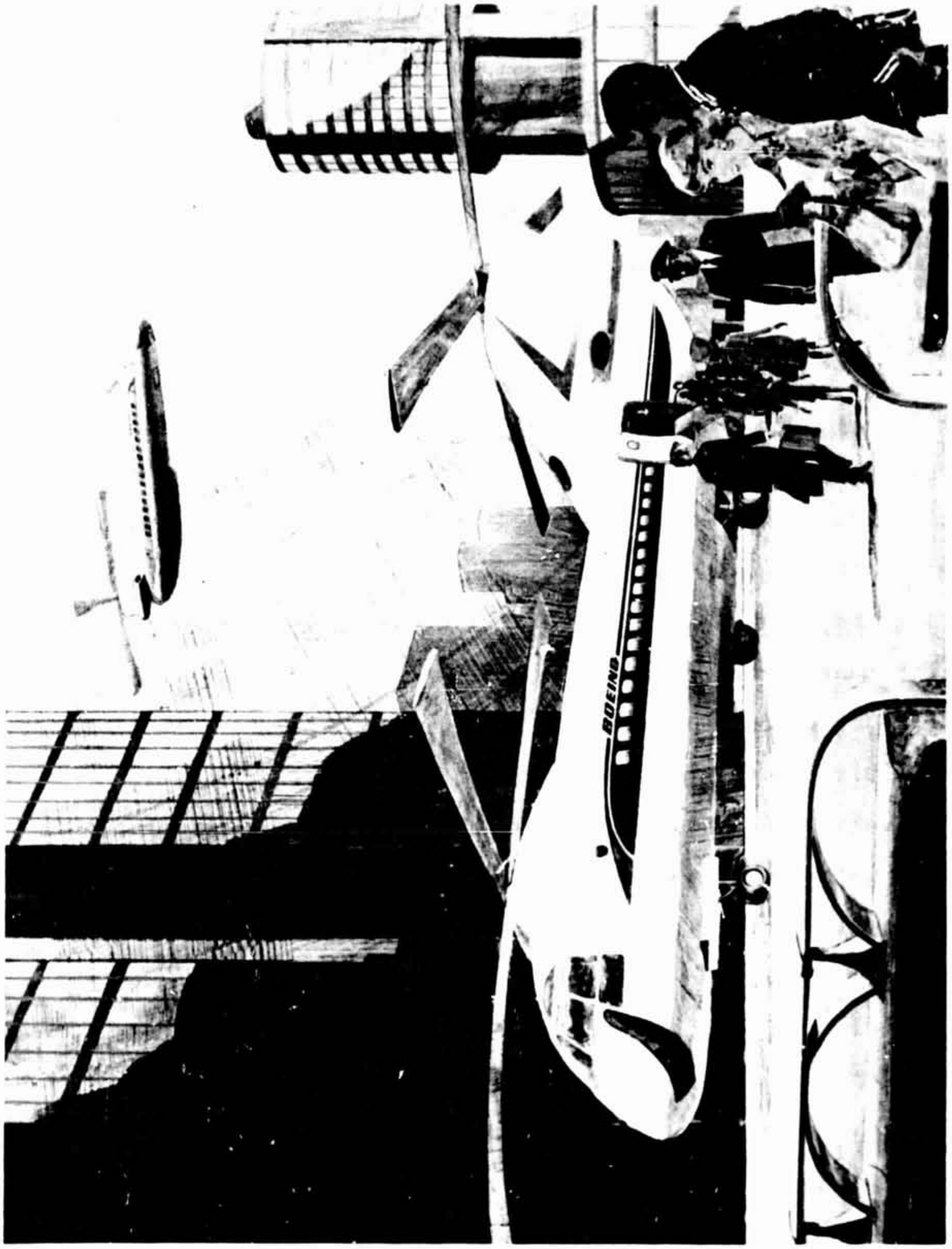


FIGURE 2.1. TANDEM HELICOPTER DESIGN POINT AIRCRAFT.

	S.I. UNITS	U.S. UNITS
WEIGHTS		
DESIGN GROSS WEIGHT	30,470 Kg	67,175 Lbs
WEIGHT EMPTY	18,226 Kg	40,181 Lbs
FUEL	3,178 Kg	7,007 Lbs
NO. OF PASSENGERS	100	100
ROTOR		
DISC LOADING	39.4 Kg/m^2	9 Lbs/Ft^2
DIAMETER	21 m	68.9 FT
SOLIDITY	.099	.099
NO. OF BLADES	4	4
TWIST	12 Degs	12 Degs
TIP SPEED	221 m/s	725 Ft/Sec
POWER		
NO. OF ENGINES	3	3
RATED POWER (S.L., STD.)	3.597×10^6 Watts	4824 SHP
FUSELAGE		
LENGTH	26.5 m	87 Ft
WIDTH	4.48 m	14.7 Ft
CABIN/LENGTH	15.03 m	49.3 Ft
PERFORMANCE		
NVRP	85 m/s	165 Knots (TAS)
ALTITUDE CRUISE	1524 m	5000 Ft
t BLOCK	1.337 Hours	1.337 Hours
NOISE		
500 FOOT SIDELINE (HOGE)	92.3 PNdB	92.3 PNdB

TABLE 2.1. DESIGN POINT HELICOPTER TABLE OF CHARACTERISTICS.

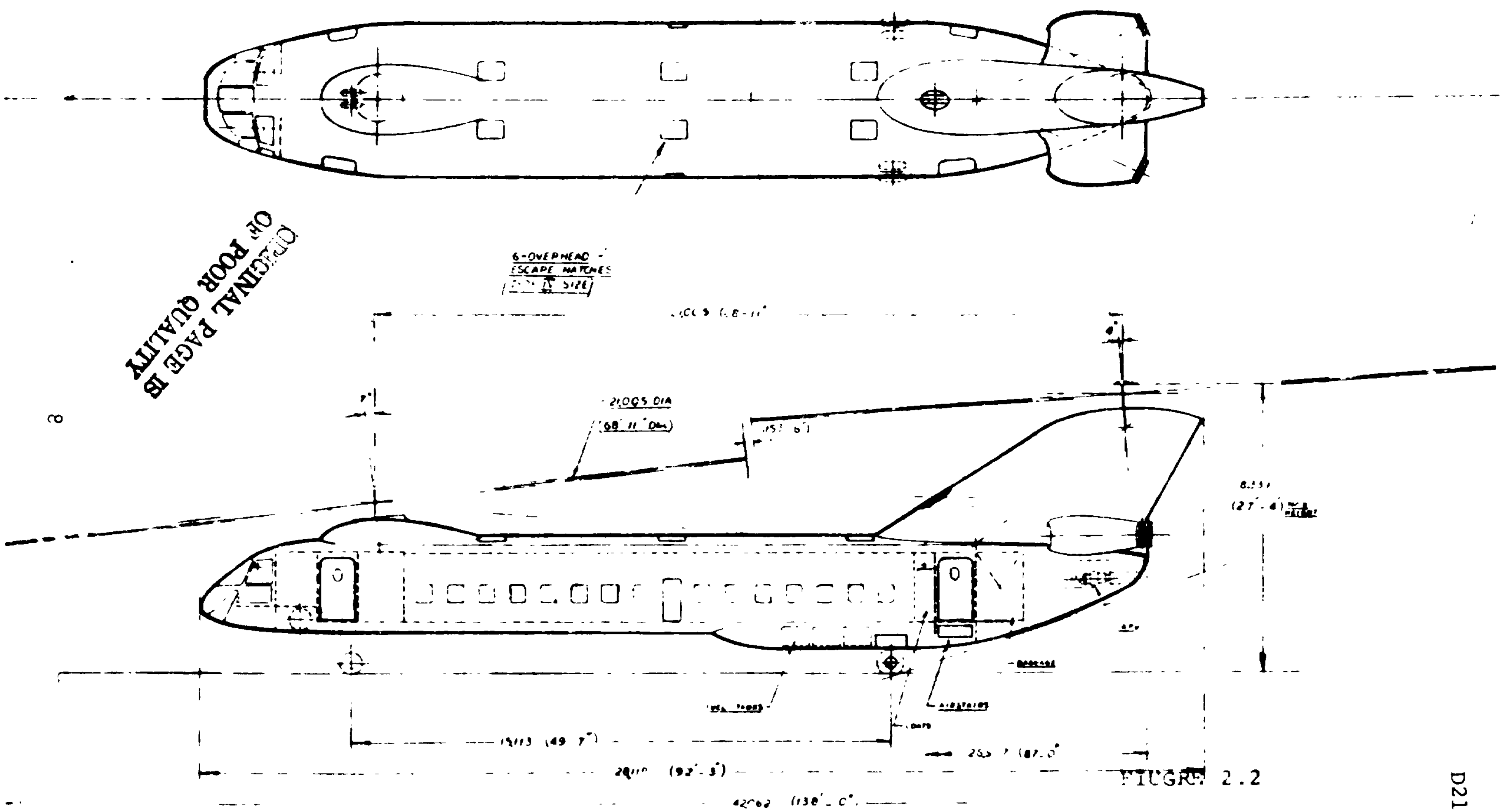


FIGURE 2.2. 1985 COMMERCIAL TRANSPORT, 100 PASSENGER - TANDEM HELICOPTER - BASELINE DESIGN.

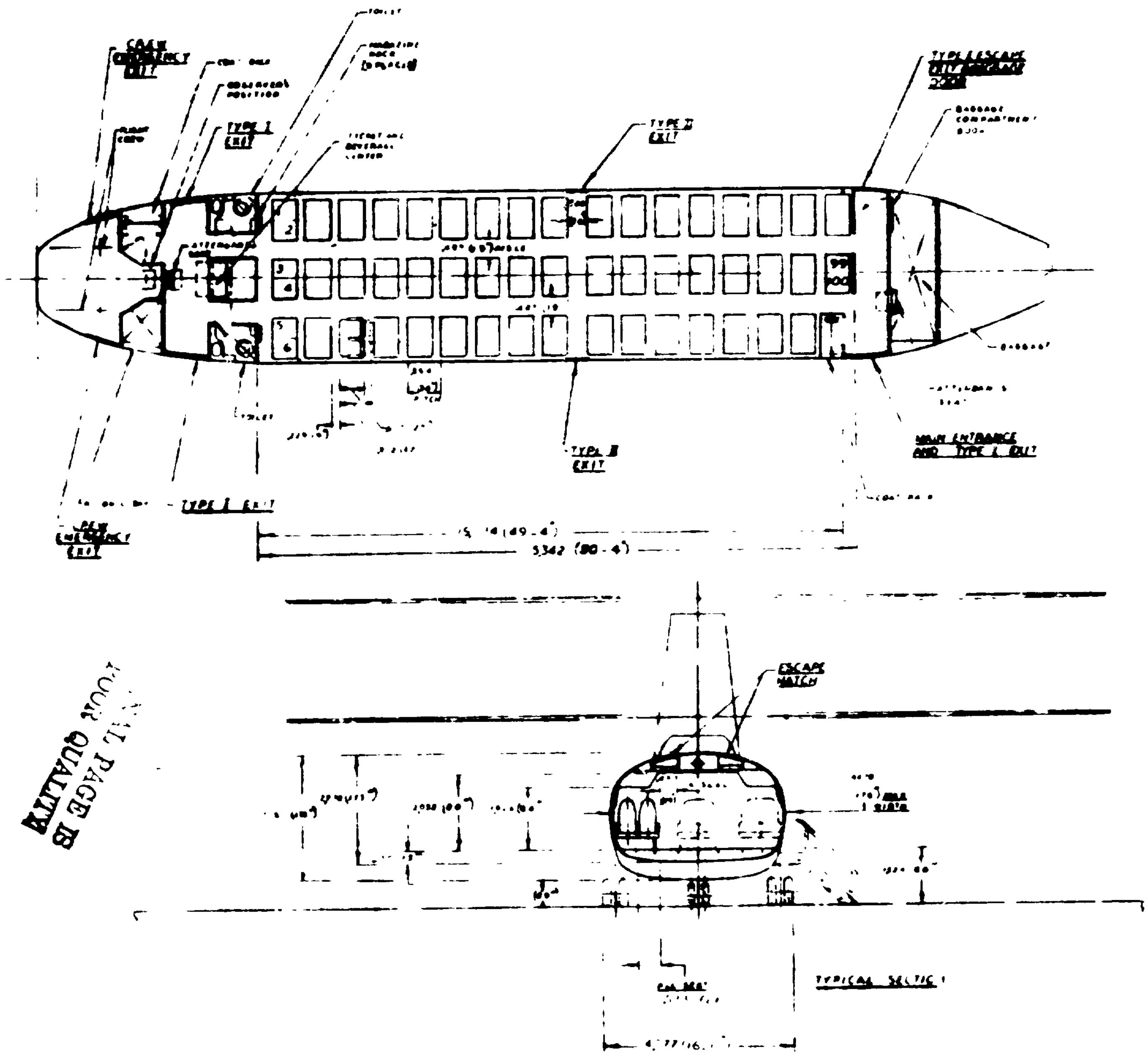


FIGURE 2.2. 1985 COMMERCIAL TRANSPORT,100 PASSENGER - TANDEM HELICOPTER - BASELINE DESIGN (CONTINUED).

cutting the trailed vortices of the other and also to eliminate the possibility of blade collision in the event of a desynchronization failure.

Both rotor shafts are swept forward (7-degrees forward rotor and 4-degrees aft rotor). This minimizes the floor angle range during hover and cruise flight, and also minimizes rotor loads. The pylon heights are arranged to provide a gap to stagger ratio of 0.1/3. This clearance is required to keep noise, rotor loads and induced power losses at a minimum.

The aircraft has three engines located aft, one on each side of the rear rotor pylon and the third buried in the pylon itself, similar to the XCH-62 (HLH). The intake for the third engine is shown in Figure 2.2, in the leading edge of the rear rotor pylon. The rationale for selecting a three-engine configuration is given in Volume II.

The transmission layout is a three gearbox arrangement where three engines drive into a combiner gearbox located aft and above the passenger cabin. The combiner box is designed for easy removal through the baggage hold ceiling.

Power is transmitted to the aft rotor by shafting in the rear pylon which drives the aft rotor transmission, and to the forward rotor by shafting along a fuselage tunnel to the forward rotor transmission located forward of the passenger cabin. The APU (Auxiliary Power Unit) is located in the aft fuselage compartment in close proximity to the engines.

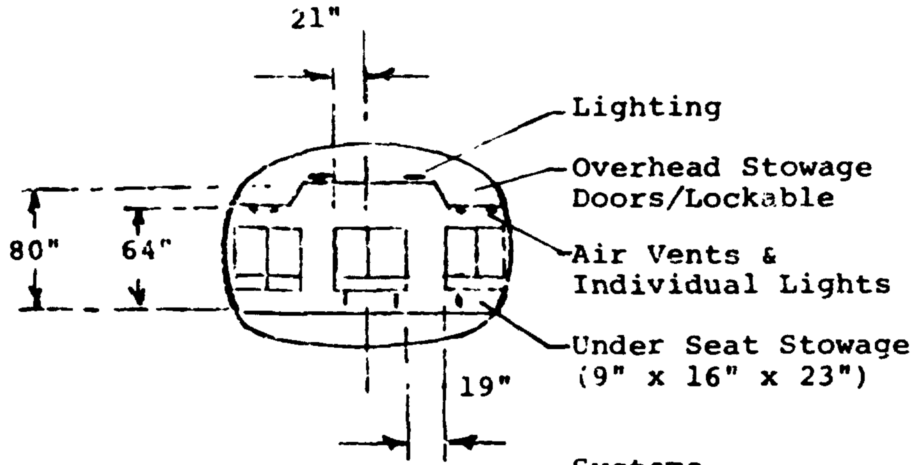
This arrangement has been selected for minimum complexity, cost, weight and performance losses as well as to minimize the effects of engine and transmission noise and vibration in the passenger cabin.

The fuel tanks are located under the rear cabin floor as shown in Figure 2.2. These tanks are "crashworthy" tanks similar to those built and tested by the Boeing Vertol Company for CH-46/47 applications (Volume II). The design philosophy is to provide adequate tank strength to ensure that no rupture will occur in the event of a 95th percentile crash. The system is designed for pressure refueling (300 gpm) with crossfeed valving, a fuel pump in each tank, and with fuel pump valves and lines routed away from the landing gear. The dual bleed conditioning system is located in the aft fuselage compartment adjacent to the APU and engine bays.

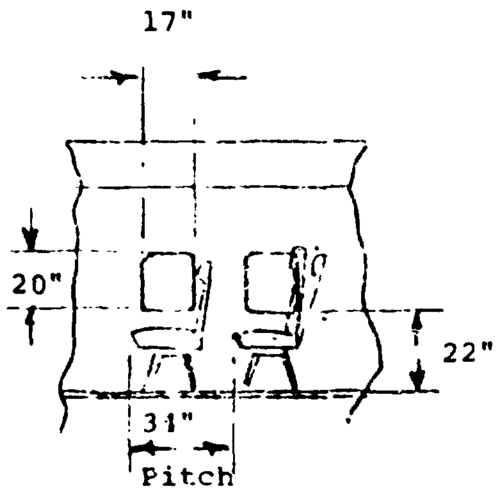
The landing gear is a tricycle layout providing excellent ground handling characteristics. The dual wheel gears are retractable into the fuselage for minimum drag and the system is designed for 500 feet per minute rate of sink on landing. The arrangement provides an overturning angle of 27-degrees and adequate fuselage clearance for flared landing.

Cabin layout and passenger accommodation details are shown in Figures 2.2 and 2.3. The aircraft cabin has two main entrances located on the port side of the aircraft. The aft entrance is equipped with an air stair in accordance with the study

Cabin



Windows



Systems

- Air Conditioning - Dual Bleed Air
- Unpressurized

Escape Provisions

- Two Type I Exits, Each Side
- One Type II Exit, Each Side
- Six Type IV Escape Hatches On Top

Entrances

- Two Main Entrances L. H. Side
- Air Stairs, Aft At Entrance
- Service Entrance, R. H. Side, Fwd.

Miscellaneous

- Coat Racks for 80 Passengers
- Two Magazine Racks
- Two Lavatories
- Beverage Service, Fwd.
- Ticket Center, Fwd.

FIGURE 2.3. TANDEM ROTOR HELICOPTER PASSENGER ACCOMMODATIONS.

guidelines. The rear entrance is the normal entrance and the exit is located adjacent to the stowed baggage compartment in the rear of the aircraft.

A third entrance is located on the starboard side of the cabin forward adjacent to the service facilities and serves the dual role as a service entrance, and an emergency exit.

A further Type I exit is located aft directly opposite the main entrance and again serves a dual role in that it can be used to load baggage by ground crew and also provides an emergency exit. This additional access provides the operator with flexibility in baggage handling procedures.

In addition to these, two additional Type II emergency exits are located amidships, one to each side. The location of these exits causes the pitch between the ninth and tenth rows of seats to be increased to 45-inches to allow a 20-inch wide access to the exit.

Six Type IV exits are provided in the cabin roof to be used in the event of an aircraft being turned over on one side.

The passenger cabin has seats for 100 passengers with an overall seat width of 21-inches and a seat pitch of 34-inches.

Each passenger has underseat stowage space (9-inches X 16-inches X 23-inches) and overhead rack stowage with lockable doors. Air vents, individual lights and a folding table are provided for each passenger in accordance with normal commercial aircraft practice.

The cabin has dual 19-inch aisles and the main cabin lights are located over the aisles.

Two coatracks are provided - one forward and one aft with provisions for 80 passengers.

Two lavatories are located in the forward end of the cabin.

In the center of the forward cabin is the beverage storage and service counter space which also incorporates ticketing facilities.

There are two cabin attendant seats. One is located forward against the forward passenger cabin bulkhead and close to the forward exits. The second is aft against the baggage hold bulkhead and close to the rear Type I exits.

The aircraft avionics and navigational gear compartment is on the port side of the aircraft just forward of the cockpit/cabin bulkhead. The cockpit space provides adequate accommodation for a flight crew of two with excellent visibility. A third "observer" seat is provided adjacent to the avionics compartment at the rear of the cockpit. This location provides the observer good forward vision, visibility over the flight crew stations and also access to the avionics/nav-aids compartment if required. The cockpit is provided with two crew emergency exits - one on each side of the cockpit.

2.1.2 Tandem Helicopter Design Point Weights

The design gross weight of the tandem rotor design point helicopter is 30,469 Kg (67,175 pounds). The aircraft empty weight is 18,221.8 Kg (40,179 pounds). Table 2.2 gives the

WEIGHT SUMMARY - PRELIMINARY DESIGN

MIL-STD-1374

D210-10858-1

	KILOGRAMS	POUNDS
WING		
ROTOR	3029.1	6678
TAIL		
SURFACES		
ROTOR		
BODY	2950.1	6504
BASIC		
SECONDARY		
ALIGHTING GEAR GROUP	1218.8	2687
ENGINE SECTION	222.7	491
PROPULSION GROUP	4401.2	9703
ENGINE INST'L	997.9	2200
EXHAUST SYSTEM *		
COOLING		
CONTROLS *		
STARTING *		
PROPELLER INST'L	*82.6	*182
LUBRICATING *		
FUEL	219.1	483
DRIVE	3101.6	6838
FLIGHT CONTROLS	1031.9	2275
AUX. POWER PLANT	288.5	636
INSTRUMENTS	191.9	423
HYDR. & PNEUMATIC	308.4	680
ELECTRICAL GROUP	378.3	834
AVIONICS GROUP	293.9	648
ARMAMENT GROUP		
FURN. & EQUIP. GROUP	3206.9	7070
ACCOM. FOR PERSON.		
MISC. EQUIPMENT		
FURNISHINGS		
EMERG. EQUIPMENT		
AIR CONDITIONING	521.6	1150
ANTI-ICING GROUP	181.4	400
LOAD AND HANDLING GP.		
WEIGHT EMPTY	18224.8	40179
CREW	299.4	660
TRAPPED LIQUIDS	52.2	115
ENGINE OIL	59.9	132
CREW ACCOMMODATIONS	68.0	150
EMERGENCY EQUIPMENT	7.3	16
PASSENGER ACCOMMO.	415.5	916
PASSENGERS (100)	8164.6	18000
FUEL	3178.3	7007
GROSS WEIGHT	30469.9	67175

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TABLE 2.2. DESIGN POINT HELICOPTER - WEIGHT SUMMARY.

weight breakdown in terms of structural components and aircraft systems. The weight of each component or system has been computed using the HESCOMP sizing program (Reference 1) which uses statistical and semi-empirical weight trend equations based on known aircraft weights. The sizing procedure is an iterative procedure in which the aircraft weight is varied until the mission fuel required is equal to the allocated fuel weight.

Weights of all structural components have been reduced by 25% from the trend curve data in keeping with the guideline directive on the use of composite materials.

Several standard item weights were also specified as shown in Table 2.3.

TABLE 2.3. WEIGHTS SPECIFIED BY STUDY GUIDELINES.

ITEM	WEIGHT
WHEELS, TIRES, AND BRAKES	COMPANY OPTIMUM
INSTRUMENTS (Flight and Navigation) ELECTRICAL (Excluding Generating Equipment) ELECTRONICS (Communication, Flight, and Navigation) AUXILIARY POWER UNIT INSTALLATION	1200 Pounds
SEATS AND BELTS	
PASSENGER: DOUBLE	16 Lbs/Passenger
PASSENGER: TRIPLE	16 Lbs/Passenger
CREW SEATS: CABIN CREW	16 Lbs/Crew Member
CREW SEATS: FLIGHT CREW	40 Lbs/Crew Member
LAVATORY	300 Lbs/Unit
BEVERAGE ONLY	200 Lbs Total
AIR STAIR	400 Lbs

The 544.2 Kg (1,200 pounds) allocated for auxiliary power unit, instruments, electrical and electronics has been assumed to be an uninstalled weight and an additional 440.8 Kg (9,721 pounds) has been included to account for installation.

The engine weights are based on a projected specific weight of .15 pounds per shaft horsepower which is expected to be available for application to a 1985 commercial aircraft.

The control system is a fly-by-wire system and the weight estimate for the controls is based upon recent Boeing experience with fly-by-wire controls on the Model 347 helicopter.

The rotor gearboxes are designed for maximum engine power and torque under sea level standard day conditions.

The landing gear is designed for a 500 foot per minute rate of descent and is 4% of weight empty.

Passenger and crew accommodations are based on Boeing 737 aircraft data since it will be necessary to provide passenger comfort to at least this standard by 1985.

The overall aircraft is sized for a maneuver load factor of 3.5 and an ultimate load factor of 5.25 as recommended in FAR Part 29.

The aircraft center of gravity and inertias for both design gross weight and weight empty are shown in Table 2.4. The aircraft CG envelope is shown in Figure 2.4. There is no need to restrict seating arrangements.

	WEIGHT EMPTY	GROSS WEIGHT
WEIGHT	18,224.8 Kg (40,179 LBS)	30,469.9 Kg (67,175 LBS)
CENTER OF GRAVITY*		
FUSELAGE STATION	15.25 M (600.4 IN.)	14.53 M (572.0 IN.)
WATER LINE	3.59 M (141.5 IN.)	2.83 M (111.5 IN.)
MOMENT OF INERTIA		
I_{xx} (ROLL)	89,392.5 Kg M^2 (10,143.5 Slug Ft^2)	96,121 Kg M^2 (10,907 Slug Ft^2)
I_{yy} (PITCH)	1,513,958.3 Kg M^2 (1,116,826.9 Slug Ft^2)	1,627,912 Kg M^2 (1,200,889 Slug Ft^2)
I_{zz} (YAW)	1,462,026.2 Kg M^2 (1,078,523.9 Slug Ft^2)	1,572,081 Kg M^2 (1,159,703 Slug Ft^2)

*FUSELAGE STATION 0 IS AT NOSE OF BODY, CENTERLINE OF FORWARD ROTOR 5.0 METERS ABOVE WATER LINE.

TABLE 2.4. WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA - DESIGN POINT HELICOPTER.

In order to provide ready comparison of this aircraft design weight with other designs with different fixed weights, the aircraft growth data are shown in Figure 2.5. This curve allows an aircraft weight to be obtained for a variation in a fixed weight item and allows reasonable comparison of weight with other designs based on different fixed equipment, etc. assumptions.

Detailed justification of the component and subsystem weights is provided in Volume II.

2.1.3 Vehicle Performance

Mission Performance

The design point tandem helicopter has been sized to fly the mission shown in Table 2.5 and Figure 2.6, with a range of 200 nautical miles.

A performance summary of the design point aircraft flying this mission is shown in Tables 2.6 and 2.7. The aircraft initial weight is 67,175 pounds. The aircraft is taxied with the engine at the ground idle engine rating, for a one minute period and 12.3 pounds of fuel is used. An additional 107 pounds of fuel is required to execute the takeoff, initial air maneuver and acceleration to climb speed. The aircraft then climbs to 5,000 feet altitude at a rate of climb of approximately 1,800 feet per minute.

The climb segment is accomplished in 2.76 minutes and requires 190.4 pounds of fuel and a distance of 4.26 nautical miles is covered.

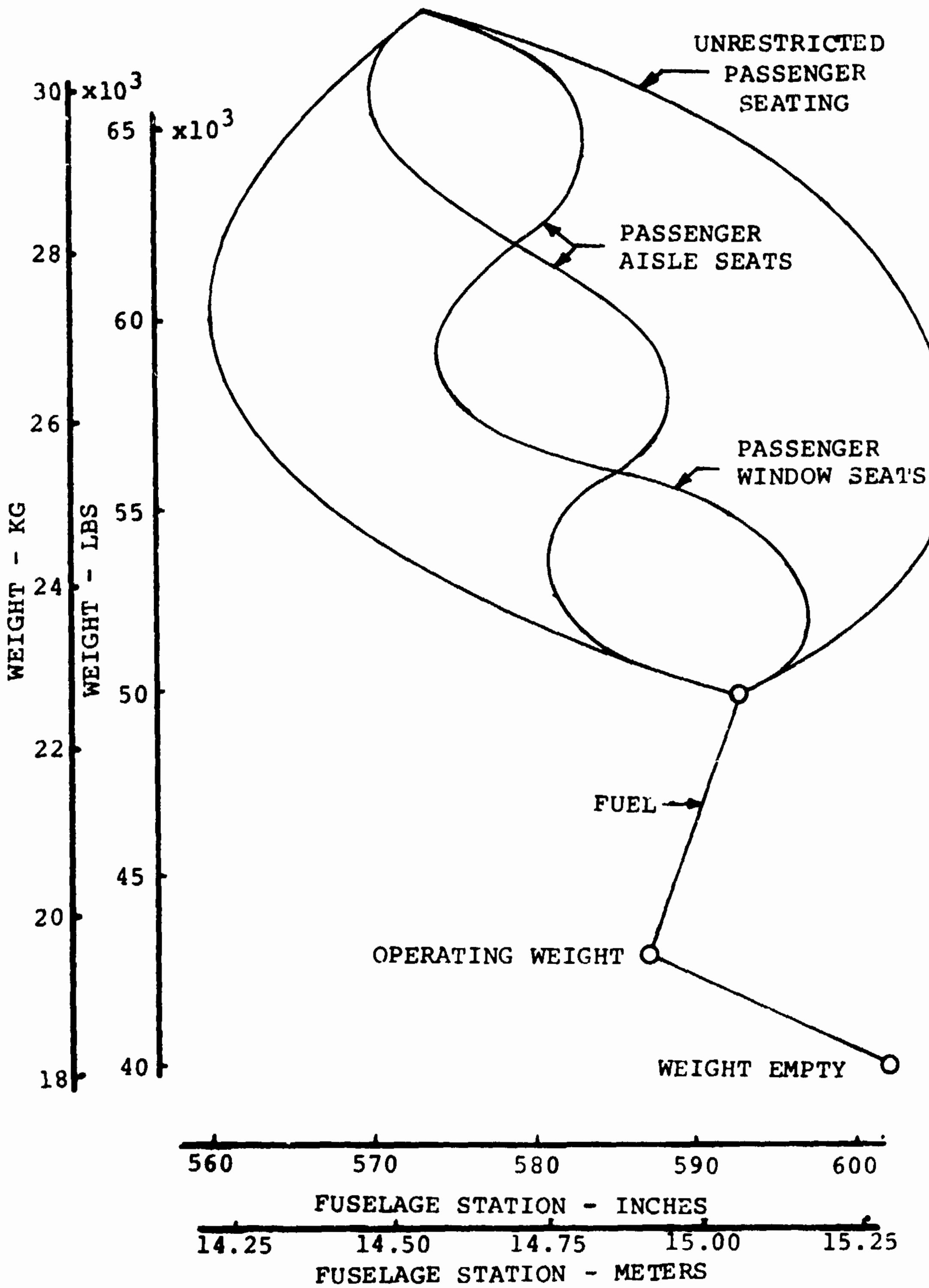


FIGURE 2.4. BASELINE TANDEM HELICOPTER - CENTER OF GRAVITY ENVELCPE.

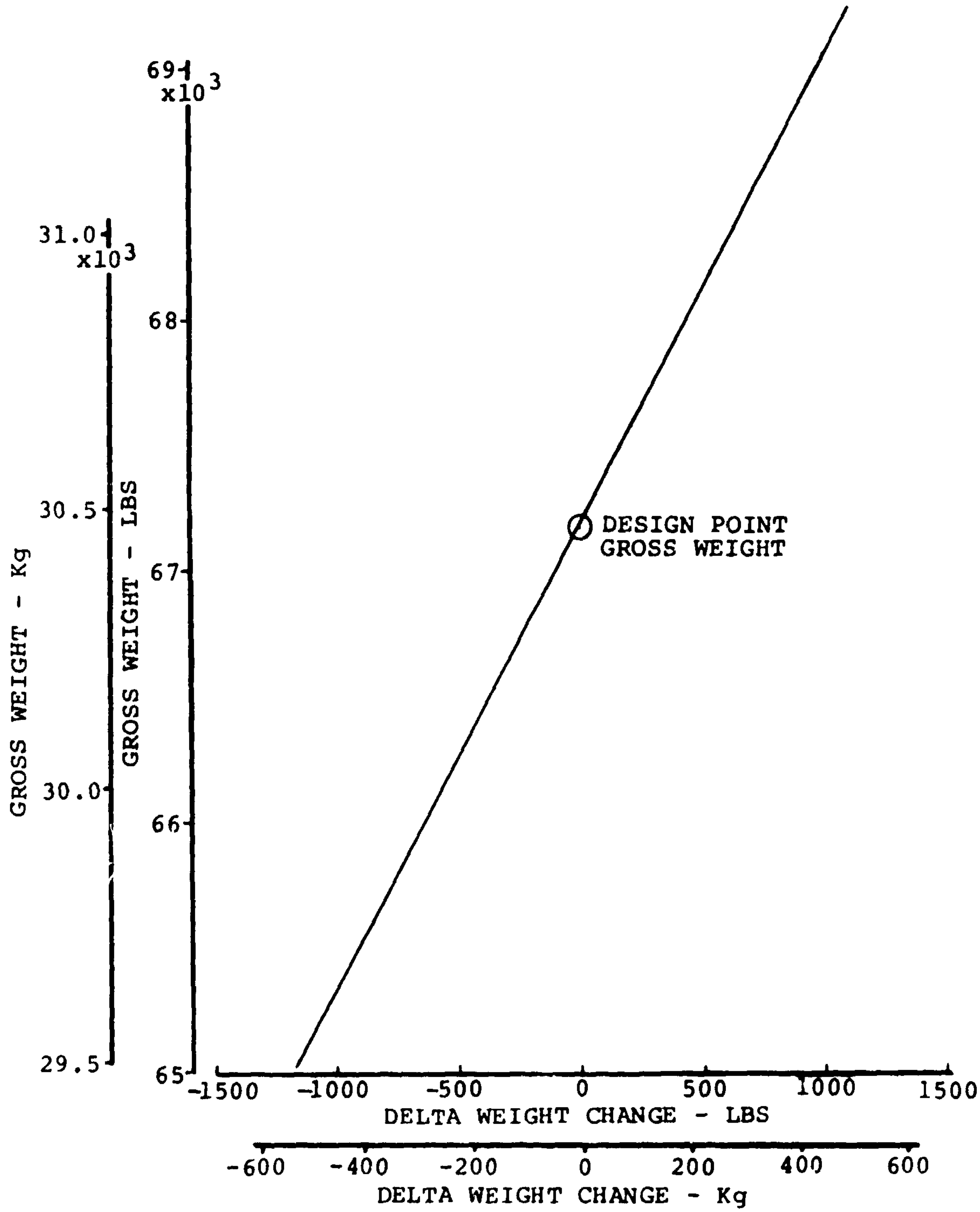


FIGURE 2.5. BASELINE HELICOPTER WEIGHT GROWTH AT CONSTANT PERFORMANCE AND STRENGTH.

SEGMENT	TIME VTOL	DISTANCE VTOL	REMARKS
Taxi Out	1 min.	0	
Takeoff, Transition & Conversion to Conventional Flight	0.5 min.	0	
Air Maneuver (Origin)	0.5 min.	0	
Acceleration to Climb Speed	As Calculated		
Climb	As Calculated		At optimum Climb Spd
Cruise	As Calculated		At Constant Integral 1000 ft. Altitudes(No Enroute Altitude Change
Descent to 2000 ft.	As Calculated		5000 fpm maximum rate of Descent
Air Maneuver at 2000 ft. (destination)	1.5 min.	0	
Decelerating Approach and Conversion to Powered Lift Flight 2000 ft. to 1000 ft.	As Calculated	0	1000 fpm maximum Rate of Descent
Transition and landing from 1000 ft. to Touchdown	As Calculated	0	1000 fpm maximum Rate of Descent Down to 35 ft. 600 fpm Maximum Rate of Descent Below 35ft.
Taxi In	1 min.	0	

TABLE 2.5 V/STOL MISSION PROFILE DEFINITION.

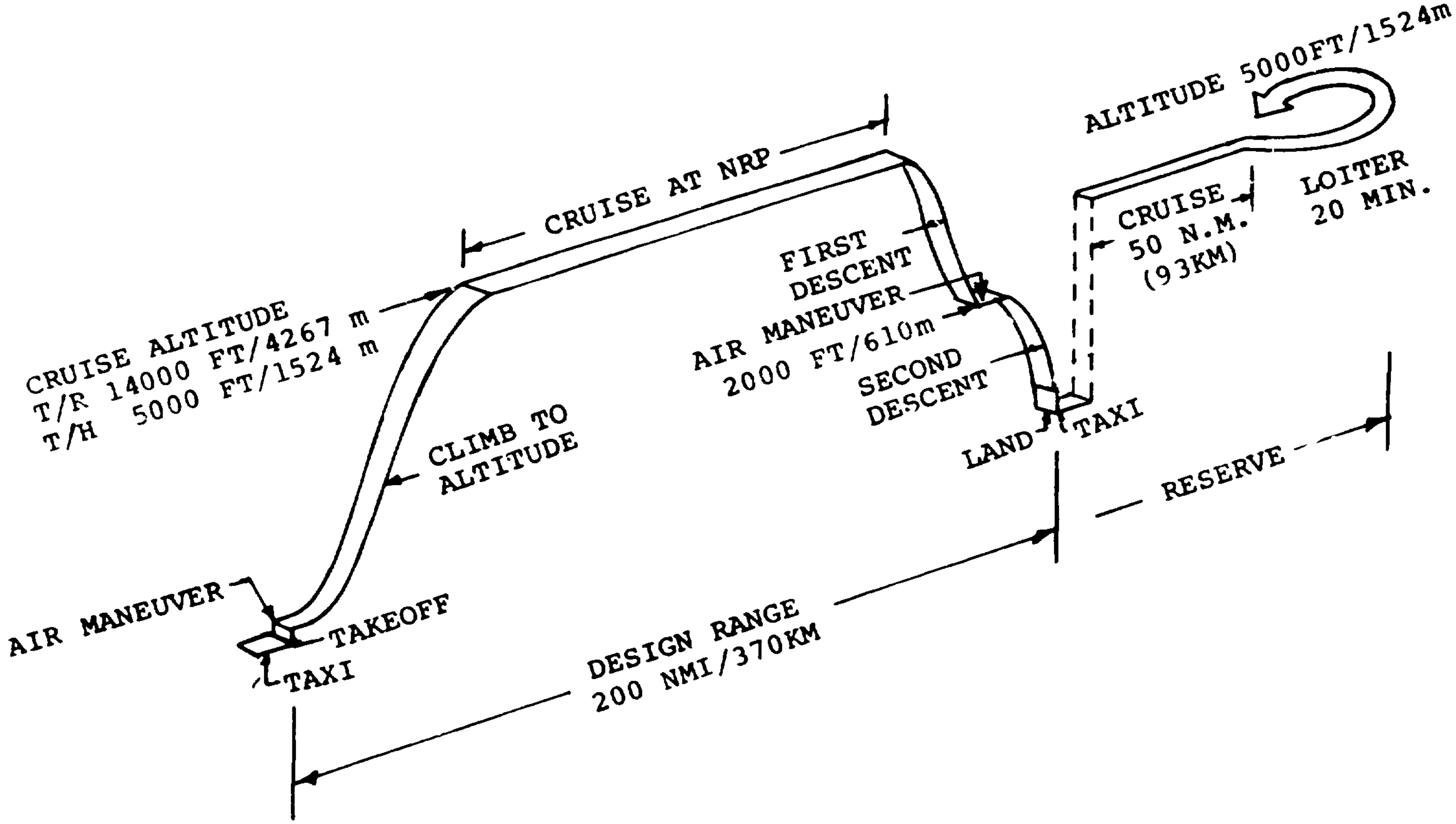


FIGURE 2.6. DESIGN SHORT HAUL MISSION.

BASELINE HELICOPTER - MISSION BREAKDOWN

	TIME (HOURS)	DISTANCE (N.MI.)	WEIGHT (LBS)	FUEL (LBS)	V (KNOTS)	R/C (FT/MIN)
	0	0	67,175		0	0
TAXI				12		
	.017	0	67,163			
TAKEOFF				107		
	.042	0	67,056			
CLIMB				191	92	1800
	.088	4.3	66,865			
CRUISE				4,582	165/170	0
	1.242	197.7	62,283			
DESCENT				25	115	-2460
	1.262	200	62,258			
AIR MANEUVER				65	92	0
	1.287	200	62,193			
DESCENT				35	70	-1000
	1.304	201	62,158			
LANDING				65	0	0
	1.321	201	62,093			
TAXI				12	0	0
	1.337	201	62,081			
RESERVE				1,916	150/93	0
	2.004	250	60,165			

TABLE 2.6. BASELINE HELICOPTER DESIGN MISSION PERFORMANCE (U.S. UNITS)

BASELINE HELICOPTER - MISSION BREAKDOWN

	TIME (HOURS)	DISTANCE (Km)	WEIGHTS (Kg)	FUEL (Kg)	V (KNOTS)	R/C (m/s)
	0	0	30,470			
TAXI				6	0	0
	.017	0	30,464			
TAKEOFF				49	0	0
	.042	0	30,416			
CLIMB				87	92	9.14
	.088	7.97	30,329			
CRUISE				2165	165/170	0
	1.242	366.34	28,251			
DESCENT				11	115	-12.5
	1.262	370.60	28,240			
AIR MANEUVER				29	92	0
	1.287	370.60	28,210			
DESCENT				16	70	-5.1
	1.304	372.45	28,194			
LANDING				29	0	0
	1.321	372.45	28,165			
TAXI				6	0	0
	1.337	372.45	28,159			
RESERVE				869	150/93	0
	2.004	463.25	27,290			

TABLE 2.7. BASELINE HELICOPTER DESIGN MISSION PERFORMANCE (S.I. UNITS).

The cruise segment of the mission is done at 5,000 feet altitude. At the start of the cruise segment of the mission the aircraft weight is 66,865 pounds and the airspeed is 165.6 knots (TAS). After cruising for 1.154 hours the aircraft has travelled a total of 197 nautical miles and the aircraft weight has reduced to 62,294 pounds, and the cruise speed has increased to 170 knots (TAS). The fuel for the cruise segment is 4670.4 pounds.

The descent segment to 2,000 feet altitude completes the range to 200 nautical miles at rate of descent of 2,460 feet per minute (within the specified maximum of 5,000 feet per minute (Table 2.5).

The air maneuver at 2,000 feet altitude has been computed as a loiter for 1.5 minutes and requires 64.9 pounds of fuel. This is followed by the final descent to 1,000 feet altitude on a spiral descent flight path at 1,000 feet per minute rate of descent.

Descent from 1,000 feet and landing takes 1.5 minutes and is followed by a taxi segment at ground idle engine rating for one minute.

This completes the 200 nautical mile mission with a block time of 1.337 hours and a fuel burn-off of 5,092.5 pounds and a final aircraft weight of 62,082 pounds.

The reserve fuel is calculated for a range increment of 50 nautical miles at 99% best range speed and a loiter for 20 minutes. The reserve fuel required is 1,914 pounds giving a total fuel load of 7,006.81 pounds.

Hover Performance

The hover performance of the design point tandem rotor helicopter is shown in Figures 2.7 and 2.8.

Data given for both all engines operating (AEO) and one engine inoperative (OEI) as well as in and out of ground effect (IGE, OGE) is included.

The design point aircraft is sized to a 90-degrees F sea level condition OEI. This point is shown on Figure 2.7 at a hover weight of 67,175 pounds. The OEI data assumes a force-to-weight ratio (F/W) of 1.03.

The requirement to size the transmission to maximum sea level shaft horsepower provides OEI performance which is power limited. In the all engines operating case the torque limit is set such that both power and torque limit coincide at 59-degrees F ambient temperature.

Maintaining a one engine out requirement and operating at standard day out of ground effect, the aircraft can take off at a gross weight of 74,700 pounds, an increase of 7,525 pounds. This would not be allowable as extra payload since the FAA takeoff gross weight certification would limit the aircraft to 67,175 pounds. The higher weight would isolate the design load factor capability. This extra lift represents increased force-to-weight capability (F/W = 1.16) at sea level standard.

BASELINE AIRCRAFT PERFORMANCE

TANDEM HELICOPTER/100 PASSENGER/92.3 PNdB

ALL ENGINES OPERATING

F/W = 1.05

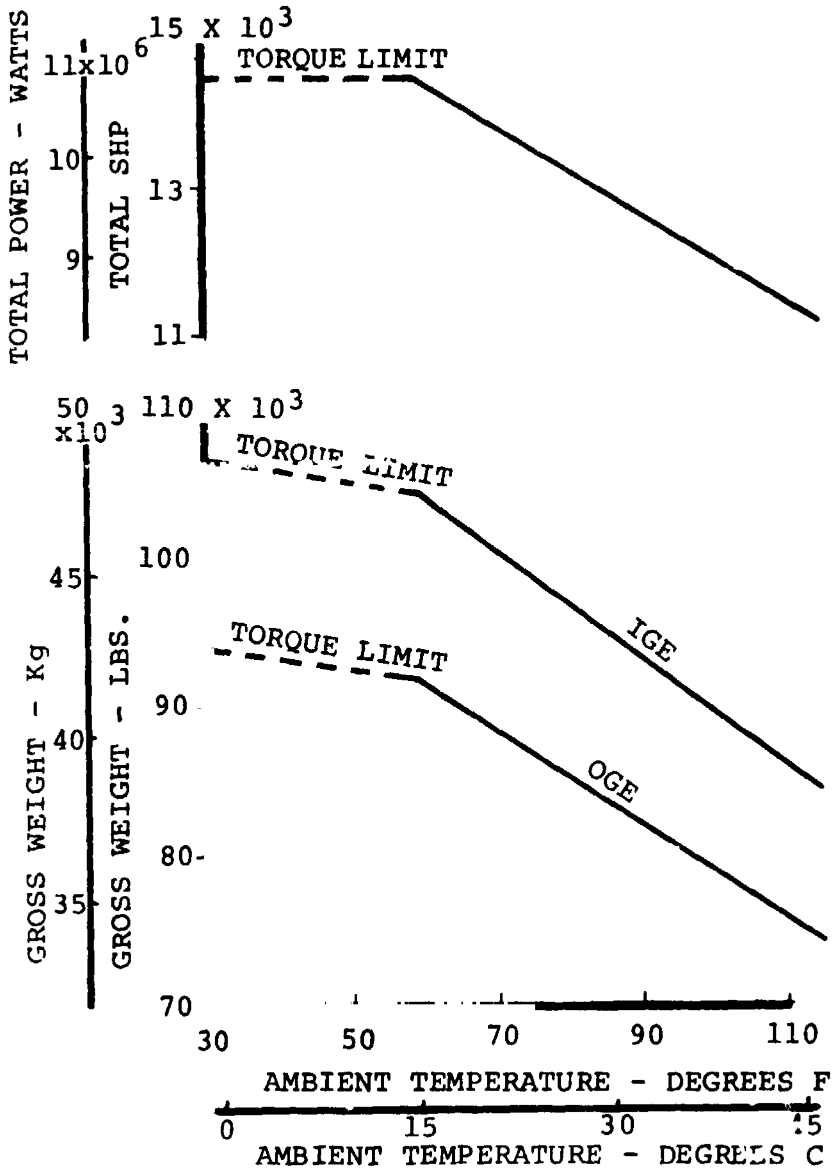


FIGURE 2.7. EFFECT OF AMBIENT TEMPERATURE ON SEA LEVEL TAKEOFF WEIGHT

BASELINE AIRCRAFT PERFORMANCE

TANDEM HELICOPTER/100 PASSENGER/92.3 PNdB

ONE ENGINE INOPERATIVE

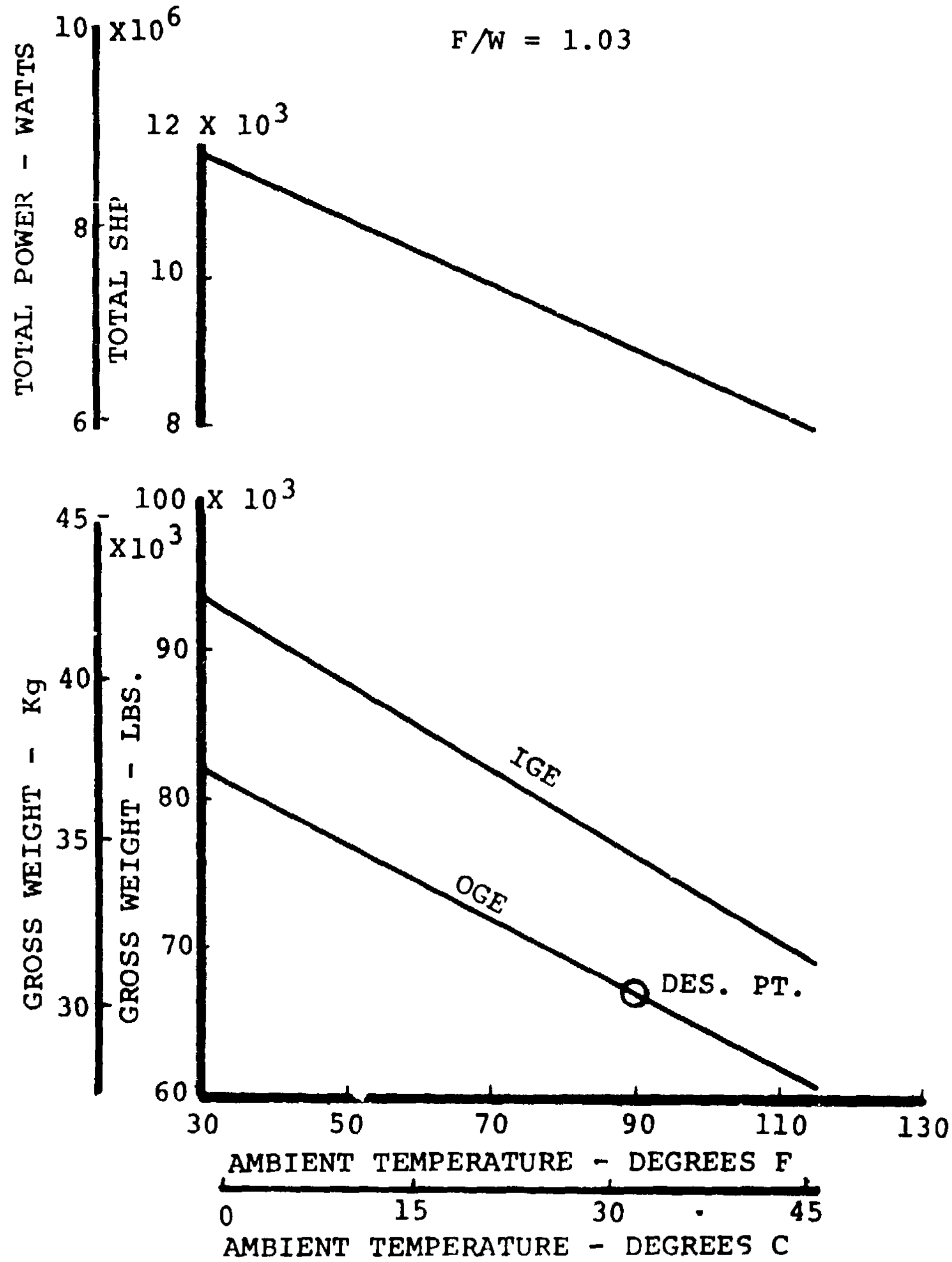


FIGURE 2.8. EFFECT OF AMBIENT TEMPERATURE ON SEA LEVEL TAKEOFF WEIGHT

With all engines operating out of ground effect, the aircraft lift capability provides an allowable force-to-weight ratio of 1.47 at 59-degrees F; at 90-degrees F this reduces to 1.31.

Data are provided for hover in ground effect. This demonstrates increased static lift capability better than the certified value. Again this increased capability can only be considered as an additional force-to-weight capability on takeoff or as a ground cushion in a landing flare.

The effect of altitude on hover performance is shown in Figure 2.9 for all engines operating. The fully loaded (100 passengers) aircraft could hover up to an altitude of 11,500 feet on a standard day and 8,000 feet for an ambient of standard plus 31-degrees F. The operating altitude is significantly less than this.

The altitude performance with one engine inoperative is shown in Figure 2.10. The design point aircraft is shown at sea level 90-degrees. For a standard day the OEI altitude capability increases to 4,500 feet.

Hover Download

An important issue in the prediction of hover performance and in the sizing of the design point aircraft installed power is the estimation of the download on the aircraft fuselage due to the downwash from the rotors. This effect has been computed using a semi-empirical technique described in Section 3.1, Volume II.

BASELINE AIRCRAFT PERFORMANCE

TANDEM HELICOPTER/100 PASSENGER/92.3 PNdB
STANDARD DAY AND STANDARD DAY + 31°F (+17.2°C)
ALL ENGINES OPERATING

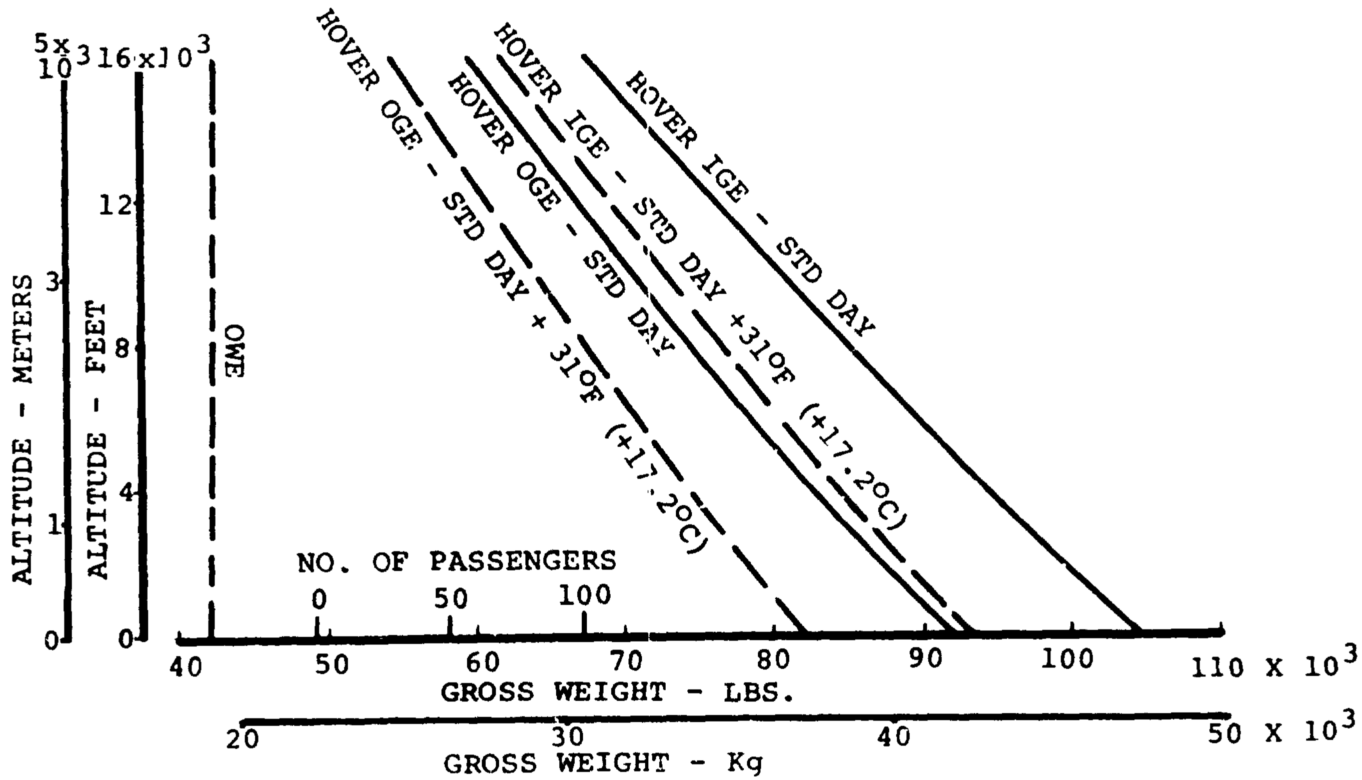


FIGURE 2.9. EFFECT OF ALTITUDE ON HOVER GROSS WEIGHT CAPABILITY.

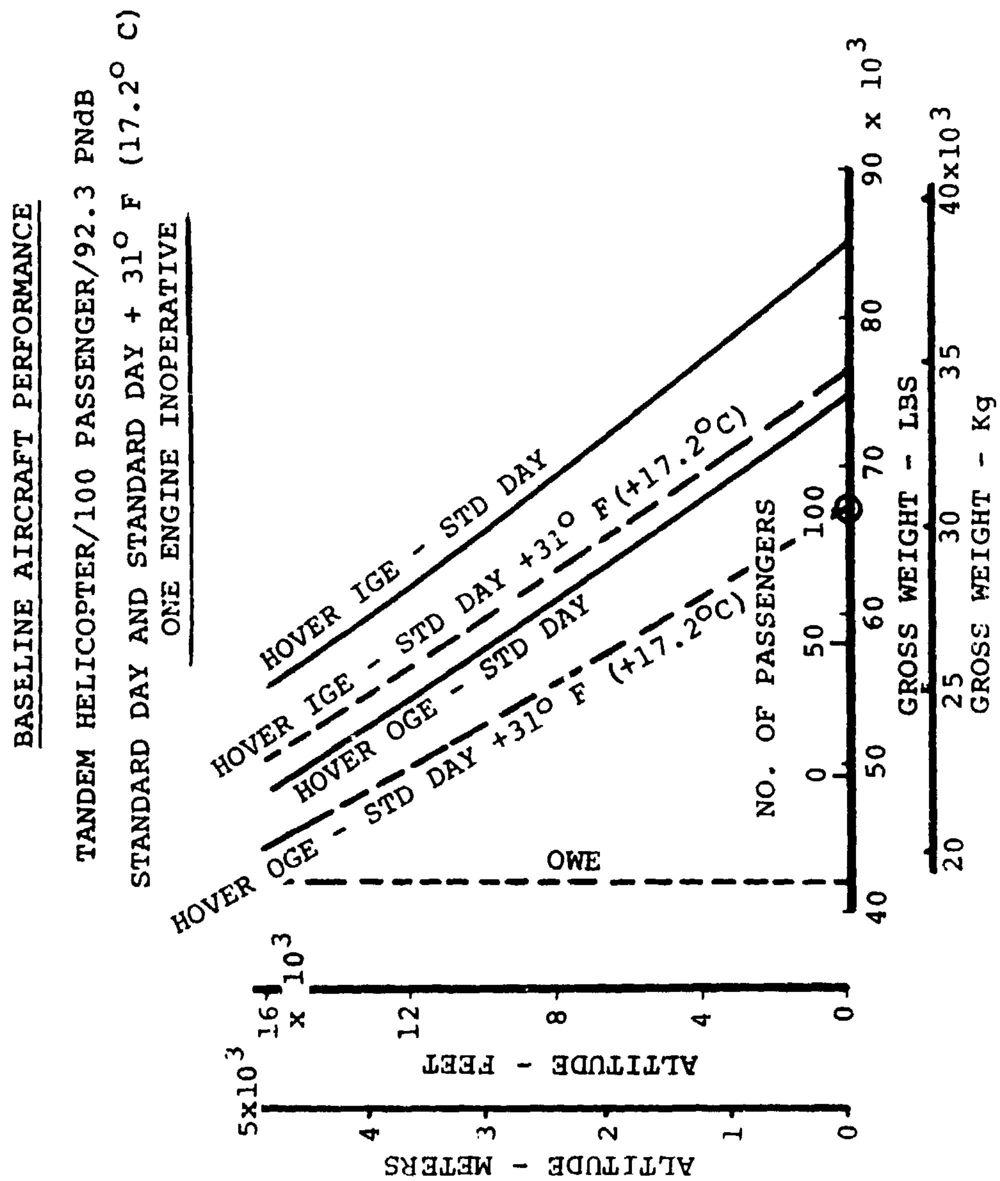


FIGURE 2.10. EFFECT OF ALTITUDE ON HOVER GROSS WEIGHT CAPABILITY.

BASELINE AIRCRAFT PERFORMANCE

TANDEM HELICOPTER/100 PASSENGER/92.3 PNdB

DGW = 67,175 LBS/30,470 Kg
MIDWT = 59,175 LBS/26,841 Kg
OWE = 42,168 LBS/19,127 Kg

ALTITUDE = SEA LEVEL
STANDARD DAY

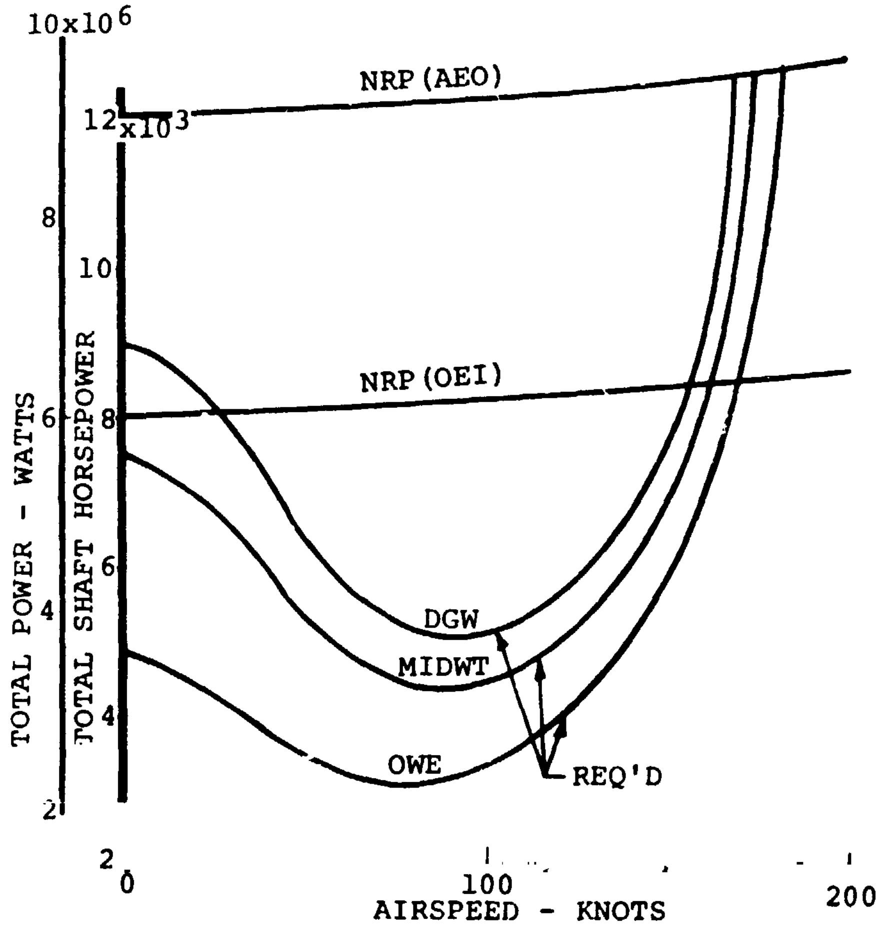


FIGURE 2.11. CRUISE PERFORMANCE - POWER REQUIRED/AVAILABLE, STANDARD DAY.

BASELINE AIRCRAFT PERFORMANCE

TANDEM HELICOPTER/100 PASSENGER/92.3 PNdB

DGW = 67,175 LBS/30,470 Kg
MIDWT = 59,175 LBS/26,840 Kg
OWE = 42,168 LBS/19,127 Kg

ALTITUDE = 5000 FEET (1524 m)
STANDARD DAY

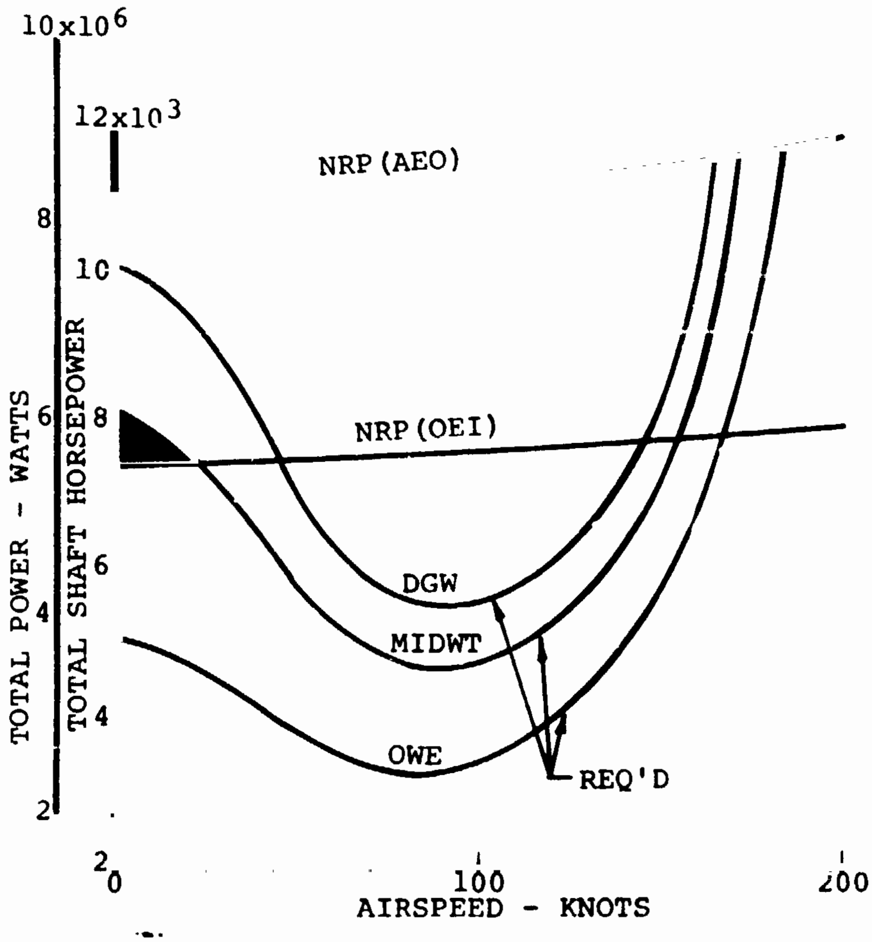


FIGURE 2.12. CRUISE PERFORMANCE - POWER REQUIRED/AVAILABLE, STANDARD DAY.

BASELINE AIRCRAFT PERFORMANCE

TANDEM HELICOPTER/100 PASSENGER/92.3 PNdB

STANDARD DAY
AEO & OEI

NORMAL RATED POWER
CRUISE RPM

DGW = 67,175 LBS/30,470 Kg
OWE = 42,168 LBS/19,127 Kg

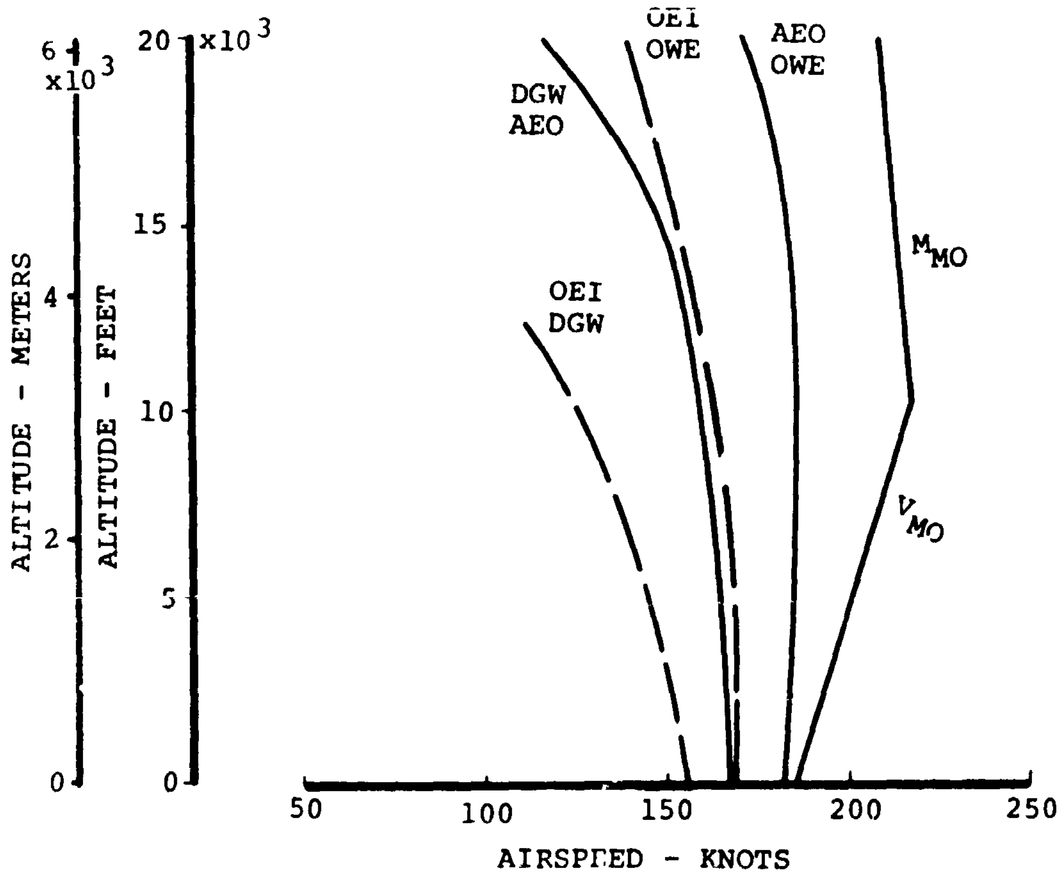


FIGURE 2.13. LEVEL FLIGHT CRUISE SPEED ENVELOPE.

The download on the aircraft at the design hover condition is 8.6% of the rotor thrust and this lift loss has been included in the sizing and performance calculations.

Performance at Forward Speed

The design point tandem helicopter power required and power available data are shown in Figures 2.11 and 2.12 for both sea level and 5,000 feet (design cruise) altitudes for standard day conditions. Power required data are given for three different aircraft weights ranging from operating weight empty to design gross weight.

At zero airspeed for both altitudes the aircraft power required is less than NRP (normal rated power).

The intersections of the power required and available lines indicate the maximum cruise speed performance capability. The design gross weight aircraft can fly at 168 knots at sea level all engines operating. This speed increases to 182 knots at operating weight empty. With one engine inoperative a maximum speed of 156 knots can be achieved at design gross weight at sea level.

At 5,000 feet altitude the NRP cruise speed is 165 knots at design gross weight and 185 knots at weight empty. These speeds decrease to 145 knots and 169 knots respectively with one engine inoperative.

The speed performance capability as a function of altitude is shown in Figure 2.13.

Rate of Climb

Rate of climb capability is shown as a function of altitude and gross weight in Figure 2.14. The two conditions of both all engines operating and one engine inoperative are shown. At design gross weight the aircraft achieves a maximum rate of climb of 3,650 feet per minute all engines operating at sea level. At design cruise altitude (5,000 feet) a climb rate of 2,910 feet per minute can be maintained.

With one engine incperative a rate of climb of 1,670 feet per minute can be maintained dropping to 1,200 feet per minute at design cruise altitude.

At minimum weight or operating weight empty the rate of climb capability increased to 6,900 feet per minute at sea level with all engines operating and 3,900 feet per minute with one engine inoperative. The engine power setting used for all climb calculations is a MIL rating.

Specific Range

The fuel consumption of the aircraft in cruise at both sea level and 5,000 feet altitudes, all engines operating, is given in Figure 2.15 for the range of aircraft weights. At design gross weight the aircraft achieves a maximum specific range of 0.0425 nautical miles per pound fuel at 140 knots at sea level. This improves to .044 nautical miles per pound at 500 feet.

BASELINE AIRCRAFT PERFORMANCE

TANDEM HELICOPTER/100 PASSENGER/92.3 PNdB

CLIMB CAPABILITY TAKEOFF RPM
STANDARD DAY MIL POWER

AEO & OEI

DGW = 67,175 LBS/30,470 Kg
OWE = 42,168 LBS/19,127 Kg

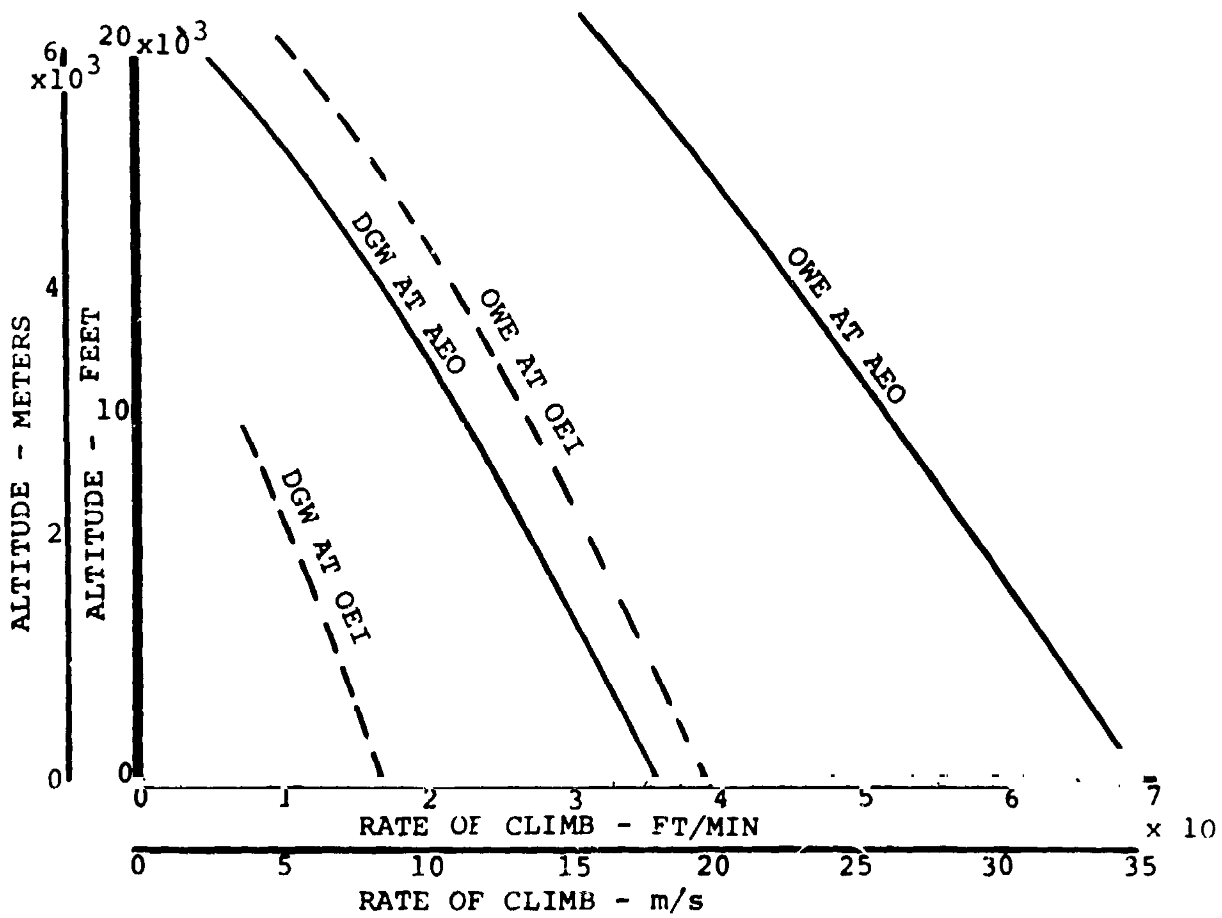


FIGURE 2.14 . BASELINE HELICOPTER DESIGN POINT AIRCRAFT-CLIMB CAPABILITY.

BASELINE AIRCRAFT PERFORMANCE

TANDEM HELICOPTER/100 PASSENGER/92.3 PNdB

DGW = 57,175 LBS/30,470 Kg
MIDWT = 59,175 LBS/26,841 Kg
OWE = 42,168 LBS/19,120 Kg

ALL ENGINES OPERATING

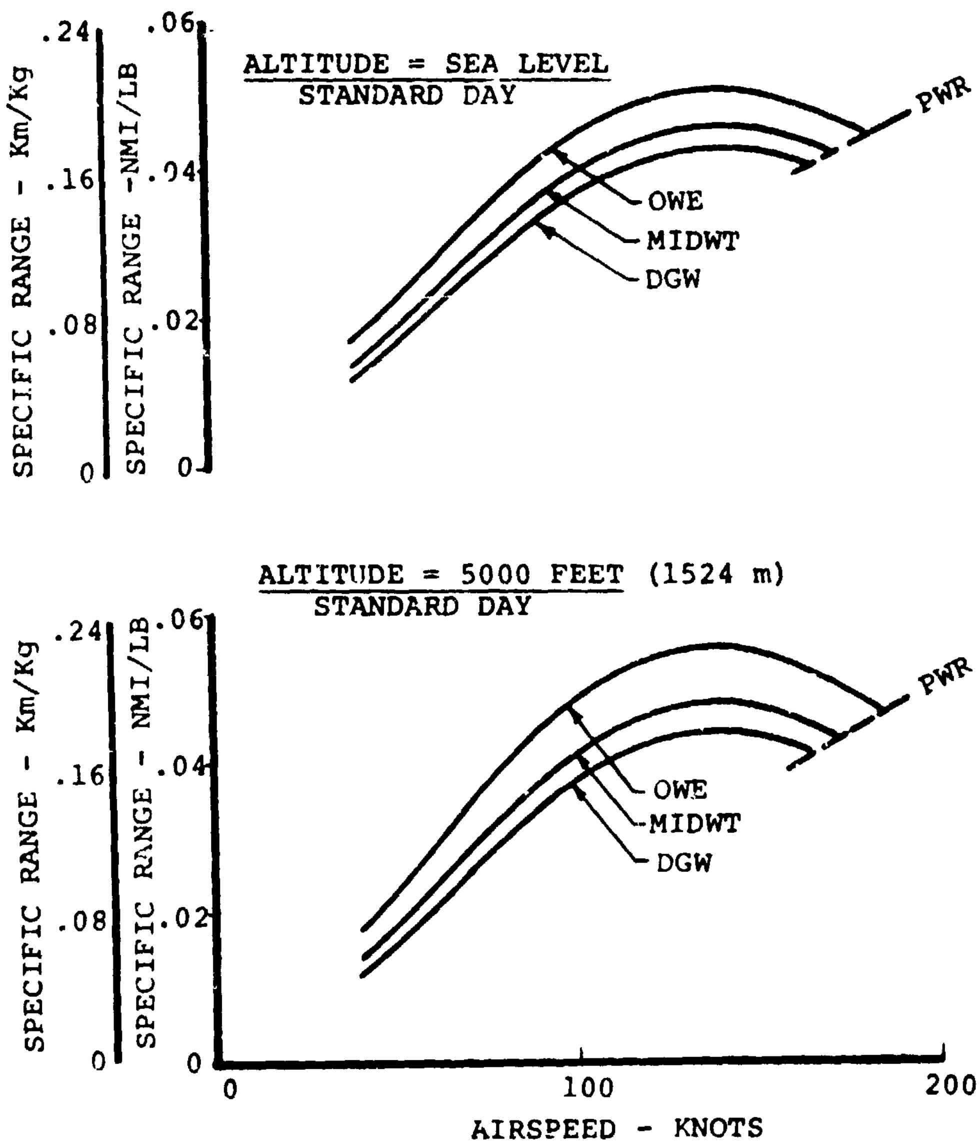


FIGURE 2.15. CRUISE PERFORMANCE - SPECIFIC RANGE - STANDARD DAY - AEO.

At the NRP cruise speeds the specific range is lower - 0.04 nautical miles per pound at design gross weight sea level and 0.041 at 5,000 feet.

With one engine inoperative (Figure 2.16) the specific range of the aircraft increases due to the increased power setting of the remaining two engines. On a standard day at sea level a specific range of 0.048 nautical miles per pound of fuel can be achieved at design gross weight at 131 knots. At 5,000 feet altitude the maximum specific range is slightly higher at 0.0485 nautical miles per pound of fuel at design gross weight.

Payload Range

The payload range performance was a specified criterion for the design point aircraft, and is shown in Figure 2.17. The design range is 200 nautical miles with a full load of 100 passengers. Reserve fuel as defined in the mission profile is still available at 200 nautical miles. The basic 200 nautical mile mission fuel limit defines the range of lighter weights such that with no passengers on board the range increases to 241 nautical miles.

An extended range version of the design point aircraft has been considered by the addition of extra fuel tanks and removing two passengers to allow for the tank weight increase. This aircraft would carry 98 passengers 200 nautical miles or could be used for 72 passengers up to 400 nautical mile range. The basic aircraft payload-range capability increases

BASELINE AIRCRAFT PERFORMANCE

TANDEM HELICOPTER/100 PASSENGER/92.3 PNdB

DGW = 67,175 LBS/30,470 K_g
MIDWT=59,175 LBS/26,841 K_g
OWE = 42,168 LBS/19,120 K_g

ONE ENGINE INOPERATIVE

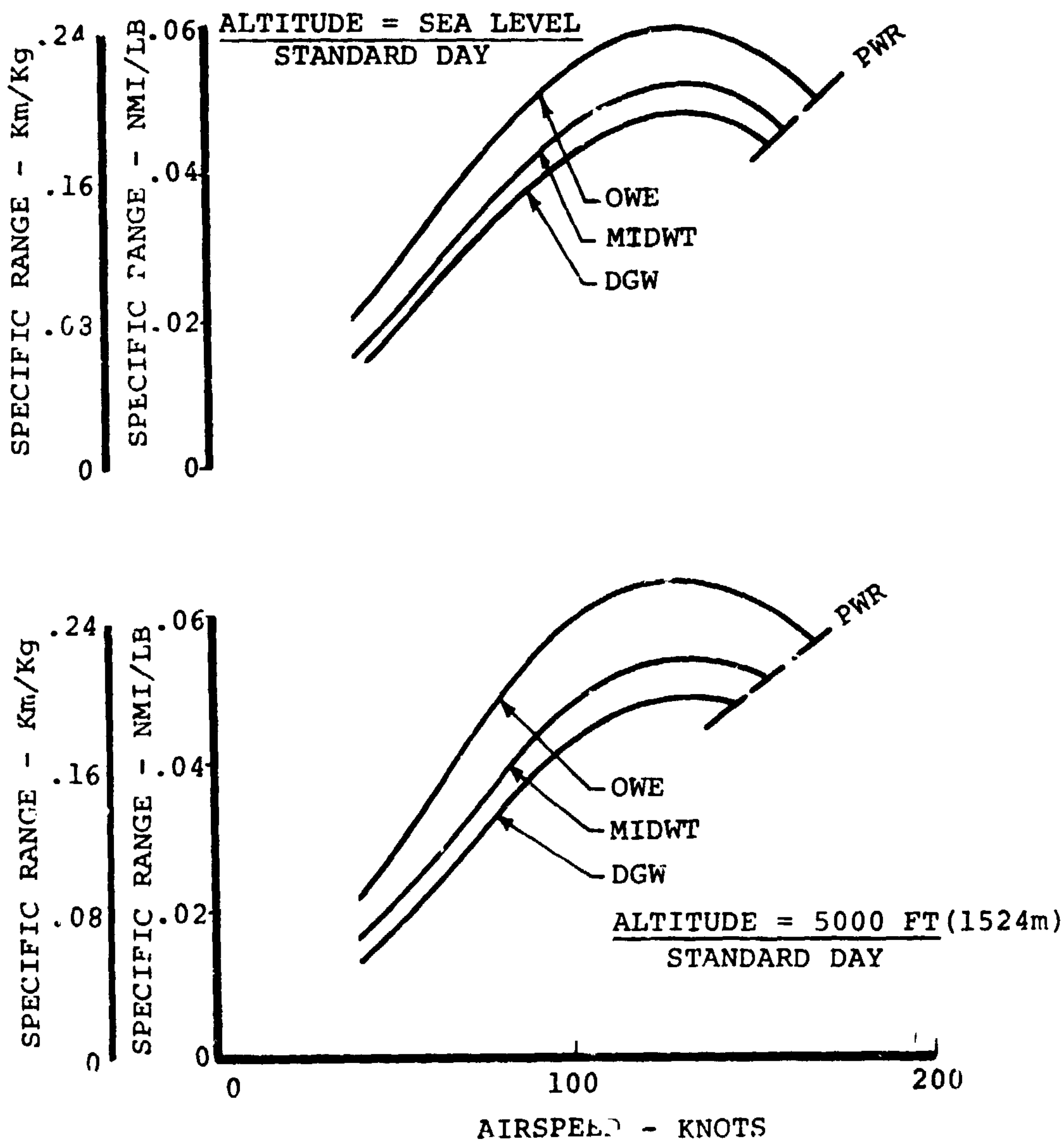


FIGURE 2.16. CRUISE PERFORMANCE - SPECIFIC RANGE - STANDARD DAY - OEI.

<u>BASELINE AIRCRAFT PERFORMANCE</u>

TANDEM HELICOPTER/100 PASSENGER/92.3 PNdB

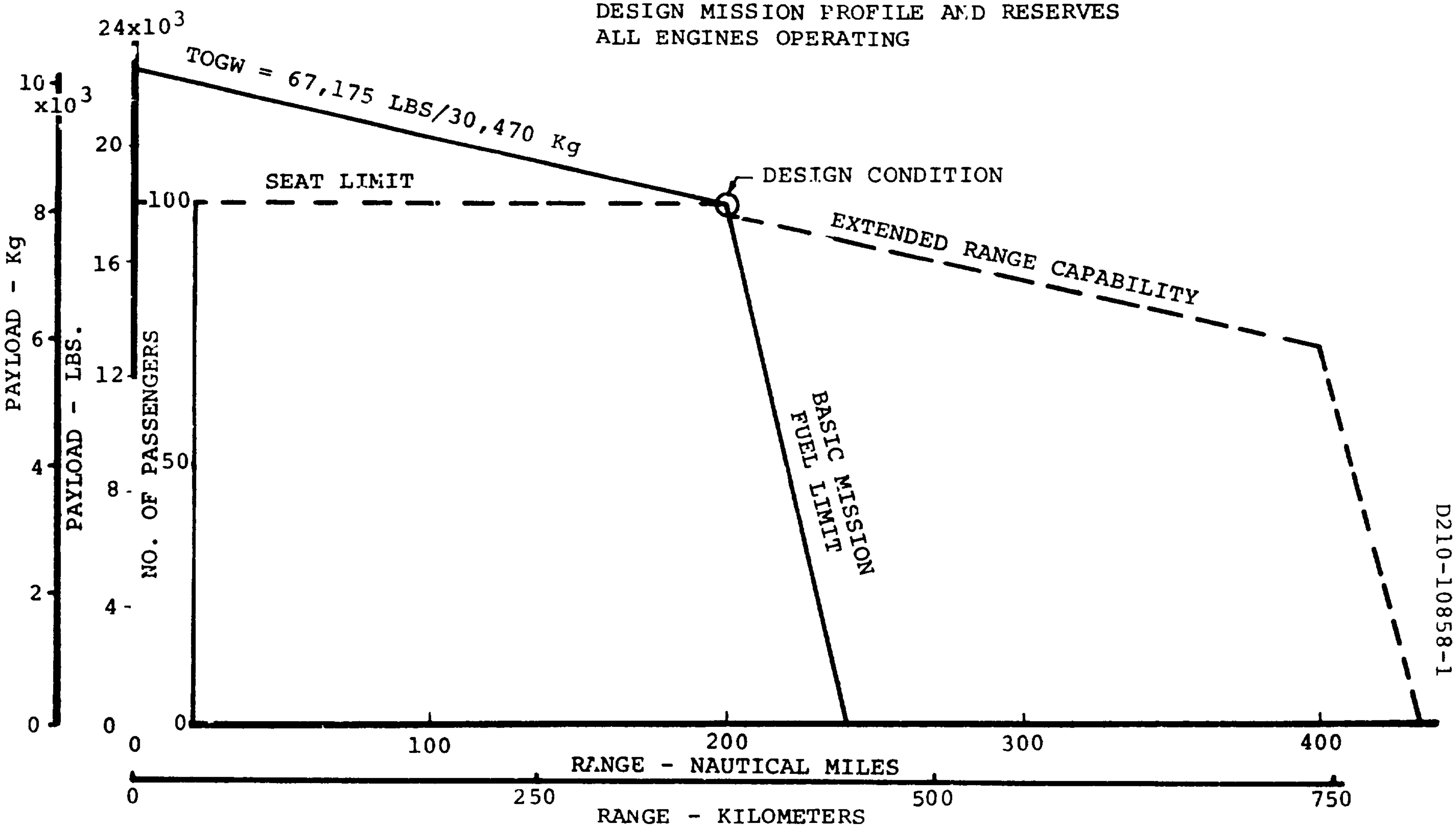


FIGURE 2.17. PAYLOAD RANGE CAPABILITY - ALL ENGINES OPERATING.

with one engine inoperative since the remaining engines are now operating at a higher percentage of power available which improves the engine SFC. Assuming cruise OEI the fully loaded aircraft range increases to 250 nautical miles and 325 nautical miles with no passenger load.

Drag

The minimum drag of the design point aircraft is shown in Table 2.8 in terms of equivalent drag area (F_e). The major contributions are from the fuselage (10.019 Ft^2), and the rotor hubs (20.2 Ft^2). The total aircraft F_e is 38.51 square feet giving a gross weight to F_e ratio of

$$\frac{GW}{F_e} = 1,765 \text{ Lbs/Ft}^2$$

A description of the drag methodology and justification for this drag estimate is given in Volume II.

2.1.4 Flying Qualities - Design Point Tandem Helicopter

Aircraft Trim

Trim data for the baseline tandem helicopter have been computed for a wide variation of aircraft weight and CG. The lightest weight considered is the operating weight empty 43,000 pounds at a 586 inch CG location. Two CG locations have been taken for a mid range aircraft weight of 57,500 pounds (FWD 556 inches and AFT 610 inches) and at design gross weight (67,175 pounds) a CG range from 560 inches (FWD) to 590 inches (AFT) has been used. The CG locations are given by the fuselage reference station locations.

TANDEM HELICOPTER DRAG SUMMARY

ITEM	DRAG AREA f_e - Ft^2
FUSELAGE	10.0193
FORWARD PYLON	2.8842
AFT PYLON	3.0609
NACELLES	1.4618
MISCELLANEOUS	
OIL COOLER MOMENTUM LOSS	.3000
AIR CONDITIONING	.5000
TRIM	.0900
SUB TOTAL	18.3162
ROTOR HUBS	20.2
TOTAL DRAG AREA	38.5162

$$\frac{GW}{f_e} = \frac{67175}{38.5162} = 1{,}744 \text{ LBS/FT}^2$$

TABLE 2.8. TANDEM HELICOPTER - BASELINE AIRCRAFT DRAG SUMMARY.

BASELINE AIRCRAFT PERFORMANCE

TANDEM HELICOPTER/100 PASSENGER/92.3 PNdB
OEI FOR CRUISE AND RESERVE SEGMENTS
DESIGN MISSION AND RESERVES

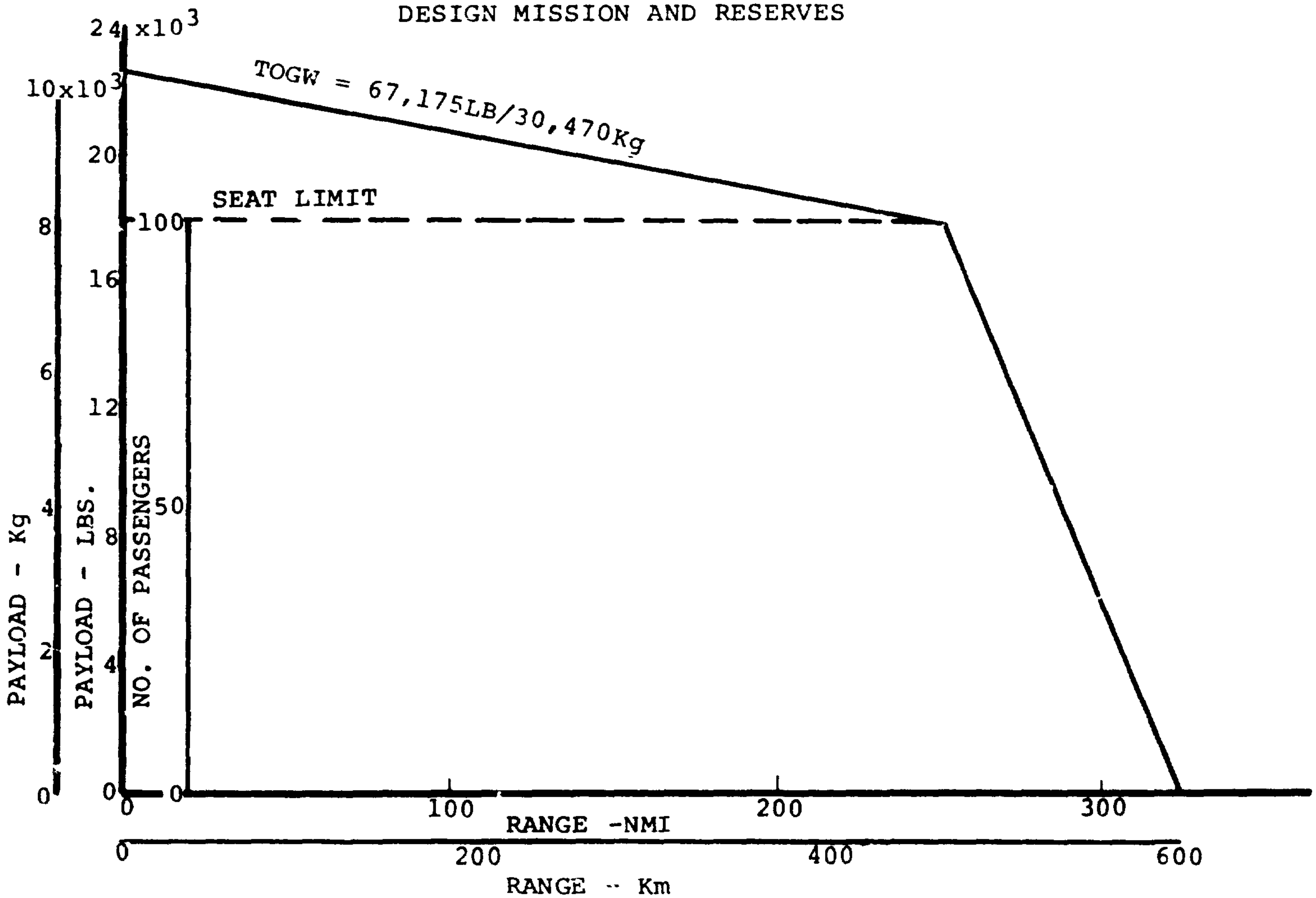


FIGURE 2.18. PAYLOAD RANGE CAPABILITY - CRUISE AT NRP AND CRUISE RPM.

The aircraft attitudes and control deflections over the entire speed range are shown in Figures 2.19 and 2.20. Data in the airspeed range from zero to 60 knots have been computed at the takeoff ambient conditions of sea level 90-degrees F. From 100 knots to maximum airspeed, the cruise altitude of 5,000 feet at 41-degrees F (standard) is assumed. The 60 knot to 100 knot airspeed range is an altitude transition.

The flight control kinematics and cumulative limit data are given in Table 2.9. These control ranges are based upon analysis of the HLH aircraft flight controls. The large collective range is selected to provide autorotative capability at light gross weight at 90% RPM and to absorb full transmission power at light gross weight for a power climb. Differential collective pitch, lateral and pedal ranges have been selected in accordance with MIL 8501A

For all gross weights and CG positions the variation of fuselage incidence over the range of airspeeds is small as a result of the large cyclic trim range available. The effect of gross weight and CG position on attitude is also small which is an inherent advantage of the tandem rotor helicopter from a passenger comfort standpoint. The aft rotor cyclic is scheduled with gross weight to minimize aft rotor flapping excursions and reduce a tendency to aeromechanical resonance.

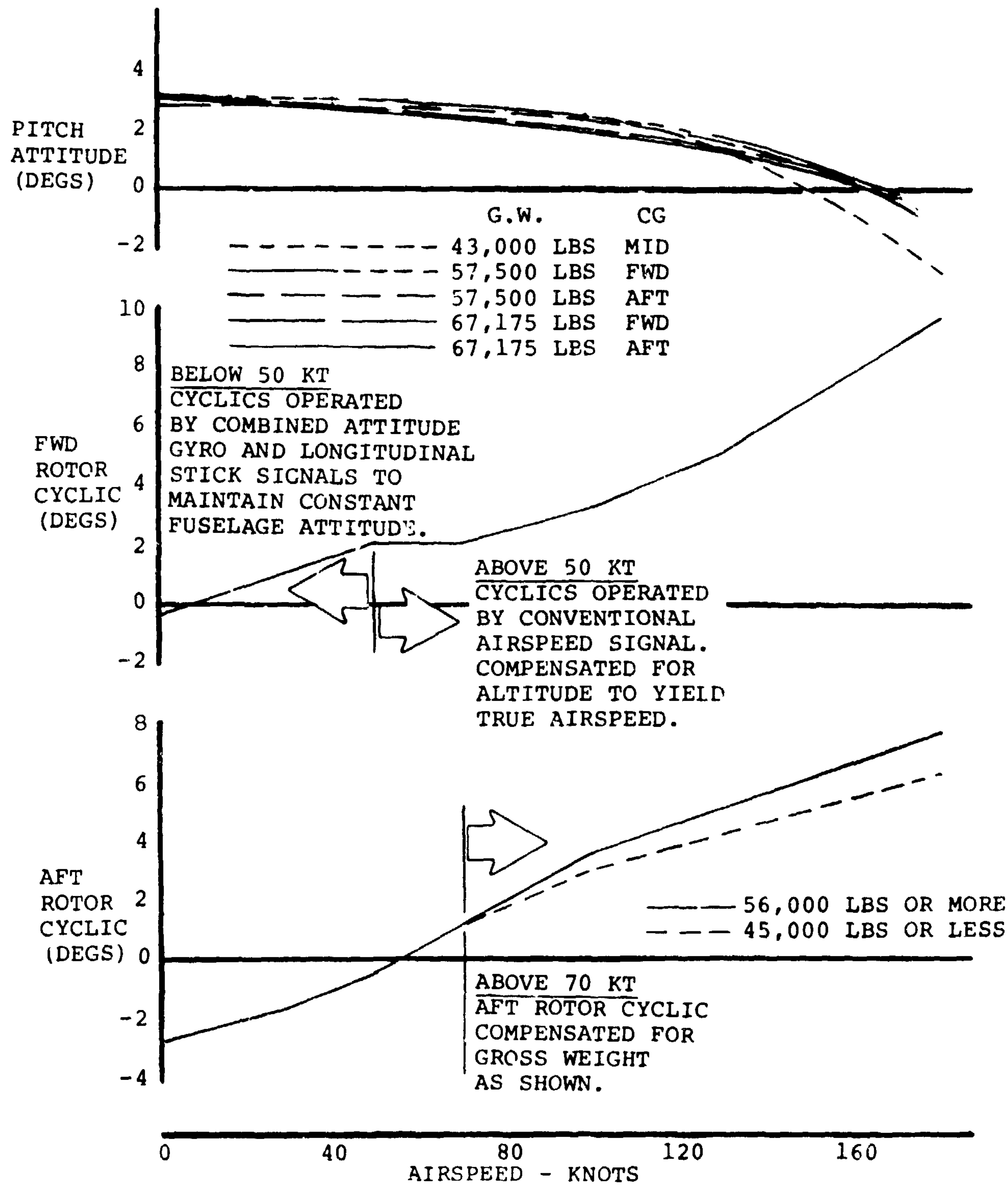


FIGURE 2.19 . LEVEL FLIGHT PITCH ATTITUDE AND CYCLIC TRIM SCHEDULES - TANDEM BASELINE CONFIGURATION.

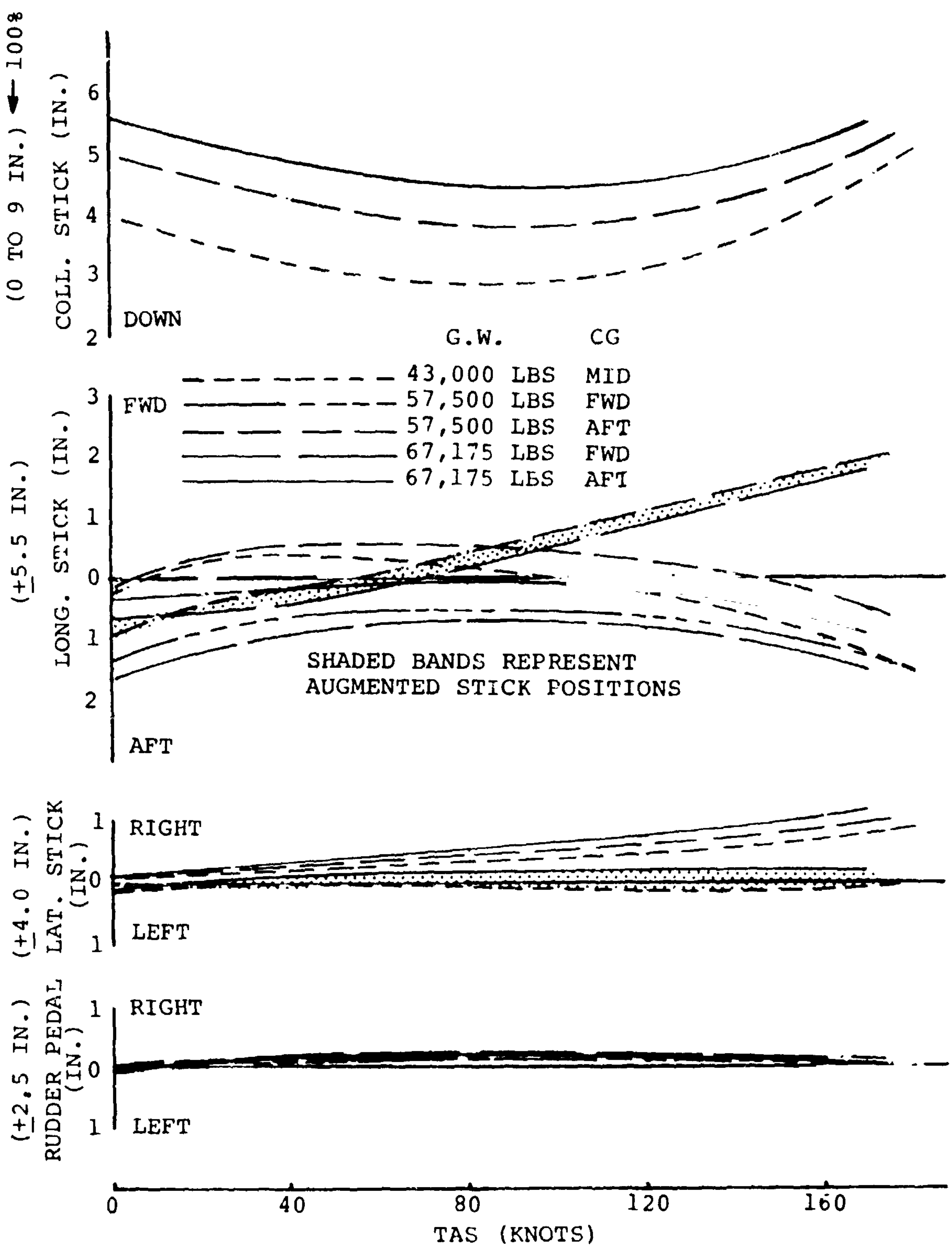


FIGURE 2.20 . CONTROL POSITIONS IN TRIMMED LEVEL FLIGHT - BASELINE TANDEM HELICOPTER.

CONTROL KINEMATICS AND CUMULATIVE LIMITS

	COCKPIT	FORWARD	AFT
COLLECTIVE	0" to 9"	-1.0° to 19.7°	-2.0° to 16.0°
DCP	± 5-1/2"	$\pm 5.0^\circ$	$\pm 3.5^\circ$
CUM COLLECTIVE		-6.0° to 23.3°	-5.5° to 18.3°
LATERAL	± 4"	$\pm 8.0^\circ$	$\pm 7.2^\circ$
PED	± 2-1/2"	$\pm 12.0^\circ$	$\pm 11.0^\circ$
CUM LATERAL		$\pm 16.1^\circ$	$\pm 14.7^\circ$
LONGITUDINAL CYCLIC		-3.2° to 12.0°	-5.4° to 10.0°
CUM CYCLIC		$\pm 20.1^\circ$	$\pm 17.8^\circ$
		(± 17.4)	(± 15.7)
TOTAL PITCH		-23.3° to 40.7°	-21.1° to 34.0°

TABLE 2.9. CONTROL KINEMATICS AND CUMULATIVE LIMIT DATA - BASELINE TANDEM HELICOPTER.

In order to meet the angle of attack stability criteria, 26.5 degrees, delta three have been introduced to the forward rotor. The effect of delta three on rudder pedal to trim is to essentially zero the pedal travel over the airspeed range.

The trim cyclic stick travels are modest compared with available control. For the SAS-ON cases a simple augmentation system is used on longitudinal and lateral stick. No SAS is applied to collective. The SAS system gains and limits are given in Table 2.10. With SAS-ON, the lateral stick excursions are essentially zero and a positive longitudinal stick gradient results.

The DASH system provides strong attitude and airspeed hold for unintentional disturbances and provides quickening in pitch for pilot command disturbances.

Control Power in Level Flight

The control powers available are shown in Figures 2.21, 2.22 and 2.23 for the range of gross weights and CG positions. Pitch control power is a minimum of 0.7 rads/sec square in hover SAS OFF and exceeds the minimum control powers defined in the guidelines at all airspeeds, gross weights and CG locations. With SAS ON, the pitch control power increases. The roll control power available is shown in Figure 2.22 and again is much higher than minimum guideline requirements. In this instance three aircraft weights are shown. CG location has almost no effect on roll control power.

<u>SAS SYSTEM</u>

DASH SYSTEM	STICK GAIN	4.5 Inch/Inch
	ATTITUDE GAIN	.28 In./Degrees
ACTUATOR LIMIT		
4" EXTEND	A/S GAIN	.092 In./Kt
1" RETREAT		
LATERAL STICK	-.20 STICK OFFSET	
	-.005 IN./KT (ABOVE 40 KT)	
PEDAL GAIN	1.5 IN./IN. (0 to 40 KTS)	
SIDESLIP GAIN	0.03 IN./DEG. (90 KTS)	
	0.05 IN./DEG. (160 KTS)	

ACTUATOR LIMIT ± 1.25 IN.

TABLE 2.10. SAS SYSTEM GAINS AND LIMITS - TANDEM HELICOPTER.

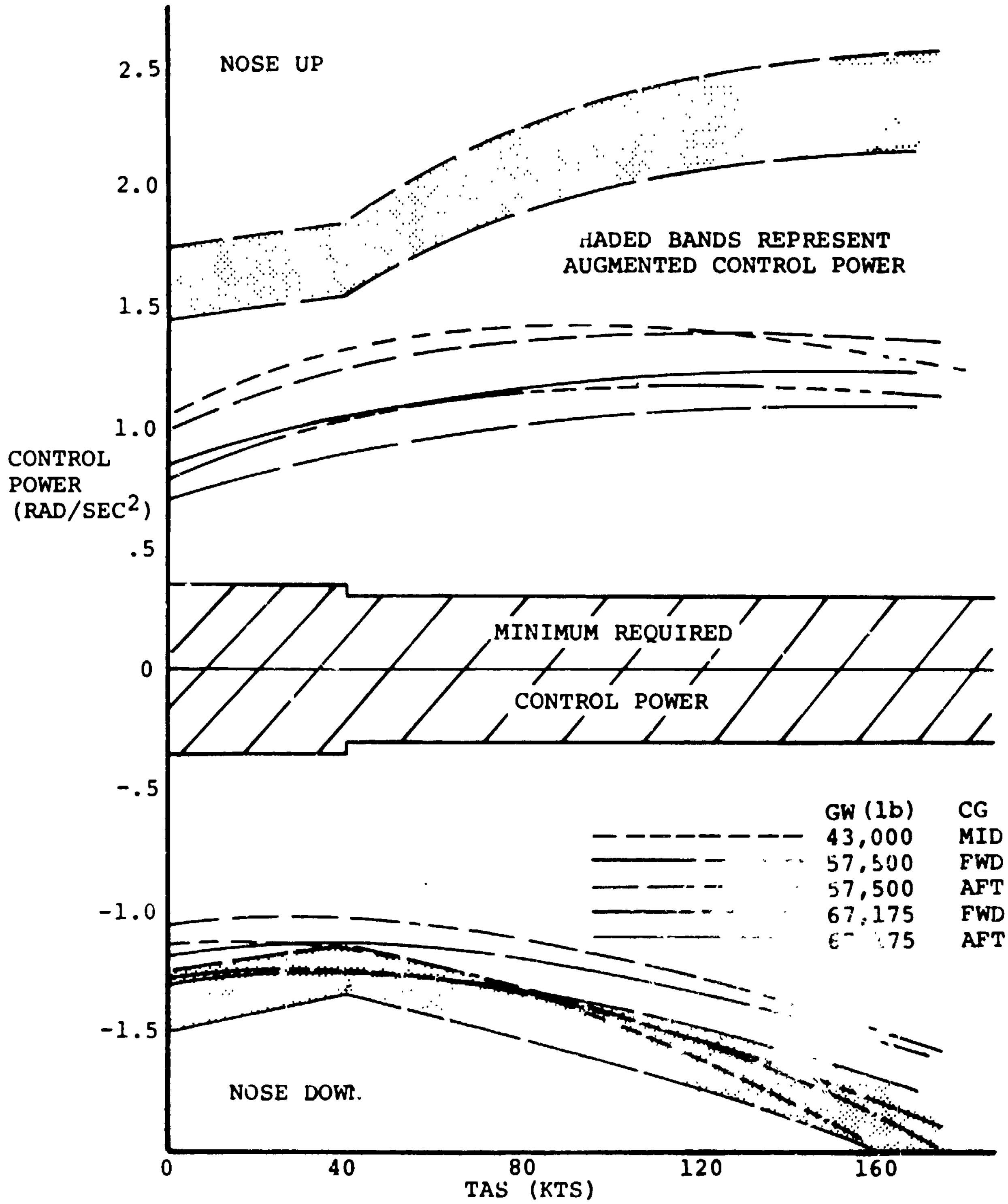


FIGURE 2.21. PITCH CONTROL POWER - LEVEL FLIGHT.

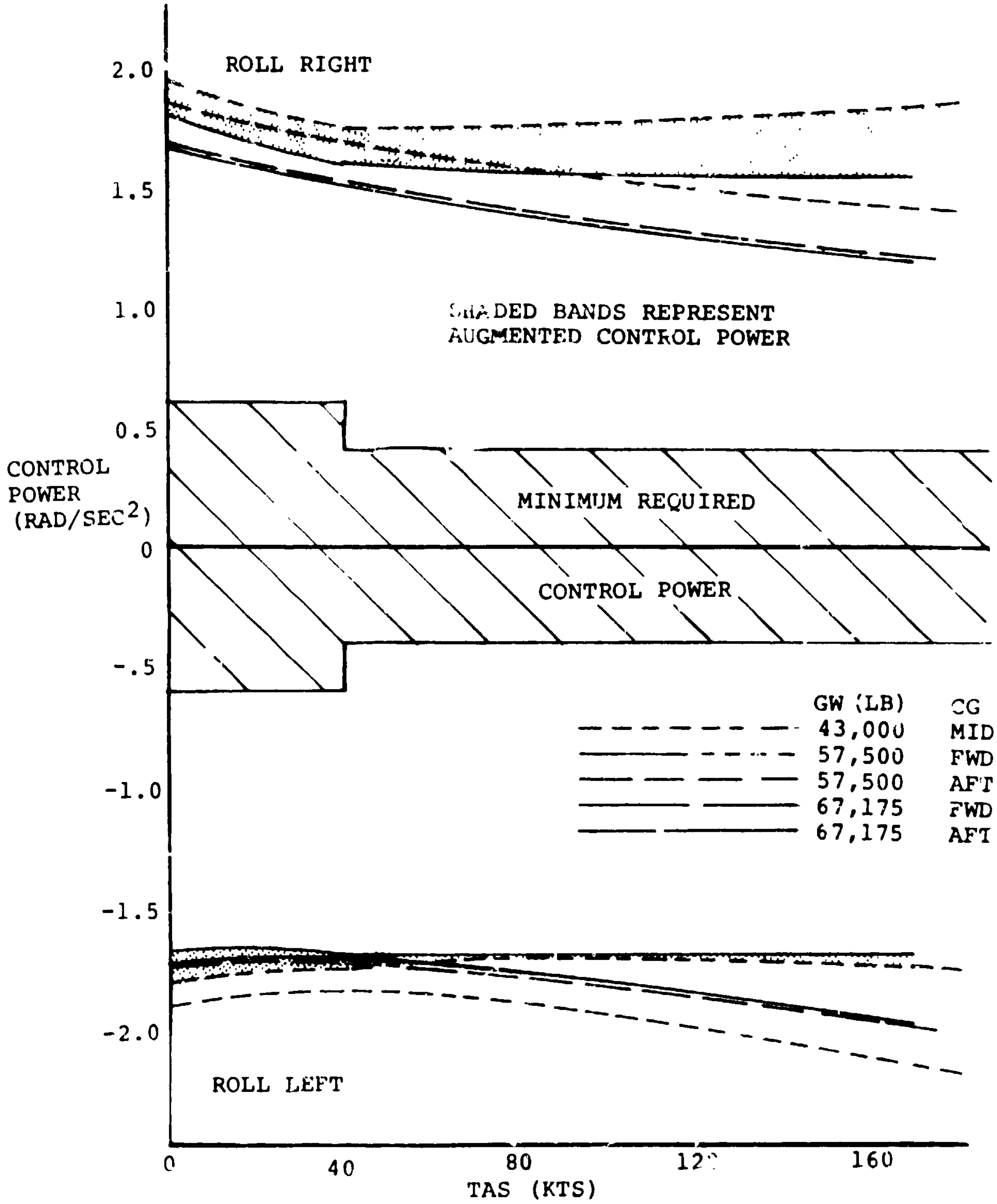


FIGURE 2.22 . ROLL CONTROL POWER - LEVEL IGHT.

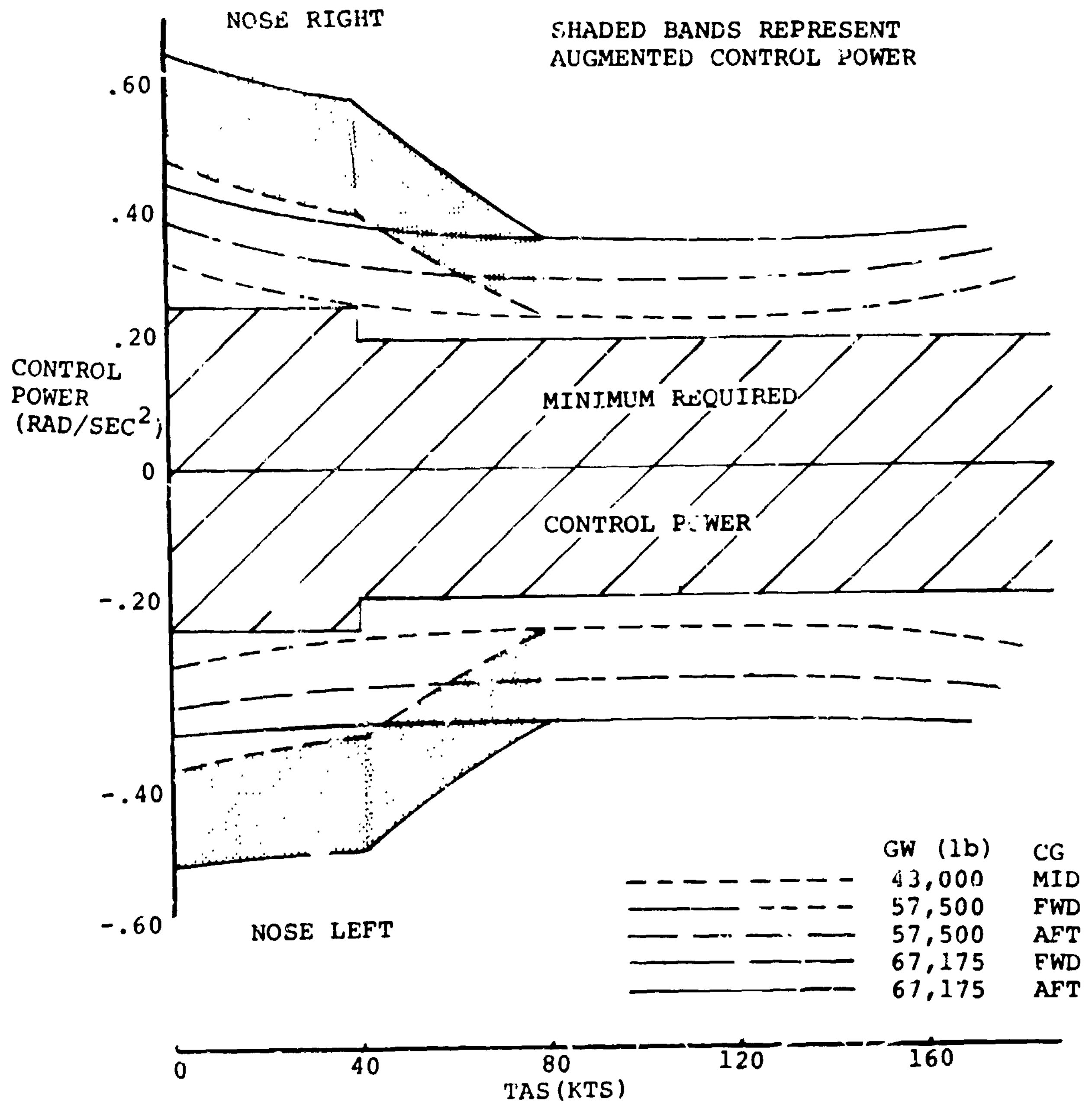


FIGURE 2.23 . YAW CONTROL POWER - LEVEL FLIGHT.

Yaw control power is shown in Figure 2.23 and again exceeds the guideline requirements, however, at the light weight the yaw control for the unaugmented aircraft is marginal at 40 knots. The yaw SAS provides quickening with a gain of 1.5 inches per inch out to 40 knots limited by ±1.25 inches actuator stroke. This quickening is washed out from 40 to 80 knots. With SAS ON the yaw control improves as shown in Figure 2.23.

Control Powers in Sideslip

For a tandem rotor helicopter, the collective and longitudinal stick positions in sideslip are not significantly different from trimmed level flight data. Hence, longitudinal control power available in sideslip is substantially unchanged from the level flight values in Figure 2.21.

Yaw control margins in sideslip, both basic and augmented, are substantial, but roll control has low margins and may be critical. The lowest roll control margins occur at 57,500 pounds with lateral CG offset. Since roll control sensitivity is also lowest for this gross weight, roll and cumulative roll/yaw control/power margins are checked for this weight with lateral CG offset, for both basic and augmented control systems (Figure 2.24).

Roll control augmentation consists solely of the speed-scheduled stick offset, while yaw control augmentation consists of quickening at hover, and sideslip stability at 80 knots and V maximum.

C-2

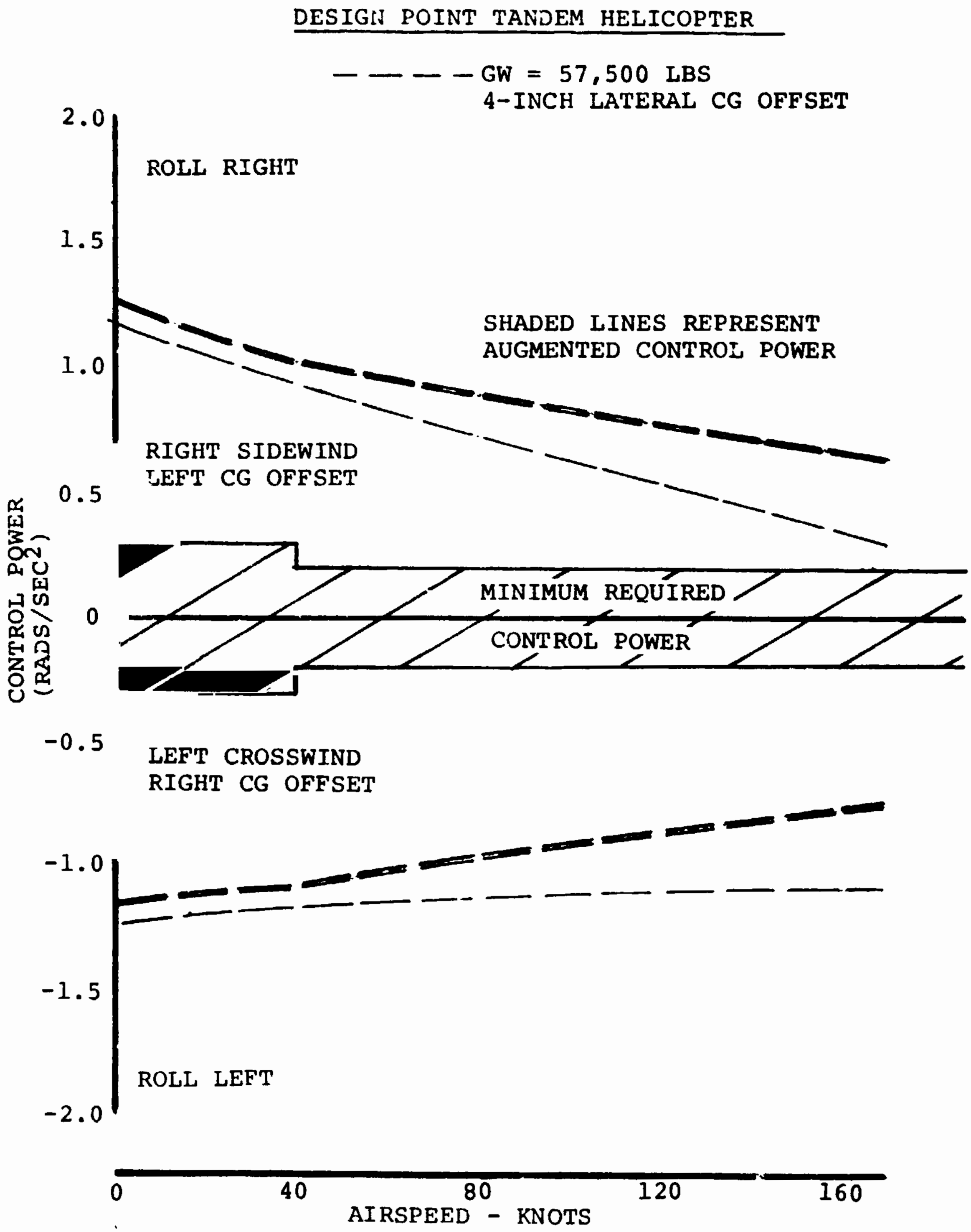


FIGURE 2.24. ROLL CONTROL POWER WITH 25 KNOT SIDEWIND.

In computing the roll and yaw control power associated with the available control margins, it is assumed that control sensitivity does not vary significantly with sideslip, and hence the level flight control derivatives are used. Although yaw control margins are not substantially reduced with respect to level flight values, the low control power available in yaw, particularly at low gross weight, as shown in Figure 2.23, indicates that yaw control power should be checked in sideslip at light weight.

The resultant yaw control power data, Figure 2.25, shows adequate margins in yaw. The inflection points in the data are due to cumulative lateral cyclic limits being encountered in the control system.

Response to Control Inputs

Typical time histories of responses to pitch, roll and yaw control input are shown in Figure 2.26 for design gross weight, aft CG at hover. The one second requirement for both basic and augmented aircraft is met.

The attitudes attained in one second for all other gross weights and airspeeds are summarized in Figures 2.27, 2.28 and 2.29. The requirements are met in all cases. Only augmented aircraft data are shown here, the unaugmented data being much higher. The unaugmented aircraft is more lively than the augmented aircraft due to the absence of artificial damping.

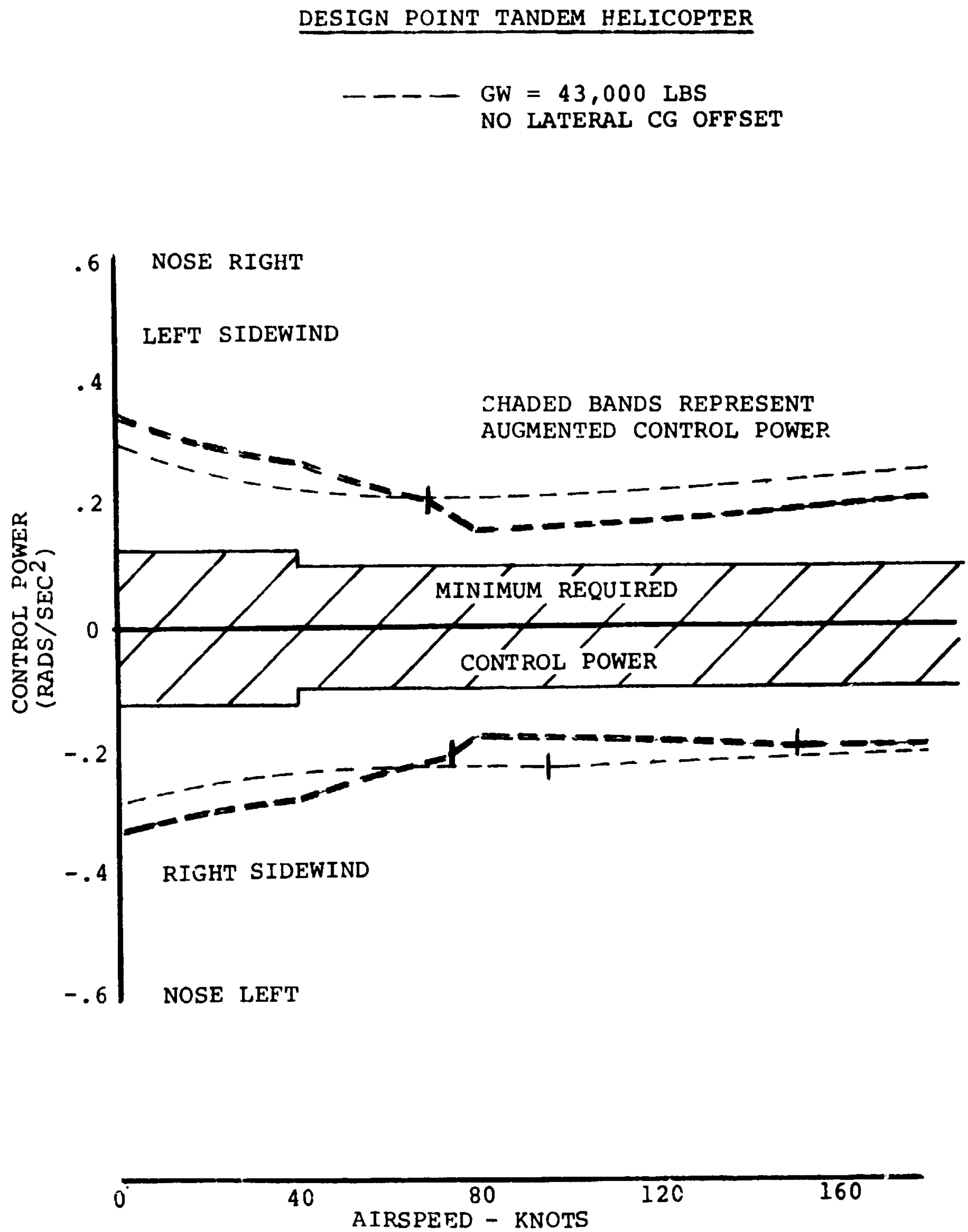


FIGURE 2.25. YAW CONTROL POWER WITH 25 KT SIDEWIND

DESIGN POINT TANDEM HELICOPTER

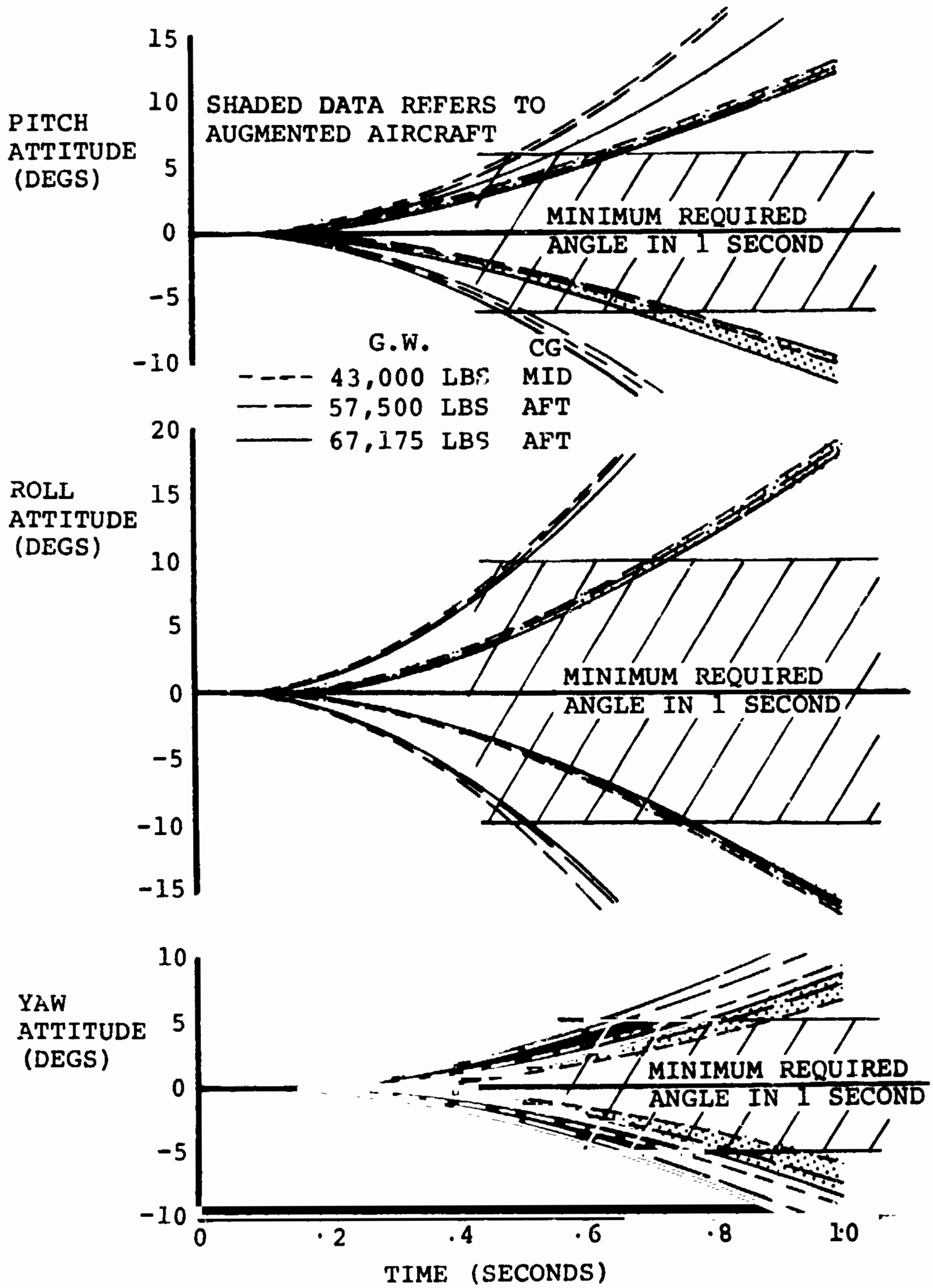


FIGURE 2.26 . RESPONSE TO FULL CONTROL INPUT AT HOVER.

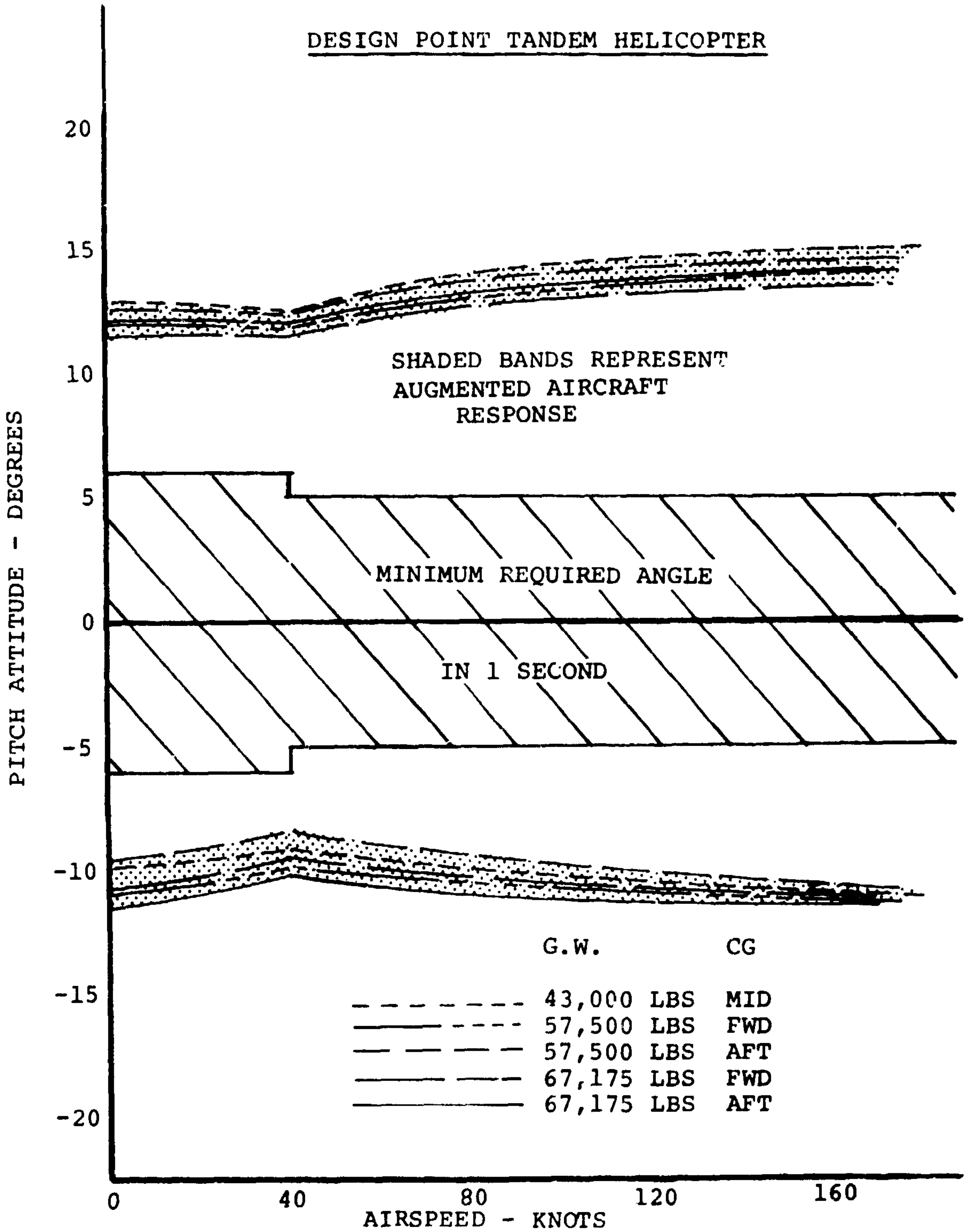


FIGURE 2.27. PITCH ATTITUDE 1 SECOND AFTER FULL LONGITUDINAL STICK INPUT.

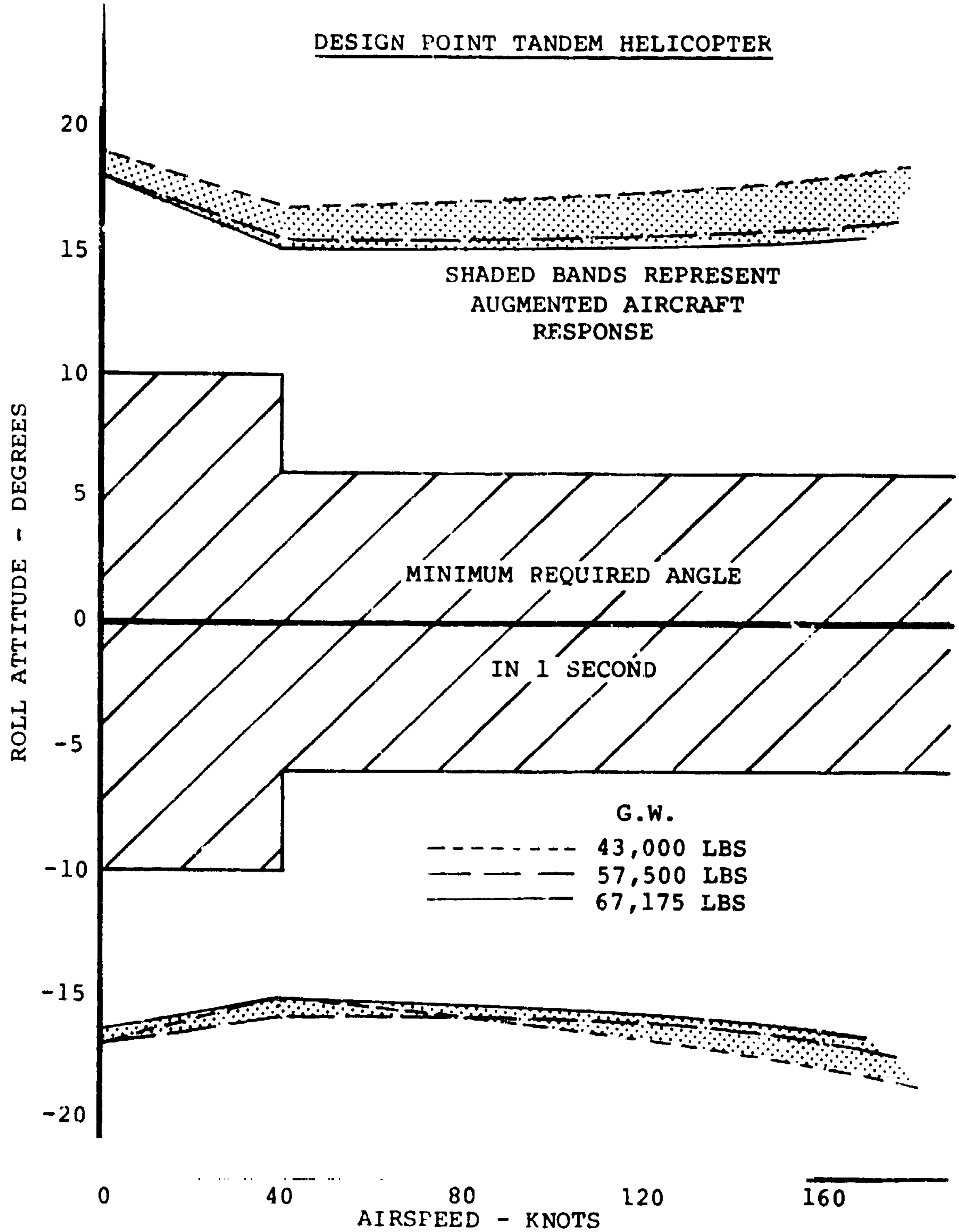


FIGURE 2.28. ROLL RESPONSE 1 SECOND AFTER FULL LATERAL STICK INPUT.

DESIGN POINT TANDEM HELICOPTER

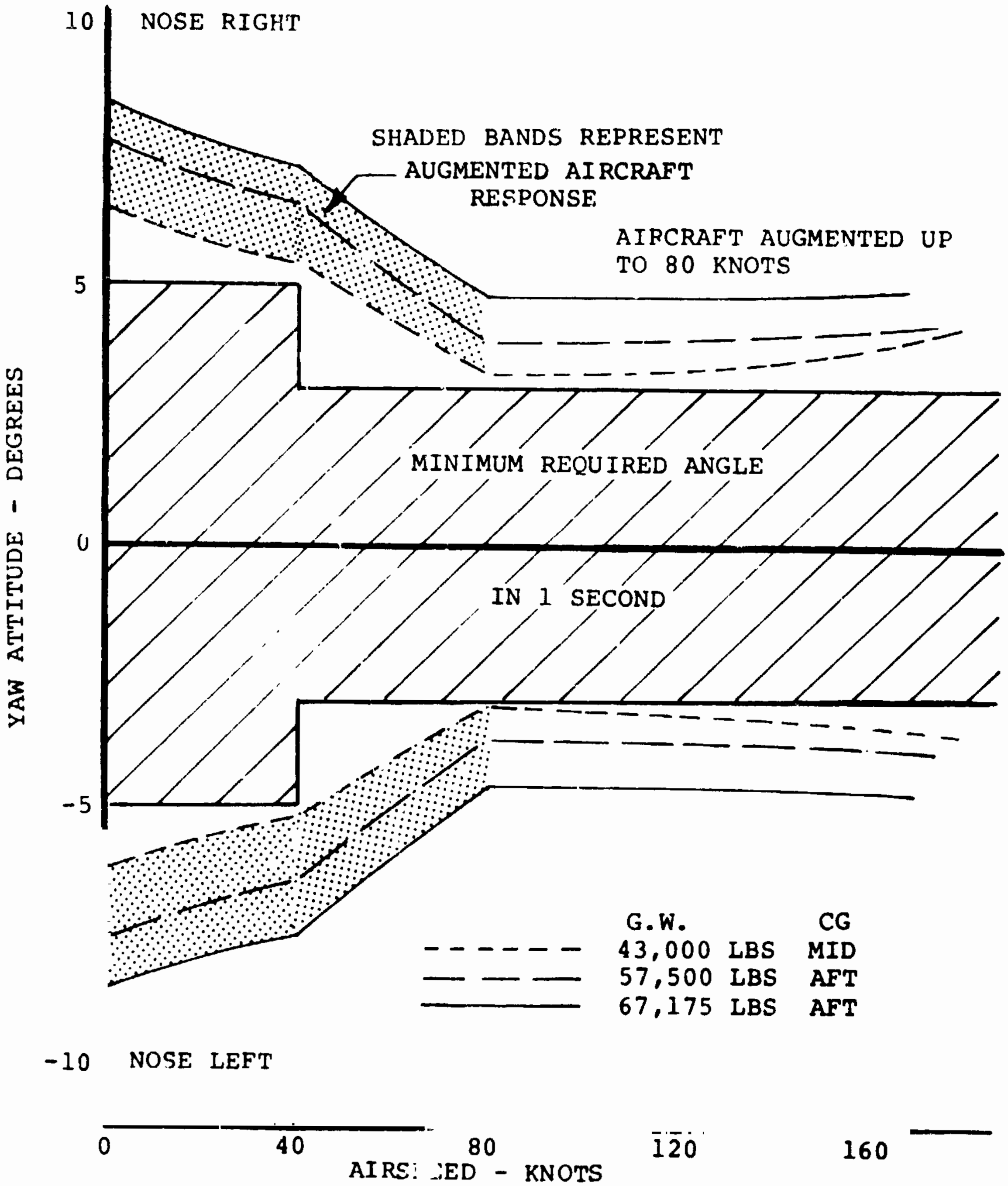


FIGURE 2.29. YAW RESPONSE 1 SECOND AFTER FULL PEDAL INPUT.

Rotor and Control System LAGS

Based on Boeing Vertol experience with the 347 aircraft, control system lag (pilot to swashplate) is estimated to be about 0.1 second for conventional control system. This is conservative for a fly-by-wire system, where only actuator lags are significant. The rotor lag is 0.055 seconds. These system characteristics meet the requirements defined in the study guidelines.

Aircraft Stability

The stability characteristics of the basic (unaugmented) vehicle are presented in Figure 2.30. These characteristics can be augmented to any desired level, to provide optimum flying qualities.

The levels of stability shown provide mildly stable characteristics in the basic vehicle, which can therefore be flown safely (pilot rating of 5 or better) after complete failure of the augmentation system. These mildly positive stability levels provide a vehicle which

(a) is readily augmented to any desired level,

(b) has no inherent instabilities to complicate AFCS design,

(c) has no inherent strong stability to be overcome by the control system in maneuvers, and

(d) has inherent minimum gust response (attitude wise).

The longitudinal static stability exceeds the $M_{\alpha}>0$ criterion

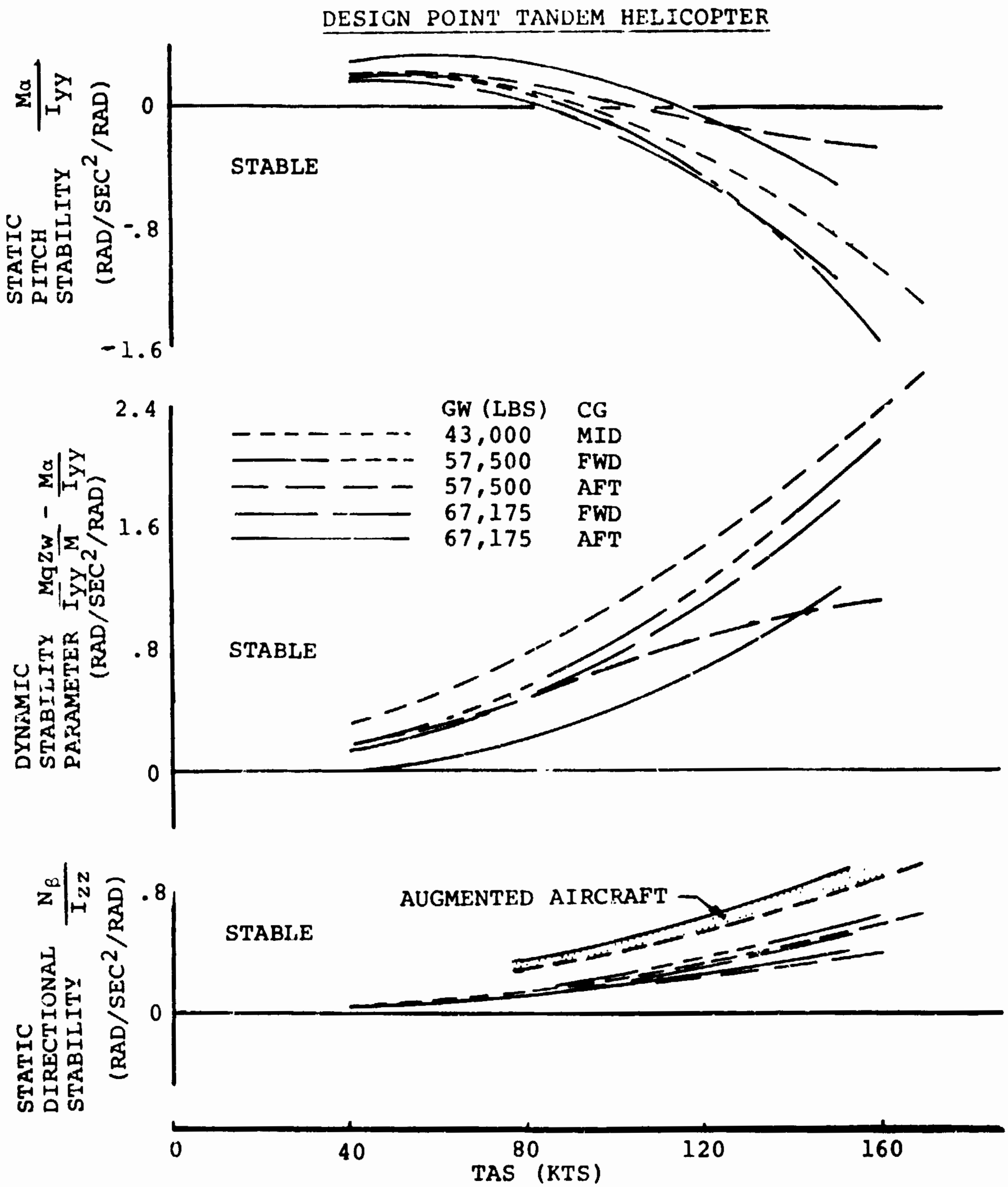


FIGURE 2.30. LONGITUDINAL AND DIRECTIONAL STABILITY.

in cruise for all airspeeds above 120 knots at aft CG.

This is achieved by 26.5 degree stabilizing delta three in the forward rotor. The mild instability indicated at lower airspeeds will present no difficulty, since the dynamic stability criterion ($M_q Z_w - M_\alpha > 0$) is met at all airspeeds. This parameter represents the stability of the aircraft in a maneuver, and the criterion corresponds to positive maneuver margin on a fixed wing aircraft.

The directional static stability exceeds the $N_\beta > 0$ criterion at all airspeeds and gross weights.

The augmented values of M_α range from 2.60 to 6.30, which is well off the graph. Similarly, for dynamic stability ($M_q Z_w - M_\alpha$), the augmented ranges is from 3.50 to 10.50.

Augmented values of N_β range from 0.30 to 1.0 as shown.

The lateral stick and directional pedal position gradients (Figures 2.31 and 2.32) are positive in sideslip for side-winds up to 25 knots and beyond. The gradients shown are for symmetrical lateral CG position. The effects of lateral CG offset are indicated by the open symbols (basic aircraft) and dark symbols (augmented aircraft). Roll attitudes are acceptable.

Lateral stick margins for the unaugmented aircraft with lateral CG offset can be as low as 0.7 inches (9%) in a high speed sideslip, but augmentation increases this margin to 1.6 inches (.9%). With no lateral CG offset, the

DESIGN POINT TANDEM HELICOPTER

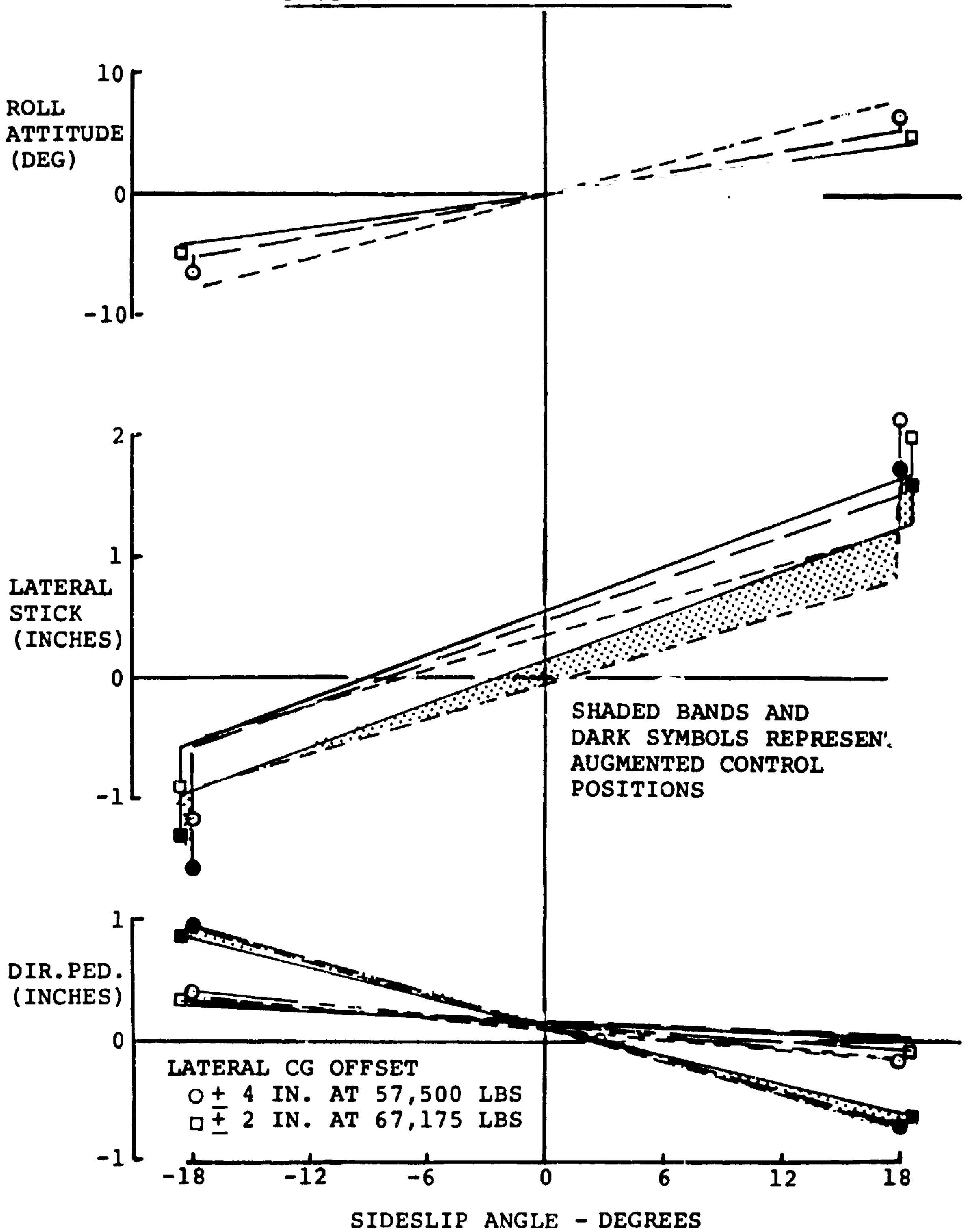


FIGURE 2.31. LATERAL-DIRECTIONAL STABILITY AT 80 KNOTS.

DESIGN POINT TANDEM HELICOPTER

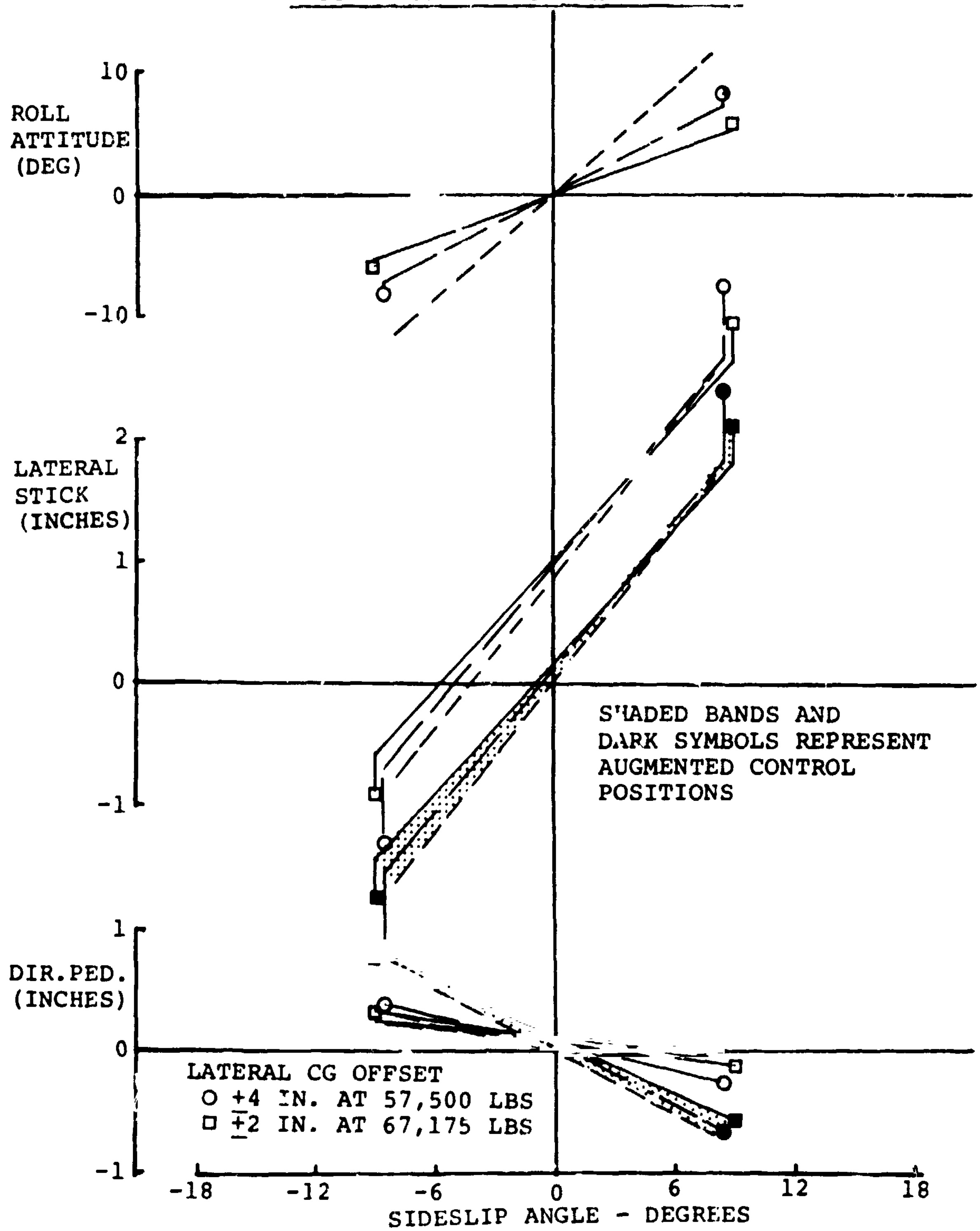


FIGURE 2.32. LATERAL/DIRECTIONAL STABILITY AT CRUISE SPEED.

augmented lateral stick margin is 2.2 inches (28%) or better. The pedal gradient of the basic aircraft in sideslip is low, but artificial stability improves this to an acceptable value.

Dynamic Stability Criterion

Compliance with dynamic stability criteria is shown in Figure 2.33. The range of desirable damping versus frequency defined in the guidelines pertain to hover and low speed. An additional requirement from AGARD 577 is shown shaded for Level 1 at high speed. The stability contours shown cover the speed range from hover to V maximum, progressing in the direction of the arrows. The intermediate points pertain to 80 knots. The unaugmented aircraft meets the Level 2 requirement at all airspeeds. The augmented aircraft falls within the Level 1 window at low speeds, and meets the Level 1 requirement of AGARD 577 at high speed.

Descending Flare Requirements

The tabulated data pertain to the unaugmented aircraft. The control positions, therefore, represent rotor control used in the specified maneuver. At all airspeeds, the control inputs required are well within rotor capabilities, as shown in the following table.

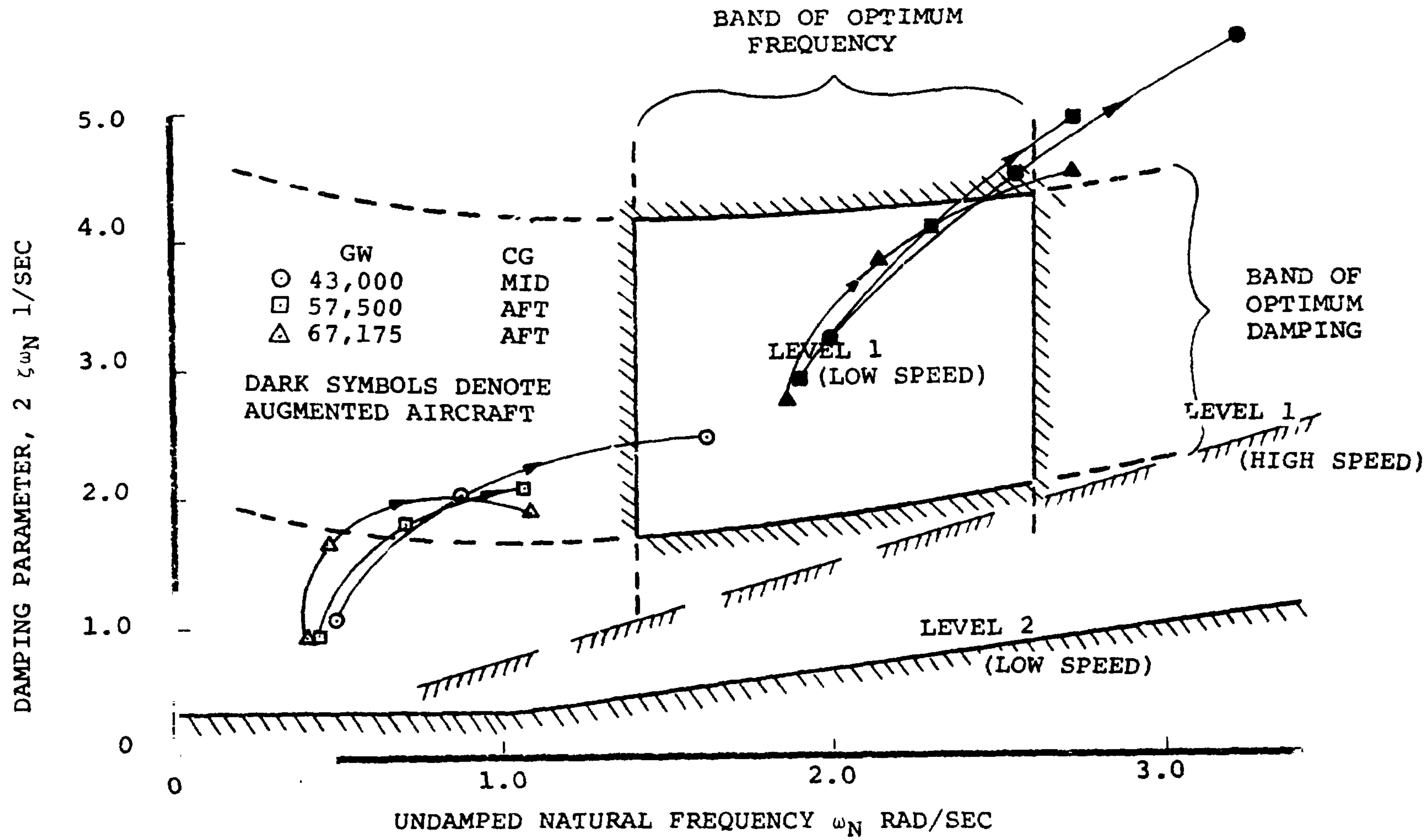


FIGURE 2.33. LONGITUDINAL DYNAMIC STABILITY.

GW = 67,175 LBS 2000 Ft/Min R/S 0.15g Decceleration	MAXIMUM RANGE AVAILABLE	ZERO SIDEWIND		25 KNOT SIDEWIND	
		MIN.	MAX.	MIN.	MAX.
COLLECTIVE	0 to 9.0	0.76	4.22	0.72	3.86
DCP	±5.50	-0.87	0.72	0.48	1.05
Lateral Stick	±4.00	0.24	0.27	0.72	0.85
Rudder Pedal	±2.50	0.19	0.27	0.11	0.20
POWER REQUIRED	0 to 9500	-1570	4960	-1500	4330

The negative horsepower required at 80 and 100 knots indicate rotor overspeed conditions of 6% and 9% excess RPM respectively for zero horsepower.

Gust Sensitivity

The tandem helicopter aircraft is naturally insensitive to gusts. Computations based on a 10 feet per second gust of varying length were performed. The worst cases are as shown in Figure 2.34. Variations in gross weight do not significantly change the gust sensitivity. The aircraft meets the specified criteria at all conditions at 10,000 feet altitude and at all except the forward CG case at high speed at 5,000 feet.

It is doubtful whether action should be taken to make this point fall within the criteria line, however, collective feedback could be used to correct this small deficiency for little more than the weight of the sensors and signal

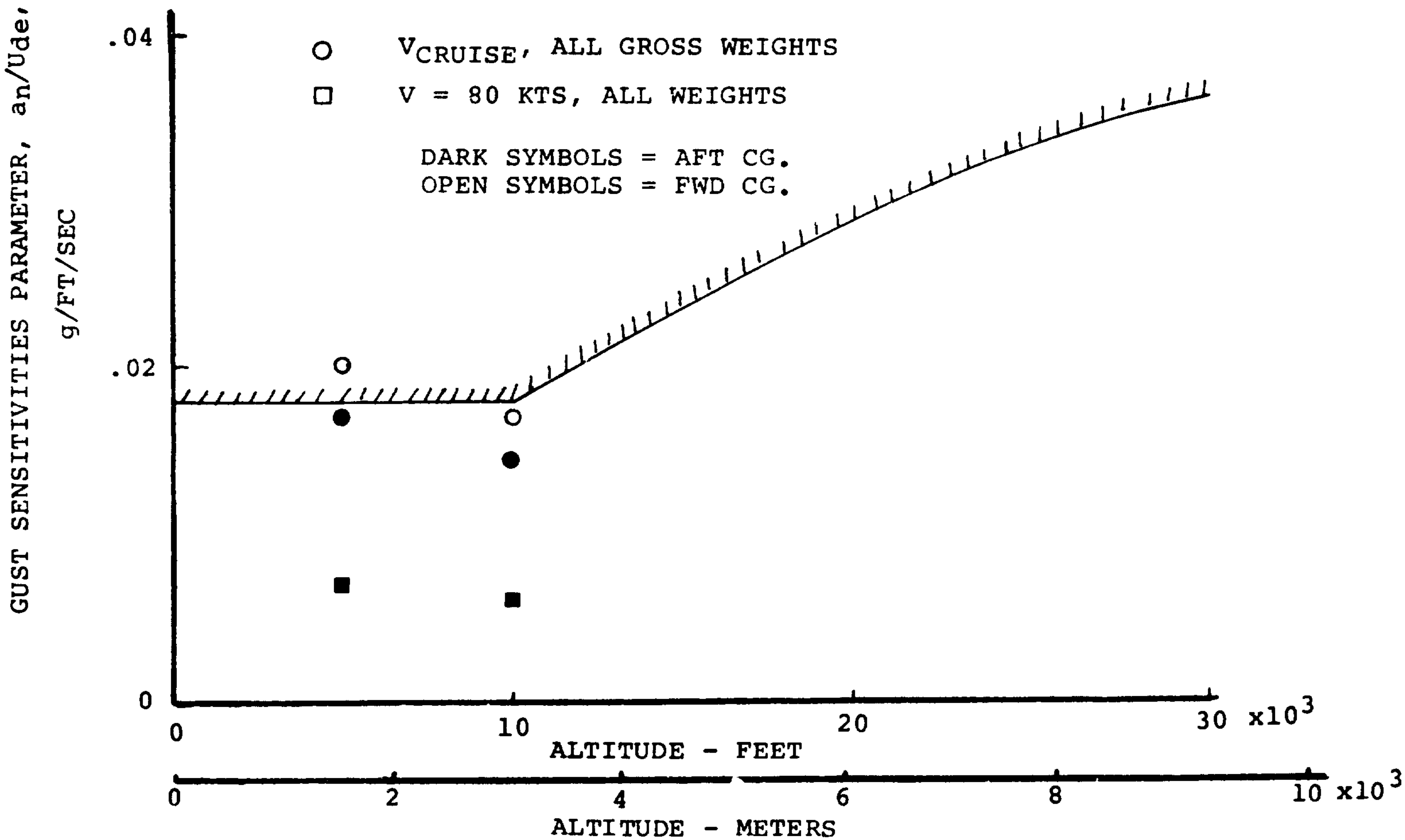


FIGURE 2.34. PASSENGER RIDE DESIGN CRITERION (HELICOPTER).

conditioning electronics (i.e., less than 50 pounds).

Hover Gust Control

The control ranges available after trim in hover at both zero wind and 25 knot sidewind are given in Table 2.11 for three gross weights. The percent of the available control required to counteract a 15 feet per second gust upset are shown for gust upsets in any direction. In no case is more than 20% of the remaining control required to counteract the gust.

2.1.5 Tandem Helicopter - Design Point Noise

The design criteria for external noise is that the 500-foot sideline noise level in hover at 100 feet altitude is to be between 90 and 100 PNdB. The design point tandem helicopter is relatively quiet with a 500-foot sideline perceived noise level of 92.3 PNdB.

The noise criterion was stated in terms of perceived noise level (PNdB) to provide some means of comparison with aircraft designed to similar criterion in other studies. It was recognized, however, that the validity of PNdB as a community acceptance indicator may not be valid since the noise signature (the distribution of absolute sound pressure level as a function of frequency) is markedly different for large helicopters than for jet engined aircraft.

For this reason the absolute sound pressure levels as a function of octave band frequency are also provided in Figure 2.35 for the noise producing components as well as the overall aircraft noise. The overall aircraft SPL is set for most

HOVER - DESIGN POINT TANDEM HELICOPTER

GROSS WEIGHT	GUST AXIS	ZERO SIDEWIND						25 KT SIDEWIND					
		PITCH		ROLL		YAW		PITCH		ROLL		YAW	
		IN.	%	IN.	%	IN.	%	IN.	%	IN.	%	IN.	%
43,000	LONG.	5.26	3.6	----	----	----	----	5.26	3.6	----	----	----	----
	VERT.	5.26	4.4	----	----	----	----	5.26	4.4	----	----	----	----
	LAT.	----	---	3.94	4.6	2.45	7.5	----	---	3.54	5.1	2.28	8.1
57,500	LONG.	4.17	4.8	----	----	----	----	4.17	4.8	----	----	----	----
	VERT.	4.17	7.9	----	----	----	----	4.17	7.9	----	----	----	----
	LAT.	----	---	3.92	10.7	2.36	12.4	----	----	2.71	15.5	2.16	13.1
67,175	LONG.	3.84	5.5	----	----	----	----	3.84	5.5	----	----	----	----
	VERT.	3.84	8.5	----	----	----	----	3.84	8.5	----	----	----	----
	LAT.	----	---	3.92	14.2	2.36	11.4	----	---	2.86	19.5	2.17	12.1
		1	2	1	2	1	2	1	2	1	2	1	2

1 AVAILABLE CONTROL TO NEAREST STOP (IN.).

2 % OF AVAILABLE CONTROL NEEDED TO COUNTERACT GUST UPSET.

TABLE 2.11. PERCENTAGE OF AVAILABLE CONTROL NEEDED TO COUNTERACT GUST UPSET.

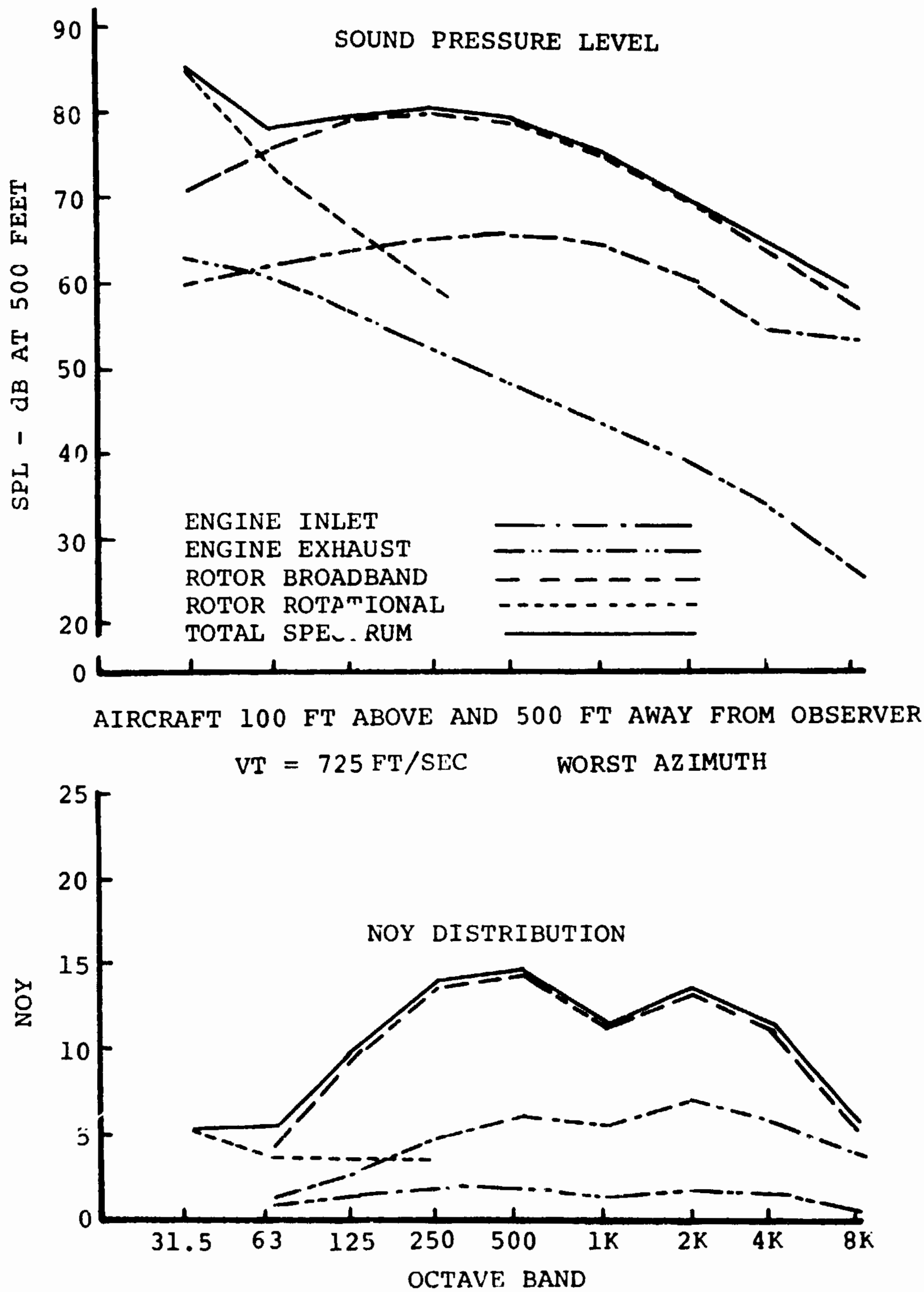


FIGURE 2.35 . HOVER NOISE SPECTRUM AND NOY DISTRIBUTION.
DESIGN POINT AIRCRAFT - TANDEM HELICOPTER
PNdB = 92.3

of the frequency range by the rotor to broadband noise, though at the very low frequencies the rotor rotational noise becomes dominant. Thus the PNdB value is set primarily by rotor noise.

Unless special noise suppression measures are adopted, the engine inlet noise becomes dominant in the 4 KHz to 8 KHz octave bands. The engine inlet is, therefore, assumed to be treated for noise reduction by installing acoustic absorption linings. The inlet absorption lining has been tuned to two bands with center of frequencies 4 KHz and 8 KHz. This matched the engine signature to that of the rotor such that the rotor signature sets the PNL value. The octave band inlet noise attenuation resulting from this treatment is shown in Figure 2.36.

A perceived noise level "footprint" for a typical takeoff is shown in Figure 2.37 for line of constant PNdB. This plot indicates that the worst noise levels occur along the flight-path of the aircraft with a perceived noise level of 100 PNdB out to 1200 feet from the point of origin. The takeoff altitude profile and the perceived noise levels at various distances along the flight path are shown in Figure 2.38. The takeoff profile assumes a vertical lift-off and acceleration to climb speed with a climb to altitude at approximately 2500 feet per minute.

The perceived noise level time histories show that at 200 feet a maximum of 112 PNdB is observed 7.5 seconds after

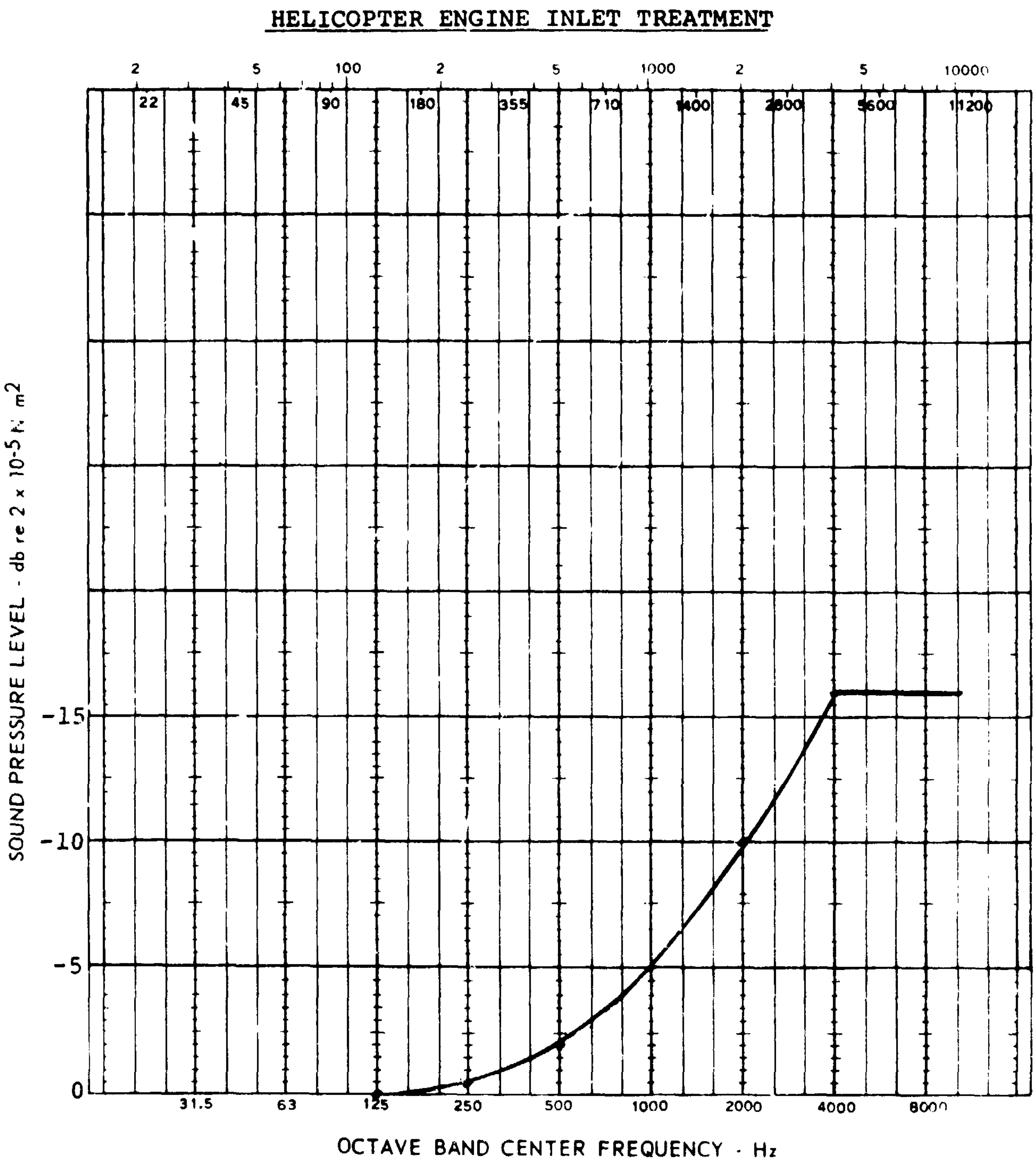


FIGURE 2.36. HELICOPTER ENGINE INLET NOISE SUPPRESSION.

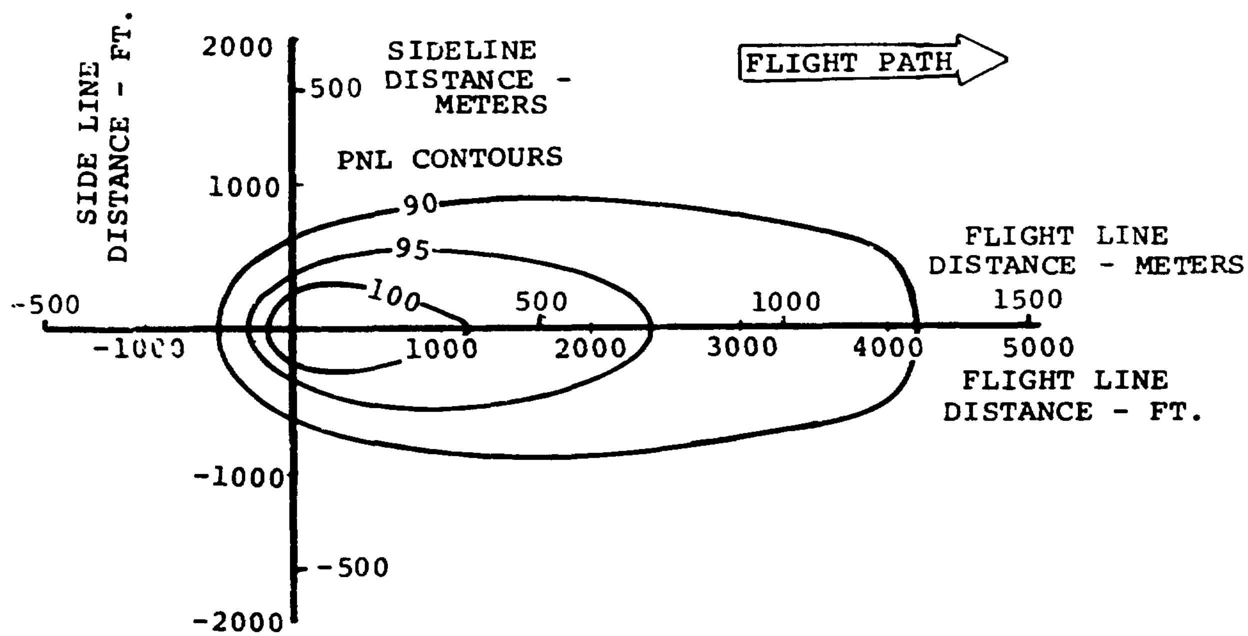


FIGURE 2.37 . BASELINE HELICOPTER DESIGN POINT - STANDARD TAKEOFF. PNL CONTOURS.

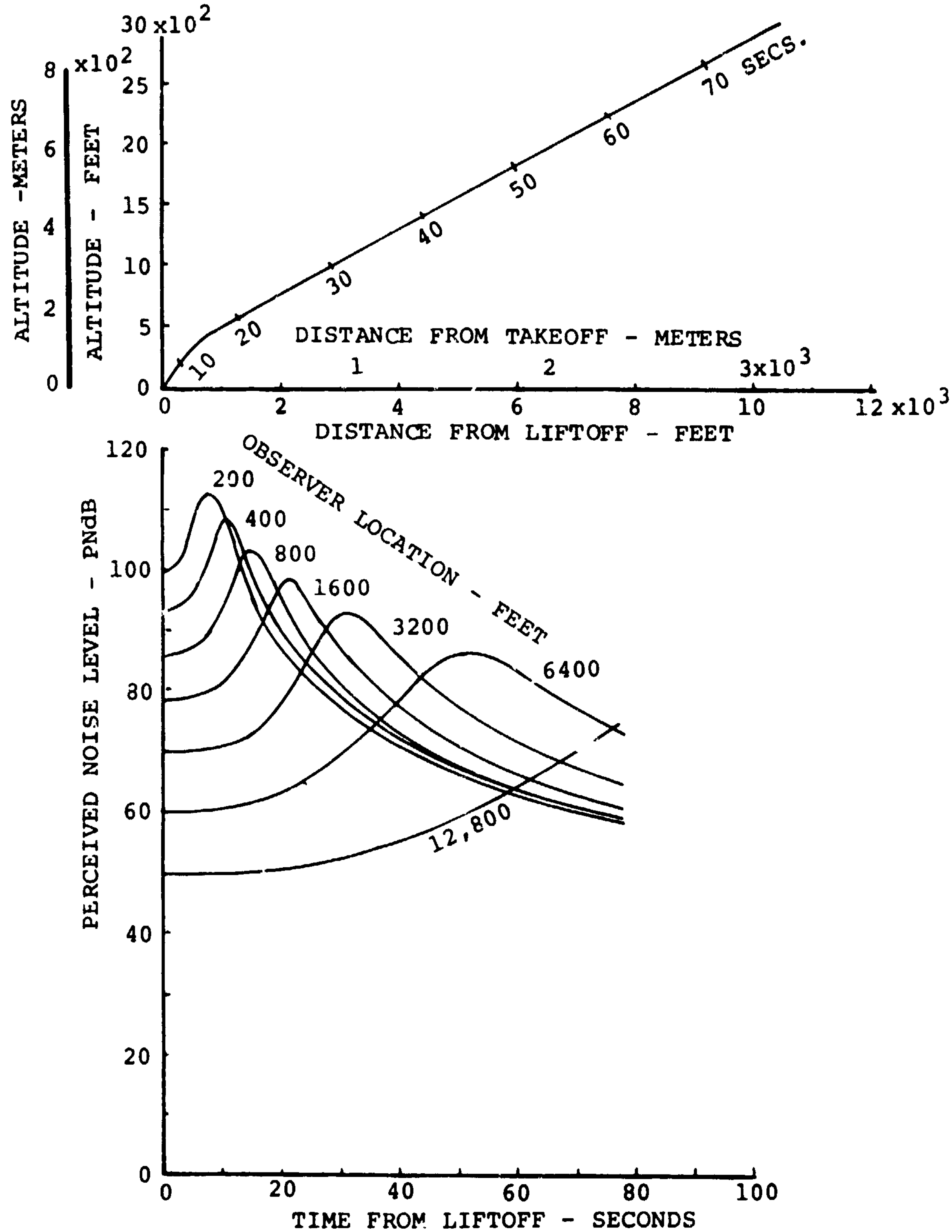


FIGURE 2.38. BASELINE HELICOPTER DESIGN POINT - STANDARD TAKEOFF. PERCEIVED NOISE.

takeoff. Another factor to be considered in assessing community acceptance is the duration of high noise levels. In this case for example the perceived noise exceeds 110 PNdB for only four seconds at 200 feet. (At each location along the flight path the noise level increases until the aircraft passes overhead and then decreases again).

The PNL contours for a typical landing profile are shown in Figure 2.39. The contours are elongated by comparison with the takeoff case. This is a result of the low rate of sink used in the landing profile.

This rate of sink is the maximum permitted by the guidelines shown in Table 2.5.

To maintain these sink rates, high power levels are required which, in combination with full tipspeed, results in the contours shown.

These contours could be reduced through use of noise abatement approach techniques available to the low disc loading V/STOL configurations. These techniques involve vertical flight at altitudes below 1000 feet with all transitions to or from forward flight accomplished above this altitude. The perceived noise levels along the flight path and the landing profile are shown in Figure 2.40.

2.1.6 Tandem Helicopter Design Point Costs

The initial or flyaway costs of the design point tandem helicopter have been computed using both \$90/pound and \$110/pound for the airframe cost. These prices are shown in Table 2.12. The initial cost is \$4.166 million at \$90/pound and

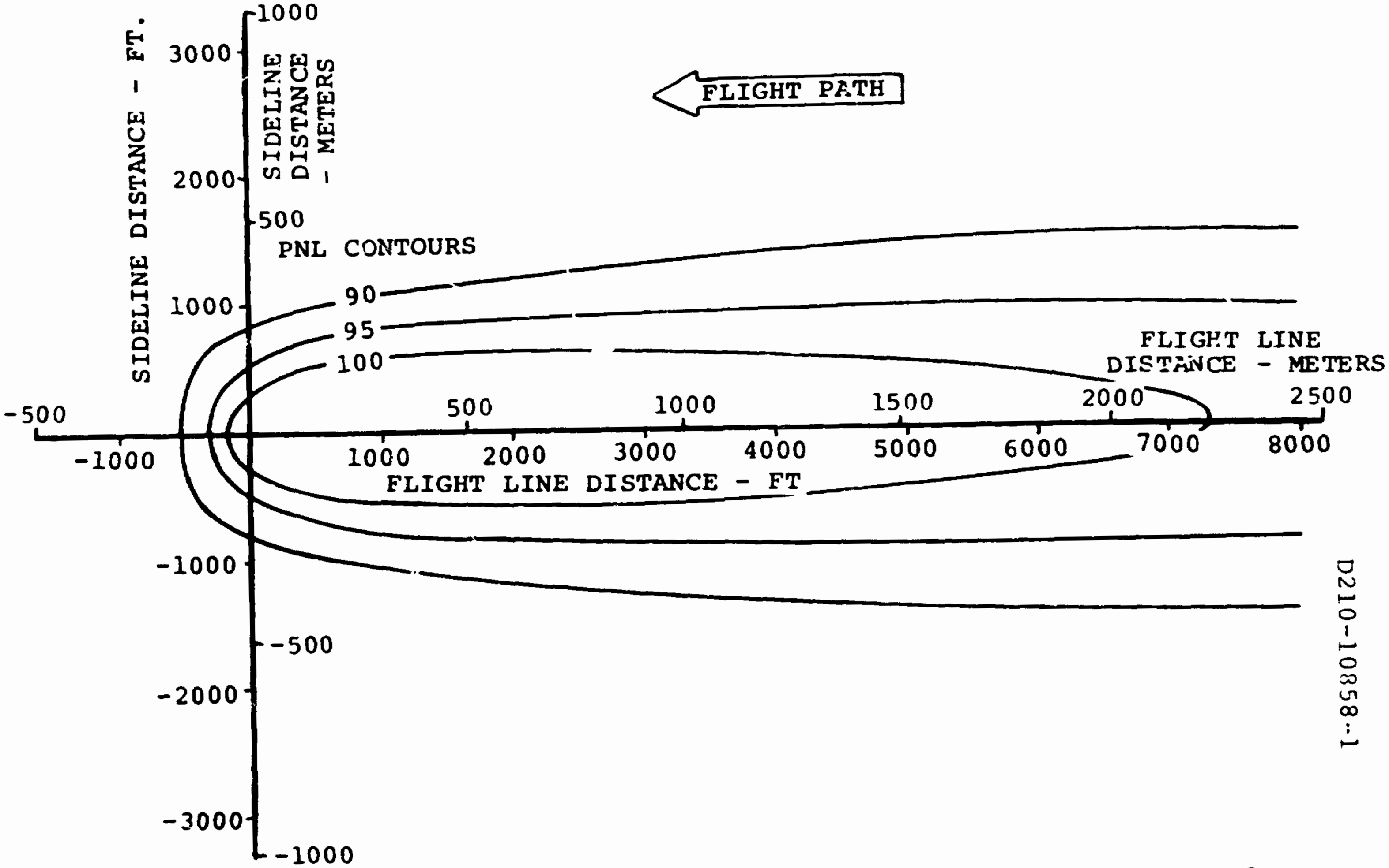


FIGURE 2.39 . BASELINE HELICOPTER STANDARD LANDING. PNL CONTOURS.

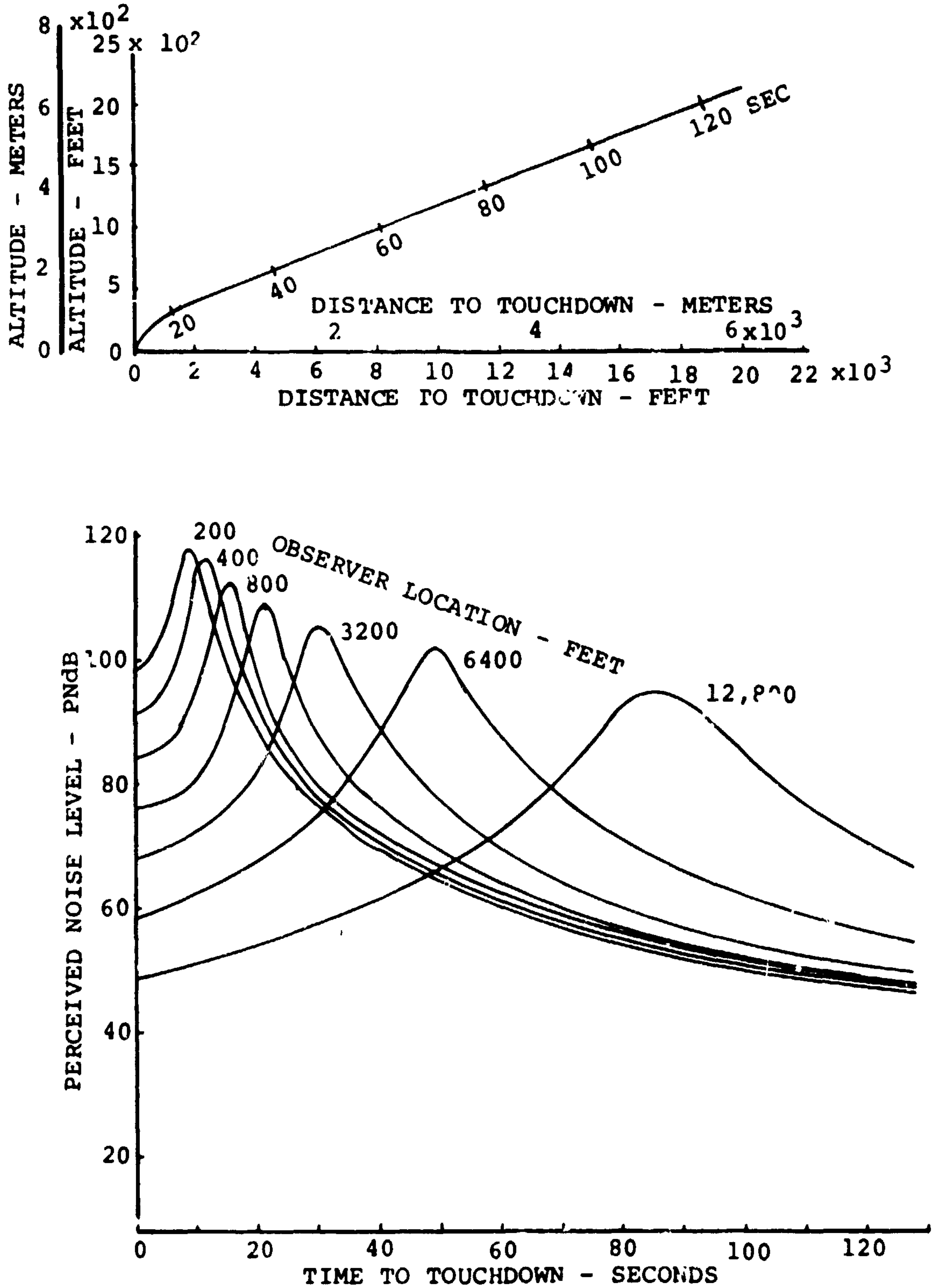


FIGURE 2.40 . BASELINE HELICOPTER DESIGN POINT - STANDARD LANDING. PERCEIVED NOISE.

TH-100(92.3)

Flyaway Costs

Airframe Cost	$90.00/Lb	$110.00/Lb
Airframe	$2,199,510	$2,688,290
Dynamic System	1,063,040	1,063,040
Engines	654,265	654,265
Avionics	250,000	250,000
Total	$4,166,815	$4,655,595

Direct Operating Costs
Dollars/Seat Mile
Block Distance = 230 St. Miles

Utilization (Hrs/Yr)	2500		3500	
Airframe Cost ($/Lb)	90	110	90	110
Flying Operations				
Flight Crew	.0081	.0081	.0081	.0081
Fuel and Oil	.0045	.0045	.0045	.0045
Hull Insurance	.0019	.0022	.0014	.0015
Total Flying Operations	.0145	.0148	.0140	.0141
Direct Maintenance				
Airframe - Labor	.0013	.0013	.0013	.0013
- Material	.0010	.0012	.0010	.0012
Engines - Labor	.0007	.0007	.0007	.0007
- Material	.0009	.0009	.0009	.0009
Dynamic System - Labor	.0011	.0011	.0011	.0011
- Material	.0017	.0017	.0017	.0017
Total Direct Maintenance	.0067	.0069	.0067	.0069
Maintenance Burden	.0047	.0047	.0047	.0047
Total Maintenance	.0114	.0116	.0114	.0116
Depreciation	.0094	.0105	.0067	.0075
Total Direct Costs	.0353	.0369	.0321	.0332

TABLE 2.12. DESIGN POINT TANDEM HELICOPTER - INITIAL AND DIRECT OPERATING COSTS.

$4.66 million at $110/pound. The airframe contribution is $2.199 million and $2.688 million at the respective rates; the rest of the initial cost being dynamic system, engines and avionics cost.

The direct operating costs are shown in Table 2.12 for two assumed aircraft utilizations of 2,500 hours per year and 3,500 hours per year.

At an airframe cost of $90/pound the direct operating cost is 3.53¢ per seat mile for 2,500 hours utilization. This cost breaks down to be 1.45¢ per seat mile for flying operations, 0.67¢ per seat mile maintenance and a depreciation of 0.94¢ per seat mile.

Increasing the airframe cost to $110/pound increases the DOC to 3.69¢ per seat mile. Most of the increase is increase in depreciation costs and the rest is insurance and maintenance material.

If the aircraft utilization is 3,500 hours per year the DOC decreases to 3.21¢ per seat mile and 3.32¢ per seat mile for airframe costs of $90/pound and $110/pound respectively.

The largest contribution to the direct operating cost is the decrease in depreciation costs per seat mile.

An extended range version of the design point tandem helicopter has also been considered with fuel tanks increased to give 400 nautical mile range. With the same takeoff gross weight the extended range version could carry 98 passengers over the design (200 NM) mission). The aircraft initial cost increases a little due to the additional tankage and the range of DOC's increase to 3.27¢ per seat mile to 3.76¢ per seat mile as shown in Table 2.13.

Direct operating costs per seat mile and seat kilometer as a function of block distance are shown in Figure 2.41 for the specified combinations of aircraft utilization and airframe costs. Figure 2.41 also illustrates the impact of extending the design range of the TH-100 (92.3) to 460 statute miles. The increase in costs at the design point range (230 statute miles) is the result of the loss of 2 available seats due to the increased weight empty for the installation of larger fuel tanks. Although not shown in Figure 2.41, it should be noted that the larger fuel tanks will result in a small increase (less than 1%) in seat mile costs at ranges less than 230 statute miles due to increases in airframe maintenance and depreciation costs. In the extended range version of the TH-100 (92.3) seat mile costs show a continuing increase beyond 230 statute miles because of the loss of available seats due to additional fuel requirements at the longer block distances.

TH-100(92.3)
EXTENDED RANGE VERSION

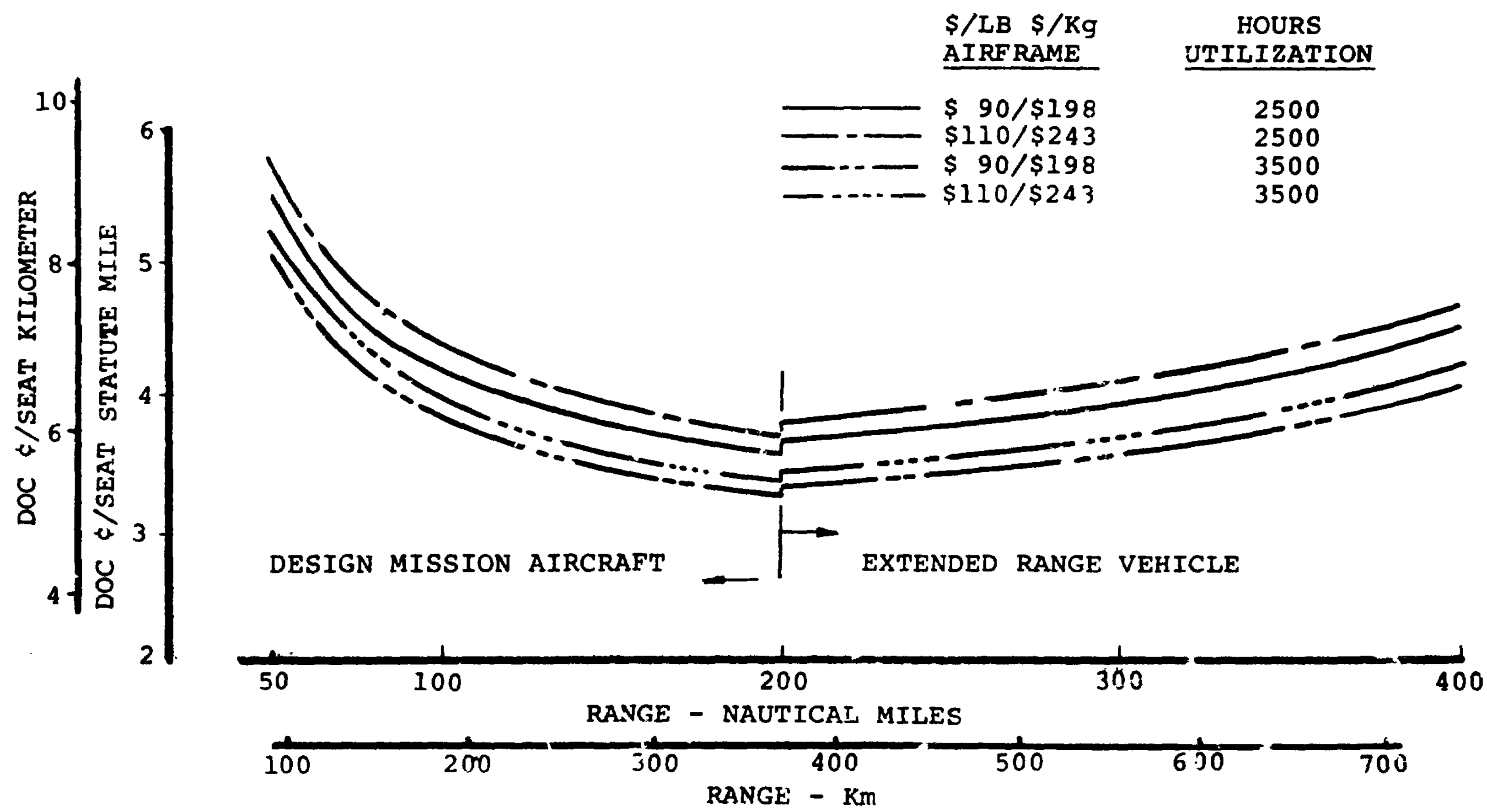
Airframe Cost	Flyaway Costs $90.00/Lb	Flyaway Costs $110.00/Lb
Airframe	$2,231,910	$2,727,890
Dynamic System	1,063,040	1,063,040
Engines	654,255	654,255
Avionics	250,000	250,000
Total	$4,199,215	$4,695,195

Direct Operating Costs
Dollars/Seat Mile
Block Distance = 230 St. Miles

Utilization (Hrs/Yr)	2500		3500	
Airframe Cost ($/Lb)	90	110	90	110
Flying Operations				
Flight Crew	.0082	.0082	.0082	.0082
Fuel and Oil	.0046	.0046	.0046	.0046
Hull Insurance	.0020	.0022	.0014	.0016
Total Flying Operations	.0148	.0150	.0142	.0144
Direct Maintenance				
Airframe - Labor	.0013	.0013	.0013	.0013
- Material	.0010	.0012	.0010	.0012
Engines - Labor	.0007	.0007	.0007	.0007
- Material	.0010	.0010	.0010	.0010
Dynamic System - Labor	.0011	.0011	.0011	.0011
- Material	.0017	.0017	.0017	.0017
Total Direct Maintenance	.0068	.0070	.0068	.0070
Maintenance Burden	.0048	.0048	.0048	.0048
Total Maintenance	.0116	.0118	.0116	.0118
Depreciation	.0097	.0108	.0069	.0077
Total Direct Costs	.0361	.0376	.0327	.0339

TABLE 2.13. TANDEM HELICOPTER - DESIGN POINT (EXTENDED RANGE VERSION) INITIAL AND DIRECT OPERATING COSTS.

FIGURE 2.41. EFFECT OF OPERATING RANGE ON DIRECT OPERATING COST - HELICOPTER



2.2 DESIGN POINT TILT ROTOR AIRCRAFT TR-100 (98.2)

The tilt rotor aircraft is essentially a conventional propeller aircraft in its cruise configuration except that its two wing tip mounted prop/rotors are larger than conventional propellers. The prop/rotors tilt to provide vertical lift in hover and transition to cruise flight. This concept has inherent qualities which make an attractive compromise between the VTOL flexibility of the helicopter and the cruise performance of a conventional aircraft. The low disc loading rotors provide good hover lift performance and agile handling qualities in low speed flight. In cruise the prop/rotor propulsive efficiency is high, which coupled with the high lift/drag ratios typical of wing borne aircraft, provides an efficient cruising vehicle.

2.2.1 Design TR-100 (98.2) - Configuration and Layout

The design point tilt rotor aircraft is shown in Figure 2.42 and a three view of the vehicle is given in Figure 2.43. Table 2.14 provides a list of the major aircraft dimensions and characteristics.

This aircraft has a takeoff gross weight of 74,749 pounds. The rotors are three-bladed and are of hingeless fiberglass construction. The rotor diameter is 56.3 feet and the solidity ratio is 0.089. In hover and low speed flight, cyclic pitch control is applied to the rotor to provide control power and trim. These rotors are highly twisted (34 degrees) by comparison with helicopter blades to provide for efficient operation at high advance ratio as well as in hover. The

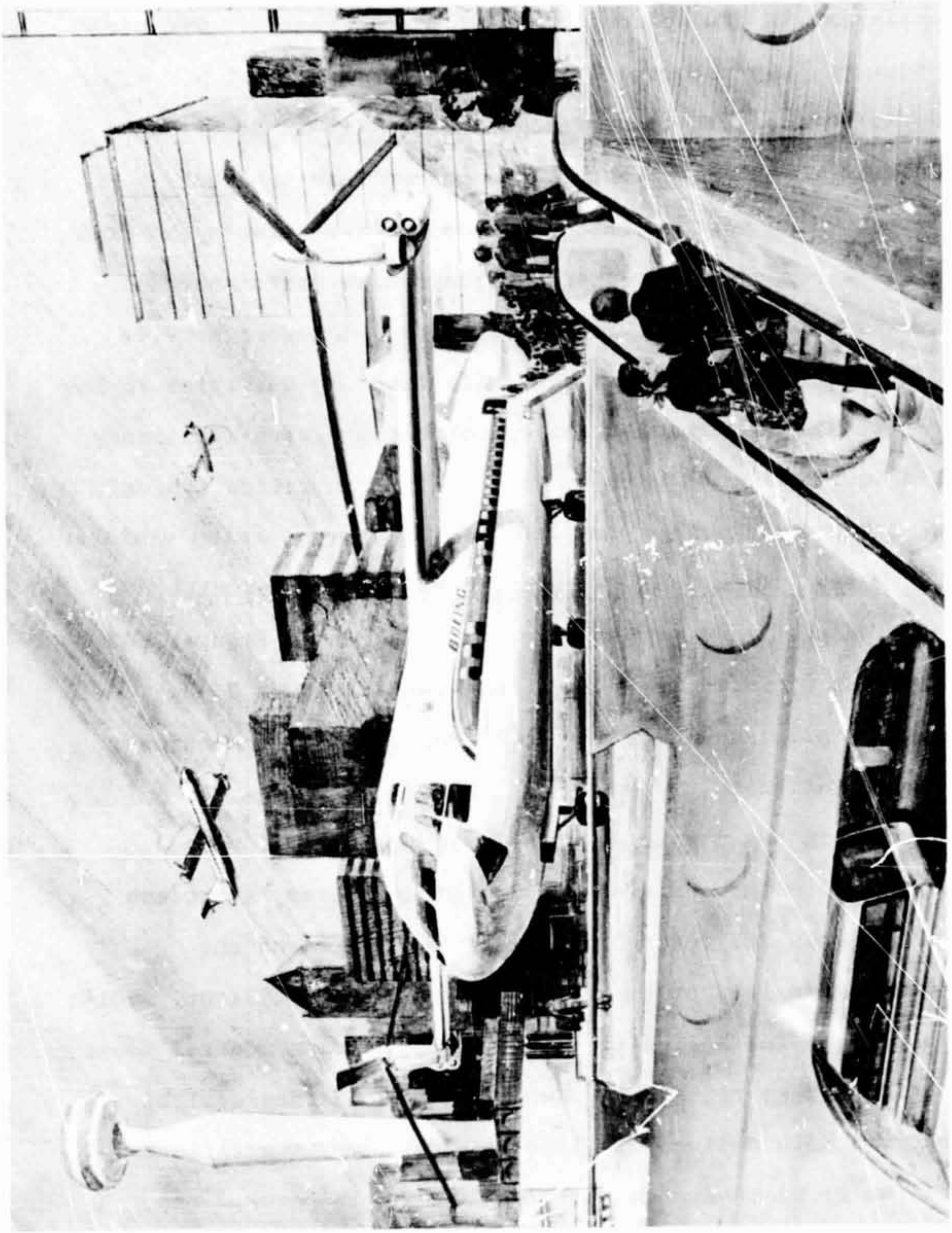


FIGURE 2.42. DESIGN POINT TILT ROTOR AIRCRAFT.

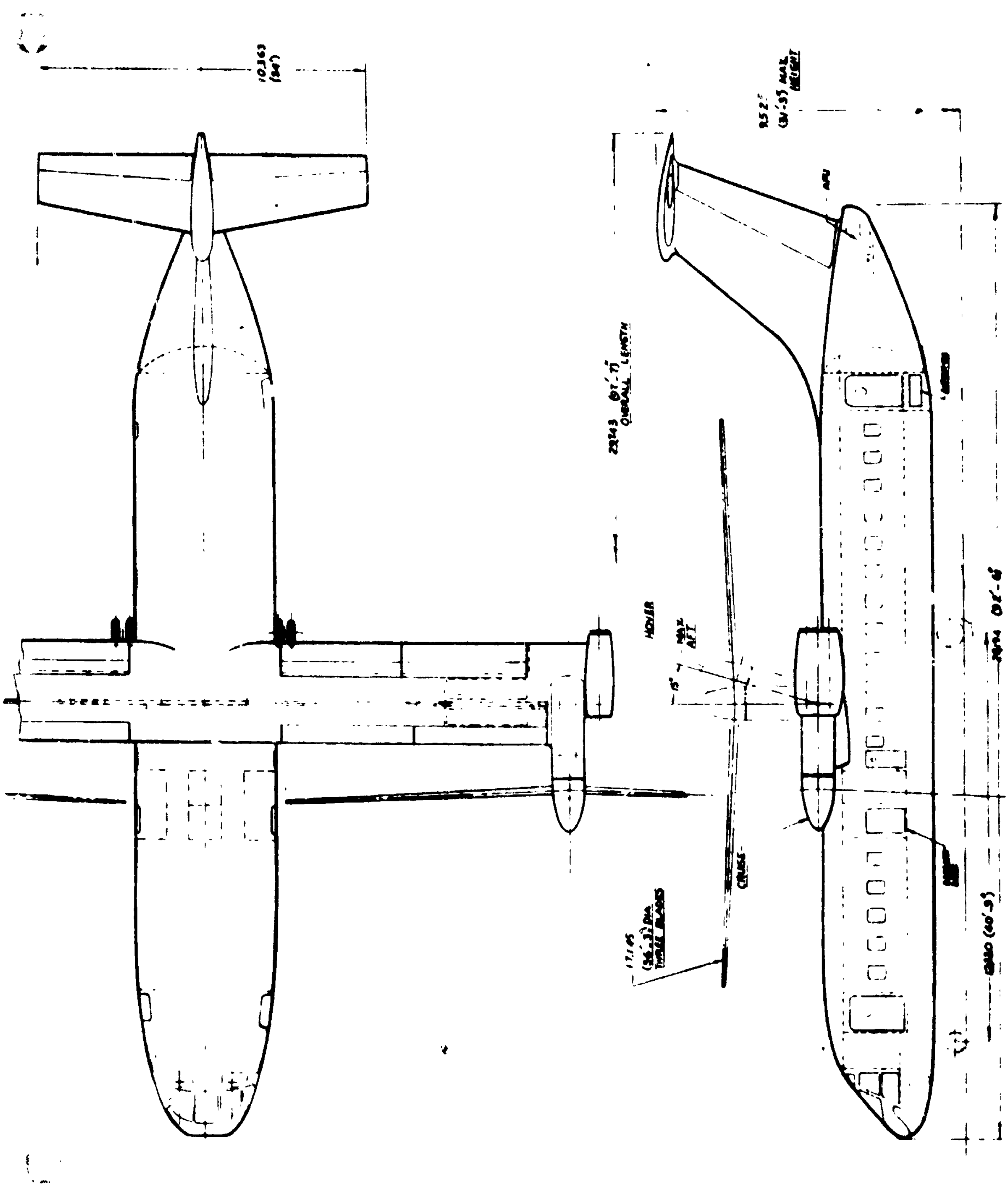


FIGURE 2.43. 1985 COMMERCIAL VTOL,100 PASSENGER TILT ROTOR - BASELINE DESIGN.

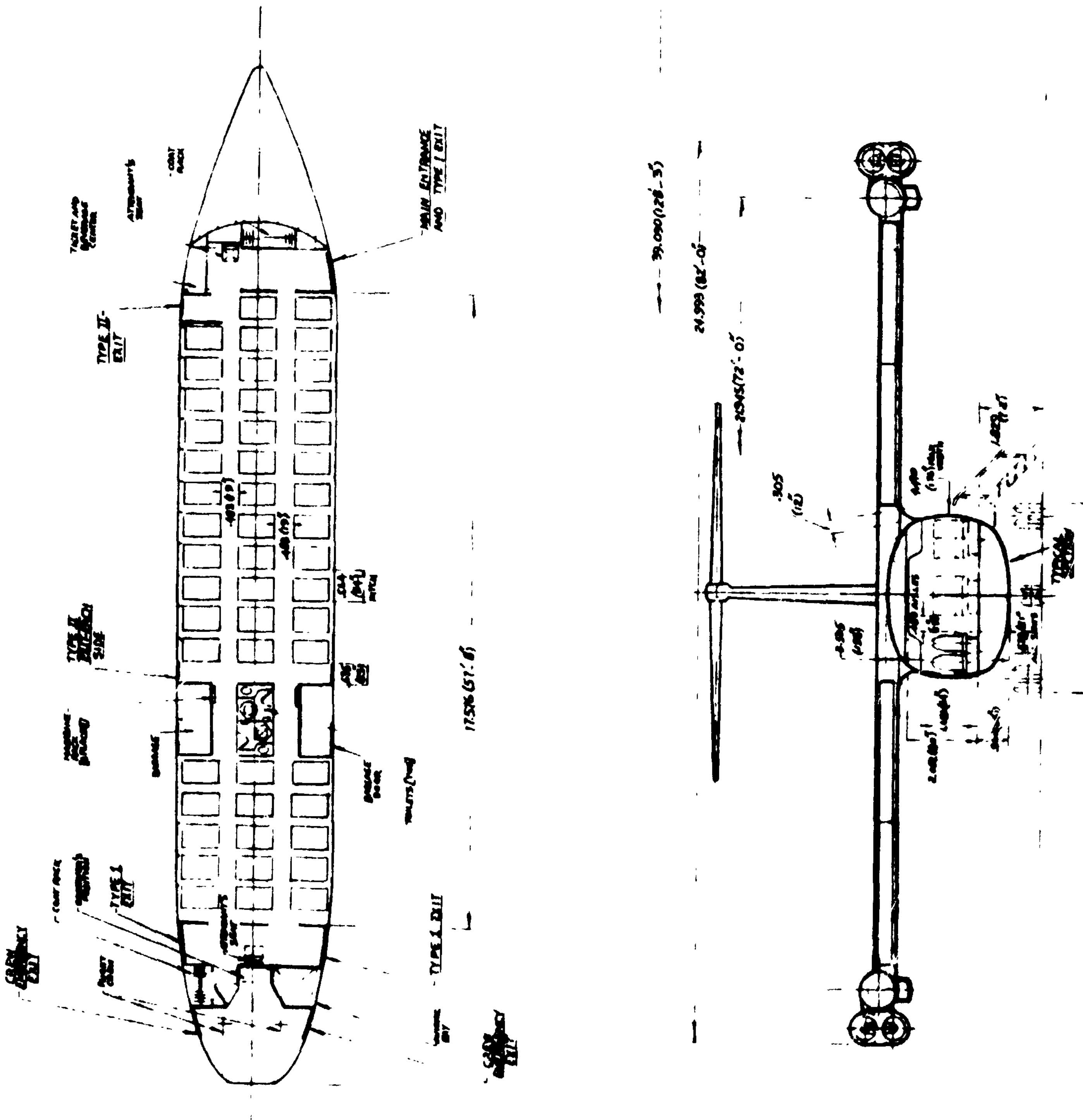


FIGURE 2.43. 1985 COMMERCIAL VTOL, 100 PASSENGER TILT ROTOR - BASELINE DESIGN (CONTINUED).

	S.I. UNITS	U. S. UNITS
WEIGHTS		
DESIGN GROSS WEIGHT	33,905 Kg	74,749 Lbs
WEIGHT EMPTY	22,710 Kg	50,068 Lbs
FUEL WEIGHT	2,111.9 Kg	4,656 Lbs
NUMBER OF PASSENGERS	100	100
ROTORS		
DISC LOADING	73.26 Kg/m^2	15 Lbs/Ft^2
DIAMETER	17.16 m	56.3 Feet
SOLIDITY	0.089	0.089
BLADE NUMBER	3	3
TWIST	36 Degs	36 Degs
TIP SPEED HOVER/CRUISE	23.622/165.506 m/s	775/543 Ft/Sec
POWER		
NO. OF ENGINES	4	4
RATED POWER/ENGINE	3.091×10^6 Watts	4145 HP
FUSELACE		
LENGTH	28.19 m	92.5 Feet
WIDTH (MAX)	4.511 m	14.8 Feet
CABIN LENGTH	17.602 m	57.75 Feet
WING		
AREA	69.44 m^2	747.5 $Feet^2$
SPAN	22.28 m	73.1 Feet
TAPER RATIO	1.0	1.0
CHORD	3.109 m	10.2 Feet
ASPECT RATIO	7.14	7.14
AIRFOIL t/c	0.21	0.21
HORIZONTAL TAIL		
AREA	18.75 m^2	204 $Feet^2$
SPAN	10.668 m	35.0 Feet
TAIL VOLUME RATIO	1.47	1.47
ASPECT RATIO	5.16	5.16
VERTICAL TAIL		
AREA	20.53 m^2	221 $Feet^2$
SPAN	5.364 m	17.6 Feet
TAIL VOLUME RATIO	.159	.159
ASPECT RATIO	1.32	1.32
PERFORMANCE		
NRP CRUISE SPEED	179.54 m/s	349 KTAS
CRUISE ALTITUDE	4267 m	14,000 Feet
BLOCK TIME	.742 Hours	0.742 Hours
NOISE		
SIDELINE NOISE - 500 FEET/HOVER	98.2 PNdB	98.2 PNdB

TABLE 2.14. DESIGN POINT TILT ROTOR TABLE OF CHARACTERISTICS.

The rotors and forward rotor transmission tilt; however, the engines mounted outboard of the tilt package, remain stationary. This arrangement does not require the engines to be requalified for vertical operation and also reduces the inertia of the tilt package.

The aircraft has four engines, two on each wing tip. The rotors and engines are connected by means of a cross-shaft which provides for torque transmission across the aircraft in event of engine failure.

The location of the engines outboard of the tilt package provides easy access to the engine bays for maintenance or engine removal.

The span of the aircraft is 82 feet. The wind is straight and untapered with a NACA 634221 section with a wing setting angle of 2 degrees relative to the fuselage. The wing aspect ratio is 7.14.

The wing flaps are full span 30% chord plain flaperons and are used as both flaps and ailerons. The leading edge is provided with an umbrella flap which opens for hover and low speed "helicopter" flight to alleviate the rotor download on the wing. This device is also used to ensure that the transition from separated to attached flow over the wing lower surface occurs simultaneously on both wings.

The empennage is a T tail configuration to reduce the impact of rotor downwash on the horizontal stabilizer in transition flight. The horizontal tail volume ratio is 1.47 and the vertical tail volume ratio is 0.159.

The landing gear is a tricycle configuration to provide good ground handling characteristics and is retractable into the lower fuselage. The undercarriage provides an overturning angle of 27 degrees.

Cabin layout and passenger accommodation details are shown in Figures 2.43 and 2.44. The aircraft cabin has two main entrances located on the port side of the aircraft. The aft entrance is equipped with an air stair in accordance with NASA guidelines. The rear entrance is the normal entrance/exit.

A third Type I entrance is located on the starboard side of the forward cabin.

Two Type II exits are provided mid-cabin immediately aft of the baggage/toilet facilities.

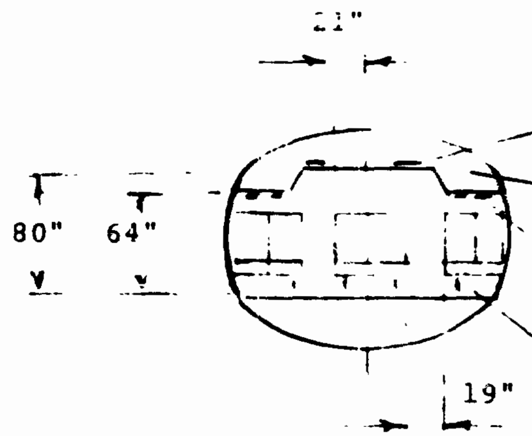
A further Type II exit is located aft directly opposite the main entrance.

The passenger cabin has seats for 100 passengers with an overall seat width of 21 inches and a seat pitch of 34 inches.

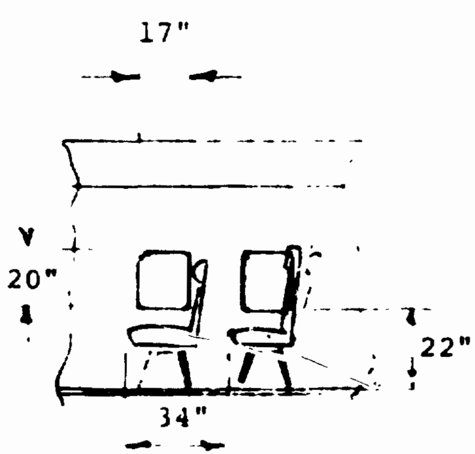
Each passenger has under-seat stowage space (9 inches X 16 inches X 23 inches) and overhead rack stowage with lockable doors. Airvents, individual lights and a folding table are provided for each passenger in line with normal commercial aircraft practice.

The cabin has dual 19 inch aisles and the main cabin lights

Cabin



Windows



Systems

- Air Conditioning - Dual Bleed Air
- Pressurized
- Emergency Oxygen

Escape Provisions

- Two Type I Exits, L.H. Side
- One Type I/One Type II Exits, R. H. Side
- One Type II Exit, Each Side Mid-Cabin

Entrances

- Two Main Entrances, L. H. Side
- Air Stairs, Aft At Entrance
- Service Entrance, R. H. Side, Fwd.

Miscellaneous

- Coat Racks for 80 Passengers
- Two Magazine Racks
- Two Lavatories
- Beverage Service, Aft
- Ticket Center, Aft

FIGURE 2.44. PASSENGER ACCOMMODATIONS - DESIGN POINT TILT ROTOR AIRCRAFT.

are located over the aisles.

Two coat racks are provided - one forward and one aft with provisions for 80 passengers.

Two lavatories are provided in the center of the cabin in line with the baggage stowage area. The location of the baggage and toilet facilities in this areas is to keep passenger seats away from the prop/rotor tip path plane in cruise to minimize noise and vibration. External baggage loading doors are provided to give ground crew access if desired.

The beverage storage and service facilities are located aft. This unit is located adjacent to the service door/emergency exit which is larger than the minimum required Type II exit. Ticketing facilities are located in the same service unit.

The cabin attendants' seats are located - one forward against the forward passenger cabin bulkhead and close to the forward exits, and the second, aft against rear bulkhead and close to the rear exits.

The aircraft avionics and navigational gear compartment is on the port side of the aircraft just forward of the cockpit/cabin bulkhead. The cockpit space provides adequate accommodation for a flight crew of two with excellent visibility. A third "observer" seat is provided adjacent to the avionics bay at the rear of the cockpit. This location provides the observer good forward vision, visibility over the flight crew stations, and also access to the avionics/non-aids bay in required. The cockpit is provided with two crew emergency exits - one on each side of the cockpit.

2.2.2 Tilt Rotor Design Point Aircraft - Weight

The design point tilt rotor aircraft design gross weight is 33,905 Kg (74,749 pounds). The weight breakdown in terms of the structural and system categories is shown in Table 2.15.

In the aircraft sizing procedure, weight trend curves developed at Boeing were used to establish the component and system weights as functions of configuration, size, flight envelope, etc. The fixed useful load, fixed equipment and payload is added and the mission fuel required computed. The aircraft size is iterated until the mission fuel required equals the fuel weight available.

The component and system weights are verified in Volume II by comparison with trend line data.

The calculation of aircraft weight is based upon several guidelines. The guidelines for the study and their impact on weight estimation are discussed in Volume II.

The major guideline requirements are summarized below:

1. The maximum takeoff weight and maximum landing weight shall be the same.
2. Passenger weight shall be 180 pounds (160 pounds passenger and 20 pounds of non-revenue baggage).
3. No revenue cargo is assumed.

	KILOGRAMS	POUNDS
WING	1960.9	4323
ROTOR	2379.5	5246
TAIL	636.8	1404
SURFACES	636.8	1404
ROTOR		
BODY	3853.2	8495
BASIC		
SECONDARY		
ALIGHTING GEAR GROUP	1356.2	2990
ENGINE SECTION	430.0	948
PROPULSION GROUP	4751.8	10476
ENGINE INST'L	1184.3	2611
EXHAUST SYSTEM *		
COOLING		
CONTROLS *		
STARTING *		
PROPELLER INST'L	*367.4	*810
LUBRICATING *		
FUEL	99.3	219
DRIVE	3100.8	6836
FLIGHT CONTROLS	1835.2	4046
AUX. POWER PLANT	288.5	636
INSTRUMENTS	191.9	423
HYDR. & PNEUMATIC	308.4	680
ELECTRICAL GROUP	378.3	834
AVIONICS GROUP	293.9	648
ARMAMENT GROUP		
FURN. & EQUIP. GROUP	3273.6	7217
ACCOM. FOR PERSON.		
MISC. EQUIPMENT		
FURNISHINGS		
EMERG. EQUIPMENT		
AIR CONDITIONING	612.3	1350
ANTI-ICING GROUP	254.0	560
LOAD AND HANDLING GP.		
WEIGHT EMPTY	22804.7	50276
CREW	299.4	660
TRAPPED LIQUIDS	52.2	115
ENGINE OIL	59.9	132
CREW ACCOMMODATIONS	68.0	150
EMERGENCY EQUIPMENT	23.6	52
PASSENGER ACCOMMO.	415.5	916
PASSENGERS (100)	8164.6	18000
FUEL	2017.6	4448
GROSS WEIGHT	33905.4	74749

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TABLE 2.15 DESIGN POINT TILT ROTOR WEIGHT SUMMARY.

4. Accommodation and equipment shall be provided for a flight crew of two and for one cabin attendant per 50 passengers. In addition, some provision shall be made on the flight deck for an occasional flight observer. Each crewman plus gear weighs 190 pounds, and each cabin attendant plus gear weighs 140 pounds.
5. The aircraft shall be equipped with an APU to meet the needs of starting, ground air conditioning and heating.
6. The aircraft designs are to be based on a 1985 operational time period. The Contractor shall assume the airframe structural weight will be reduced by 25% by the use of composite materials.

It is to be assumed that by 1985, a system to permit all weather operation will have been established and that the V/STOL short haul transport system will use it.

<u>Standard Weight Items</u>

The weights of specified standard items shall be as provided in Table 2.16.

<u>Fly-By-Wire Control Systems</u>

Fly-by-wire control systems are permitted. Control configured vehicles (CCV), such as a tailless tilt rotor configuration are not permitted.

<u>Gearboxes</u>

The rotor gearboxes shall be designed for the maximum rated engine power and torque under sea level standard day conditions.

Engines

Rubberized versions of existing engine designs are permitted, as appropriate for commercial service in 1985. The engine specific weight shall be 0.15 pounds per shaft horsepower. The guideline weight of (544.2 Kg) 1,200 pounds for instrumentation, electrical, electronics and auxiliary power unit installation has been assumed to be the uninstalled weight and an additional weight of 440.8 Kg (972 pounds) has been added to reflect installation.

ITEM		WEIGHT
WHEELS, TIRES, AND BRAKES		COMPANY OPTIMUM
INSTRUMENTS (FLIGHT AND NAVIGATION) ELECTRICAL (EXCLUDING GENERATING EQUIPMENT) ELECTRONICS (COMMUNICATION, FLIGHT, AND NAVIGATION) AUXILIARY POWER UNIT INSTALLATION		1200 LBS
SEATS AND BELTS		
PASSENGER:	DOUBLE	16 LB/PASSENGER
	TRIPLE	16 LB/PASSENGER
CREW SEATS:	CABIN CREW	16 LB/CREW MEMBER
	FLIGHT CREW	40 LB/CREW MEMBER
LAVATORY		300 LB/UNIT
BEVERAGE ONLY		200 LB TOTAL
AIR STAIR		400 LB

TABLE 2.16. TILT ROTOR WEIGHTS GUIDELINES.

The cockpit and passenger cabin accommodation weights have been based upon the Boeing 737 aircraft since it was considered that passenger comfort of at least current commercial quality would be required.

The landing gear was sized to take a rate of sink of 500 feet per minute and represents 4% of the gross weight.

The fly-by-wire control system weights are based upon recent Boeing experience with fly-by-wire controls in the 347 helicopter.

The aircraft structure has been sized to a maneuver load factor of 2.5 and an ultimate load factor of 3.75 as recommended in FAR Part 25.

The aircraft center of gravity locations and moments of inertia are given in Table 2.17 for both hover and cruise flight at the extremes of the weight envelope, i.e., weight empty and design gross weight.

The excursions of center of gravity travel are shown for both hover and cruise flight in Figure 2.45. The center of gravity envelope for this aircraft assumes that window seats are filled first, followed by aisle seats.

The aircraft weight resulting from this study is governed to a large extent by the selection of fixed equipment and fixed useful load weights as well as payload. In order to facilitate reasonable comparison with aircraft designed in other studies using different weights, growth factor data are given in Figure 2.46. This plot provides the change in aircraft gross

	WEIGHT EMPTY	GROSS WEIGHT
WEIGHT	22,804.7 Kg (50,276 LBS)	33,905.4 Kg (74,749 LBS)
CENTER OF GRAVITY*		
HORIZONTAL FLIGHT		
FUSELAGE STATION	12.72 M (500.8 IN.)	12.77 M (502.8 IN.)
WATER LINE	3.56 M (140.4 IN.)	3.26 M (128.5 IN.)
VERTICAL FLIGHT		
FUSELAGE STATION	13.08 M (515.0 IN.)	13.12 M (516.5 IN.)
WATER LINE	3.96 M (156.1 IN.)	3.53 M (139.0 IN.)
MOMENT OF INERTIA		
HORIZONTAL FLIGHT		
I_{xx} (ROLL)	1,199,928 Kg M^2 (885,170^{2}Slug Ft^2)	1,290,245 Kg M^2 (951,796 Slug Ft^2)
I_{yy} (PITCH)	519,241 Kg M^2 (383,037 Slug Ft^2)	558,324 Kg M^2 (411,868 Slug Ft^2)
I_{zz} (YAW)	1,398,099 Kg M^2	1,503,382 Kg M^2
VERTICAL FLIGHT		
I_{xx} (ROLL)	1,261,339 Kg M^2 (930,473 Slug Ft^2)	1,356,279 Kg M^2 (1,000,509 Slug Ft^2)
I_{yy} (PITCH)	562,151 Kg M^2 (415,622 Slug Ft^2)	604,464 Kg M^2 (446,905 Slug Ft^2)
I_{zz} (YAW)	1,512,572 Kg M^2 (1,115,805 Slug Ft^2)	1,626,422 Kg M^2 (1,199,790 Slug Ft^2)

*FUSELAGE STATION O IS NOSE OF BODY, CENTERLINE OF ROTOR IN HORIZONTAL FLIGHT IS 4.6 METERS ABOVE WATER LINE O.

TABLE 2.17 . WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA - DESIGN POINT TILT ROTOR.

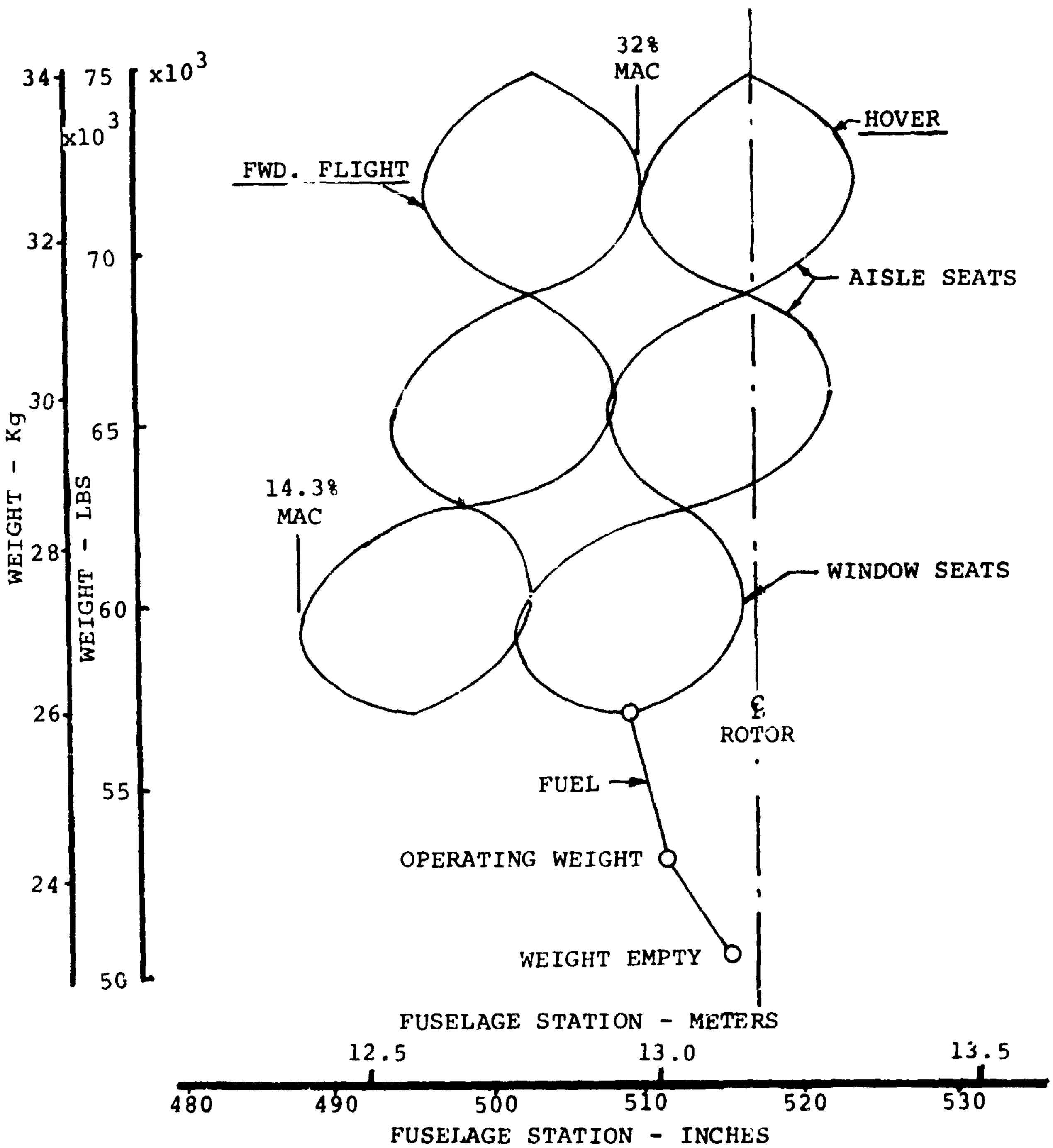


FIGURE 2.45 . BASELINE TILT ROTOR - CENTER OF GRAVITY ENVELOPE.

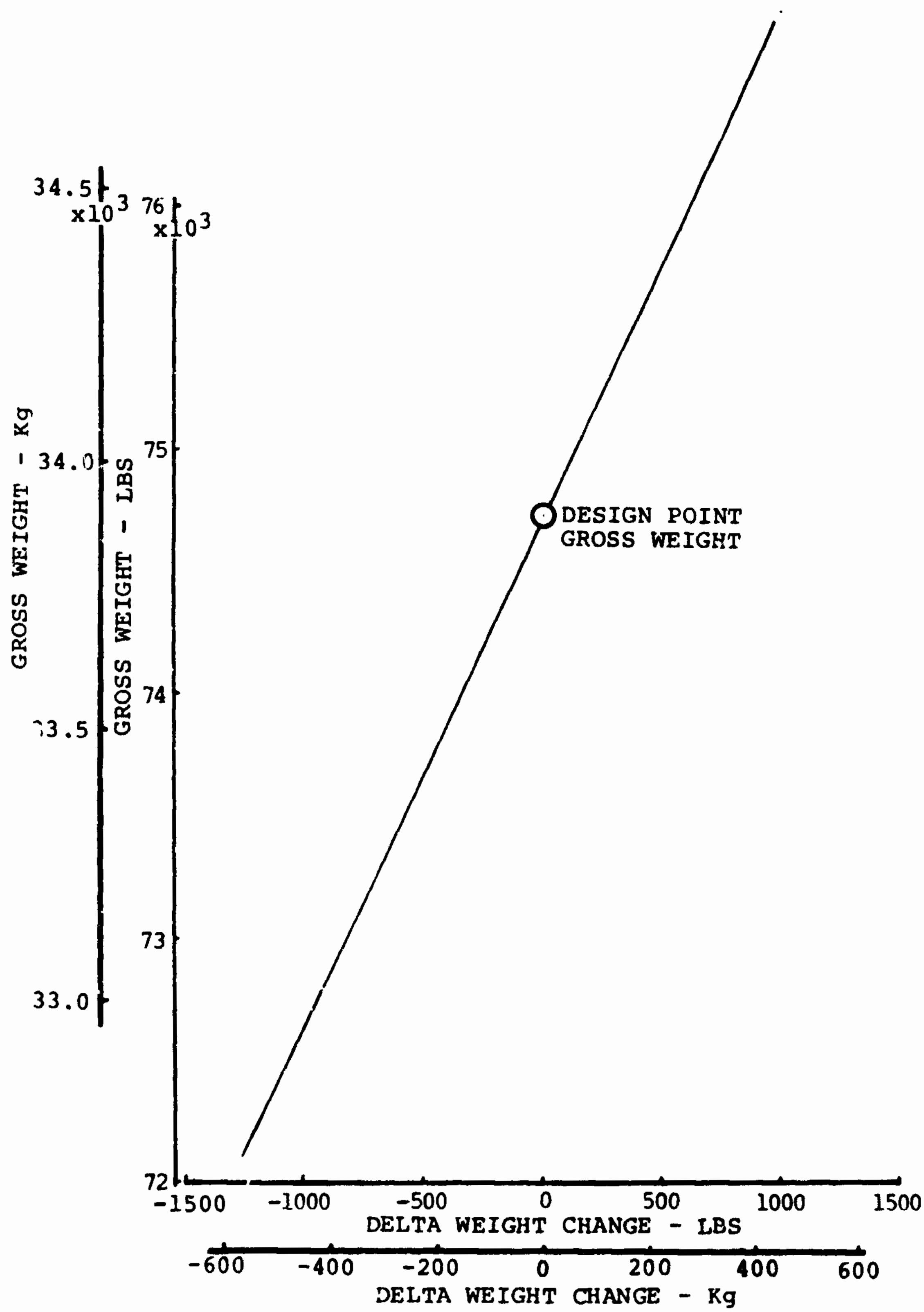


FIGURE 2.46 . TILT ROTOR WEIGHT GROWTH AT CONSTANT PERFORMANCE AND STRENGTH.

weight for increasing or decreasing fixed weight items.

2.2.3 Design Point Tilt Rotor - Vehicle Performance

The design point tilt rotor aircraft has been sized to the mission defined in Table 2.5. This aircraft carries 100 passengers over a short haul range of 371 Kilometers (200 nautical miles).

A summary of the mission performance is given in Tables 2.18 and 2.19.

The initial phases of the mission including taxi, takeoff, initial air maneuver and conversion to cruise flight require 193.1 pounds of fuel. The aircraft then climbs to 14,000 feet at an initial rate of climb of 4,227 feet per minute and a final rate of climb of 2,265. At the end of the climb segment the aircraft has burned 600.7 pounds of fuel and has travelled 12.45 nautical miles down range.

The cruise segment is done at 14,000 feet at an initial aircraft weight of 74,148 pounds and a true airspeed of 349 Knots. At the end of the cruise segment the aircraft fuel used is 2,799.8 pounds and the distance travelled has increased to 171.82 nautical miles. The aircraft speed at the end of cruise is 351 Knots TAS.

The descent to 2,000 feet altitude is initially at 4,073 feet per minute rate of descent falling to 2,027 feet per minute at 2,000 feet altitude. The fuel used at the end of descent amounts to 2,938.4 pounds for a range of 200 nautical miles.

BASELINE TILT ROTOR - MISSION BREAKDOWN

	TIME (HOURS)	DISTANCE (N.MI.)	WEIGHT (LBS)	FUEL (LBS)	V (KNOTS)	R/C (FT/MIN)
	0	0	74,749			
TAXI				14	0	0
	.017	0	74,735			
TAKEOFF				179	0	0
	.050	0	74,556			
CLIMB				408	178	+3200
	.121	12.45	74,148			
CRUISE				2,200	351	0
	.576	171.82	71,948			
DESCENT				139	305/250	-2033
	.672	200	71,809			
AIR MANEUVER				65	138	0
	.697	200	71,744			
DESCENT				12	255	-2033
	.705	202	71,732			
LANDING				128	0	0
	.730	202	71,604			
TAXI				14	0	0
	.747	202	71,590			
RESERVE				1,511	143/242	0
	1.278	250	70,079			

TABLE 2.18. BASELINE TILT ROTOR DESIGN MISSION PERFORMANCE (U.S. UNITS).

	TIME (HOURS)	DISTANCE (KM)	WEIGHT (Kg)	FUEL (Kg)	V (KNOTS)	R/C m/s
	0	0	33,905			
TAXI				6	0	
	.017	0	33,899			
TAKEOFF				82	0	
	.050	0	33,817			
CLIMB				185	178	21.5/11.5
	.121	23	33,632			
CRUISE				998	351	
	.576	319	32,634			
DESCENT				63	305/250	-20.7/-10.3
	.672	371	32,571			
AIR MANEUVER				29	138	
	.697	371	32,542			
DESCENT				6	255	
	.705	374	32,536			
LANDING				58	0	-10.2
	.730	374	32,478			
TAXI				6	0	
	.747	374	32,472			
RESERVE				685	143/242	
	1.228	463.2	31,787			

TABLE 2.19. MISSION SUMMARY DESIGN POINT TILT ROTOR (S.I. UNITS).

The final air maneuver or loiter for 1.5 minutes increases the fuel used to 3,003.9 pounds. The descent to 1,000 feet altitude is done at an average rate of descent of 2,092 feet per minute followed by the descent from 1,000 feet conversion and landing. At touchdown the aircraft has used 3,143.5 pounds of fuel and after a final taxi segment completes the mission for 3,157.4 pounds of fuel.

Table 2.18 also shows the computation of reserve fuel which is 1,511 pounds for a total fuel load of 4,668.64 pounds. The mission block time is 0.747 hours.

Hover Performance

The hover performance of the aircraft is shown in Figures 2.47 and 2.48 in terms of the gross weight lifting capability of the aircraft as a function of ambient temperature.

Data are shown for "all engines operating" (AEO) and also "one engine inoperative" (OEI) both in and out of ground effect (IGE,OGE). The power level shown for the all engines operating case is the normal design takeoff power setting. For the one engine inoperative data a 9% power increase per engine has been allowed.

Allowance has been made in the computations for force/weight ratios of F/W = 1.05 (AEO) and F/W = 1.03 OEI in accordance with guideline requirements.

The aircraft sizing condition was the OEI case at 90-degrees F sea level. This point is shown on Figure 2.47 giving a design condition lift capability of 74,749 pounds of out ground effect.

BASELINE AIRCRAFT PERFORMANCE

TILT ROTOR/100 PASSENGER/98.2 PNdB

ONE ENGINE INOPERATIVE

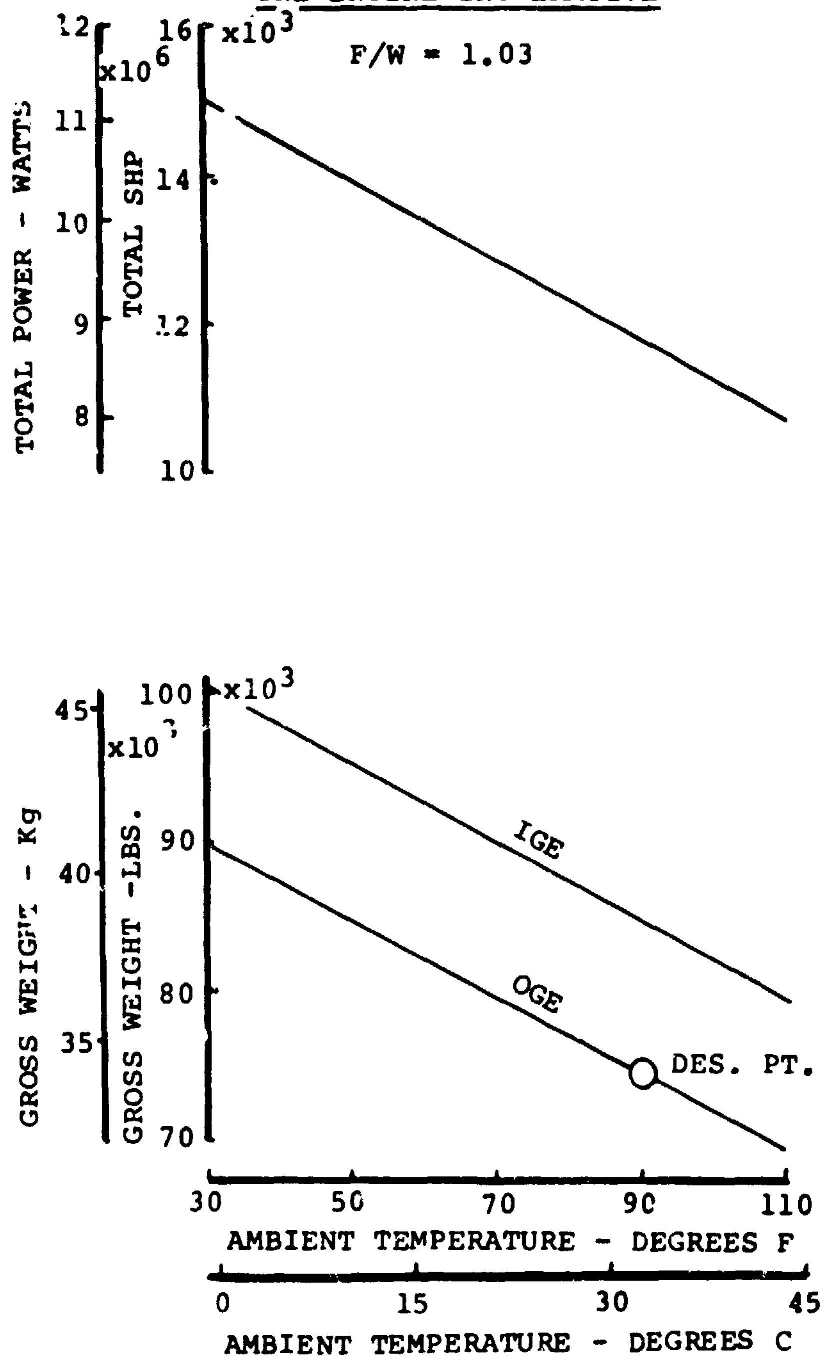


FIGURE 2.47 EFFECT OF AMBIENT TEMPERATURE ON SEA LEVEL - HOVER PERFORMANCE, ONE ENGINE INOPERATIVE.

BASELINE AIRCRAFT PERFORMANCE

TILT ROTOR/100 PASSENGER/98.2 PNdB

ALL ENGINES OPERATING
F/W = 1.05

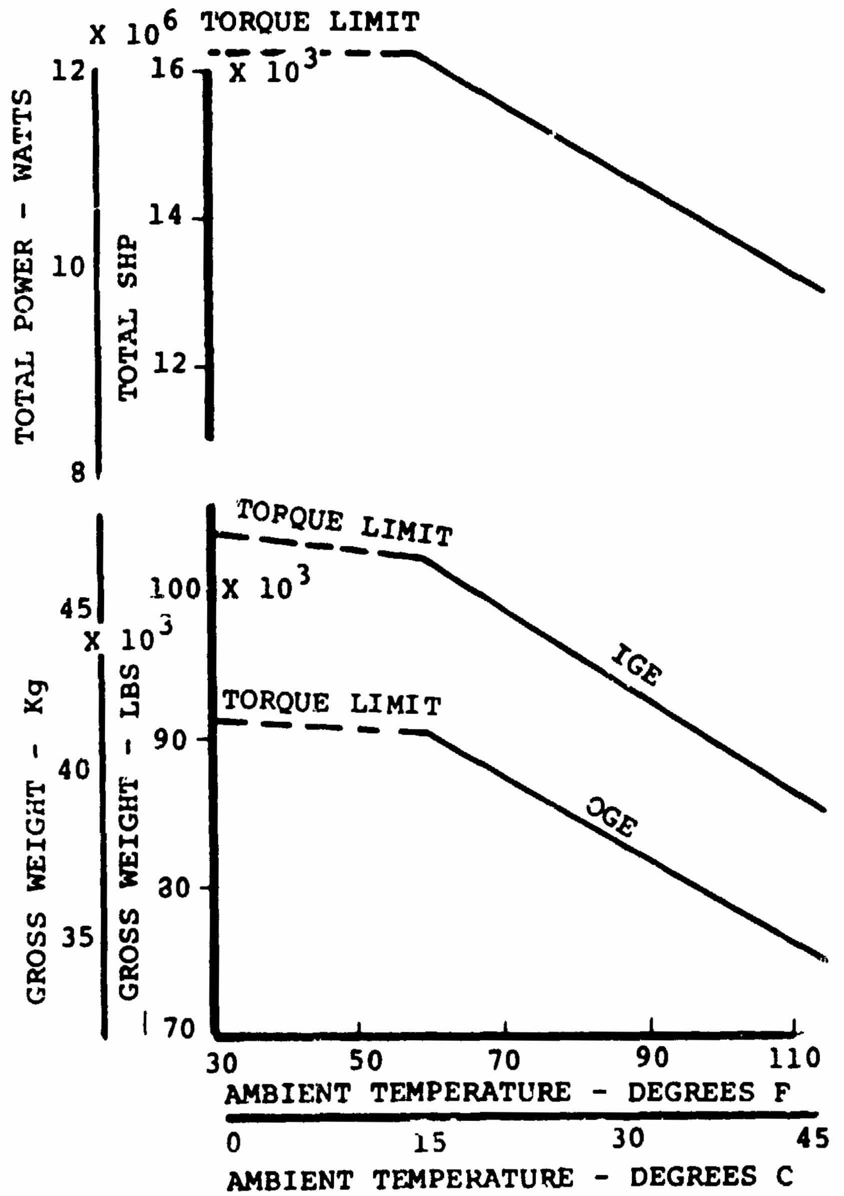


FIGURE 2.48 EFFECT OF AMBIENT TEMPERATURE ON SEA LEVEL - HOVER PERFORMANCE, ALL ENGINES OPERATING.

With all engines operating the main drive train sets the torque limit, and is shown to be adequate for sea level, standard day, all engines operating with a force-to-weight ratio of 1.05. The torque limit for the main transmission was set by cruise at normal rated power at 14,000 feet.

The additional lift capability at temperatures below the aircraft sizing condition is not normally used since the OEI requirement sets the FAA gross weight certification.

The IGE data shown reflect the undercarriage just clear of the ground condition and at the OEI condition 90-degrees F ground effect provides an additional 10,450 pounds of lift. This extra lift capability will not be used operationally as payload; however it provides a useful cushion for deceleration or flare of the aircraft on landing, and an additional initial force-to-weight capability on lift-off (F/W initial = 1.36 all engines operating IGE at design gross weight).

The lift performance of the aircraft at altitude is shown in Figure 2.49.

At design gross weight (74,749 pounds) the aircraft can hover (OGE) at 3,600 feet altitude, all engines operating at an ambient temperature of standard plus 31-degrees F and on a standard day can maintain hover at 7,600 feet fully loaded.

BASELINE AIRCRAFT PERFORMANCE

TILT ROTOR/100 PASSENGER/98.2 PNdB

STANDARD DAY AND STANDARD DAY PLUS 31°F (+17.2°C)

ALL ENGINES OPERATING

F/W = 1.05

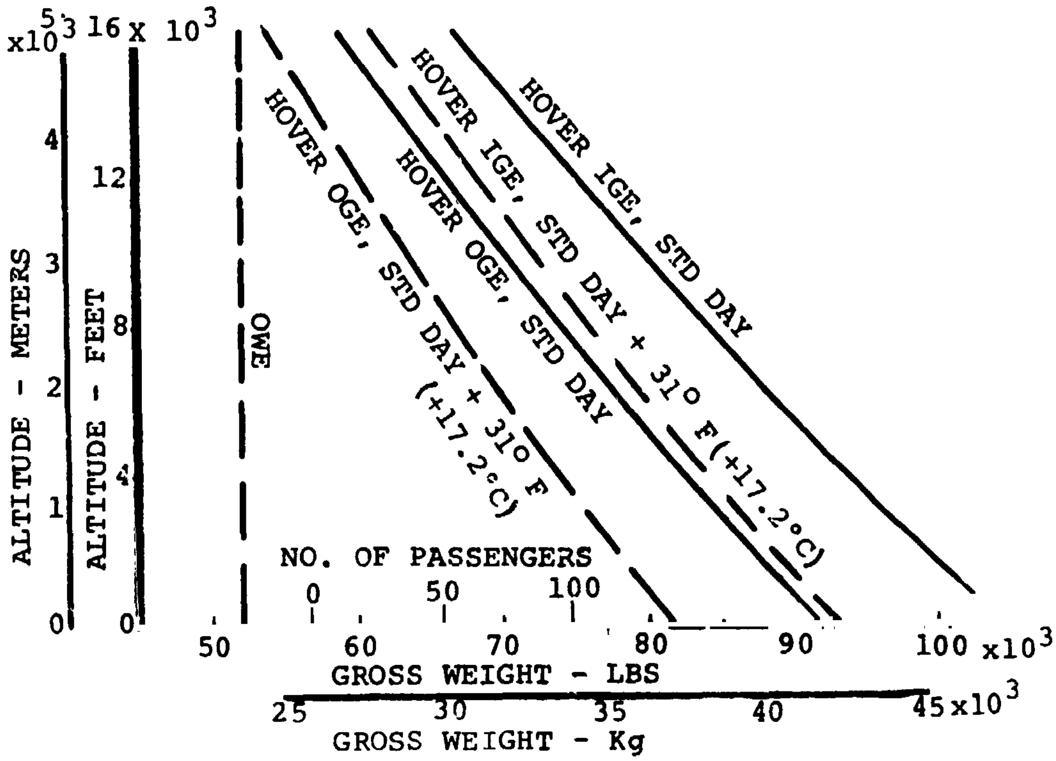


FIGURE 2.49. EFFECT OF ALTITUDE ON HOVER GROSS WEIGHT CAPABILITY

With zero payload the maximum hover altitudes increase to 14,200 feet (standard day plus 31 degrees) and to 17,000 feet standard day.

The one engine inoperative (OEI) case is shown in Figure 2.50. The design point sizing condition is at sea level 90 and is again shown at 74,749 pounds. At standard day conditions, the fully loaded aircraft can maintain hover OEI at 4,000 feet altitude OGE.

Transition Performance

Performance in transition depends on how nacelle angle is scheduled with speed. This is in turn a function of control system details. A detailed design of the transition control scheduling has not been attempted in this conceptual study. However, the power required to fly the transition trim schedule shown in Figure 2.71 has been computed and is shown in Figure 2.51. Throughout transition the power required is much less than the power available at NRP.

Cruise Performance

In cruise flight the nacelles are fully down and the rotors are operating as propellers. The rotor RPM is decreased to 70% of the hover RPM.

Data on power required and normal rated power (NRP) available in cruise are shown for three aircraft weights at 5,000 feet and 14,000 feet altitude in Figures 2.52 and 2.53. At 5,000 feet altitude the aircraft is transmission limited to 324 Knots at design gross weight, all engines operating and to

BASELINE AIRCRAFT PERFORMANCE

TILT ROTOR/100 PASSENGER/98.2 PNdB

ONE ENGINE INOPERATIVE

F/W = 1.03

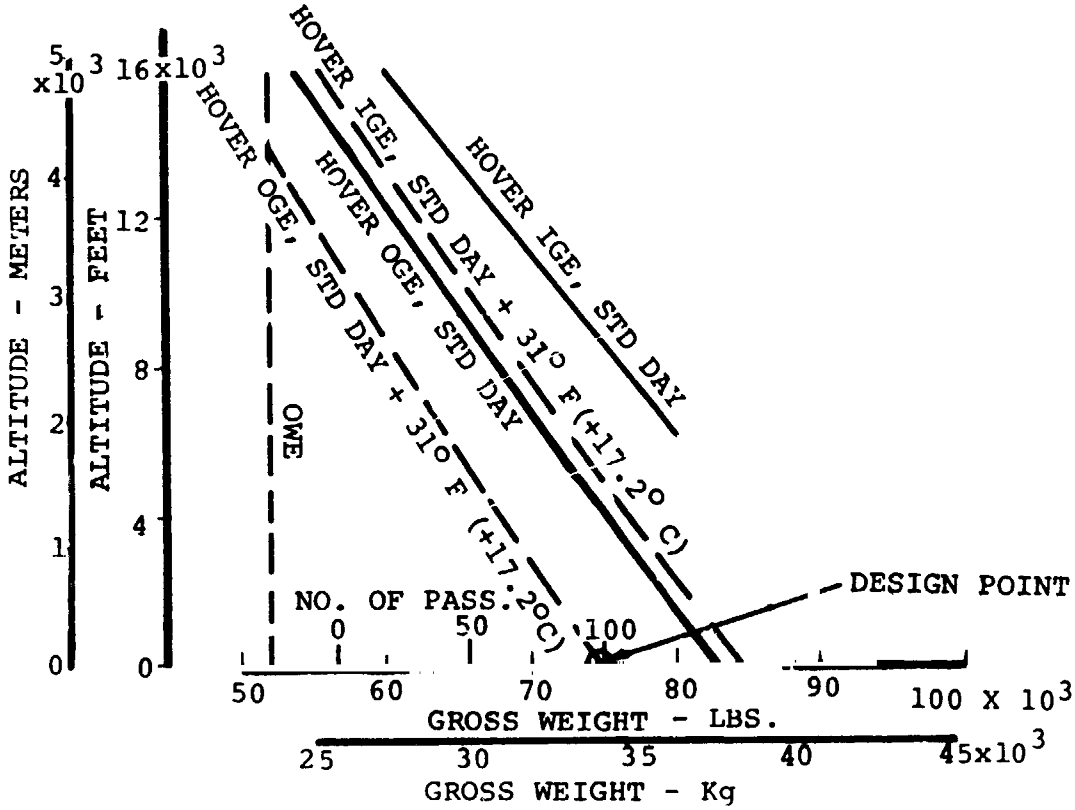


FIGURE 2.50. EFFECT OF ALTITUDE ON HOVER GROSS WEIGHT C ABILITY.

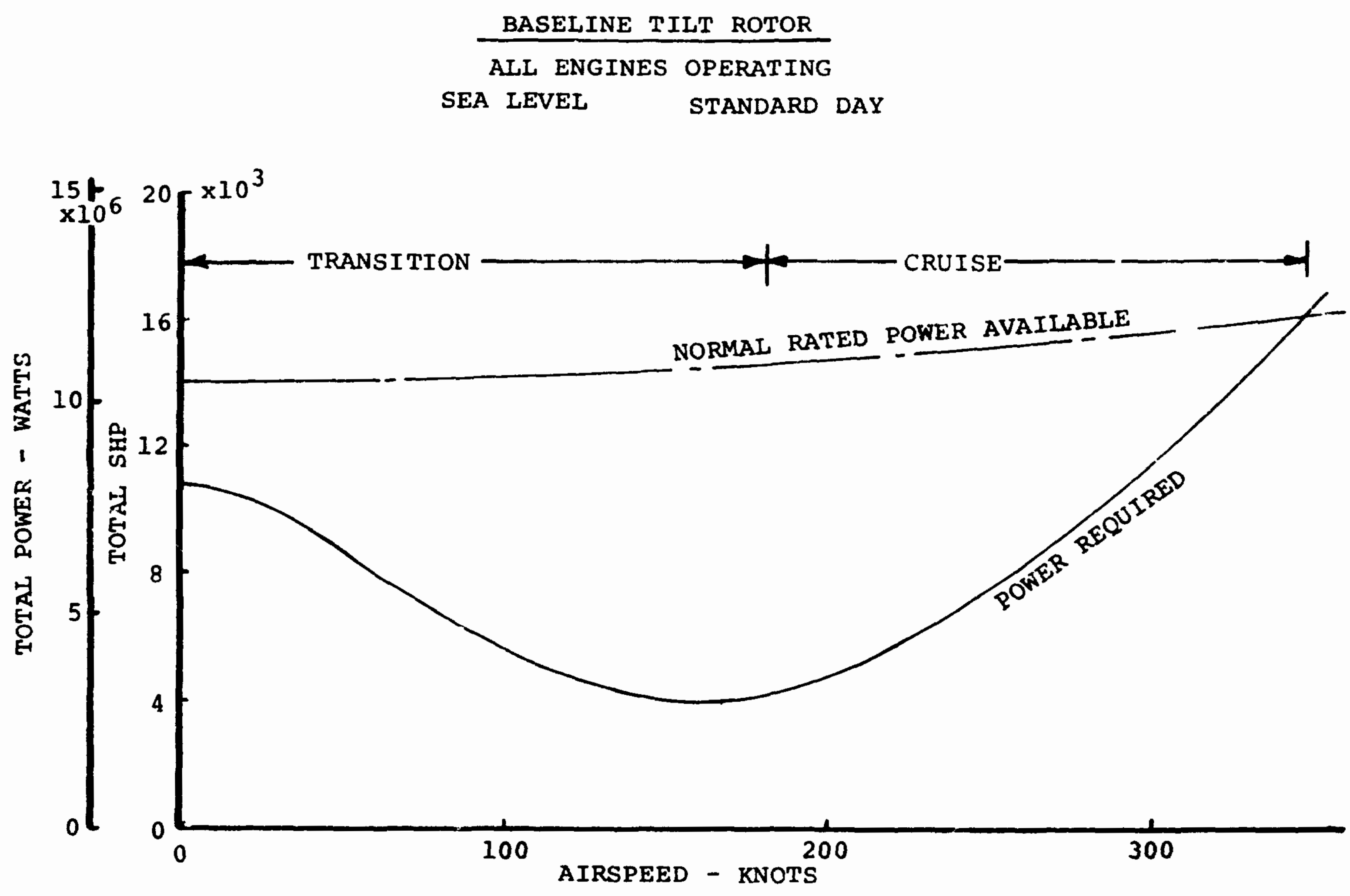


FIGURE 2.51. BASELINE TILT ROTOR POWER REQUIRED THROUGH TRANSITION.

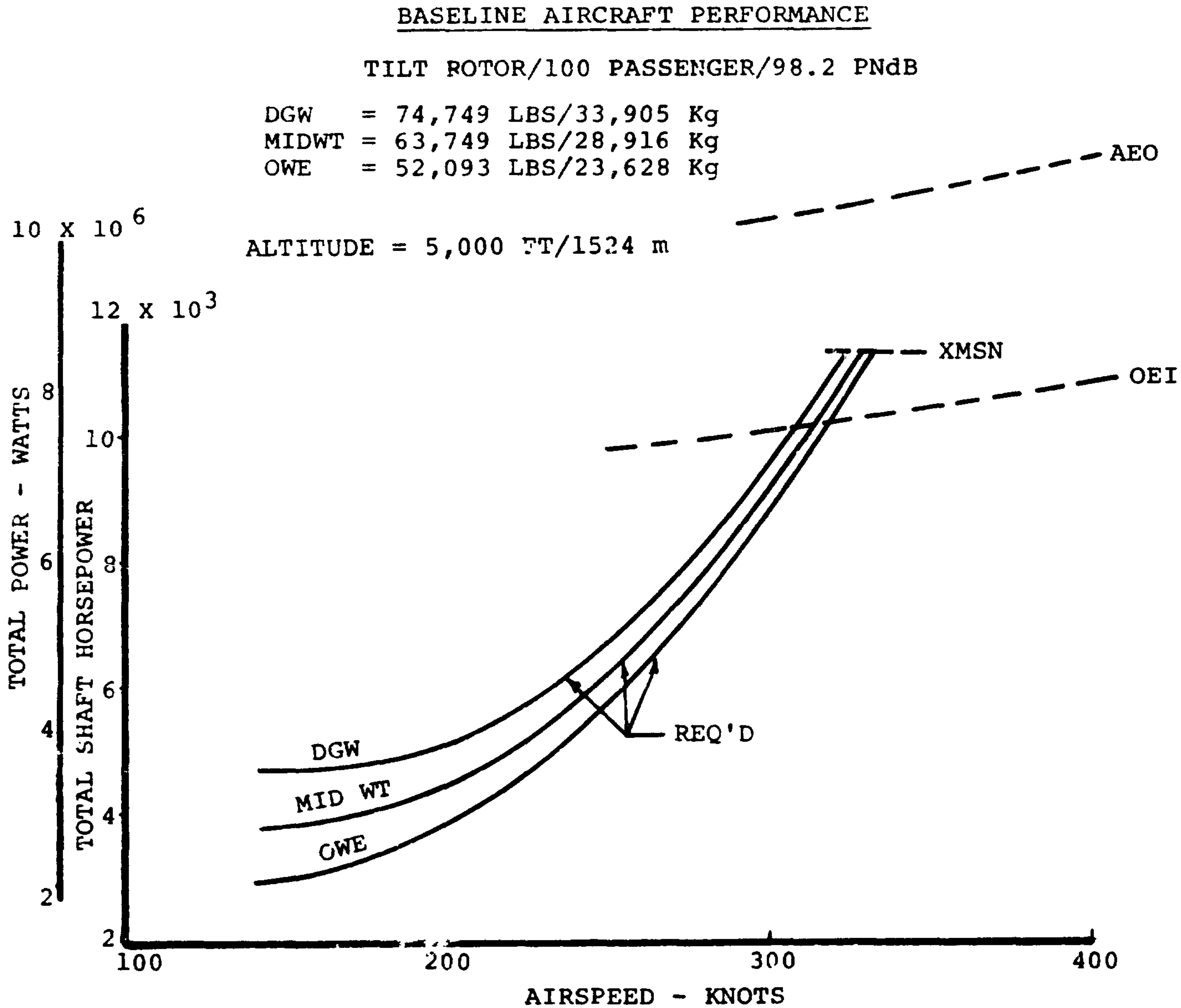


FIGURE 2.52. CRUISE PERFORMANCE - POWER REQUIRED/AVAILABLE - STANDARD DAY - CRUISE RPM.

BASELINE AIRCRAFT PERFORMANCE

TILT ROTOR/100 PASSENGER/98.2 PNdB

DGW = 74,749 LBS/33,905 Kg
MIDWT = 63,749 LBS/28,916 Kg
OWE = 52,093 LBS/23,628 Kg

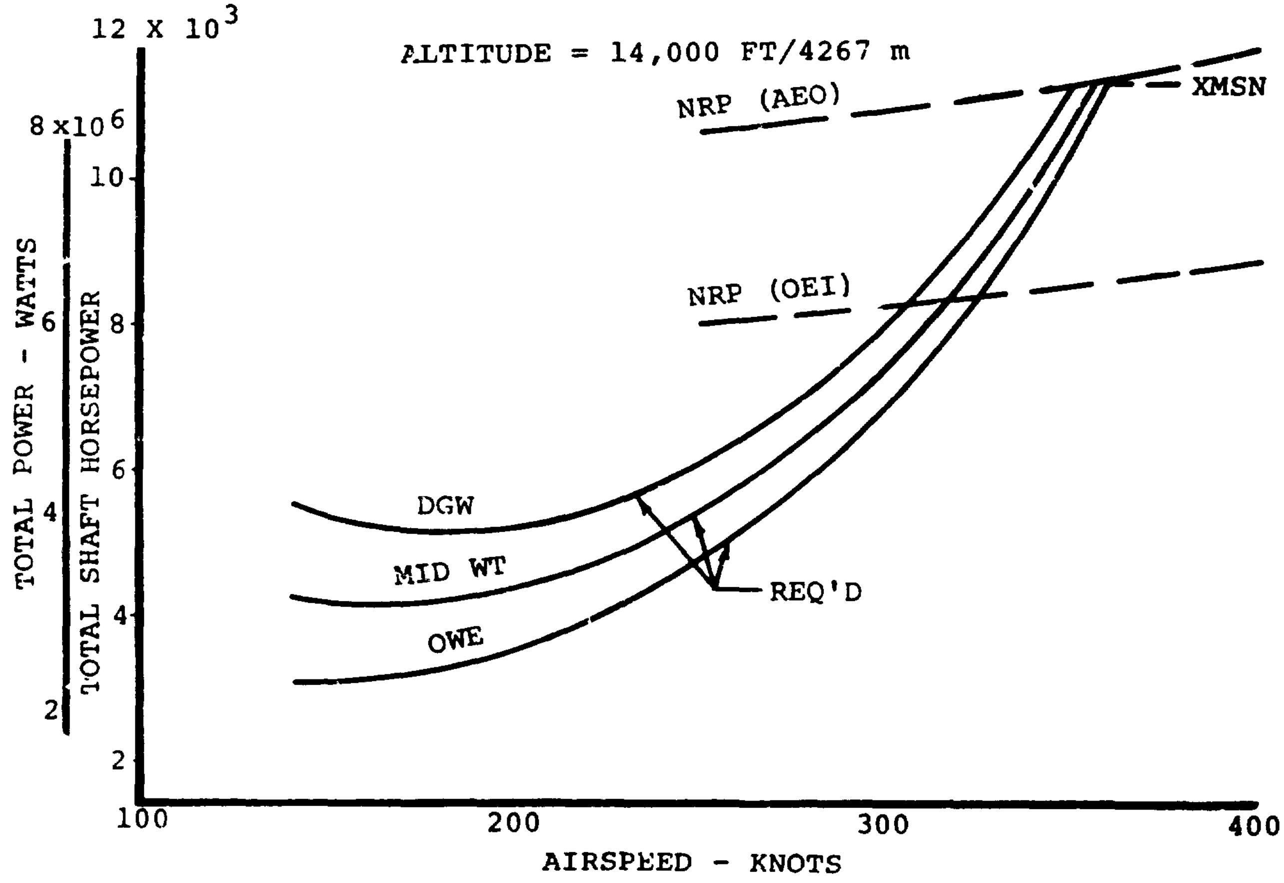


FIGURE 2.53. CRUISE PERFORMANCE - POWER REQUIRED/AVAILABLE - STANDARD DAY - CRUISE RPM.

332 Knots at operating weight empty.

The one engine inoperative power available allows a cruise speed of 310 Knots at design gross weight and 320 Knots at operating weight empty.

With all engines operating at 14,000 feet and design gross weight, the aircraft transmission limit and NRP occur simultaneously limiting the airspeed to 350 Knots. This condition was used to size the main rotor transmission.At operating weight empty this speed can be increased to 360 Knots.

The one engine inoperative case is power limited and a true airspeed of 306 Knots can be maintained at design gross weight. This speed increases to 325 Knots at operating weight empty.

The intersections of the power required - power available data define the velocity capability of the aircraft at various altitudes. This data is shown in Figure 2.54.

The aircraft maximum speed at design gross weight is 350 Knots at 14,000 feet. Below this altitude the aircraft is transmission limited and above 14,000 feet it is power limited. At minimum flying weight - operating weight empty - the maximum airspeed is 360 Knots at 14,400 feet altitude.

The one engine inoperative case is not transmission limited and results in a maximum low altitude speed of 310 Knots at 4,000 feet.

The speed capability of the aircraft is greater than the 250 Knots EAS restriction at less than 10,000 feet, and the

BASELINE AIRCRAFT PERFORMANCE

TILT ROTOR/100 PASSENGER/98.2 PNdB

STANDARD DAY — CRUISE RPM

ALL ENGINES OPERATING & ONE ENGINE INOPERATIVE — NORMAL RATED POWER

DGW = 74,749 LBS/33,905 Kg
OWE = 52,093 LBS/23,628 Kg

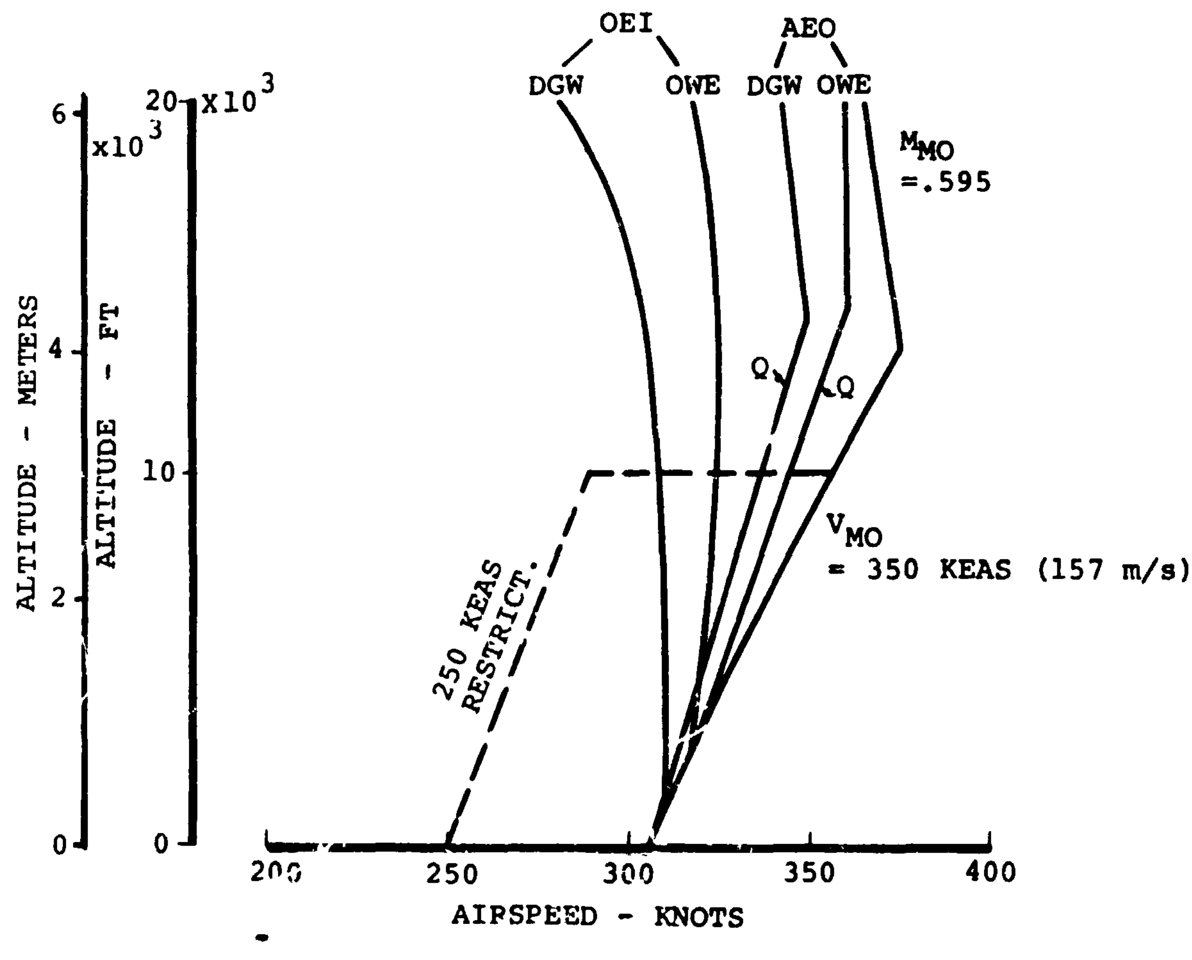


FIGURE 2.54. LEVEL FLIGHT CRUISE SPEED ENVELOPE.

vehicle would be constrained operationally to the 250 Knot EAS boundary shown in Figure 2.54.

Rate of Climb

The design point tilt rotor aircraft climb capability in the cruise flight mode is shown for both the design gross weight and operating weight empty as a function of altitude in Figure 2.55. Two sets of data are shown for both all engines operating and one engine inoperative.

At design gross weight (AEO) the aircraft can climb at 4,600 feet per minute at sea level and at normal cruise altitude 14,000 feet can maintain a rate of climb of 3,109 feet per minute.

In the one engine inoperative case the aircraft can maintain adequate climb rates in its normal operating range of altitudes (3,000 feet per minute at sea level and 1,350 feet per minute at 14,000 feet altitude) at design gross weight. At lighter weight (e.g., OWE) the climb rates increase and in some cases require a fuselage angle in excess of 20 degrees. This is shown for the OWE data in Figure 2.55 and reflects a probable normal operational maximum rate of climb.

Specific Range

Specific range data in the cruise flight configuration are shown in Figures 2.56 and 2.57. The AEO case at both 5,000 feet and 14,000 feet altitudes is given in Figure 2.56. At the design cruise speed of 348 Knots at 14,000 feet and design gross weight the aircraft achieves 0.0725 nautical miles per

BASELINE AIRCRAFT PERFORMANCE

TILT ROTOR/100 PASSENGER/98.2 PNdB

CLIMB CAPABILITY — TAKEOFF RPM
STANDARD DAY — MIL POWER
OEI & AEO

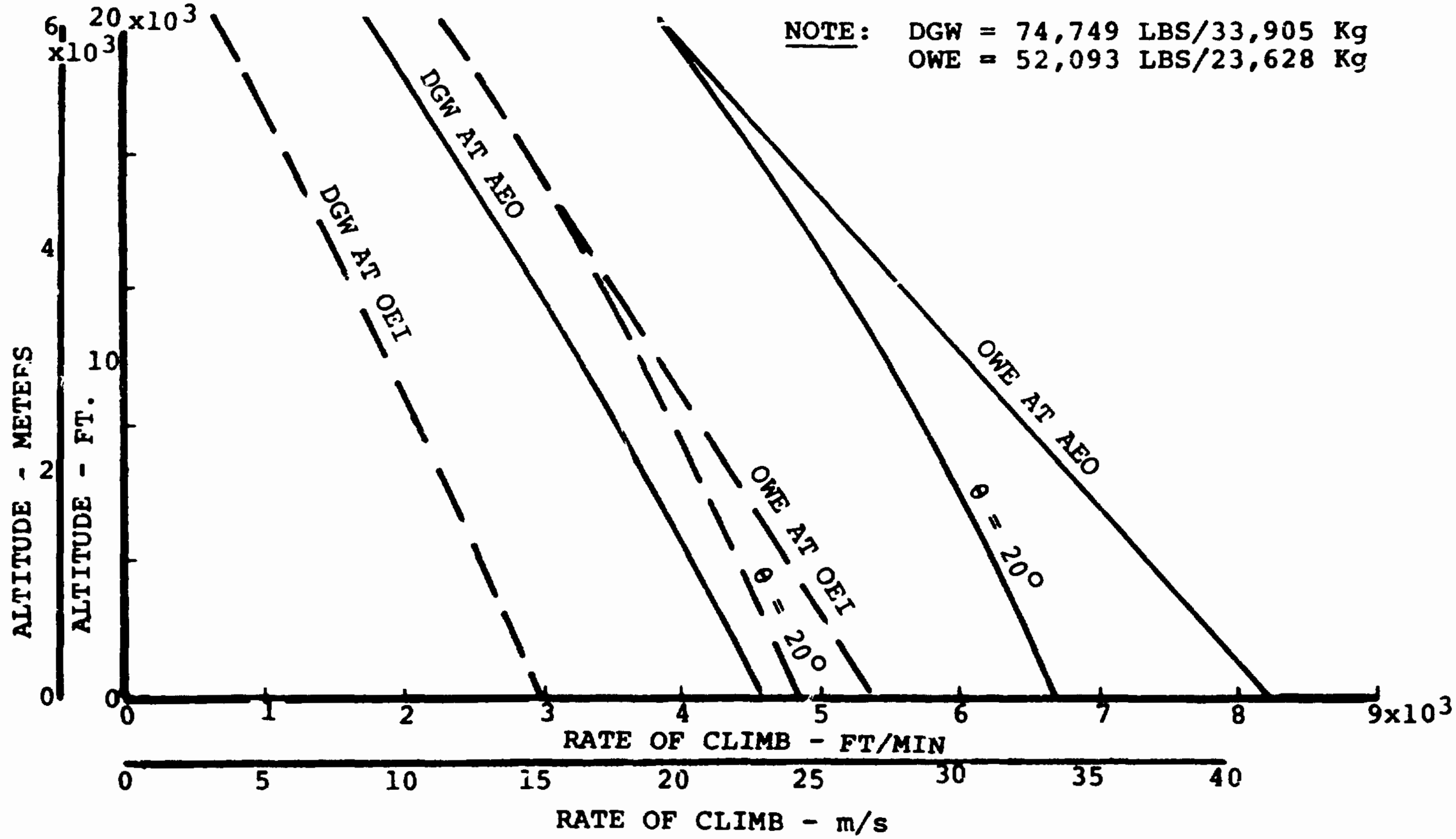


FIGURE 2.55. BASELINE TILT ROTOR DESIGN POINT AIRCRAFT - CLIMB CAPABILITY.

BASELINE AIRCRAFT PERFORMANCE

TILT ROTOR/100 PASSENGER/98.2 PNdB

DGW = 74,749 LBS/33,905 Kg
MIDWT = 63,749 LBS/28,916 Kg
OWE = 52,093 LBS/23,628 Kg
ALL ENGINES OPERATING

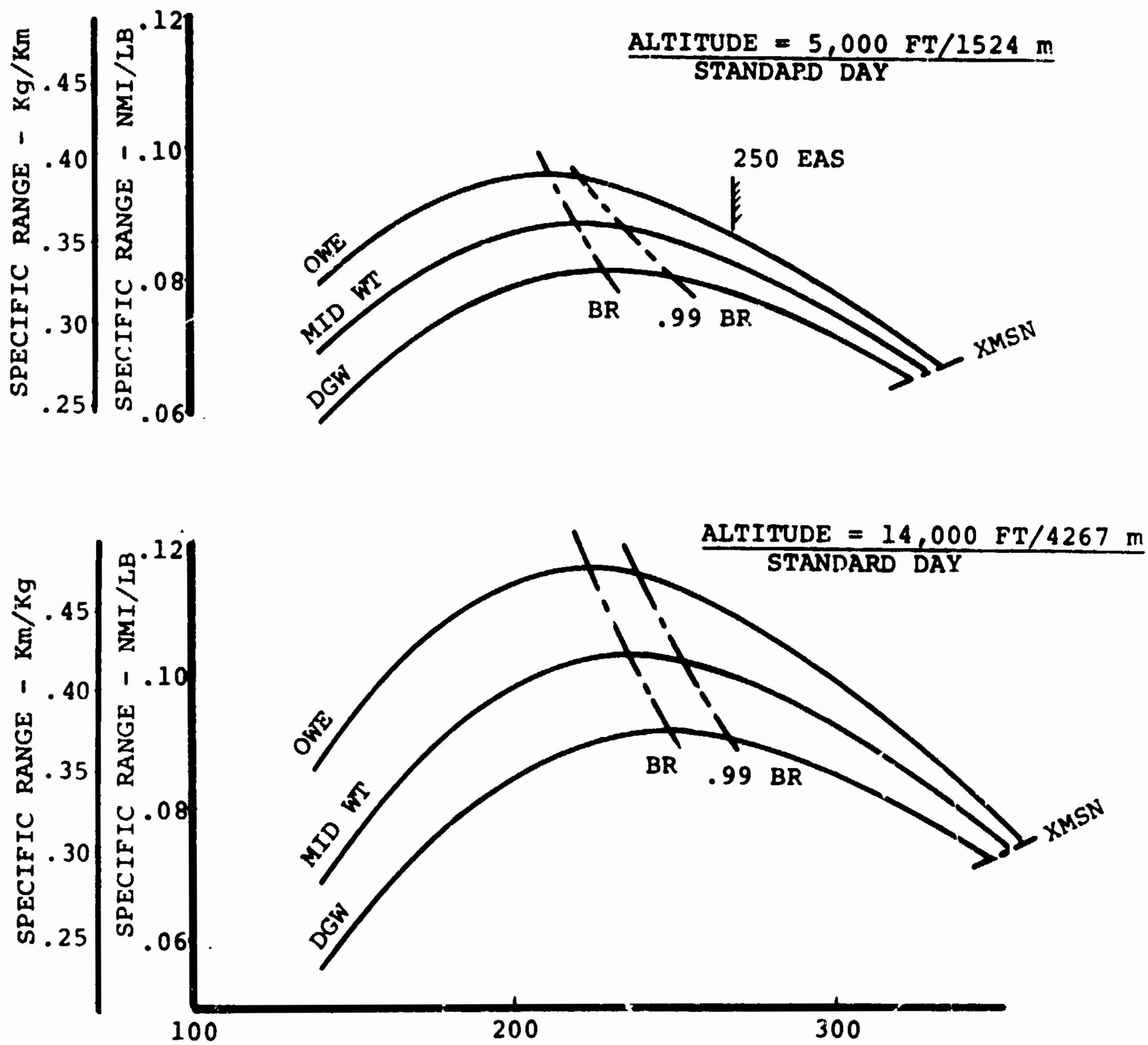


FIGURE 2.56. CRUISE PERFORMANCE - SPECIFIC RANGE - STANDARD DAY - CRUISE RPM - AEO.

BASELINE AIRCRAFT PERFORMANCE

TILT ROTOR/100 PASSENGER/98.2 PNdB

DGW = 74,749 LBS/33,905 Kg
MIDWT = 63,749 LBS/28,916 Kg
OWE = 52,093 LBS/23,628 Kg

ONE ENGINE INOPERATIVE

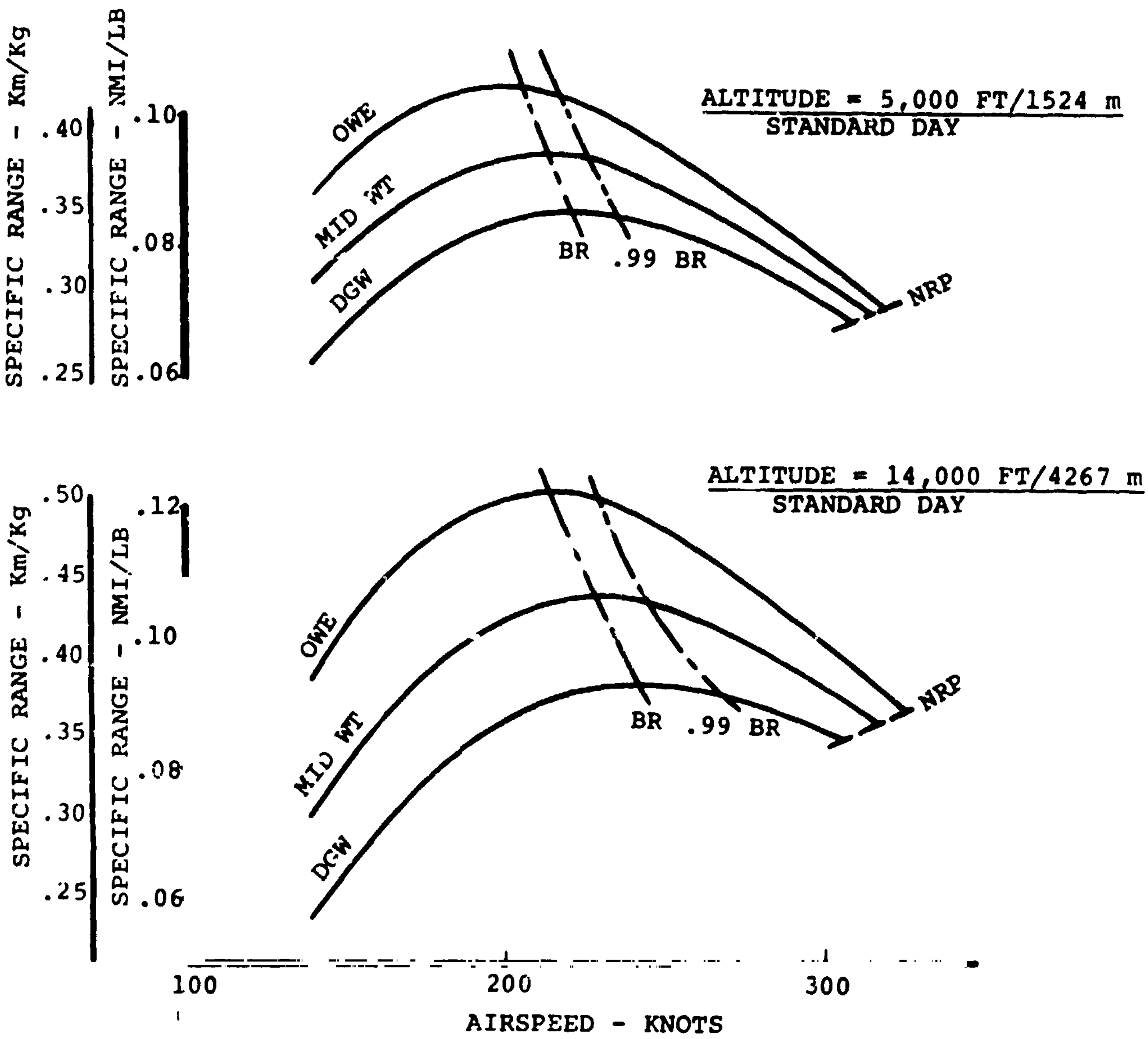


FIGURE 2.57. CRUISE PERFORMANCE - SPECIFIC RANGE - STANDARD DAY - CRUISE RPM - OEI.

pound of fuel. The best range cruise speed at this altitude and weight is 249 knots giving a specific range of 0.092 nautical miles per pound of fuel and a 99% best range speed of 268 knots.

The effect of weight is shown by comparing the three sets of data for weights between operating weight empty (OWE) and design gross weight (DGW). The maximum specific range achieved at OWE (14,000 feet) is 0.1165 nautical miles per pound of fuel at a best range speed of 224 knots and the 99% best range speed is 239 knots.

Flying at lower altitude (5,000 feet) reduces the specific range capability of the aircraft (Figure 2.56).

For example, at the transmission limit speed (DGW) of 322 knots and 5,000 feet altitude the specific range is 0.066 nautical miles per pound of fuel and the maximum specific ranges achieved are 0.0815 and 0.0965 nautical miles per pound of fuel at DGW and OWE respectively.

With one engine inoperative or one engine shut down the range performance of the aircraft improves slightly, Figure 2.57. This is due to the higher power setting required on the operating engines which provides a lower specific fuel consumption. At the normal rated power limit speed of 306 knots at 14,000 feet altitude and DGW the specific range is 0.086 and the maximum specific range achieved are 0.093 and 0.1225 nautical miles per pound of fuel at DGW and OWE respectively at 14,000 feet altitude.

The same data are provided (OEI) at 5,000 feet altitude in Figure 2.57 and again show a reduction in specific range performance compared with 14,000 feet altitude.

Range Performance

The payload range data for the design mission profile and reserves with all engines operating is shown in Figure 2.58. With a takeoff gross weight of 74,749 pounds and 100 passengers the aircraft has a design range of 200 nautical miles as shown. The design mission fuel limit defines the range at zero payload as 234 nautical miles.

The range of the aircraft can be extended by the addition of extra wing tanks. If the fuel load is increased to 7150 pounds and accounting for additional tank weight of 180 pounds the range of the aircraft becomes 400 nautical miles with payload of 85 passengers and baggage.

With design mission fuel and tanks the range performance of the aircraft (OEI) has been computed, Figure 2.59.

This data shows an increased range to 250 nautical miles with a full passenger load due to the improved specific range and SFC's which result from operating the remaining three engines at a higher fraction of available power.

Drag

The tilt rotor drag is shown in terms of evuivalent flatplate area (F_e) in Table 2.21. The method evaluates the drag of each major aircraft component and sums the components to give the vehicle F_e.

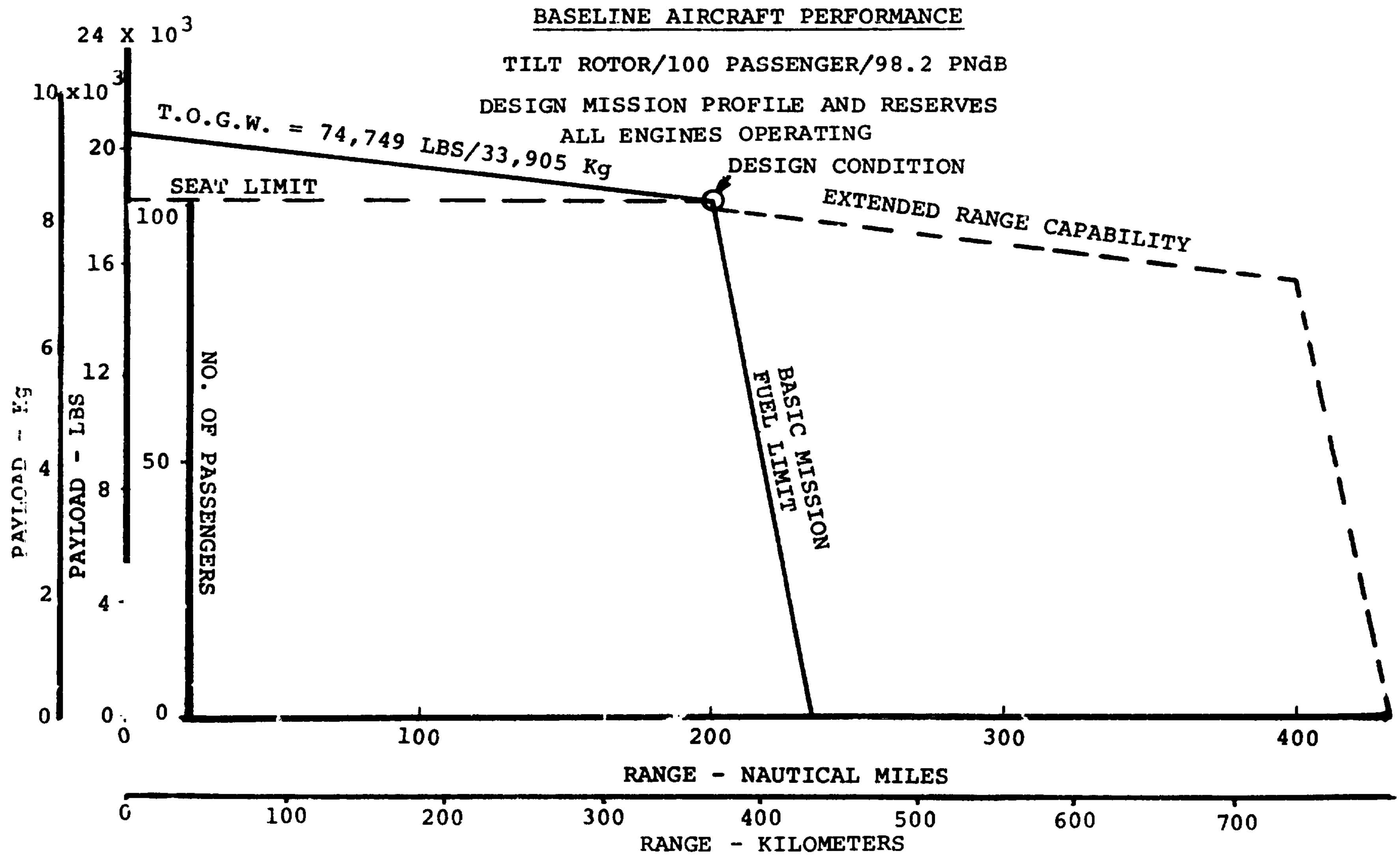


FIGURE 2.58. PAYLOAD RANGE CAPABILITY - CRUISE AT NRP AND CRUISE RPM..

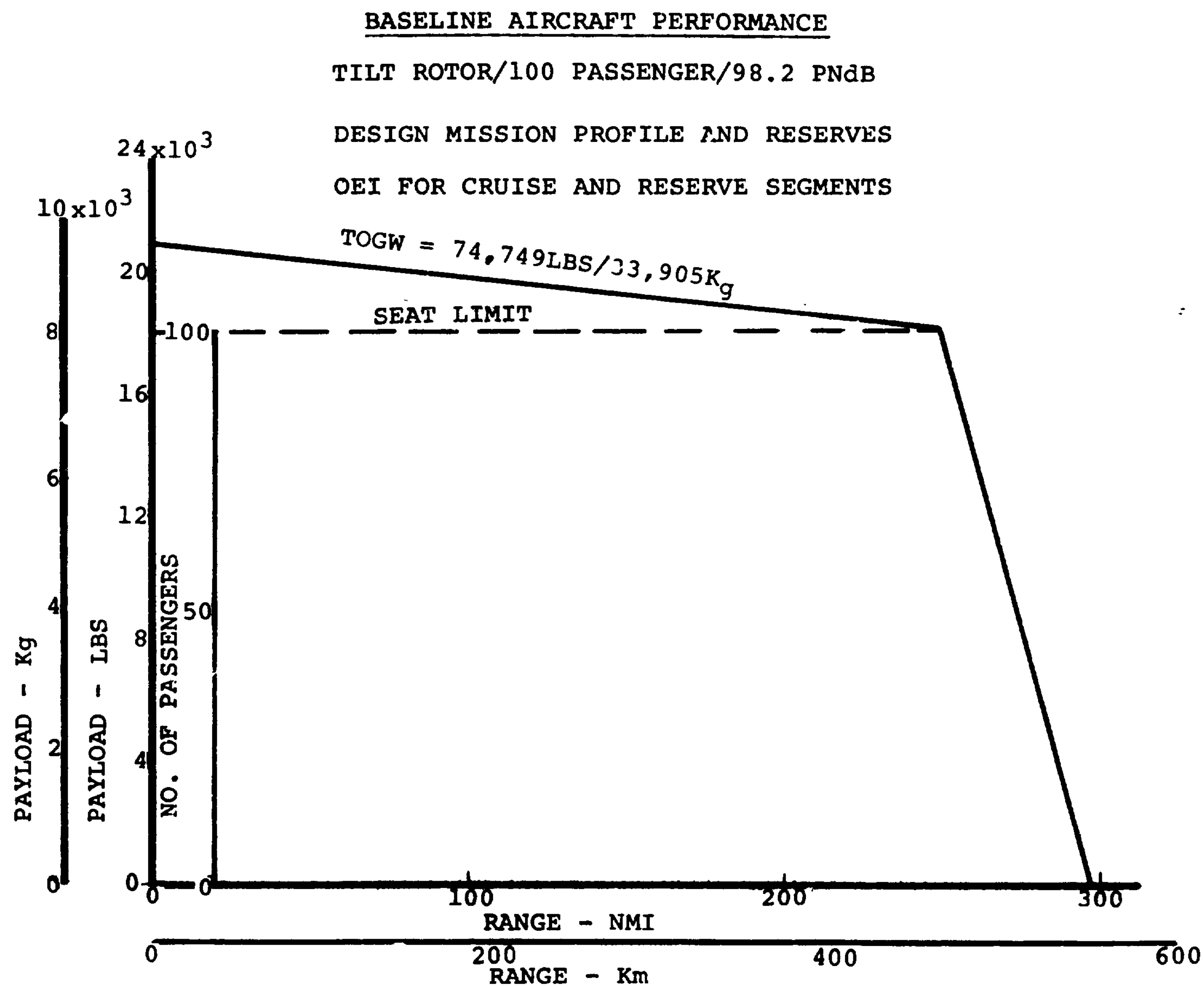


FIGURE 2.59. PAYLOAD RANGE CAPABILITY - CRUISE AT NRP & CRUISE RPM.

TILT ROTOR DRAG SUMMARY

ITEM	DRAG AREA f_e - FT^2
FUSELAGE	10.3914
WING	7.3627
VERTICAL TAIL	2.2474
HORIZONTAL TAIL	2.5998
ROTOR NACELLE	1.2946
ENGINE NACELLE	2.6573
MISCELLANEOUS	
OIL COOLER MOMENTUM LOSS	.3750
AIR CONDITIONING	.5000
TRIM	.0640
TOTAL DRAG AREA	27.4922

TABLE 2.20. TILT ROTOR BASELINE AIRCRAFT DRAG SUMMARY.

The design point tilt rotor has an equivalent drag area of 27.492 feet^2 or a gross weight/F_e ratio of 12,885 Kg/m^2 (2,750 pounds per feet^2).

Prop/Rotor Performance

The definition of the aerodynamic design of a prop/rotor for a tilt rotor aircraft is a compromise between the requirements for good hover and cruise performance. Design trade studies have been performed to optimize the rotor design parameters and are reported in Volume II.

The static and cruise performance of the selected design is shown in Figures 2.60 and 2.61. In hover a maximum figure of merit of 77% is achieved at a rotor thrust coefficient of 0.010. For 1g hover the rotor design thrust coefficient is 0.0106. The cruise performance is shown as a rotor map in Figure 2.61 giving C_T and C_P for lines of constant advance ratio.

2.2.4 Design Point Tilt Rotor - Flying Qualities

Hover

The hover trim data at design gross weight is shown in Figure 2.62. Data are given for three CG locations from 45% MAC in hover which is equivalent to 42% MAC in cruise to 25% MAC which is equivalent to 13.8% MAC in cruise. The CG shift between hover and cruise is due to nacelle tilt.

For nacelles at 90 degrees (hover) the aircraft fuselage attitude is 0.6 degrees at the aft CG and requires 0.9 degrees cyclic to trim. At the forward CG the fuselage

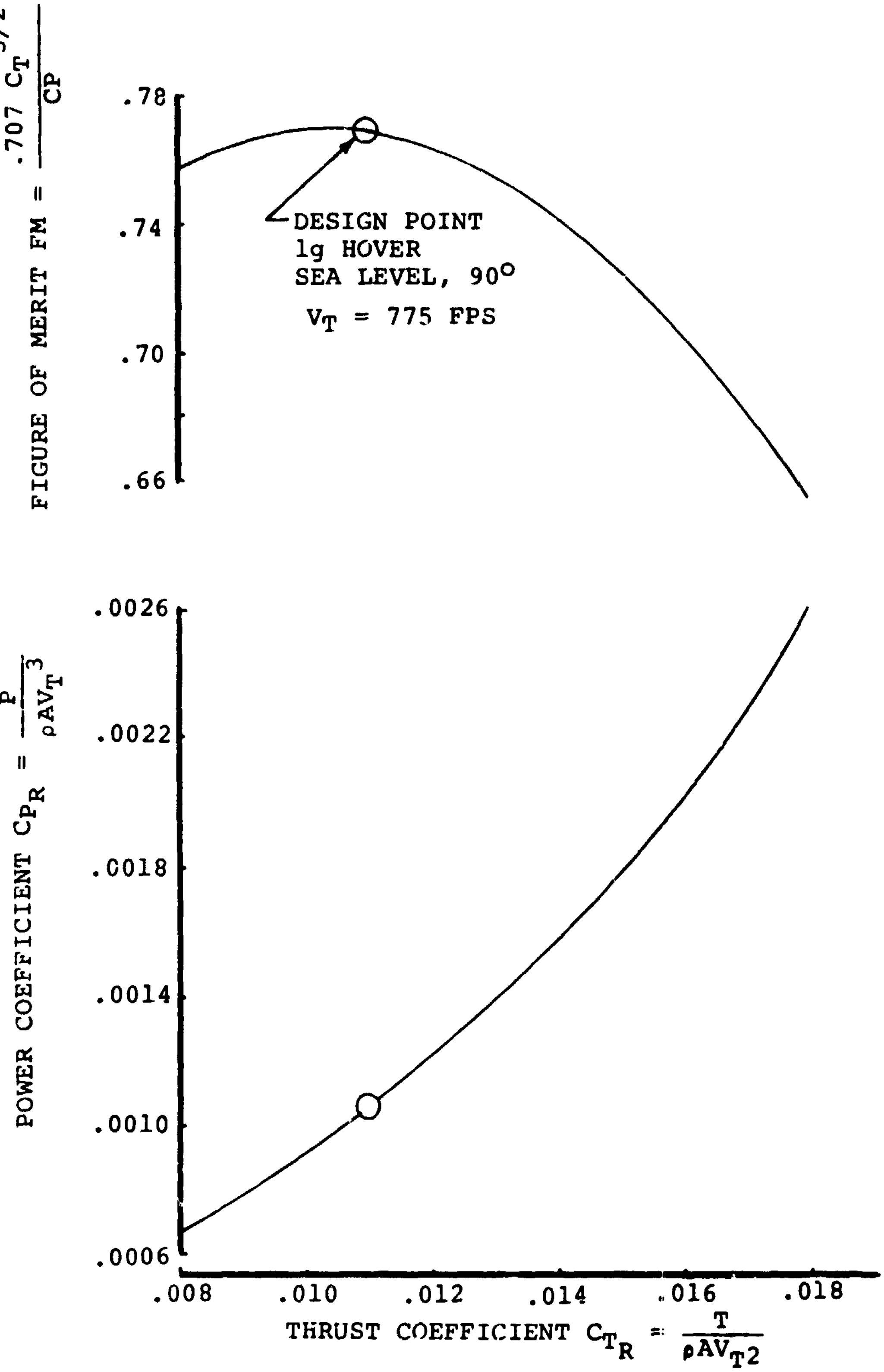


FIGURE 2.60. DESIGN POINT TILT ROTOR STATIC ROTOR PERFORMANCE.

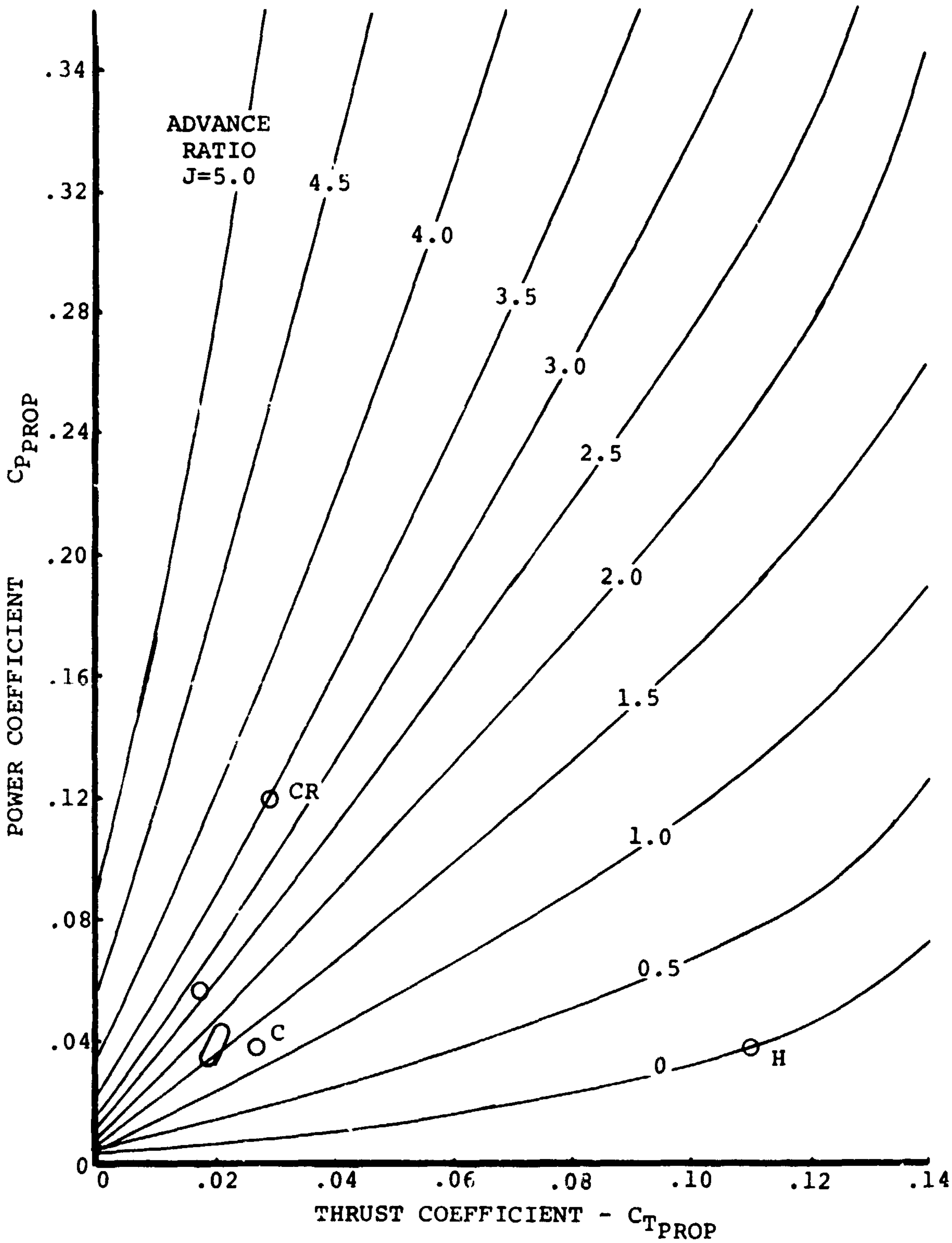


FIGURE 2.61. ROTOR CRUISE PERFORMANCE - BASELINE TILT ROTOR.

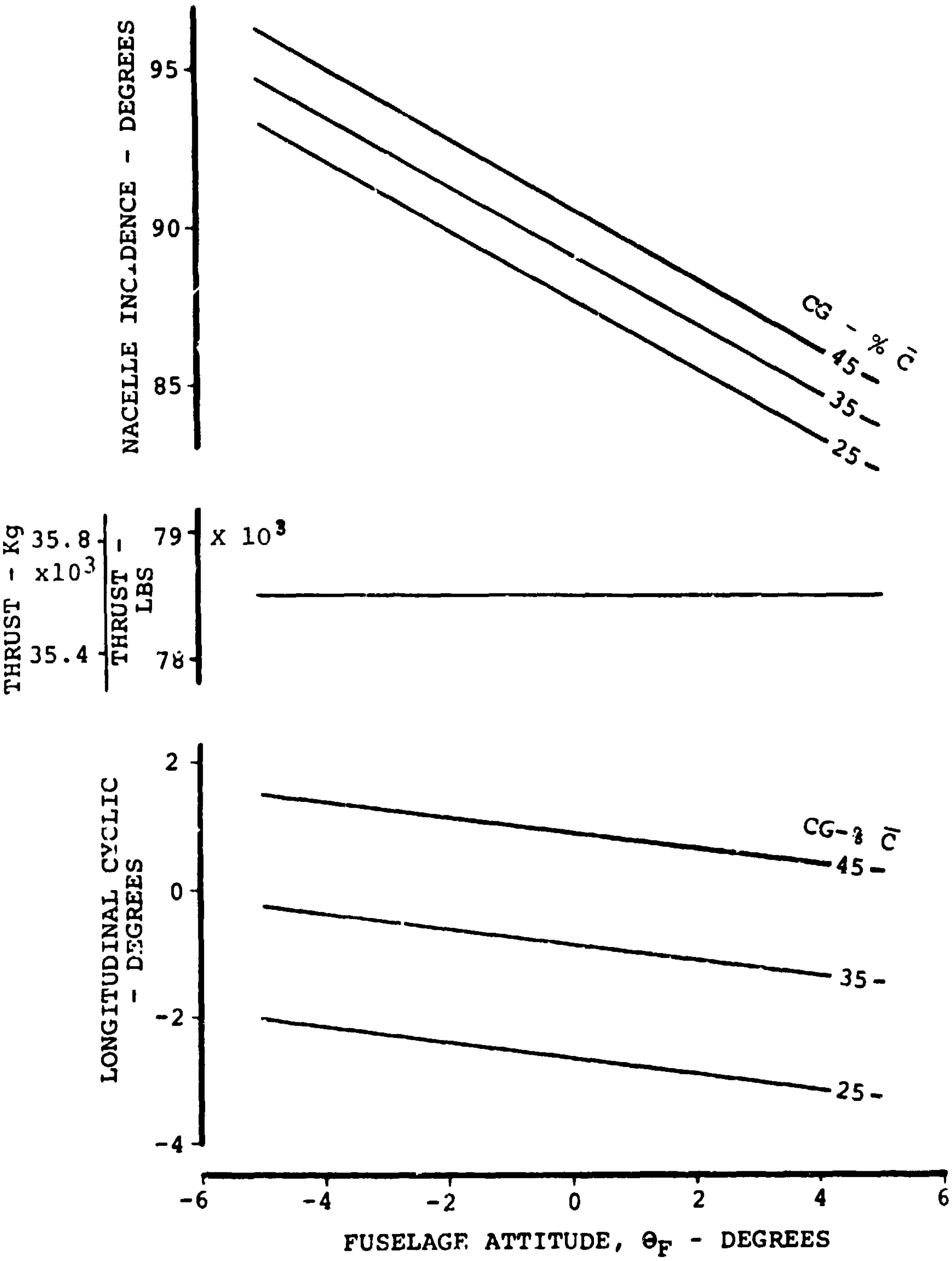


FIGURE 2.62. DESIGN POINT TILT ROTOR HOVER TRIM REQUIREMENTS. (GW = 74,749 LBS)

trims 1.9-degrees nose down and requires 2.4-degrees cyclic to trim.

Hover trim is possible with nacelle angles greater and less than 90-degrees, however, the fuselage attitude excursions increase. The cyclic required for hover trim is relatively insensitive to fuselage attitude, but is strongly dependent on CG location since this defines the moment arm for the weight.

Control Power In Hover

The aircraft control power in hover is shown in Figures 2.63, 2.64 and 2.65. Pitch control is obtained by the application of cyclic pitch. The resulting hub moment and in-plane force times the distance from the hub to the CG gives a pitch moment which is used for pitch trim and control. The sensitivity of pitch acceleration to cyclic pitch is 0.107 radians per second squared per degree of cyclic at design gross weight of 74,749 pounds. At lower weights, for example 60,000 pounds, the pitch control sensitivity decreases slightly to .1025 radians per second squared per degree. This effect is due to the reduced thrust level which decreases the in-plane force component of the aircraft pitch moment. Six degrees of cyclic are available for full pitch control.

At design gross weight and a mid range CG the cyclic required to trim in pitch is -0.9 degrees. The remaining cyclic would allow a control power of 0.58 radians per second squared. With a most forward CG and corresponding 60,000 pound weight the

BASELINE AIRCRAFT

TILT ROTOR/100 PASSENGER/98.2 PNdB

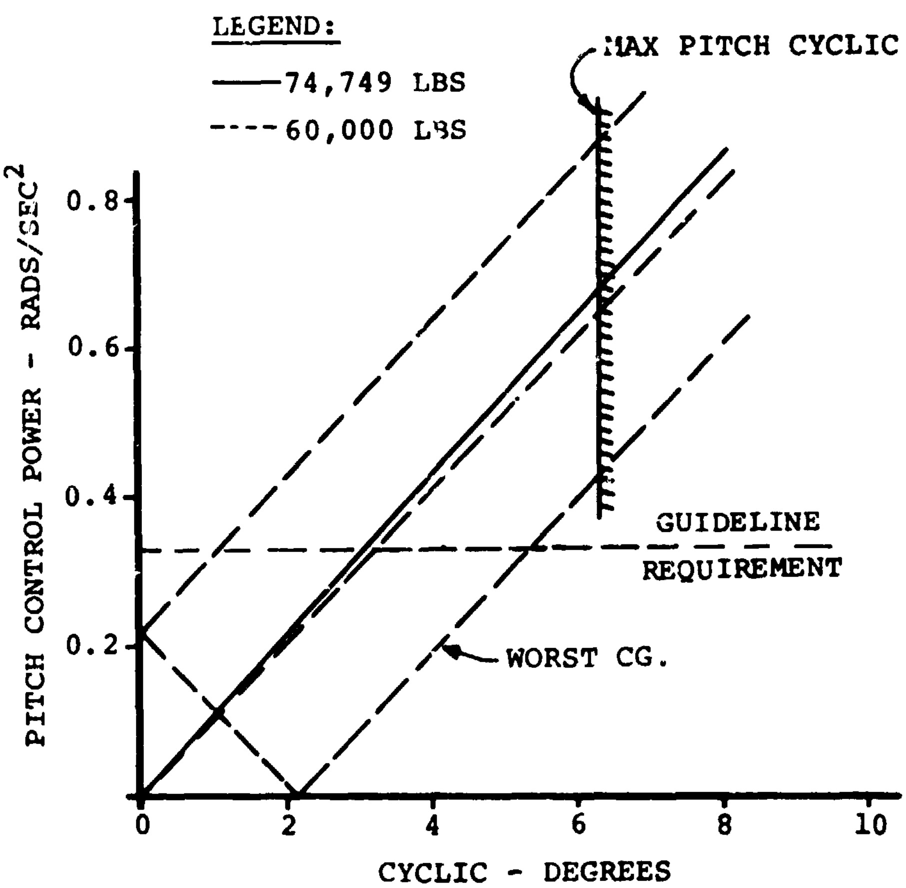


FIGURE 2.63. HOVER PITCH CONTROL POWER - DESIGN POINT TILT ROTOR.

BASELINE AIRCRAFT

TILT ROTOR/100 PASSENGER/98.2 PNdB

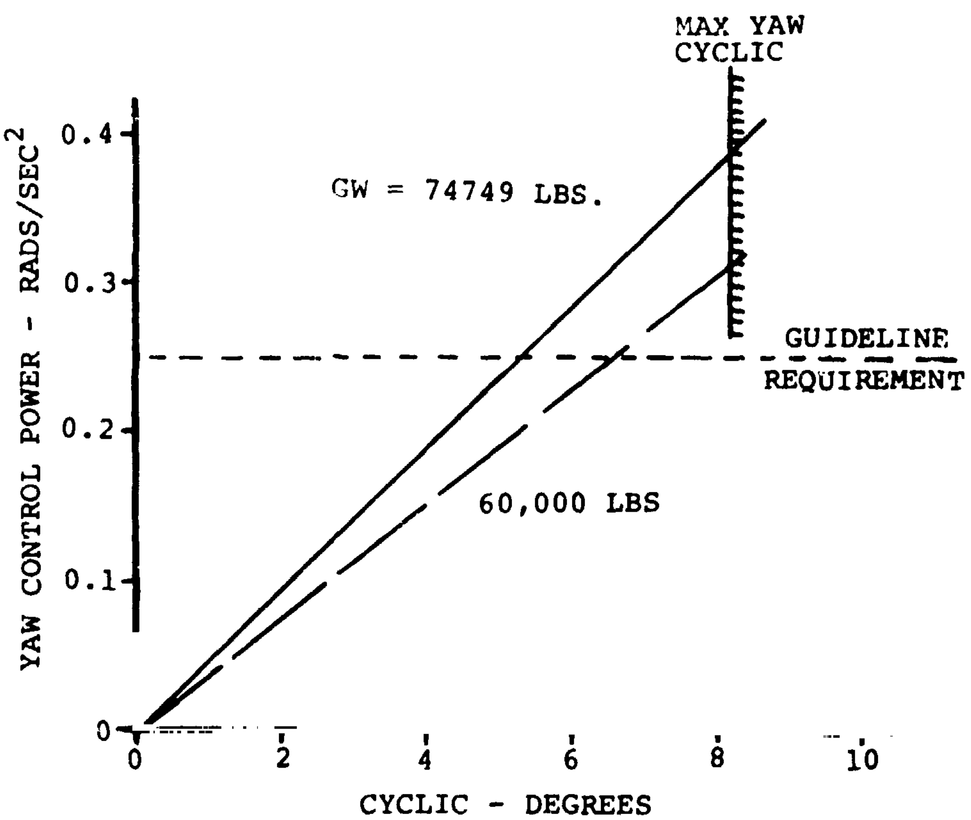


FIGURE 2.64. DESIGN POINT TILT ROTOR - HOVER YAW CONTROL.

BASELINE AIRCRAFT

TILT ROTOR/100 PASSENGER/98.2 PNdB

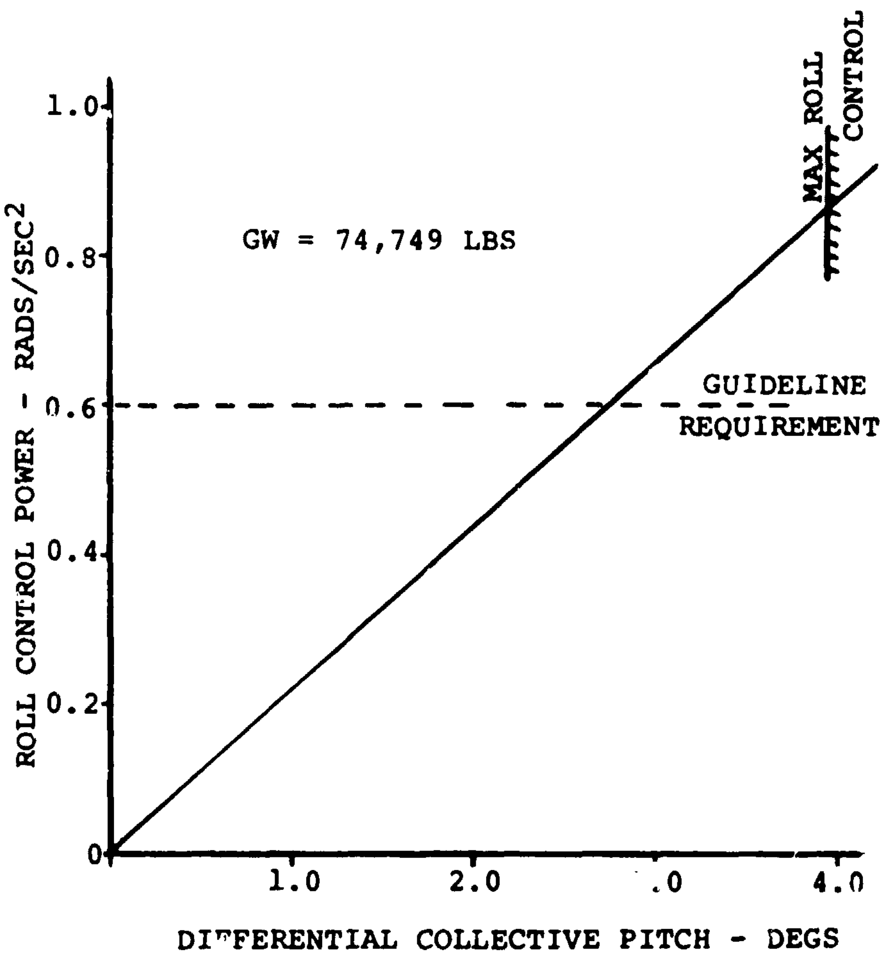


FIGURE 2.65. DESIGN POINT TILT ROTOR - HOVER ROLL CONTROL.

cyclic required for pitch trim is 2.18 degrees. At this condition the minimum control power available in pitch is 0.43 radians per second squared.

Yaw acceleration is achieved by differential cyclic pitch. Cyclic pitch input is phased to obtain the maximum in-plane force per degree of cyclic. The in-plane forces provide a couple to yaw the aircraft. Since the in-plane force produced by the application of cyclic pitch results largely from thrust vector tilt, the light weight case (i.e., reduced thrust) is the most critical yaw condition.

At design gross weight with -.9 degrees pitch to trim the maximum yaw control of 8.15 degrees can be applied which gives an initial acceleration of 0.388 radians per second squared. At 60,000 pounds gross weight with 2.18 degrees cyclic to trim the available yaw cyclic is 7.2 degrees which provides 0.278 radians per second squared acceleration.

The combined control criteria of .25 radians per second squared yaw control at 60,000 pounds gross weight requires 6.47 degrees cyclic and accounting for 2.18 degrees pitch cyclic to trim the remaining cyclic for pitch acceleration is 0.97 degrees which provides 0.1 radians per second squared or 30% of 0.33 radians per second squared as specified in the guidelines.

Roll control is obtained by the use of differential cyclic pitch. The sensitivity of control power to differential collective is 0.22 radians per second squared per degree.

In hover 3.95 degrees of differential collective are available and at design gross weight full control produces 0.87 radians per second squared all engines operating.

Control Response In Hover

The control response to control input in hover is given in Figure 2.66. The control response data are computed for a 60,000 pound weight with 2.18 degrees cyclic to trim. With a yaw cyclic input of 6.42 degrees (i.e., an initial acceleration of 0.25 radians per second sqaured) the aircraft achieves a yaw angle of 8.5 degrees in one second. In roll, 15.5 degrees of roll can be achieved with full roll control. With 6.47 degrees yaw cyclic control applied and 2.18 degrees pitch cyclic to trim the pitch cyclic available for control is 0.97 degrees. This cyclic provides an initial acceleration of 0.10 radians per second squared which meets the 0.099 radians per second squared required (i.e., 30% of .33 radians per second squared). The pitch angle achieved in this instance of combined control is 2.22 degrees in one second (guideline requirement 30% of 6 degrees = 1.8 degrees in one second).

With 5.4 degrees cyclic pitch control, the pitch response in one second is 7.4 degrees which again exceed the 6 degrees guideline requirement.

Hover longitudinal dynamic stability is shown with no stability augmentation in Figure 2.67. There are two roots in evidence, an aperiodic root which is critically damped with a time to half amplitude of less than one second and aperiodic root with an undamped natural frequency of 0.195 radians per second.

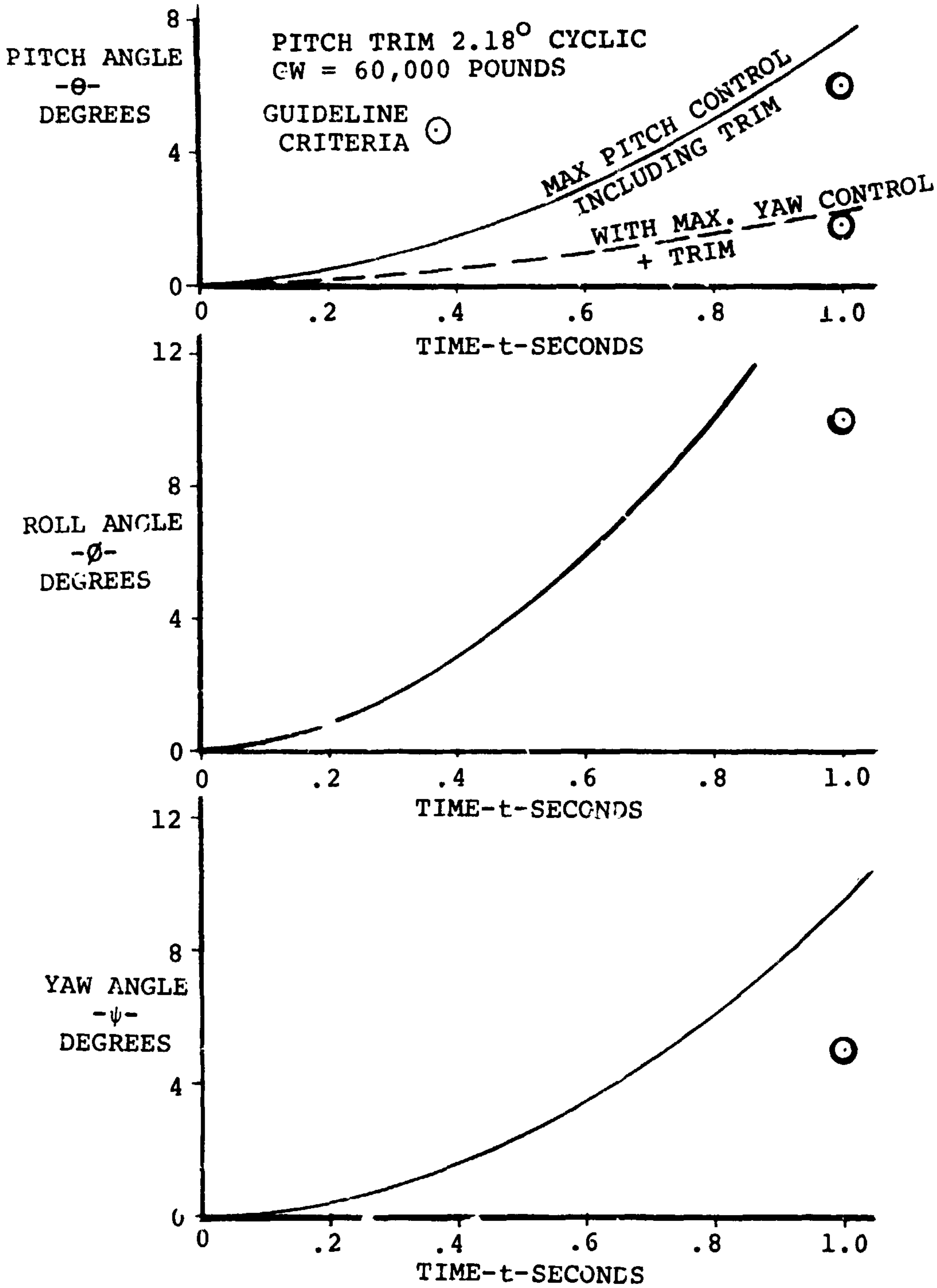


FIGURE 2.66. 1985 TILT ROTOR COMMERCIAL TRANSPORT - CONTROL RESPONSE IN HOVER.

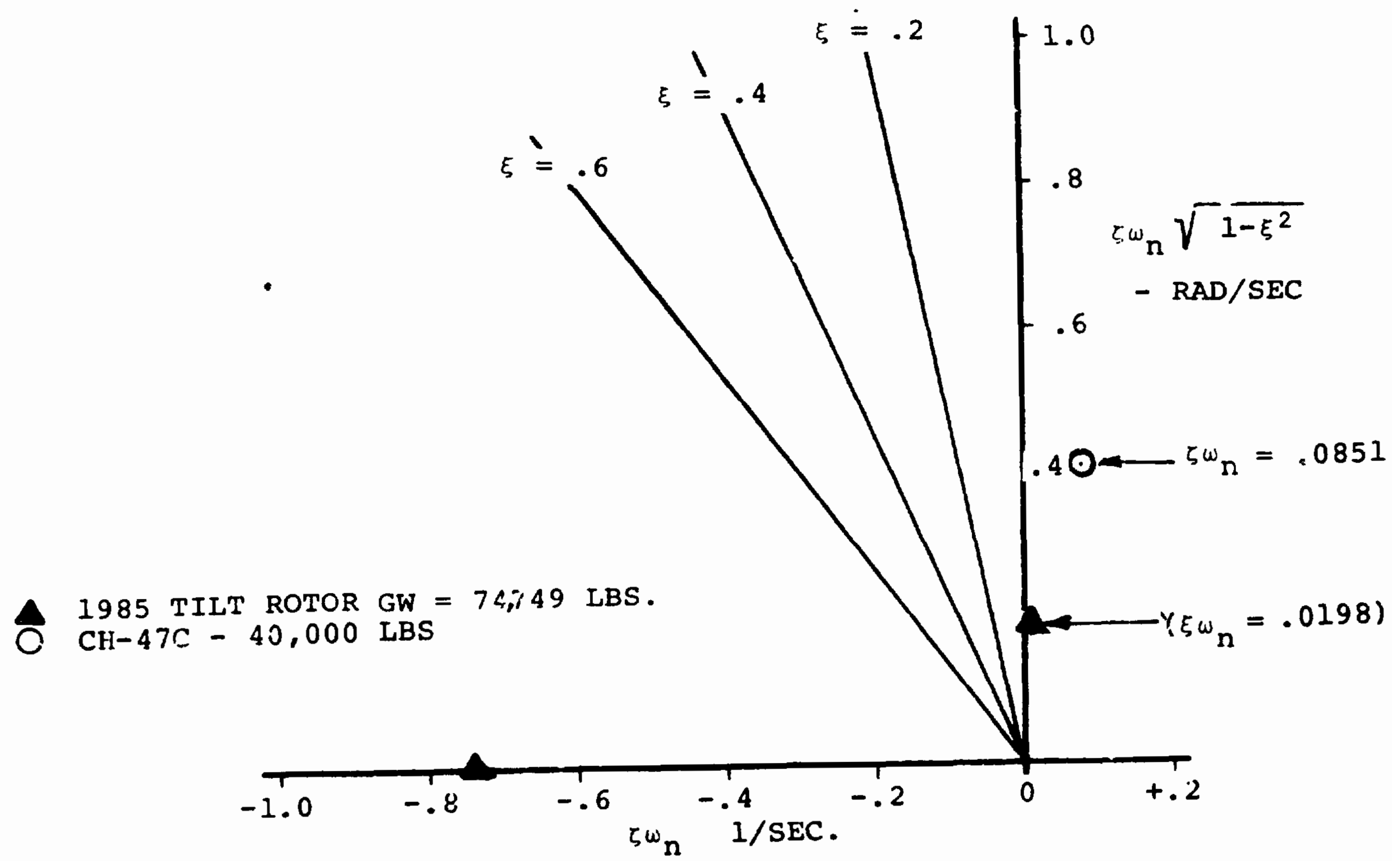


FIGURE 2.67. HOVER DYNAMIC STABILITY.

The periodic root is mildly unstable with a time to double amplitude of much greater than 12 seconds. The aircraft longitudinal response to a step input is shown in Figure 2.68.

With no SAS the aircraft meets Level 2 requirements. The inclusion of a low gain attitude stabilization will provide damping in the long period (33 seconds) oscillatory pitch mode to meet Level 1 requirements. The response in this mode with an attitude stabilization gain of 0.02 degrees cyclic per degree attitude is also shown in Figure 2.68.

Figures 2.69 and 2.70 shows the aircraft trim in hover with a 25 Knot wind for equivalent sideslip angles from zero (head wind) to 90 degrees (side wind).

With a 25 Knot side wind the aircraft trims with 2 degrees roll angle at design gross weight, aft CG and 2.5 degrees roll at 60,000 pounds forward CG. Differential collective to trim is 0.4 degrees for the aft CG design gross weight case and 0.27 degrees for 60,000 pounds with a forward CG.

At design gross weight and aft CG the cyclic required to trim is a maximum of 1.9 degrees (right rotor) and at 60,000 pounds 1.9 degrees (right rotor) and at 60,000 pounds forward CG 3.5 degrees cyclic are required on the left rotor. In these calculations the side wind is assumed from the starboard side of the aircraft.

With a 25 Knot side wind the lateral control deflections to trim are 0.7 inch rudder pedal out of a total available of

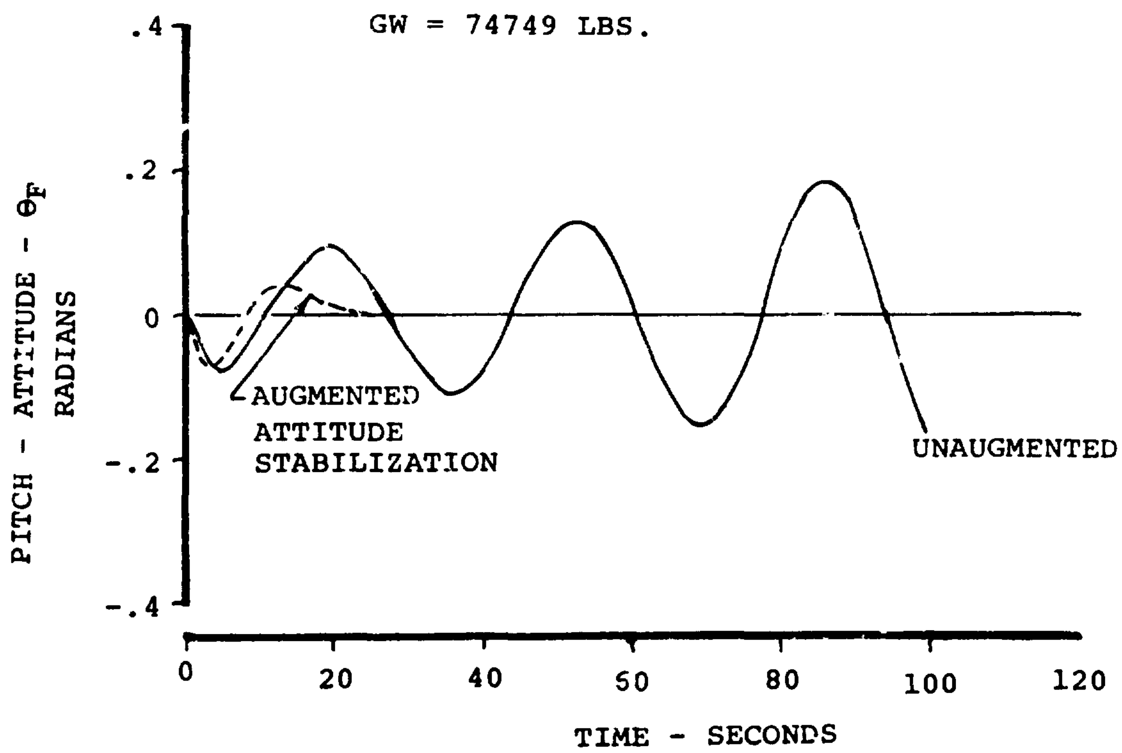


FIGURE 2.68. 1985 TILT ROTOR COMMERCIAL TRANSPORT - LONGITUDINAL DYNAMIC RESPONSE IN HOVER.

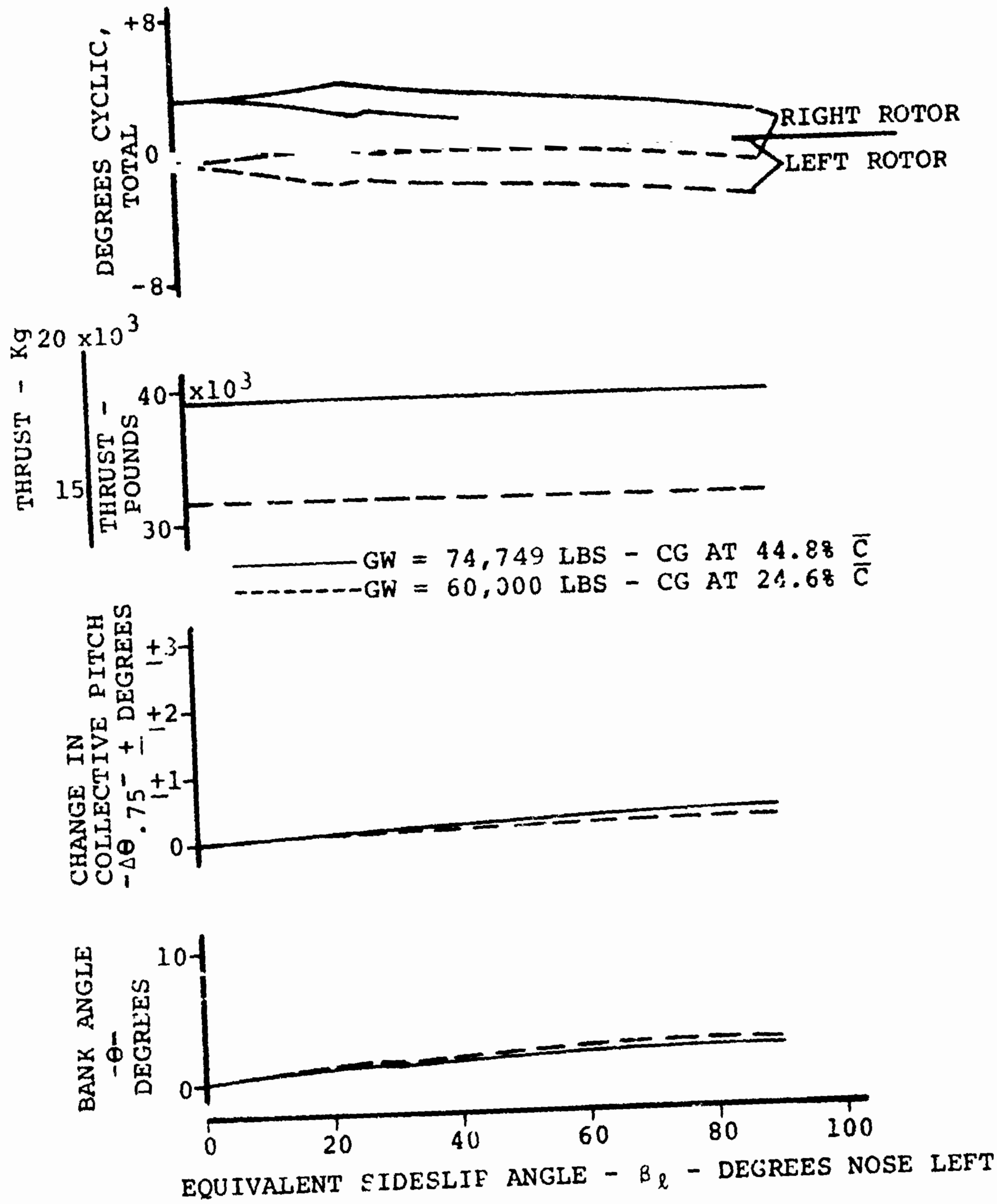


FIGURE 2.69 . DESIGN POINT TILT ROTOR TRIM IN 25 KNOT WIND DURING HOVER.

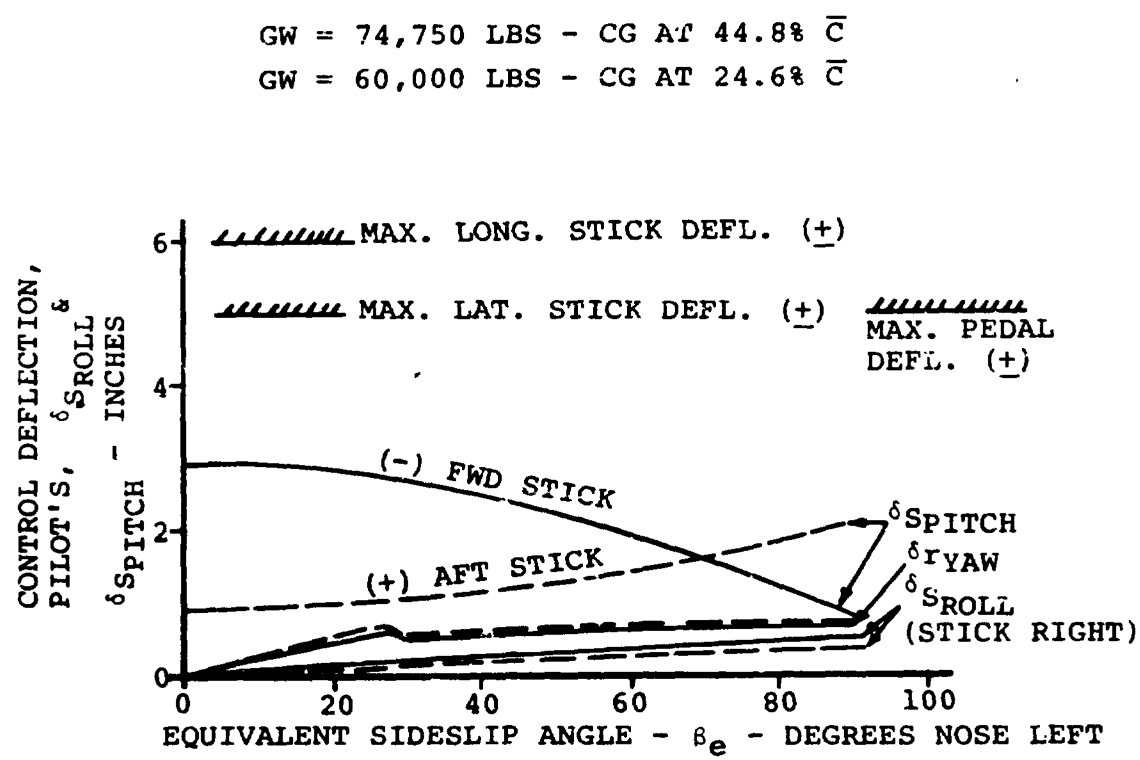


FIGURE 2.70. 1985 TILT ROTOR COMMERCIAL TRANSPORT - TRIM IN 25 KNOT WIND DURING HOVER.

2.5 inches and 0.5 inch lateral stick out of a total available 5 inches. The control authority available after trim is more than sufficient to handle a 15 feet per second gust.

In the worst longitudinal gust case while hovering in a 25 knot side wind the trim longitudinal stick is 2.1 inches aft out of a total of 6 inches available. A 15 feet per second gust would require an additional 0.93 inches of stick travel, well within the 6 inches available.

Transition

The transition corridor for a tilt rotor aircraft is bounded at high speed by blade fatigue loads and power limits and at low speed by stall. The calculation of aircraft trim and control for all of the possible trim conditions requires the detail design of the control system and is considered beyond the scope of this conceptual study. However, a typical transition schedule has been computed in order to estimate the control powers available.

The 1g trim data is shown in Figures 2.71 and 2.72 for the design gross weight aircraft with an aft CG location of 34% MAC. These trim data assume a flap schedule with nacelle incidence as shown in Figure 2.73.

The longitudinal cyclic for full stick travel is shown in Figure 2.74 as a function of nacelle incidence. The cyclic pitch per inch of stick is reduced as nacelle incidence is reduced and terminates in cruise with the cyclic authority

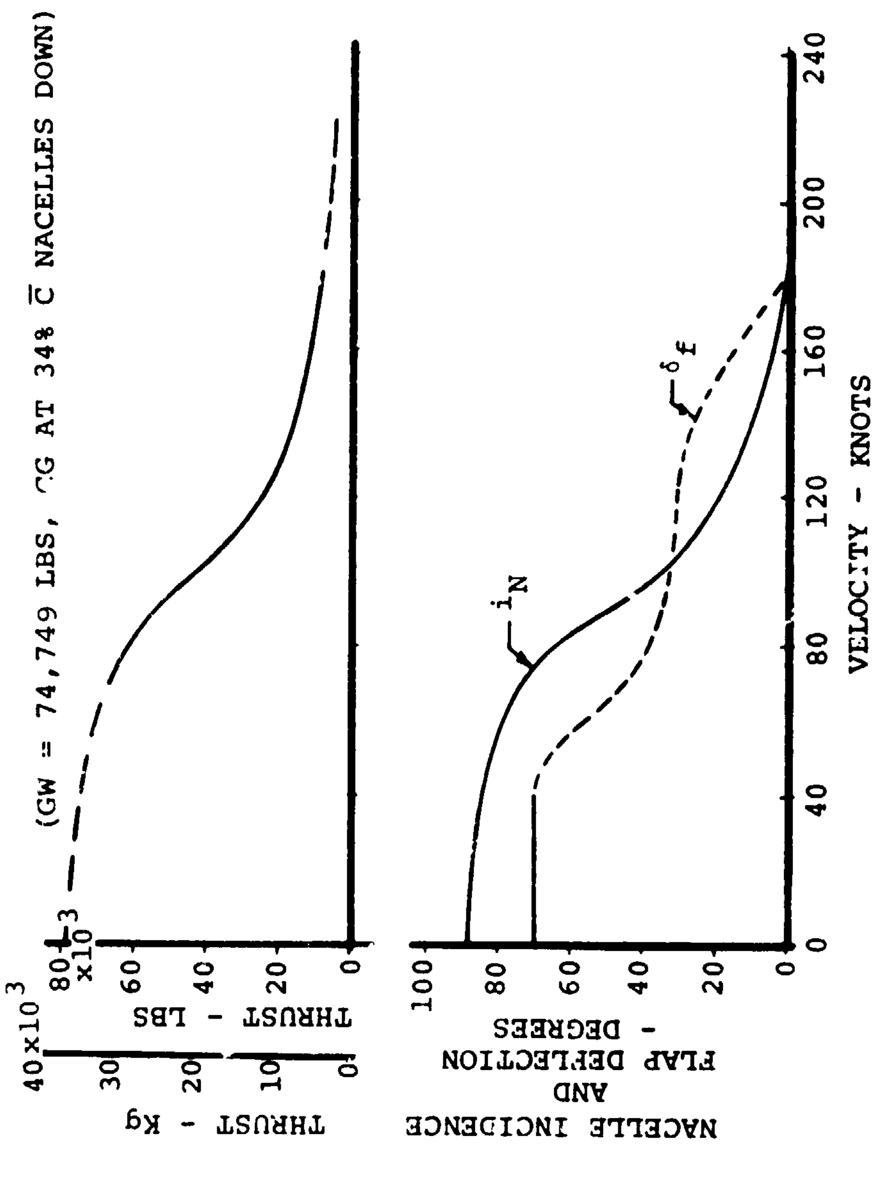


FIGURE 2.71 . DESIGN POINT TILT ROTOR TYPICAL TRANSITION TRIM CHARACTERISTICS.

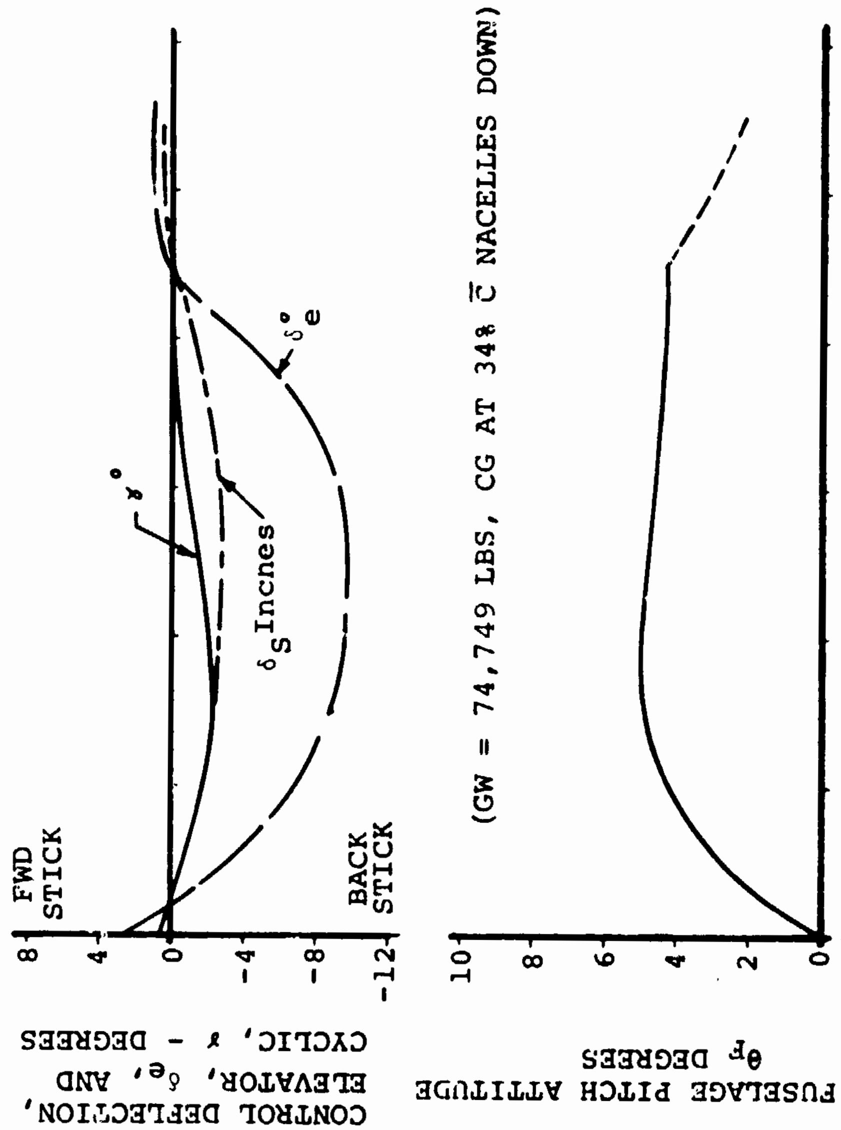


FIGURE 2.72. DESIGN POINT TILT ROTOR TYPICAL TRANSITION TRIM CHARACTERISTICS.

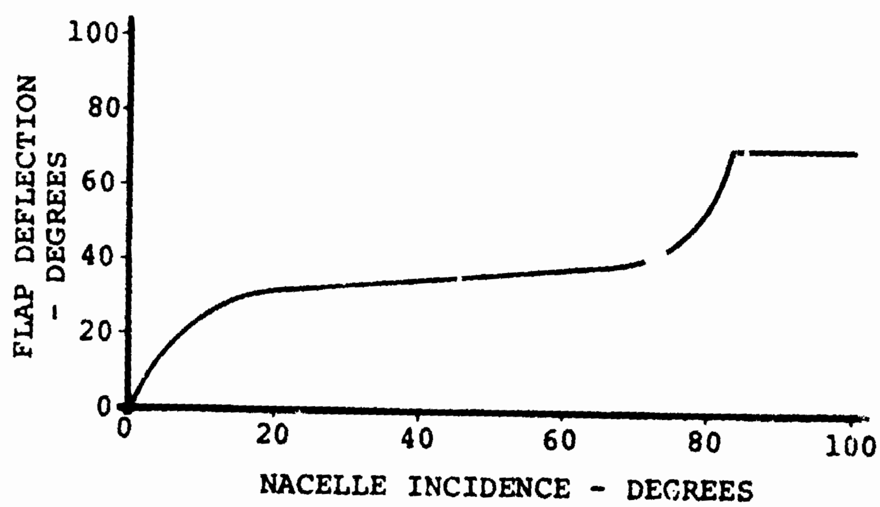


FIGURE 2.73. TRANSITION FLAP SCHEDULE.

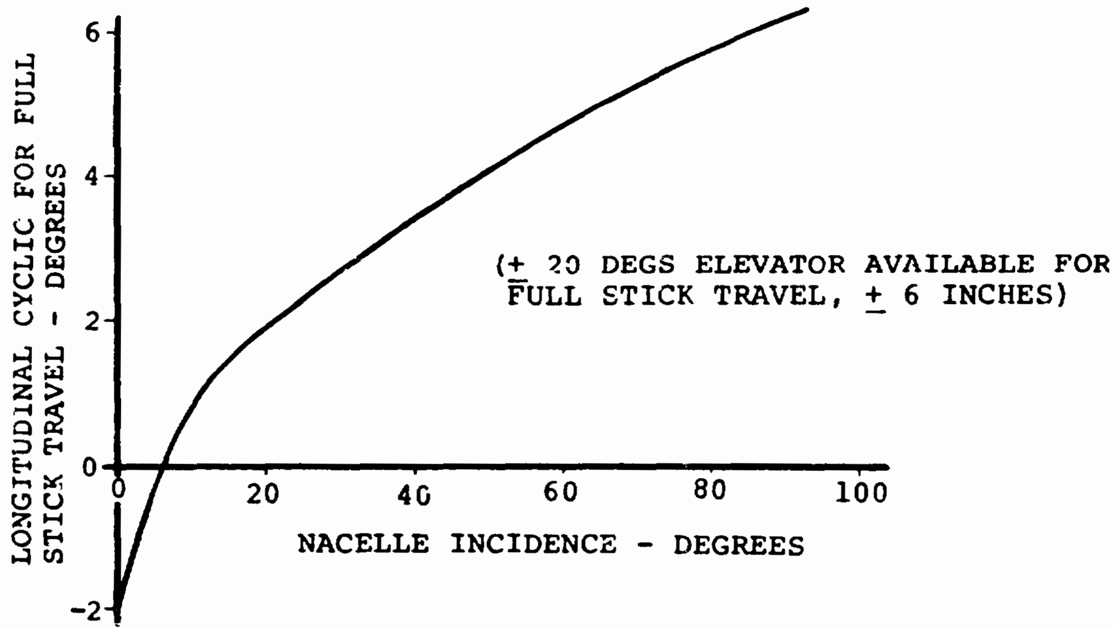


FIGURE 2.74. LONGITUDINAL CYCLIC SCHEDULE.

required for rotor loads control in cruise flight. The estimated pitch control power in transition is shown in Figure 2.75 for zero stick trim and for trim with most aft CG. The minimum pitch control is 0.4 radians per second squared at 57 Knots which exceeds the guideline design requirements of 0.3 radians per second squared.

Differential cyclic and collective are used for yaw control in transition as shown in Figure 2.76. As the rudder effectiveness grows with air speed these controls are phased out.

The yaw control power capability is shown for two gross weights. At the light weight the yaw control power is 0.265 radians per second squared at 40 Knots compared with a requirement of 0.25 radians per second squared. The minimum control power is 0.22 radians per second squared at 82 Knots compared with a requirement of 0.2 radians per second squared.

Differential collective and cyclic a[illegible] used for roll control in transition and are phased [illegible] as [illegible]eron and spoiler capability grows with airspeed. [illegible] 2.77. At 40 Knots a roll control power of 0.68 radi[illegible]r second squared is available (requirement of 0.6) and the minimum control power is 0.465 radians per second squared at 140 Knots which again exceeds the 0.4 radians per second squared guideline requirement as shown in Figure 2.78.

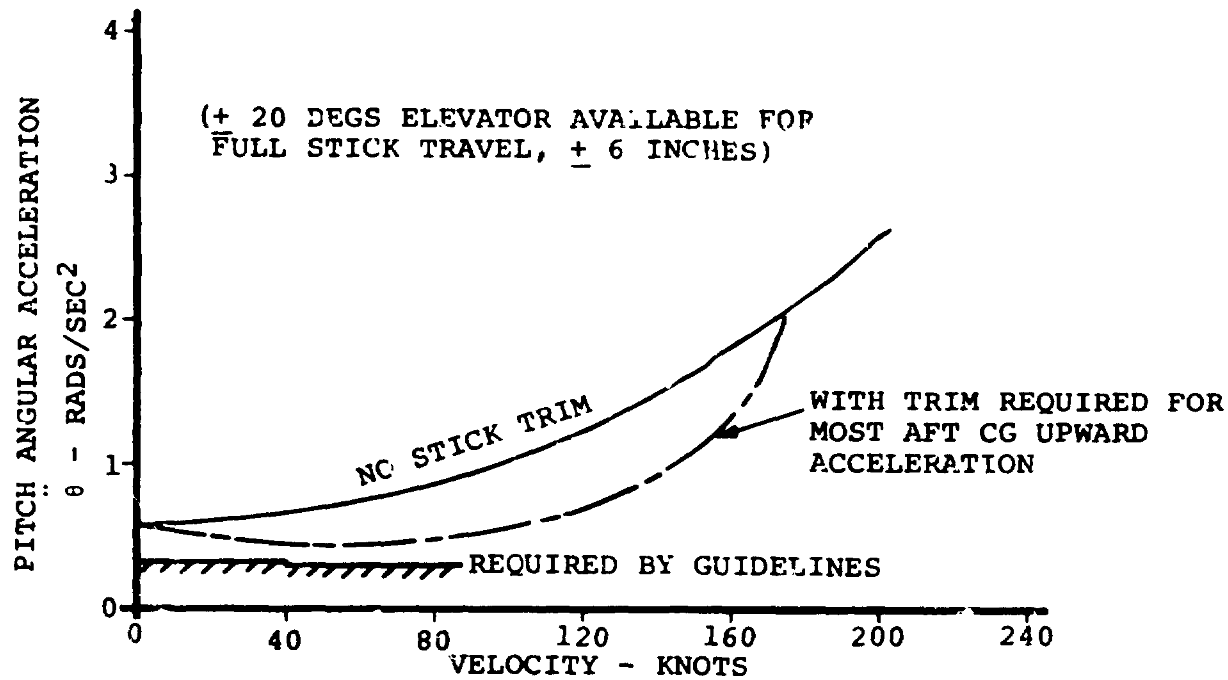


FIGURE 2.75. PITCH ANGULAR ACCELERATION CAPABILITY IN TRANSITION.

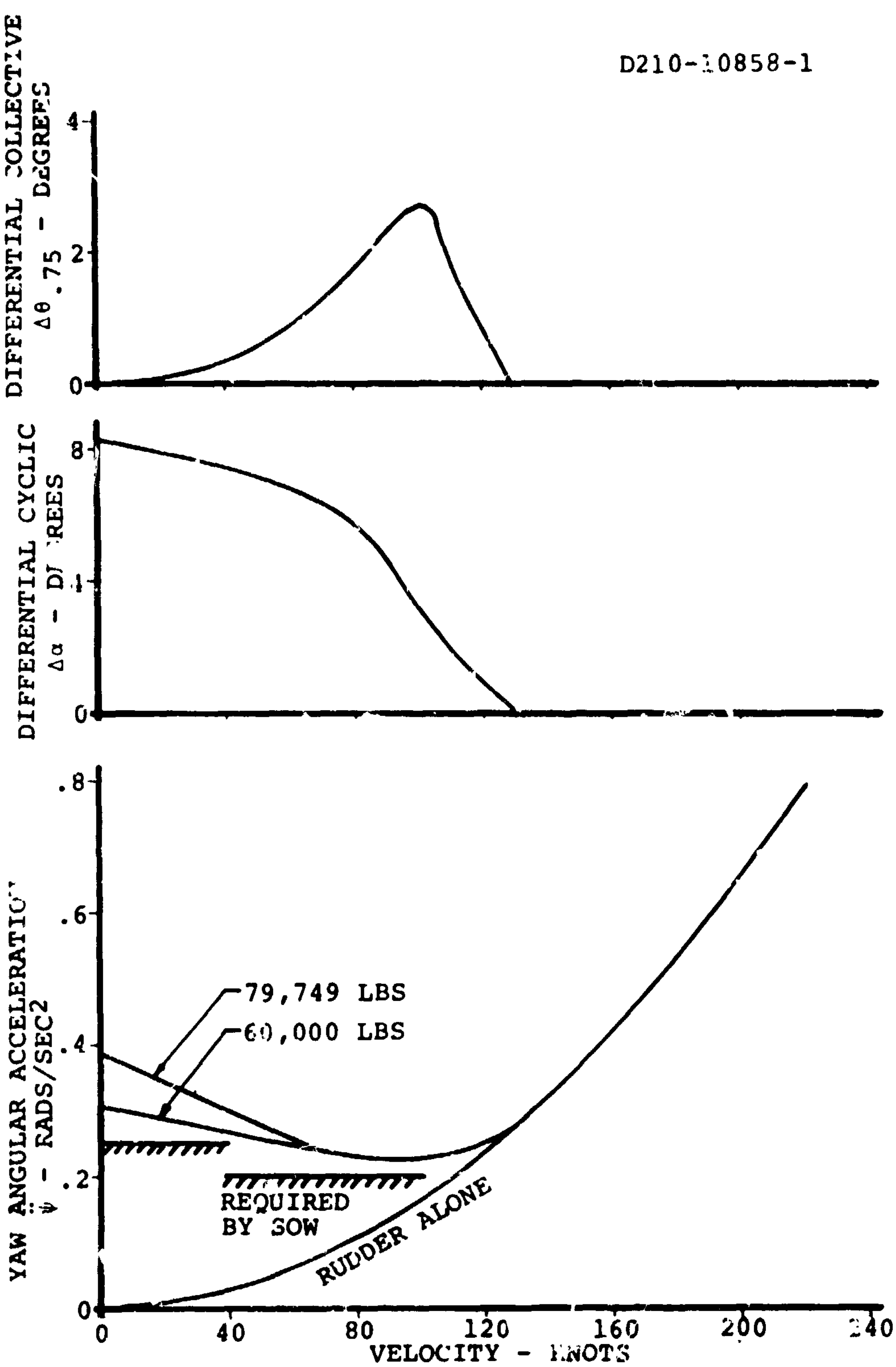


FIGURE 2.76. YAW ANGULAR ACCELERATION CAPABILITY AND CONTROL REQUIRED IN TRANSITION.

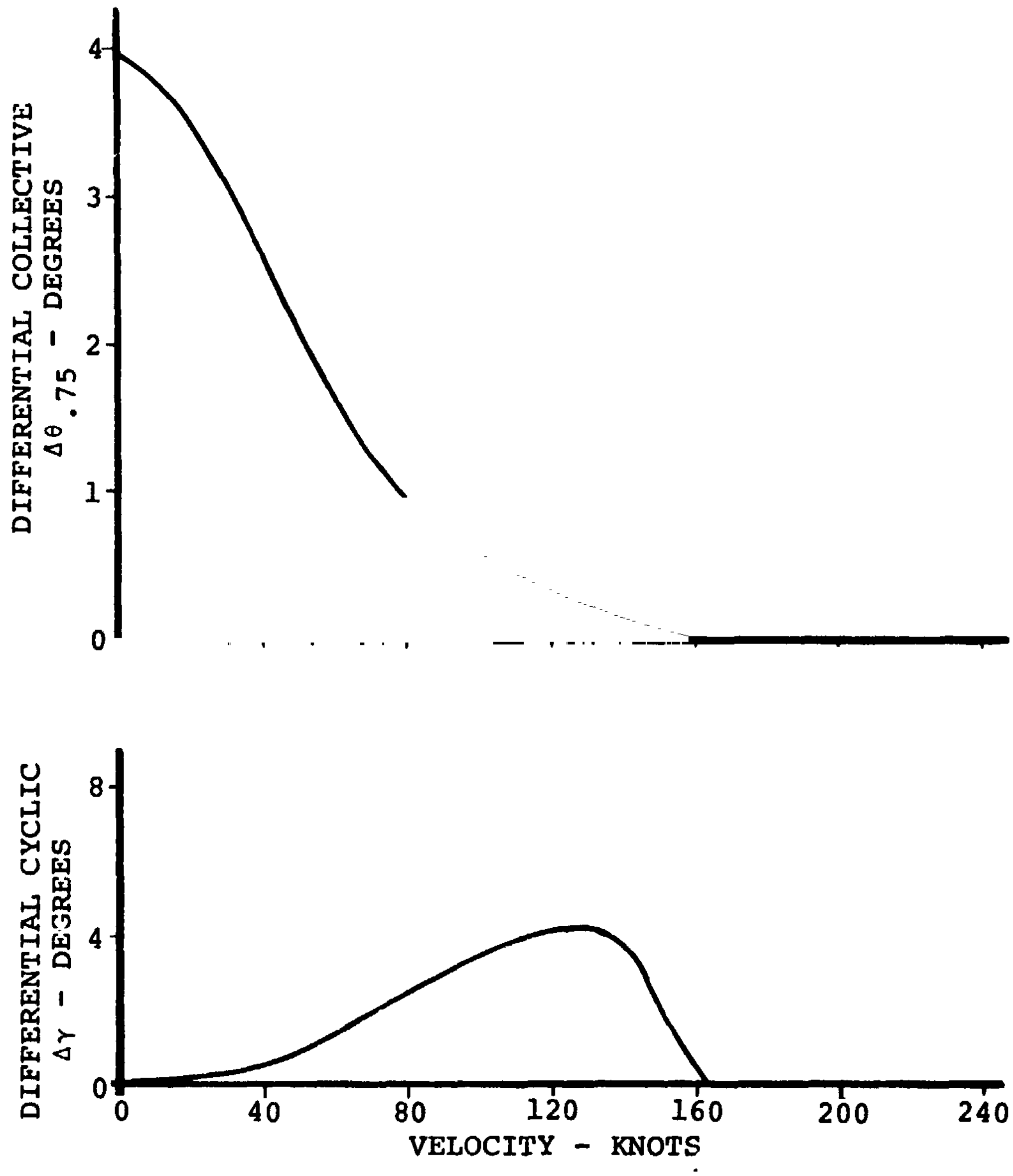


FIGURE 2.77. ROLL ANGULAR ACCELERATION CAPABILITY AND CONTROL REQUIRED IN TRANSITION.

C-3

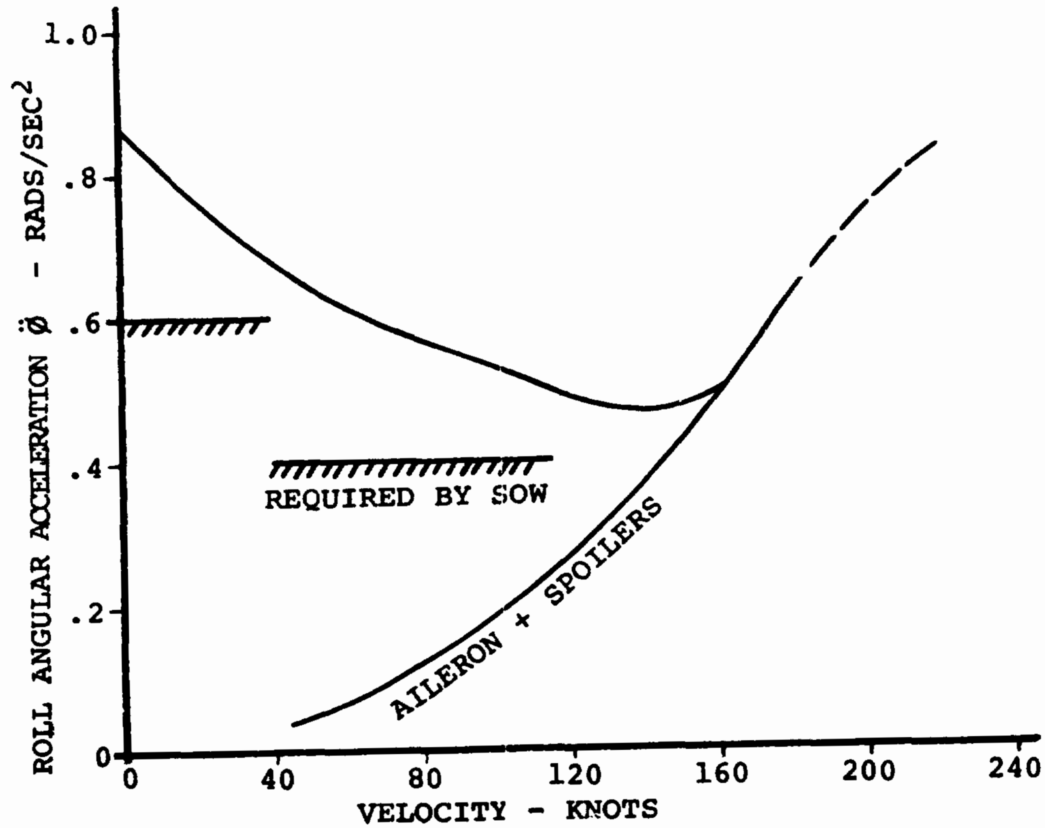


FIGURE 2.78. ROLL ANGULAR ACCELERATION CAPABILITY IN TRANSITION.

Cruise Flight - Longitudinal Stability and Control

The variation of neutral point for various cruise flight speeds is given in Figure 2.79 for a range of horizontal tail volume ratios. The most aft CG in cruise is 34% MAC which defines a most forward neutral point location of 39% MAC to provide 5% static margin. The design point tilt rotor horizontal tail volume ratio is 1.47 which gives a neutral point at 39.5% MAC at 140 Knots, i.e., 5.5% static margin. As airspeed increases the available static margin increases.

Trim

Aircraft cruise flight longitudinal trim data are shown in Figure 2.80 for design gross weight with CG aft (34% MAC) for sea level, 3,048 meters (10,000 feet) and 4,267 meters (14,000 feet) altitudes. The angles of attack to trim decreases as airspeed increases such that at design cruise speed of 349 Knots at 14,000 feet the aircraft trims at zero fuselage incidence. The design cruise condition requires 1.7 degrees of elevator to trim.

The most forward CG shown on the CG envelope in Figure 2.45 is at 60,000 pounds weight. The cruise trim data for this weight at 10% MAC CG position are shown in Figure 2.81. At the lower gross weight the angle of attack for trim is reduced and the elevator deflections required for trim are less. (i.e., more negative).

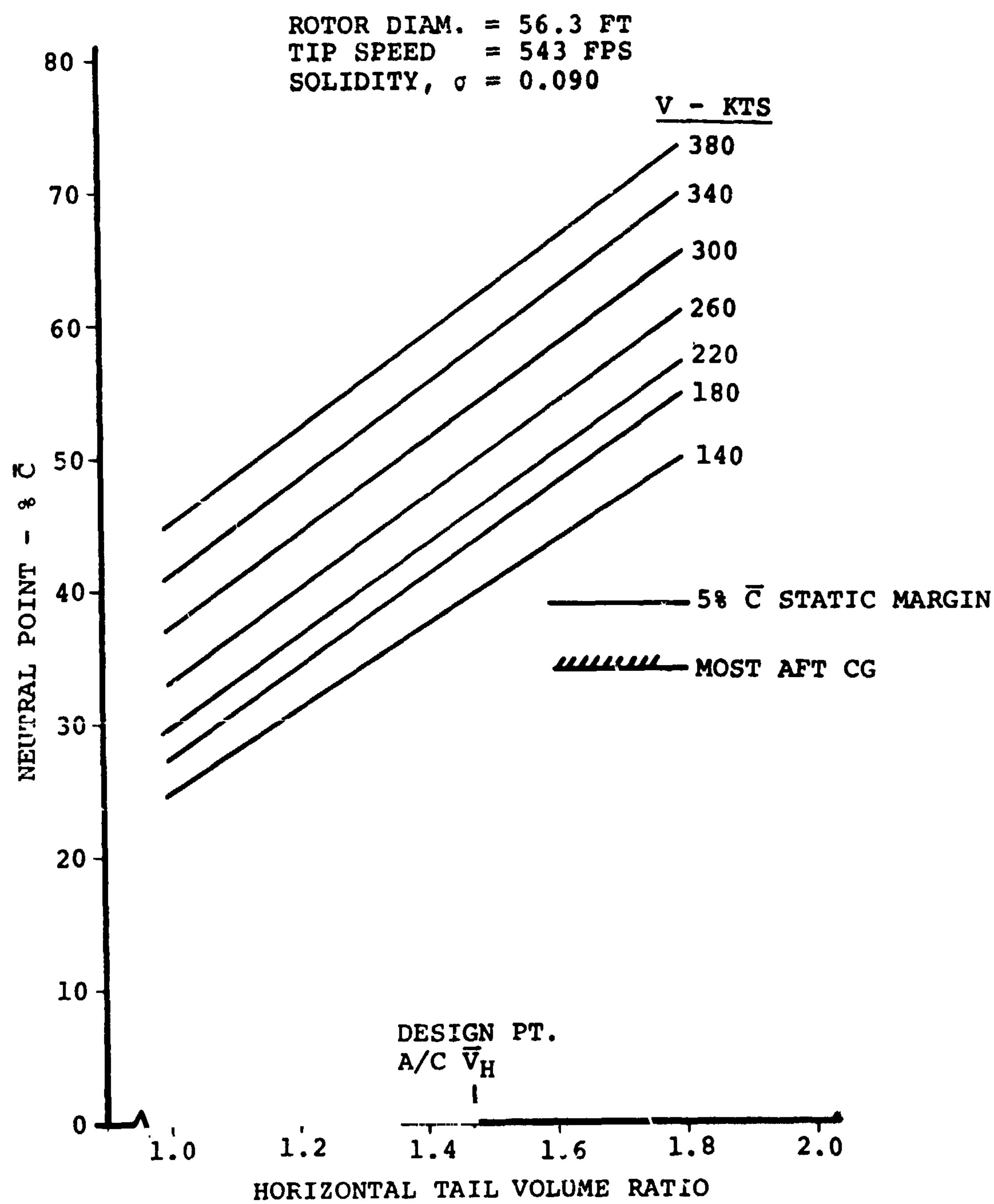


FIGURE 2.79. DESIGN POINT TILT ROTOR AIRCRAFT - STATIC STABILITY IN CRUISE.

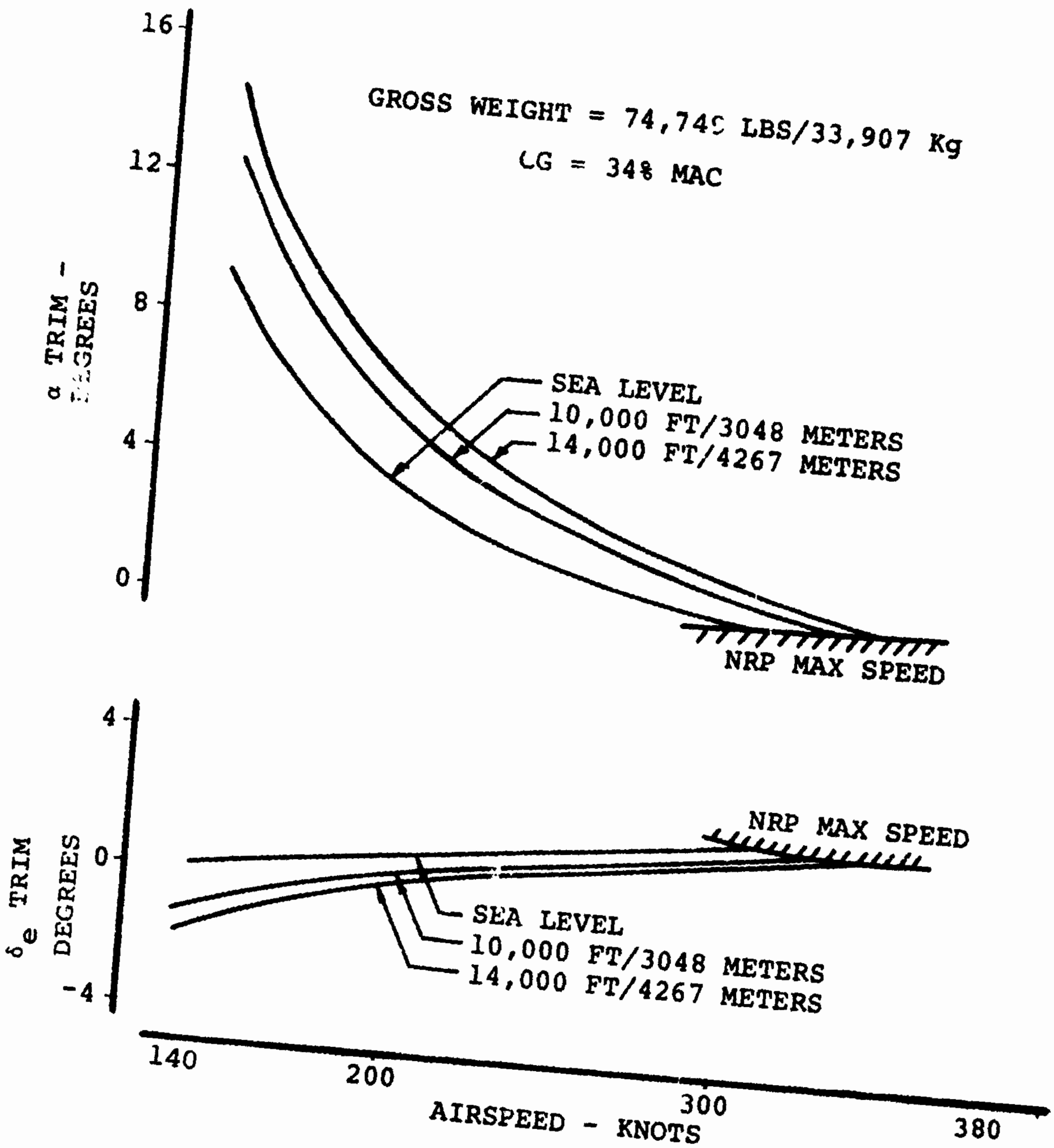


FIGURE 2.80. DESIGN POINT TILT ROTOR - CRUISE TRIM CHARACTERISTICS.

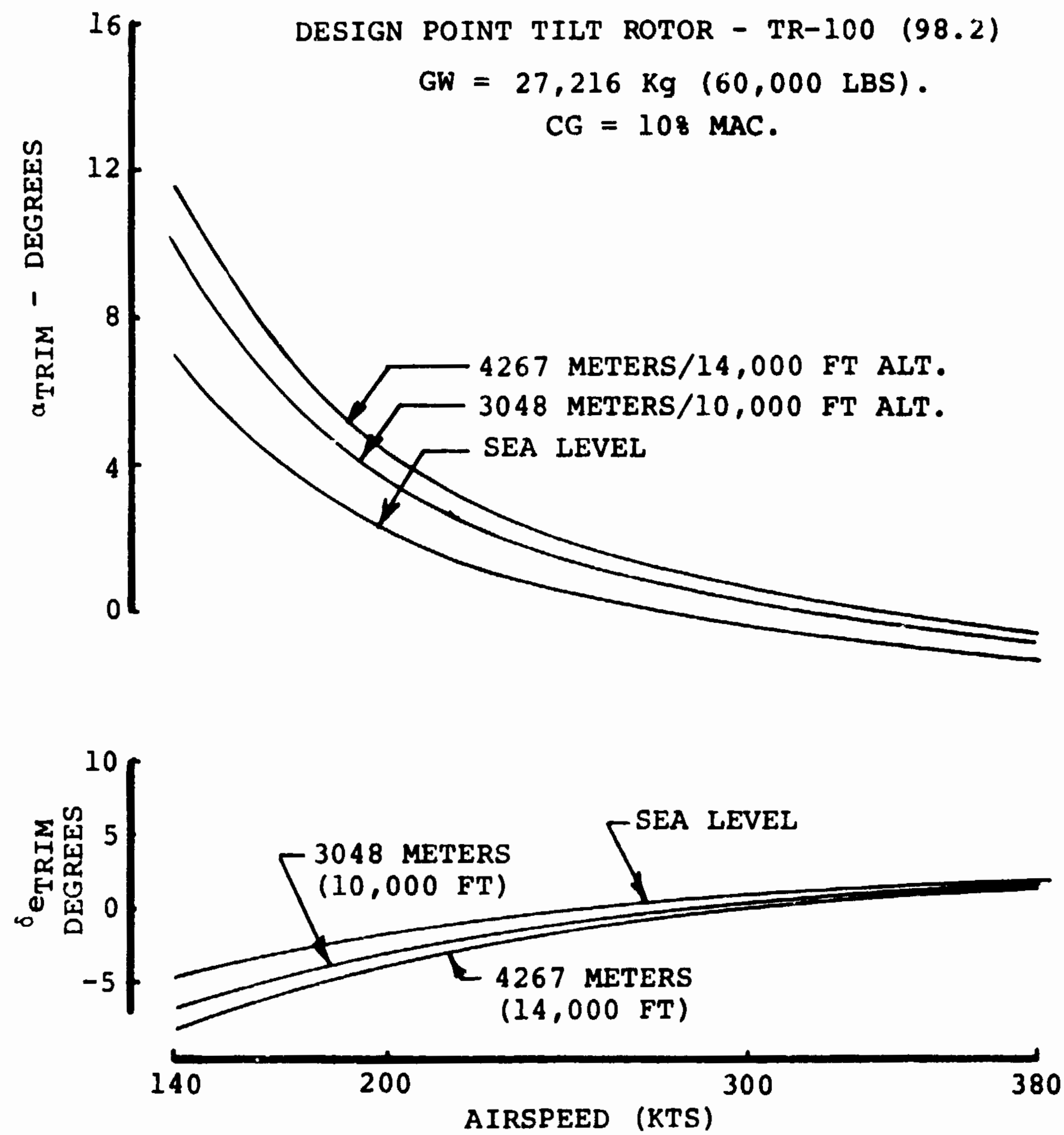


FIGURE 2.81. DESIGN POINT TILT ROTOR CRUISE TRIM.

Figures 2.82 and 2.83 present the angle of attack and elevator per g for the same gross weight, CG range and altitudes.

At high airspeeds the elevator required per g is small. The longitudinal "feel" system would need some adjustment to provide acceptable stick force per g. The small stick deflections per g result in high stick force per inch. Detailed design of the control system would provide compensation to increase the stick travel per g at high speed.

The pitch rate response to a one degree of step elevator input at 180, 300 and 380 Knots is shown in Figures 2.84, 2.85 and 2.86. For these data the elevator is stepped at time equal to one second and the data show the resulting time histories of pitch attitude and pitch rate. The elevator effectiveness is lowest at the low speed case. In one second after control input a pitch attitude of 1.4 degrees is achieved due to 1 degree elevator. Total elevator travel is ± 20 degrees. The pitch rate time constant is low (0.18 seconds to 63% maximum rate) indicating a crisp response to pilot command. The time constant shortens as airspeed increases.

These response data are for the aft CG case for which the short period roots become aperiodic as shown in Figure 2.87. Although this case indicates heavily damped real roots, the rate of response to pilot command indicated by Figures 2.84 to 2.86 is still high.

DESIGN POINT TILT ROTOR - TR-100 (98.2)
CRUISE TRIM - GW = 74,749 LBS/33,907 Kg

CG = 34% MAC

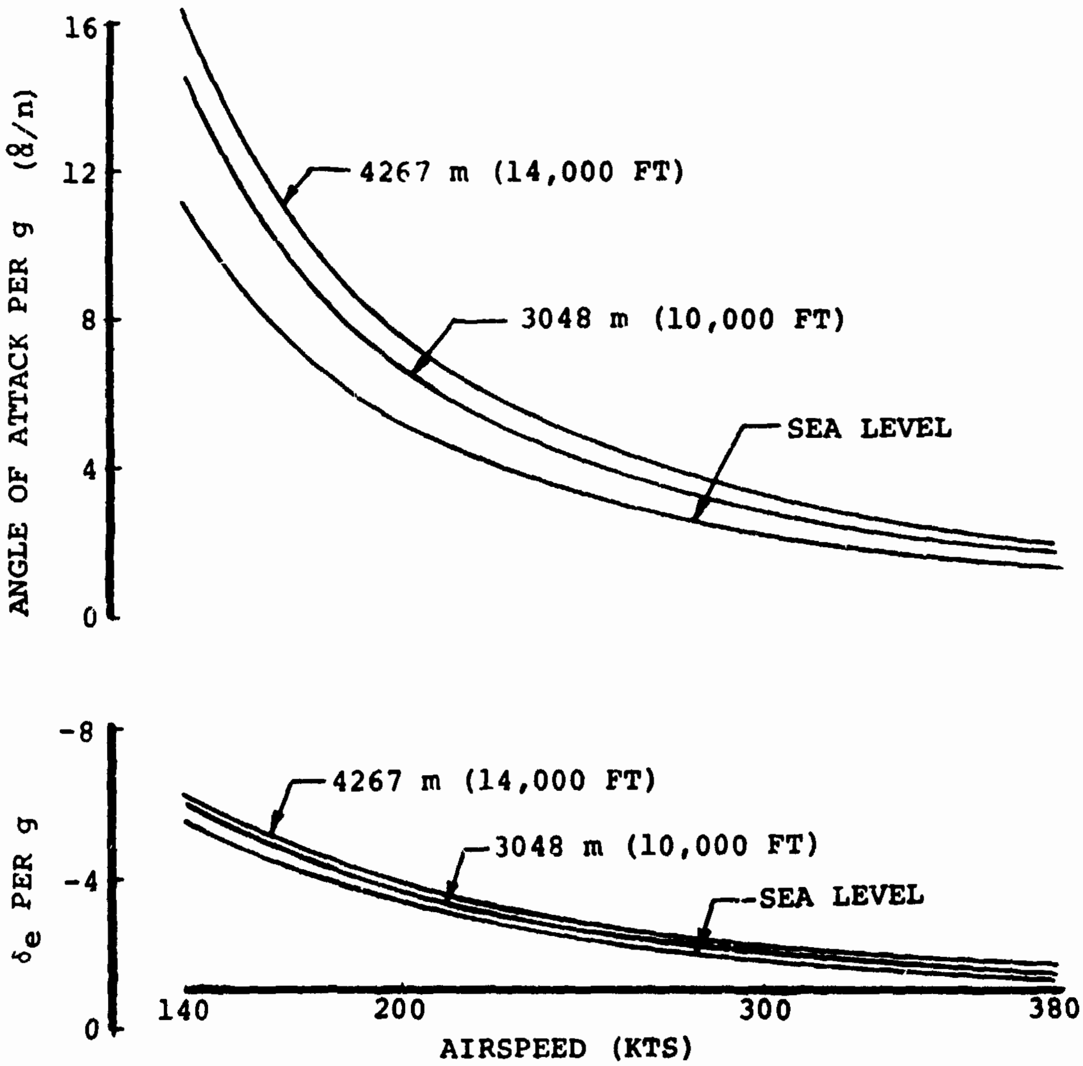


FIGURE 2.82 . DESIGN POINT TILT ROTOR - ANGLE OF ATTACK AND ELEVATOR PER g - CRUISE - 34% MAC CG.

DESIGN POINT TILT ROTOR - TR-100 (98.2)
CRUISE TRIM - GW = 27,216 Kg (60,000 LBS)

CG = 10% MAC

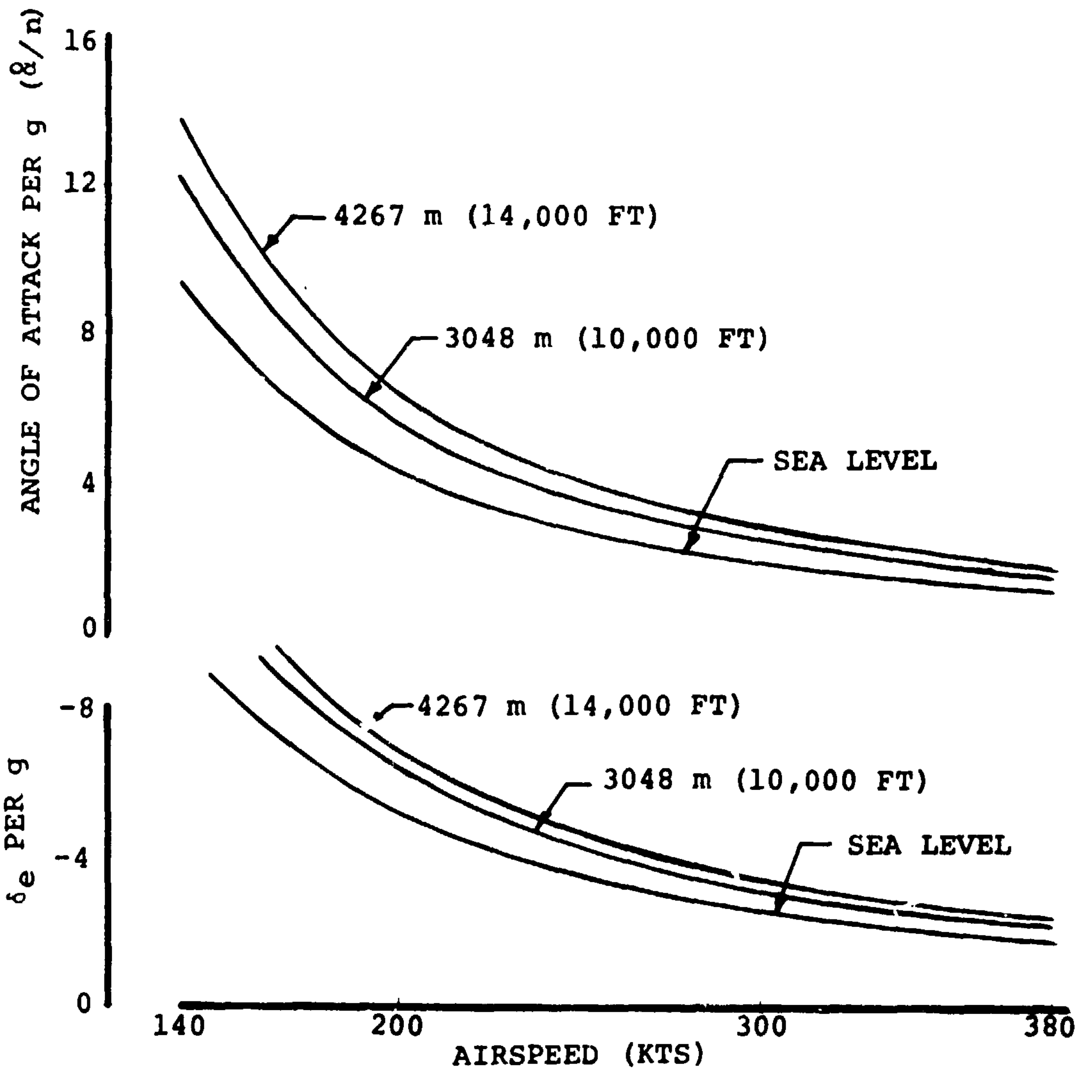


FIGURE 2.83. DESIGN POINT TILT ROTOR - ANGLE OF ATTACK AND ELEVATOR PER g - CRUISE - 10% MAC CG.

A/S = 180 KTS
GW = 74,749 LBS
CG = 34%
ALT. = SEA LEVEL

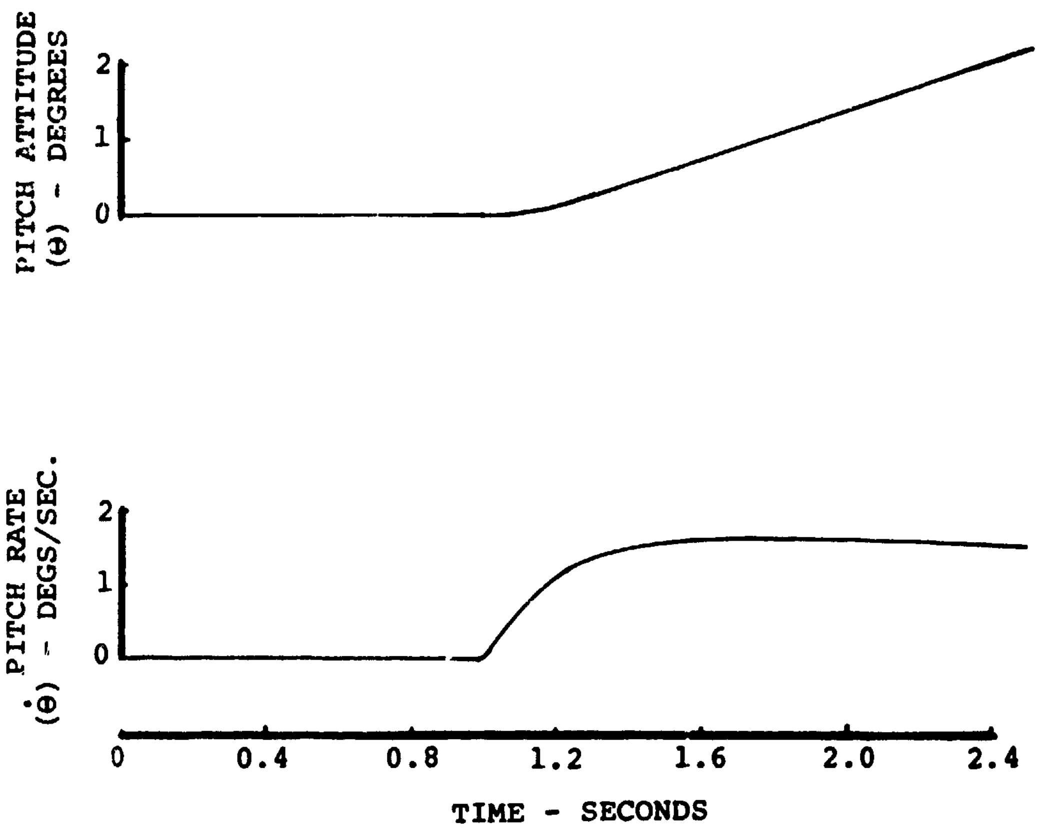


FIGURE 2.84. DESIGN POINT TILT ROTOR LONGITUDINAL RESPONSE TO 1 DEGREE STEP FLEVATOR INPUT.

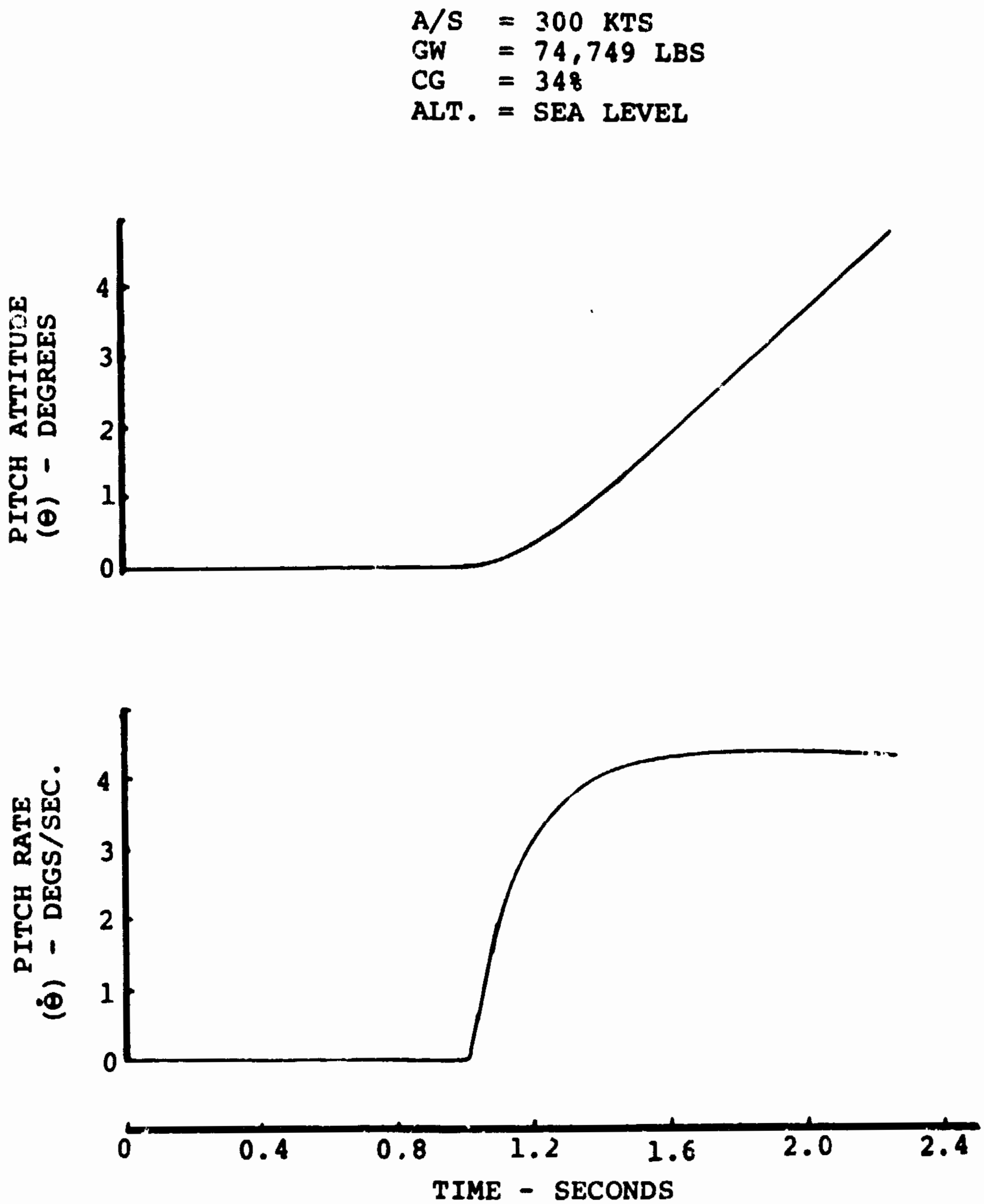


FIGURE 2.85. DESIGN POINT TILT ROTOR LONGITUDINAL RESPONSE TO 1 DEGREE STEP ELEVATOR INPUT.

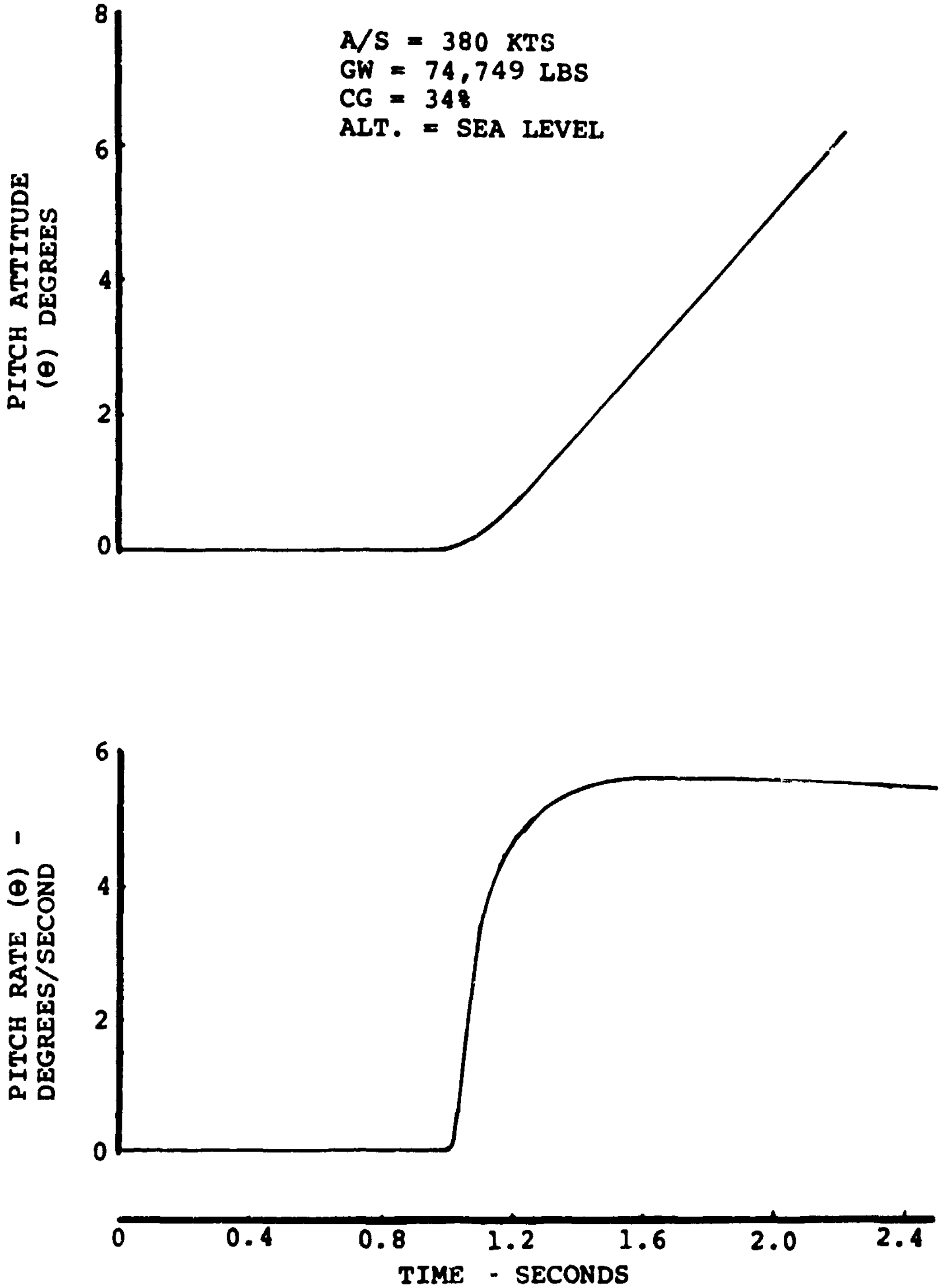


FIGURE 2.86. DESIGN POINT TILT ROTOR LONGITUDINAL RESPONSE TO ONE DEGREE STEP ELEVATOR INPUT.

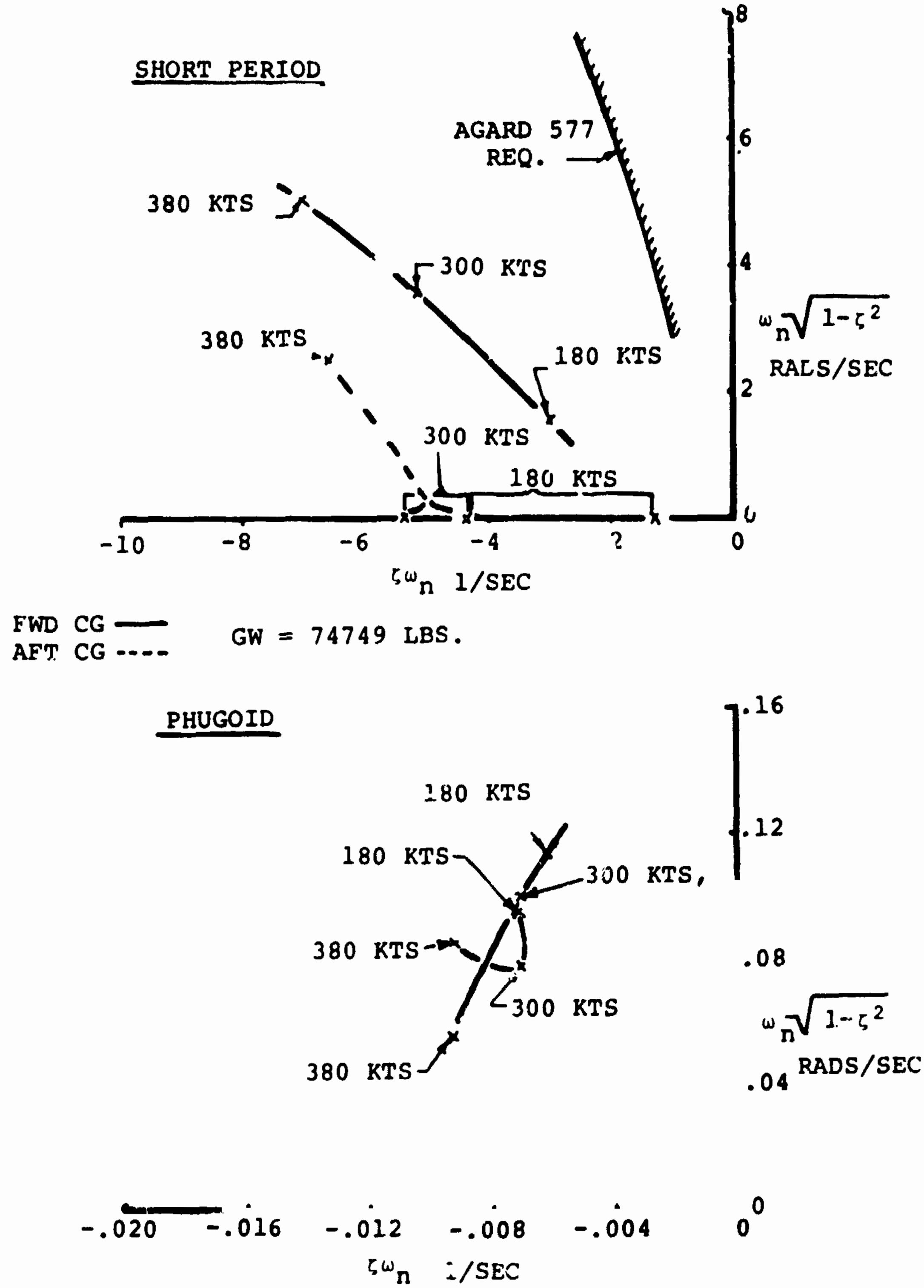


FIGURE 2.87. LONGITUDINAL DYNAMIC CHARACTERISTICS - CRUISE.

At 380 Knots the data indicates a normal short period mode with high damping. For the forward CG case the roots are periodic and damped well in excess of the recommendations of AGARD 577.

The phugoid mode is shown in Figure 2.87 and is a long period, well damped mode.

Figure 2.88 shows the short period frequency as a function of g's per angle of attack and the aircraft meets Level 1 criteria for both forward and aft CG cases.

Lateral-Directional - Stability and Control

Stability Derivatives

The lateral-directional derivatives due to sideslip are shown as a function of airspeed in Figure 2.89.

At speeds in excess of 187 Knots the rotors decrease the dihedral effect, however, a positive dihedral effect (negative $C_{\ell\beta}$) is maintained over the entire speed range. At low speeds the ratio of $C_{N\beta}/C_{\ell\beta} = 0.91$. As airspeed increases the increasing $C_{N\beta}$ and reduction in dihedral effect tends to improve the aircraft dutch roll stability and reduce the stability of the spiral mode.

The rotor contribution to the sideforce derivative due to sideslip is large and the resulting total $C_{Y\beta}$ will require a relatively large roll angle to compensate in flying a straight-ground track in sideslip.

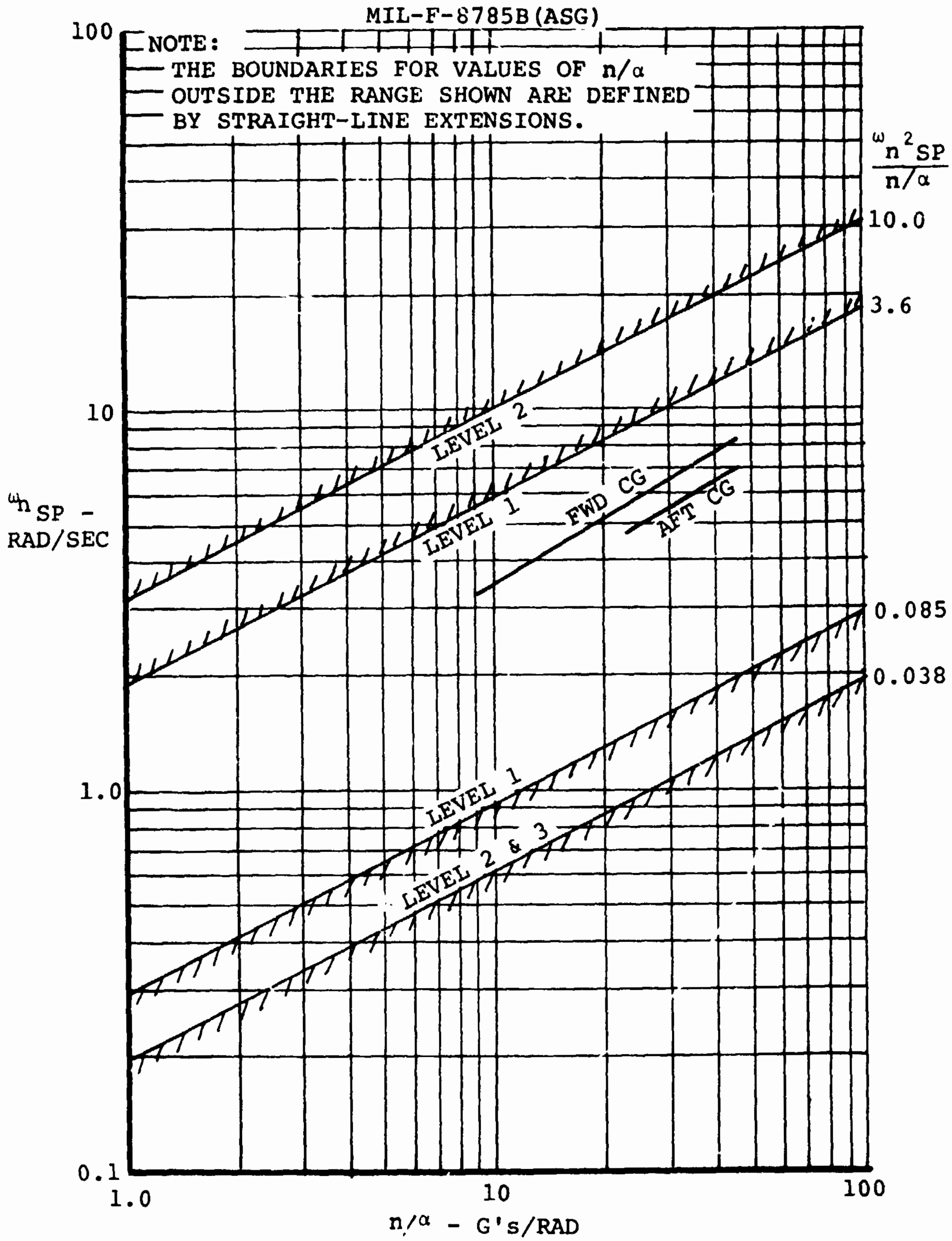


FIGURE 2.88. SHORT-PERIOD FREQUENCY REQUIREMENTS - CATEGORY B FLIGHT PHASES.

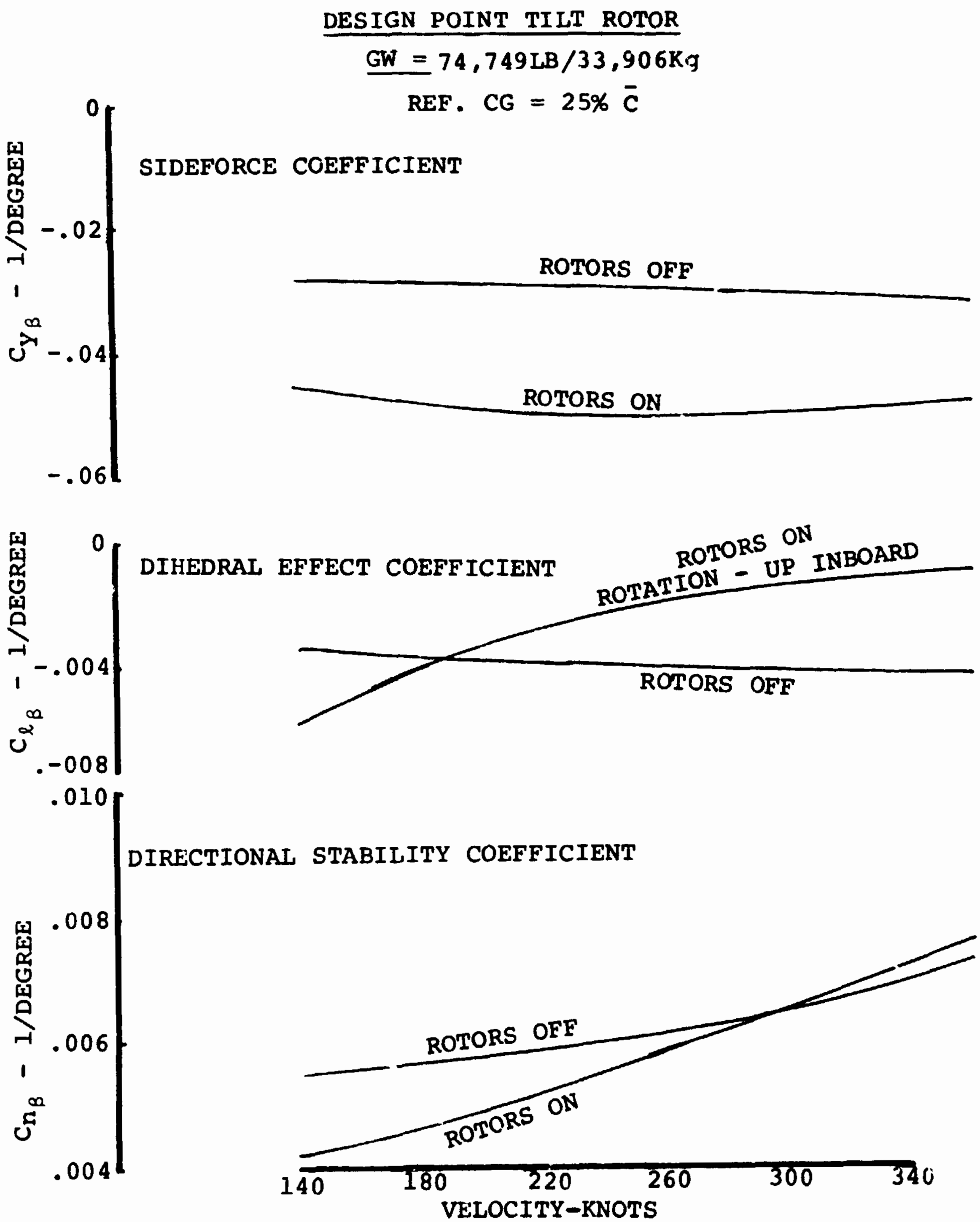


FIGURE 2.89. VARIATION OF LATERAL-DIRECTIONAL DERIVATIVES DUE TO SIDESLIP WITH VELOCITY

The sideslip characteristics in cruise are shown in Figure 2.90 and this effect is in evidence. The lateral stick per degree of rudder crosses zero at 285 Knots indicating a requirement for a small amount of aileron to rudder pedal coupling to provide normal control direction in sideslip. The rudder effectiveness is high and decreases slightly with airspeed.

The roll rate derivatives are shown in Figure 2.91. The roll rate damping is high ($C_{\ell p}$ 1.1 to -1.4) due to the influence of the rotors. The yaw moment coefficient due to roll rate decreases as airspeed increases.

The yaw rate derivatives are shown in Figure 2.92 and indicate a high C_{nr} or yaw rate damping. The rotors again contribute substantially to the yaw damping.

The lateral-directional dynamic stability is shown in Figures 2.93 and 2.94. The dutch roll is stable and well damped at both aircraft weights and CG's considered. The spiral mode is slightly unstable at high airspeeds, but has an acceptable minimum time to double amplitude of 1.85 minutes.

The roll mode time constant is less than one second for all velocities at sea level and slightly higher at low speeds at 14,000 feet altitude. Roll rate response data are presented in Figures 2.95 and 2.96. At 180 Knots a roll angle of 14 degrees can be achieved in one second after a step input. At higher airspeeds the roll rate response increases.

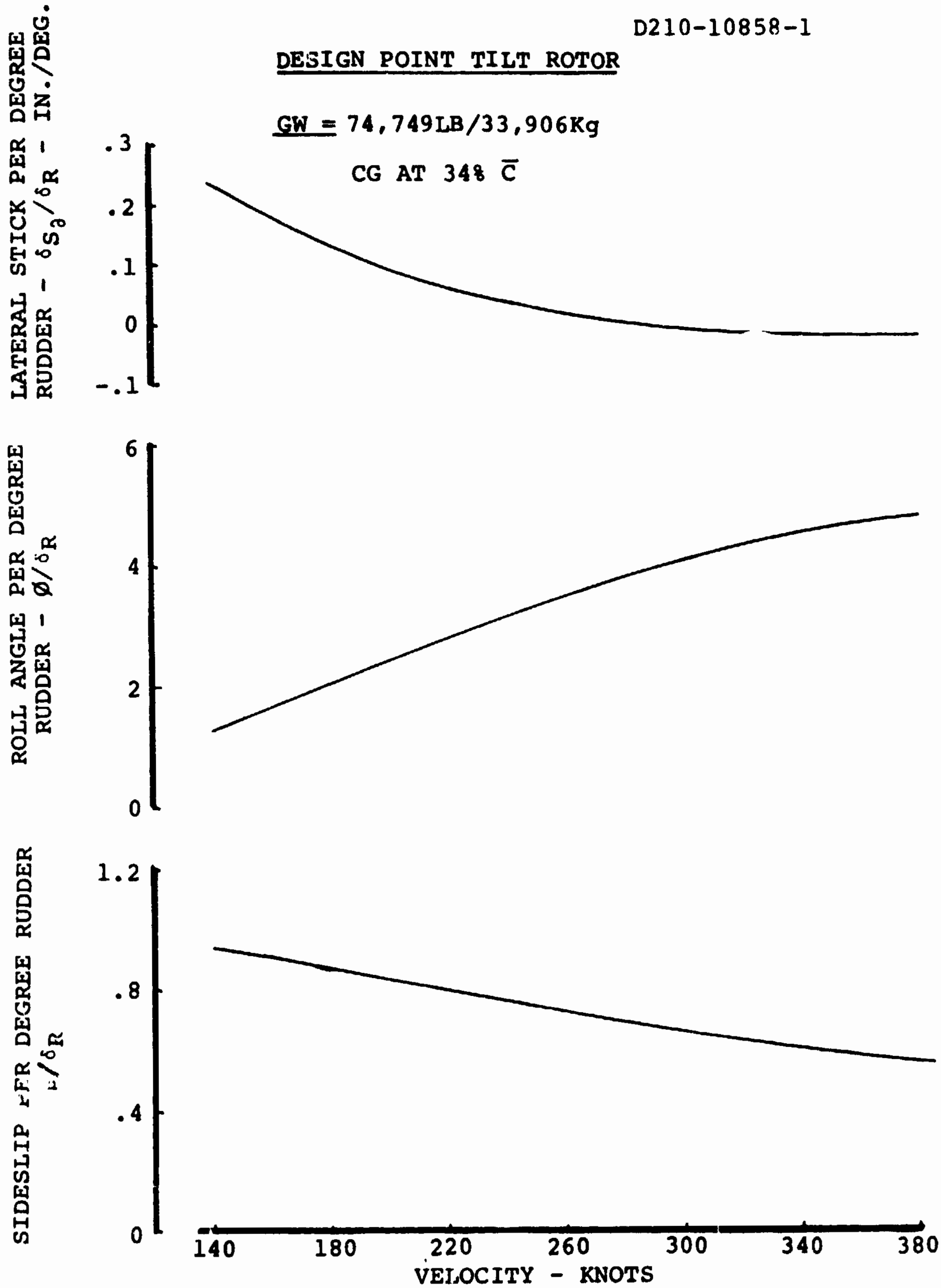
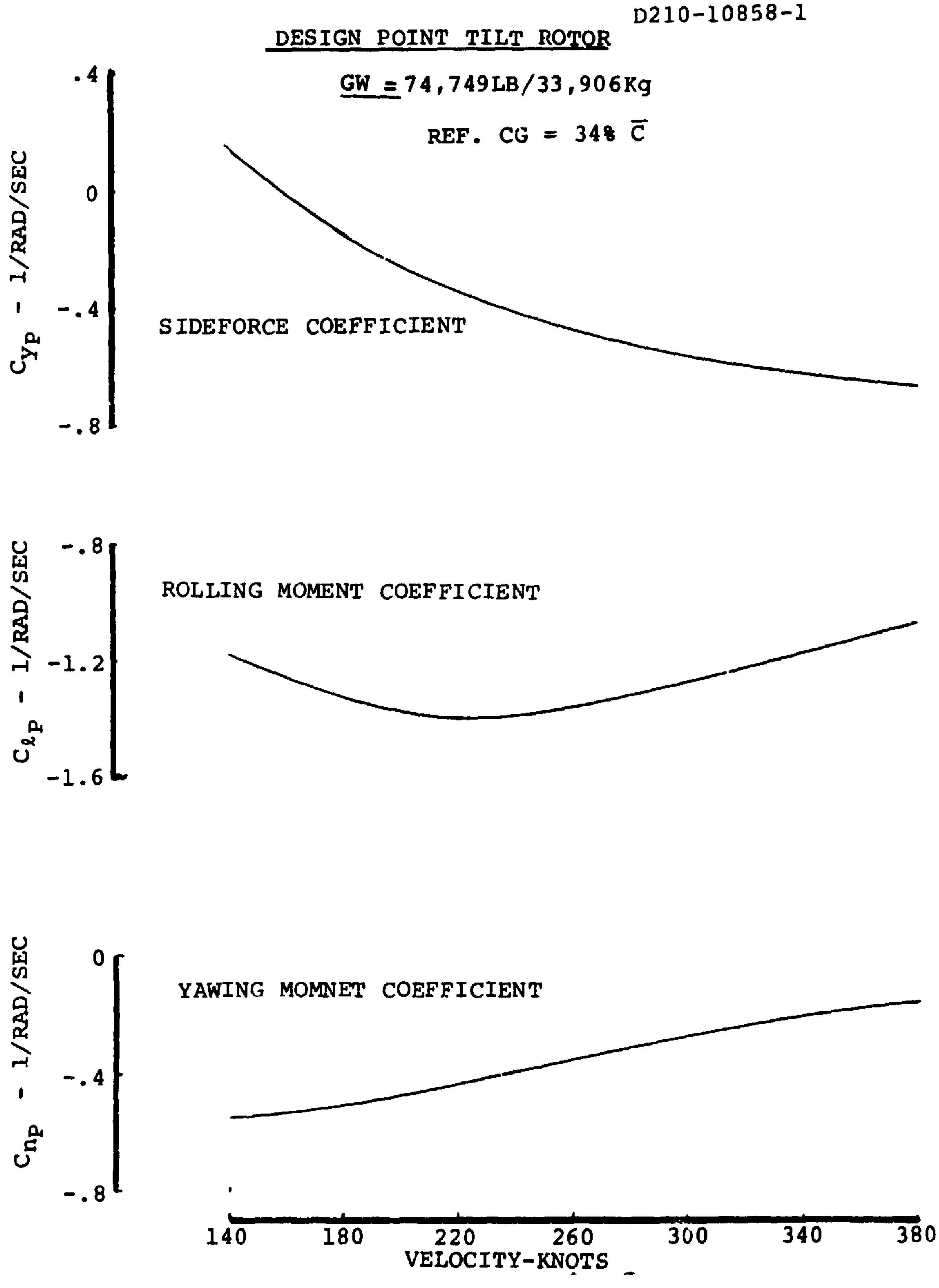


FIGURE 2.90 . SIDESLIP CHARACTERISTICS IN CRUISE.
(STRAIGHT GROUND TRACK).



FIUGRE 2.91. STABILITY COEFFICIENTS DUE TO ROLL RATE.

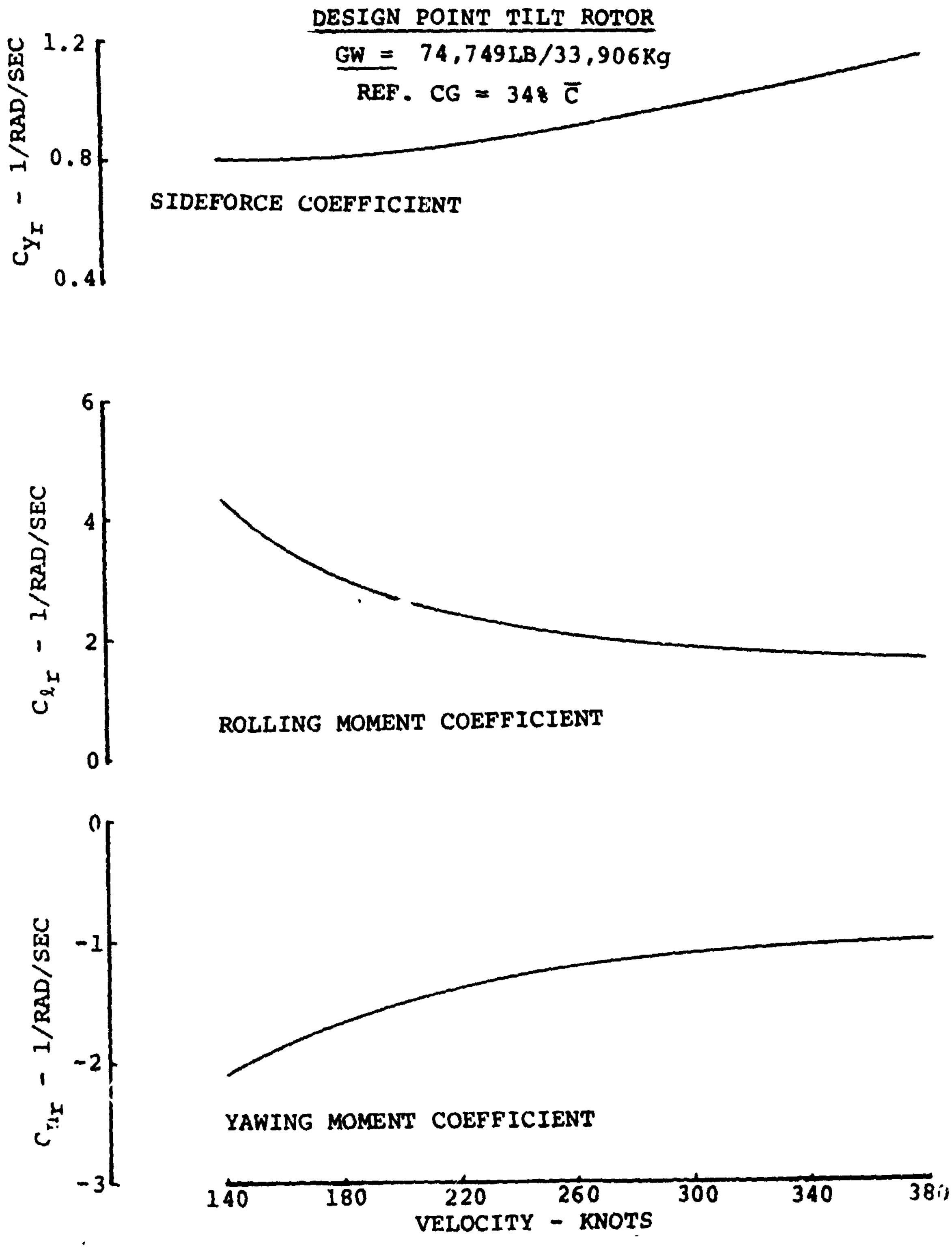


FIGURE 2.92 . STABILITY COEFFICIENTS DUE TO YAW RATE.

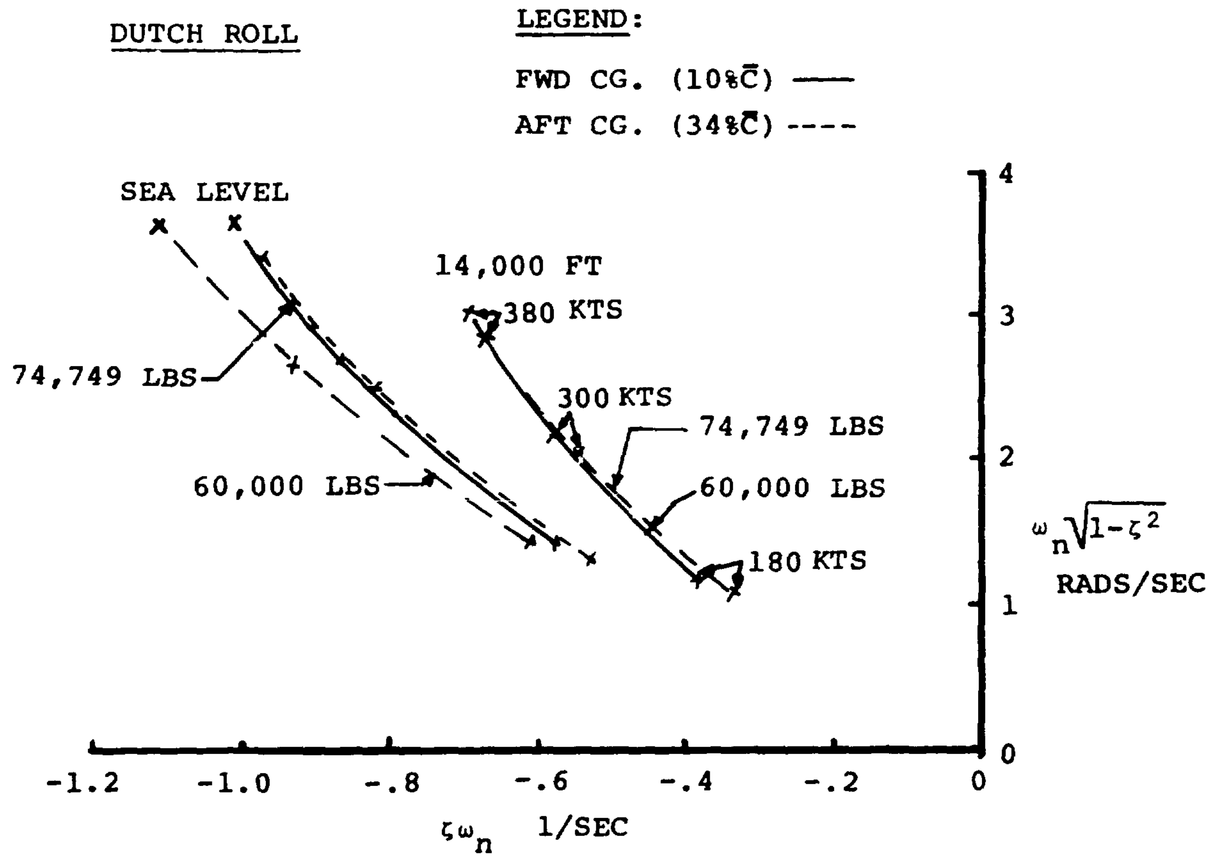
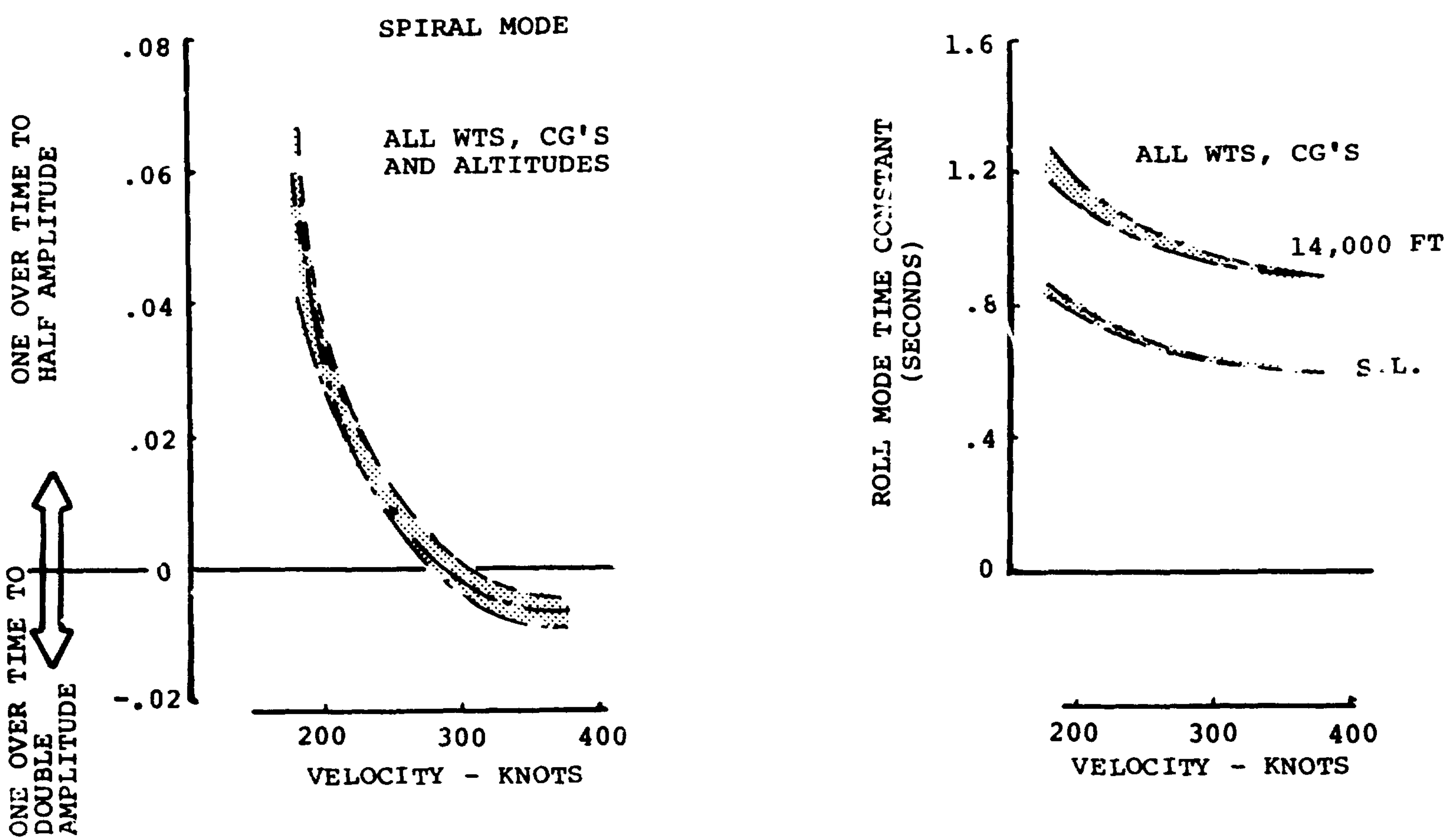


FIGURE 2.93 . DESIGN POINT TILT ROTOR - LATERAL-DIRECTIONAL DYNAMICS IN CRUISE.

FIGURE 2.94 . DESIGN POINT TILT ROTOR - LATERAL-DIRECTIONAL DYNAMICS.



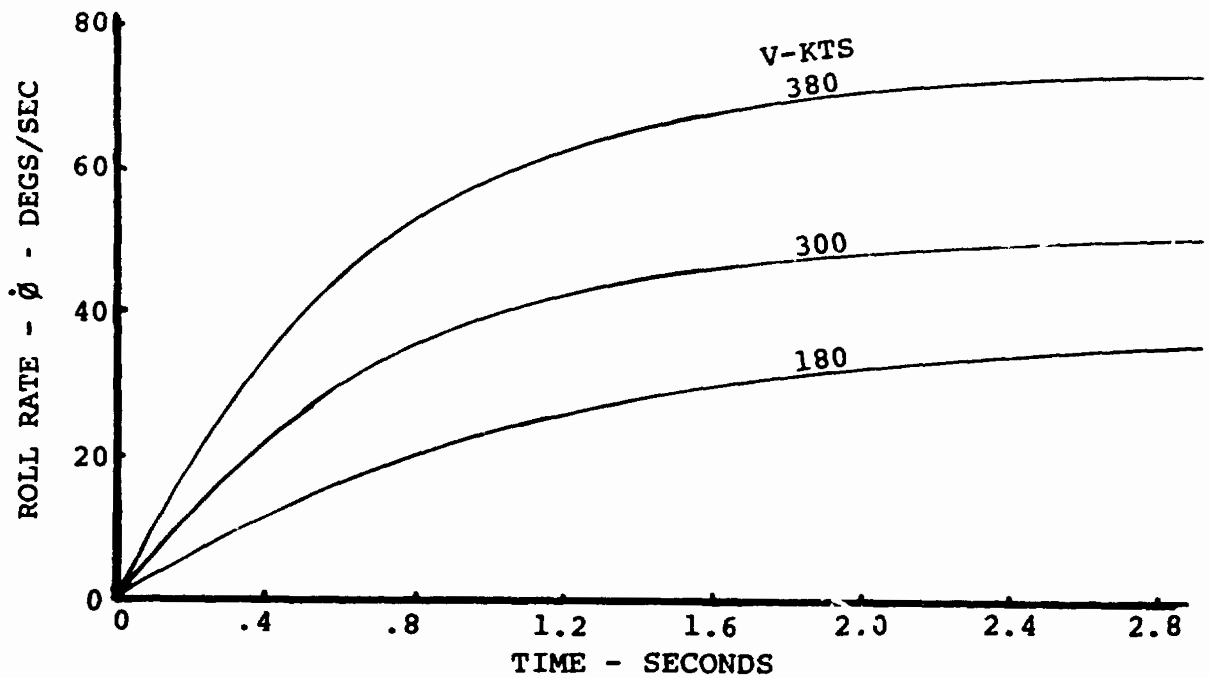


FIGURE 2.95. DESIGN POINT TILT ROTOR ROLL RESPONSE - CRUISE.

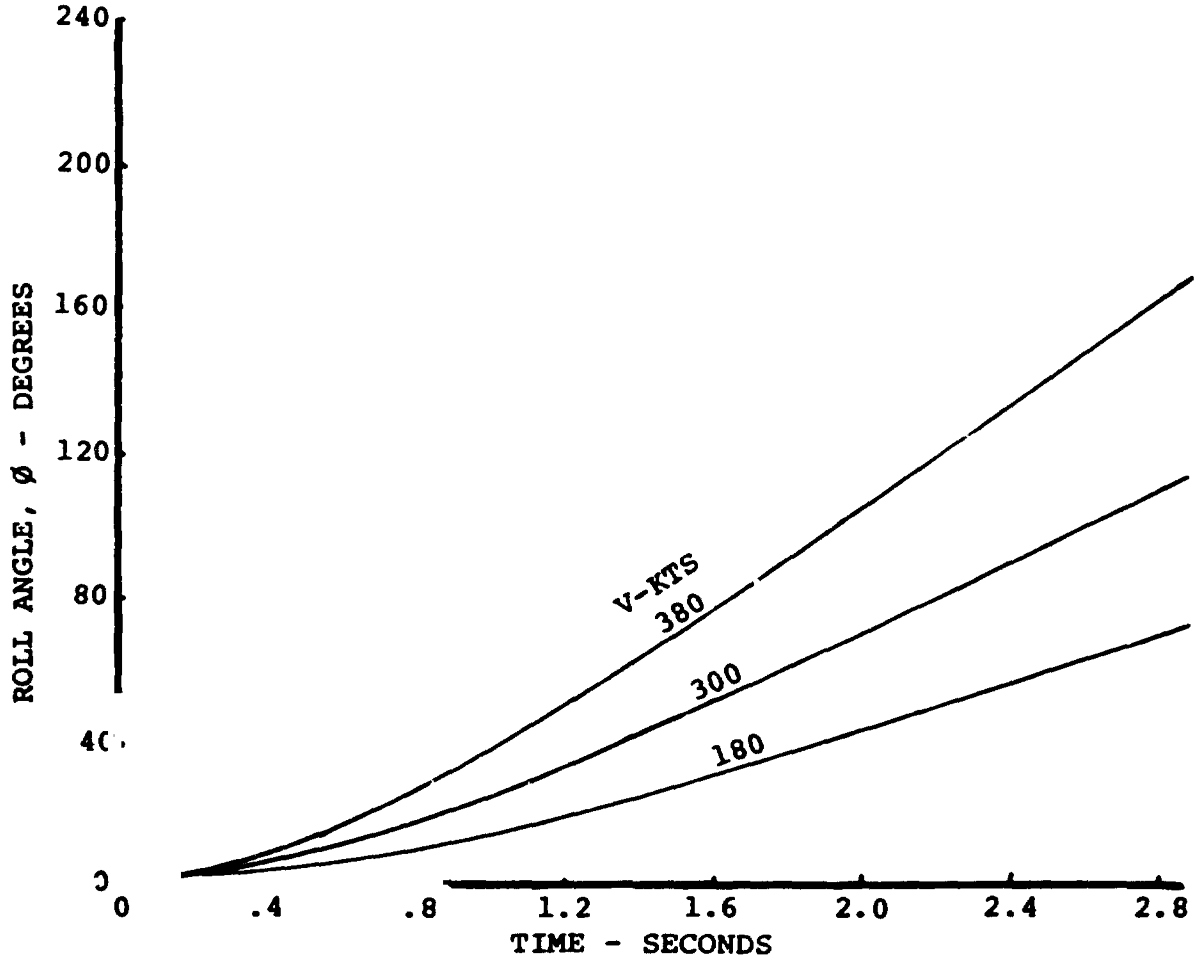


FIGURE 2.96. DESIGN POINT TILT ROTOR ROLL RESPONSE - CRUISE.

Gust Sensitivity and Direct Lift Control

Reduction of gust sensitivity using direct lift control is required to bring the 100 passenger tilt rotor transport with the limits given in the study guidelines. The situation without alleviation shown in Figure 2.97 for minimum and maximum operating gross weights at 10,000 and 14,000 feet indicates that substantial amounts of lift must be dumped, if the criterion is to be met. It is envisioned that this will be primarily accomplished by the automatic symmetric application of spoilers or flaps in amounts proportional to the angle of attack change produced by the gust. Further alleviation would be produced by similar operation of the rotor cyclic pitch controls; elevator controls would also be applied to counteract unwanted pitching moments.

In this study the approximate requirements of the flap and spoiler system have been found to be less demanding than the normal control system requirements, so that no structural weight penalties are associated with the direct lift control system. For example, at 55,726 pounds gross weight (minimum fuel, 10 passenger) at 10,000 feet, at a maximum cruise speed of 296 Knots EAS, the control applications required to reduce sensitivity to the criterion level in a 15 feet per second gust are:

	FLAP	SPOILER
ANGLE	6.8 degrees	12.0 degrees
MAXIMUM RATE	50 degrees/sec	85 degrees/sec

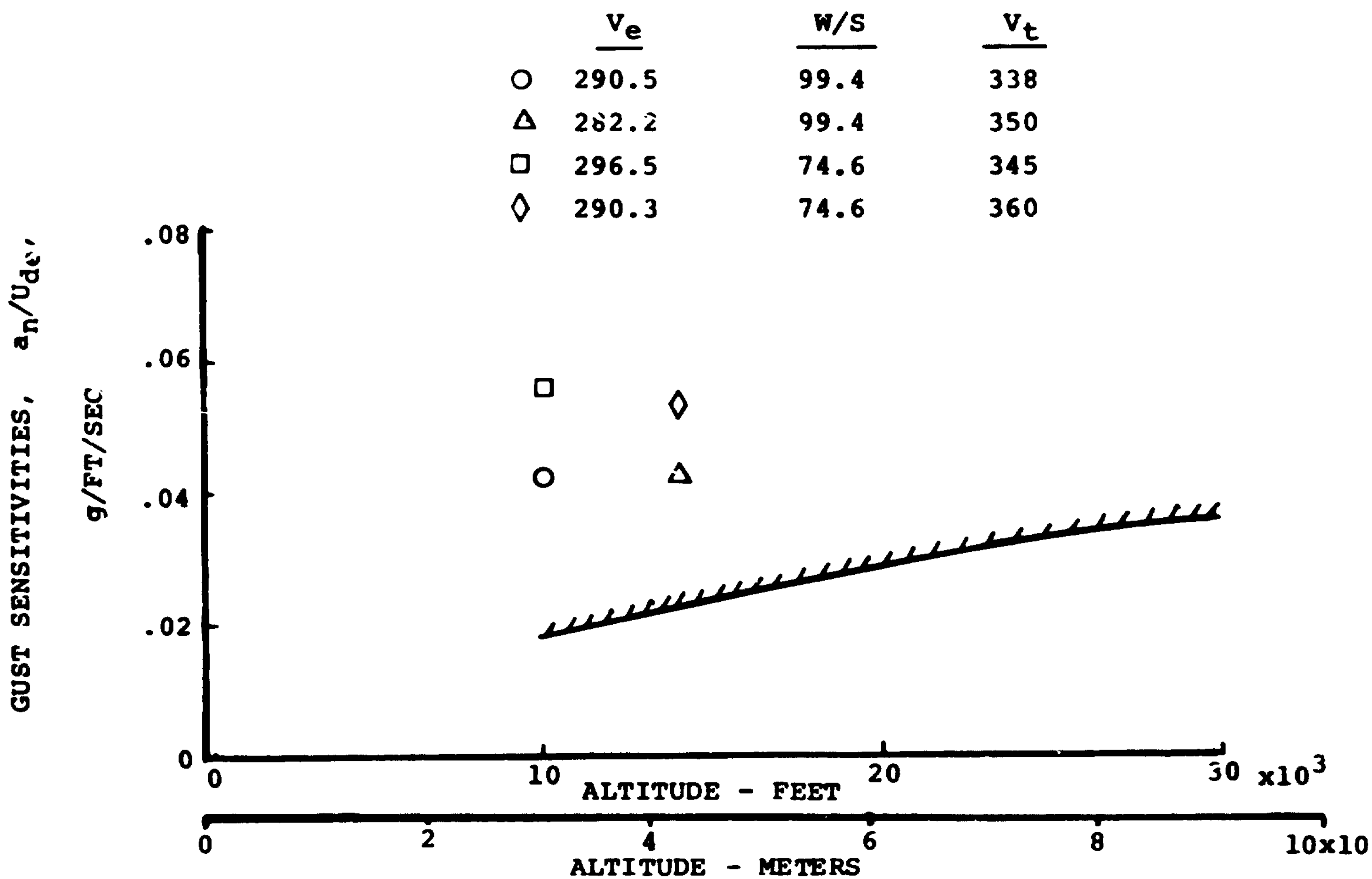


FIGURE 2.97. PASSENGER RIDE DESIGN CRITERON (TILT ROTOR)

These may be compared with the design control angles and rates implied by the time constant criteria of Paragraph 4.1.1.3 of the study guidelines. Flap and spoiler maximum travels are 20 degrees and 45 degrees respectively and the average rates implied by the time constant criteria are 6^ degrees per second for the flaps and 142 degrees per second for the spoilers. It is concluded that the only major additional system requirements and weight penalties would be those associated with gust sensing equipment and avionics for signal conditioning and transmission of commands to the control surface actuators. These are estimated to be approximately 35 pounds.

2.2.5 Design Point Tilt Rotor - Noise

The external noise design criteria for the baseline aircraft was that the sideline noise at 500 feet in hover at 100 feet altitude was to be between 90 and 100 PNdB. It is recognized that the use of PNdB as a means of expressing noise annoyance may not be a valid comparison between rotary wing aircraft and conventional jet engined aircraft because of the different spectra of the two types. The highest sound pressure levels for the tilt rotor occur at lower frequencies. The sound pressure level data at 500 feet sideline are shown in Figure 2.98 as a function of octave band frequency. Using the NOY weightings the aircraft perceived noise level at 500 feet is 98.2 PNdB in hover.

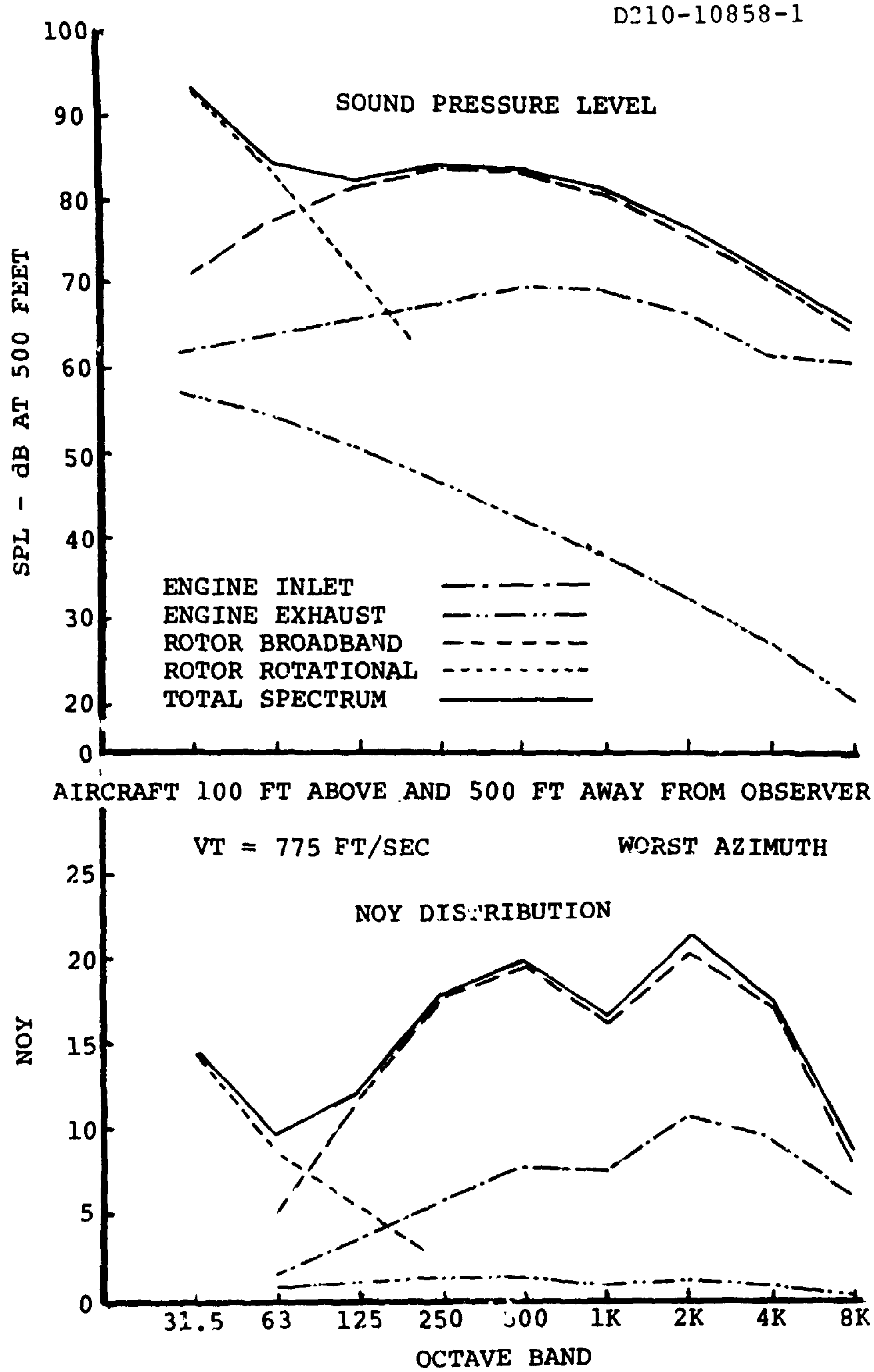


FIGURE 2.98 TILT ROTOR BASELINE HOVER SPECTRUM AND NOY DISTRIBUTION. PNdB = 98.2

The components contributing to the overall sound pressure levels are also given in Figure 2.98, and over most of the frequency range, the noise level is set by rotor broadband noise.

Rotor rotational noise is dominant at the low frequencies, however, this has only a small impact on the aircraft PNdB value.

In order that the PNL be set by rotor noise components, the engine inlets were treated to attenuate inlet noise at high frequencies. The inlet lining is tuned to 4 KHz and 8 KHz in order to match the predicted rotor signature. The inlet treatment area necessary to achieve the inlet attenuation shown in Figure 2.99 is 7.5 square feet per engine.

Contours of constant perceived noise levels are shown in Figure 2.100 for a typical takeoff. The noise contours are symmetrical about the flight path axis with the highest noise levels beneath the flight path. These PNL contours are based upon a takeoff profile as shown in Figure 2.101 and on the same figure the perceived noise levels at various points on the flight path are given as a function of noise duration. For example, an observer 200 feet from the point of origin would perceive 105 PNdB at time zero, 116 PNdB at 7.5 seconds, and less than 90 PNdB at 20 seconds from takeoff. An observer at 2,450 feet from the point of origin would never perceive more than 90 PNdB.

TILT ROTOR INLET TREATMENT

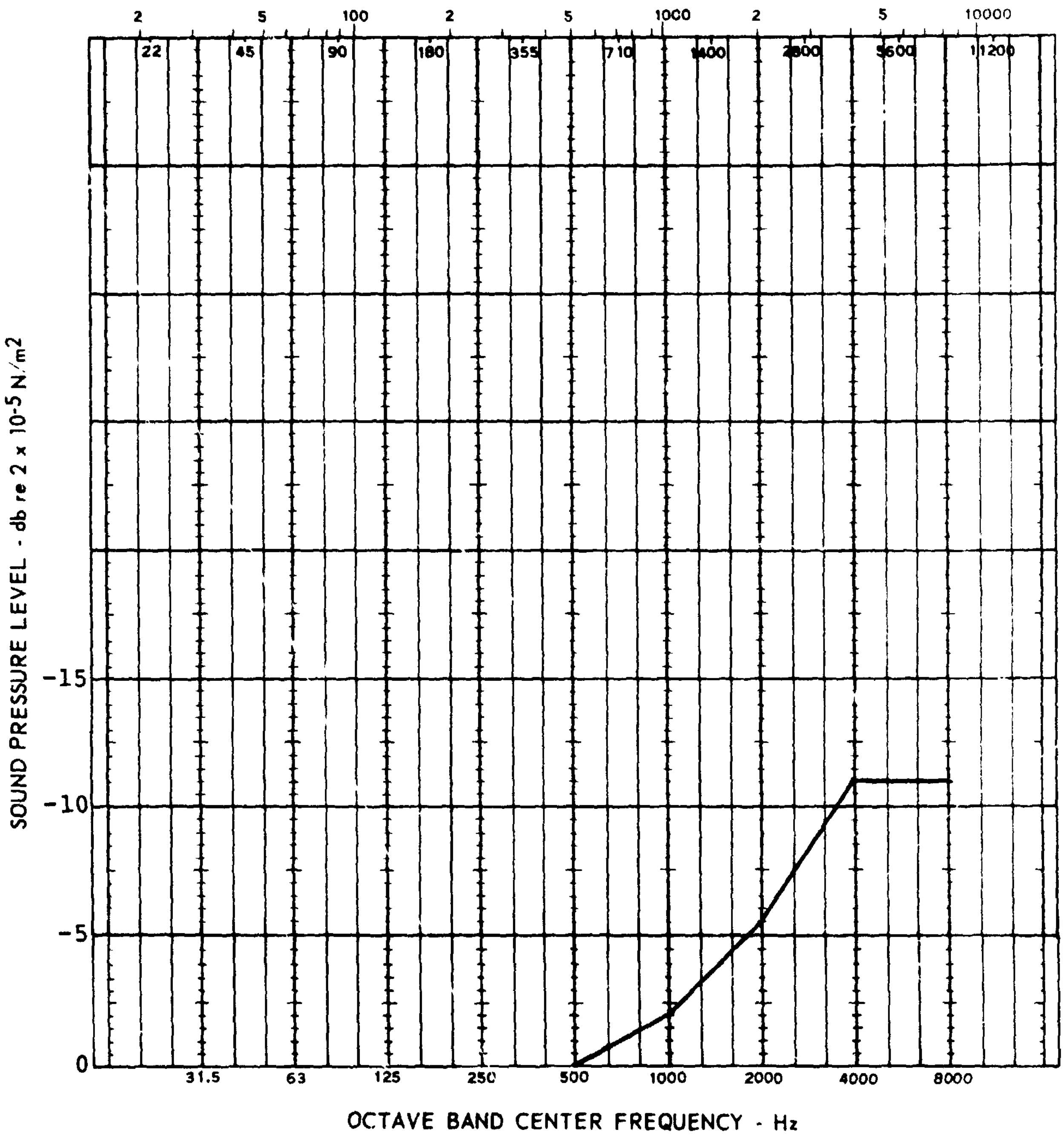


FIGURE 2.99. INLET NOISE SUPPRESSION.

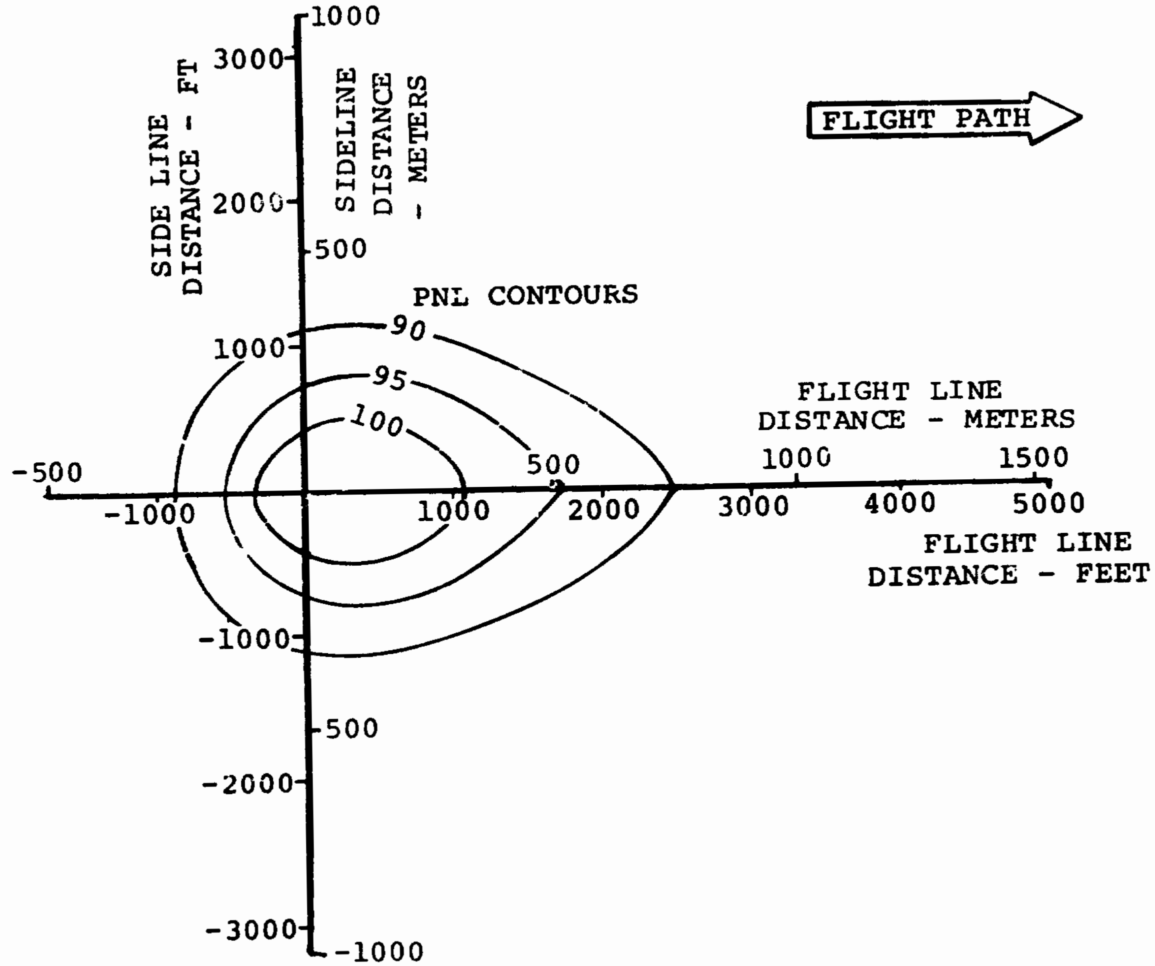
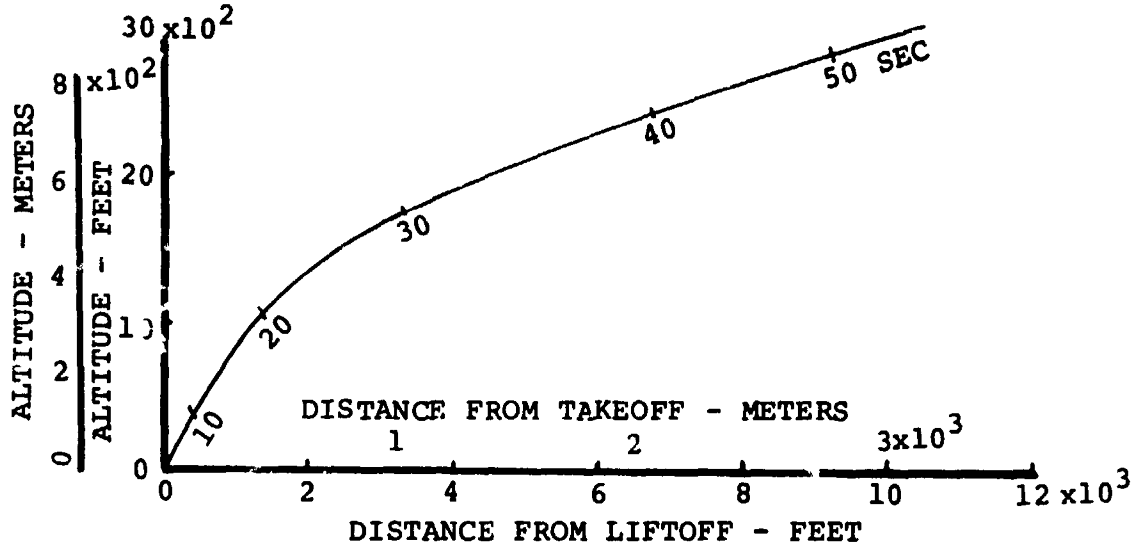


FIGURE 2.100. BASE LINE TILT ROTOR DESIGN POINT - STANDARD TAKEOFF. PNL CONTOURS.



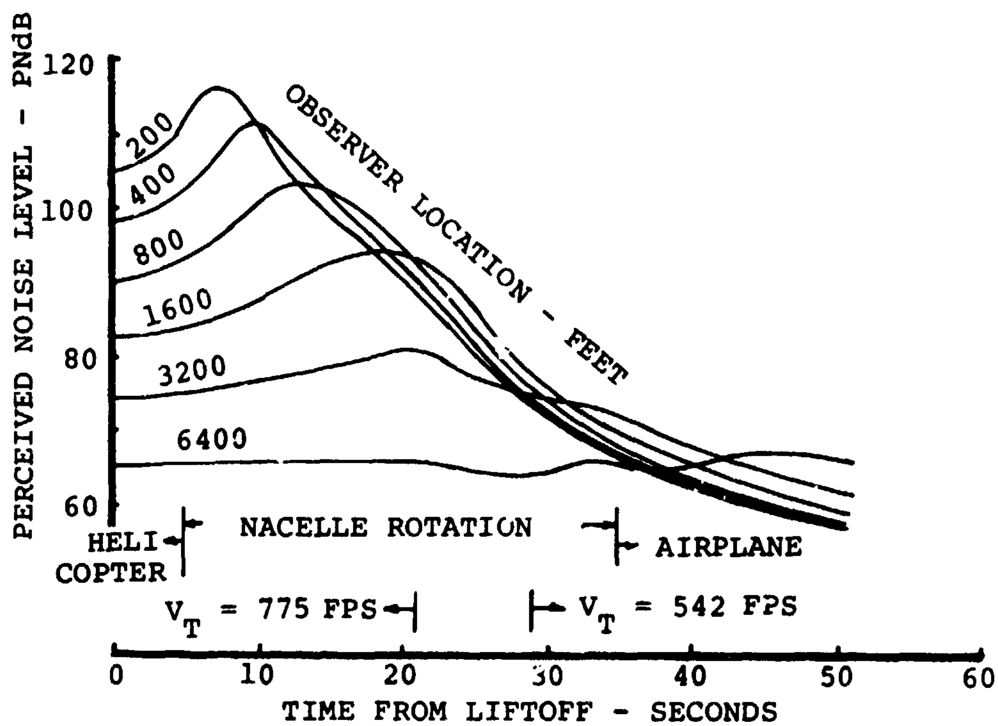


FIGURE 2.101. BASELINE TILT ROTOR DESIGN POINT - STANDARD TAKEOFF. PERCEIVED NOISE.

A similar set of data is shown in Figures 2.102 and 2.103 for a typical landing case. Again observers on the flight path are subject to the highest PNL and the perceived noise is greater for the landing case than for takeoff.

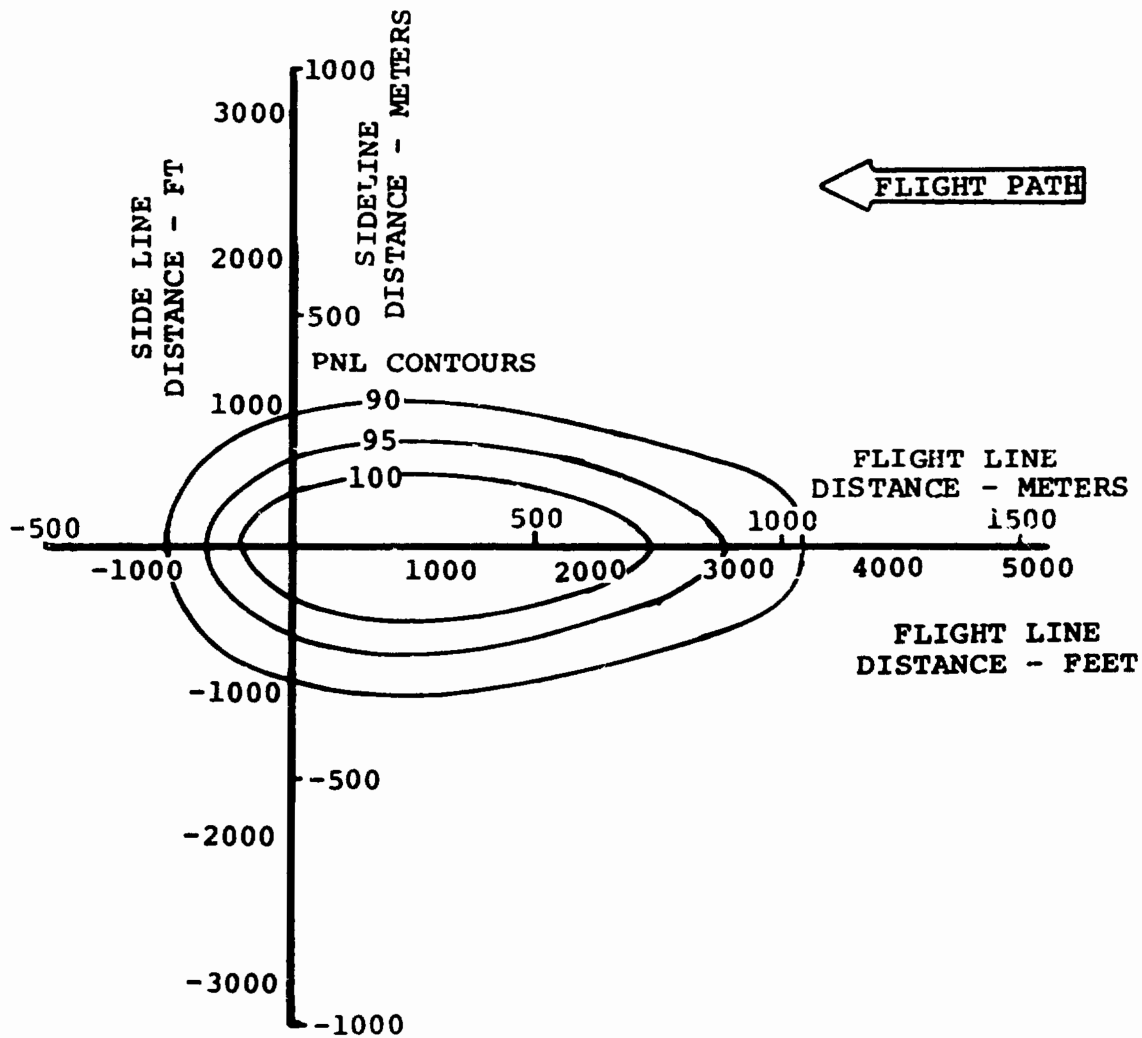


FIGURE 2.102. BASELINE TILT ROTOR DESIGN POINT - STANDARD LANDING. PNL CONTOURS.

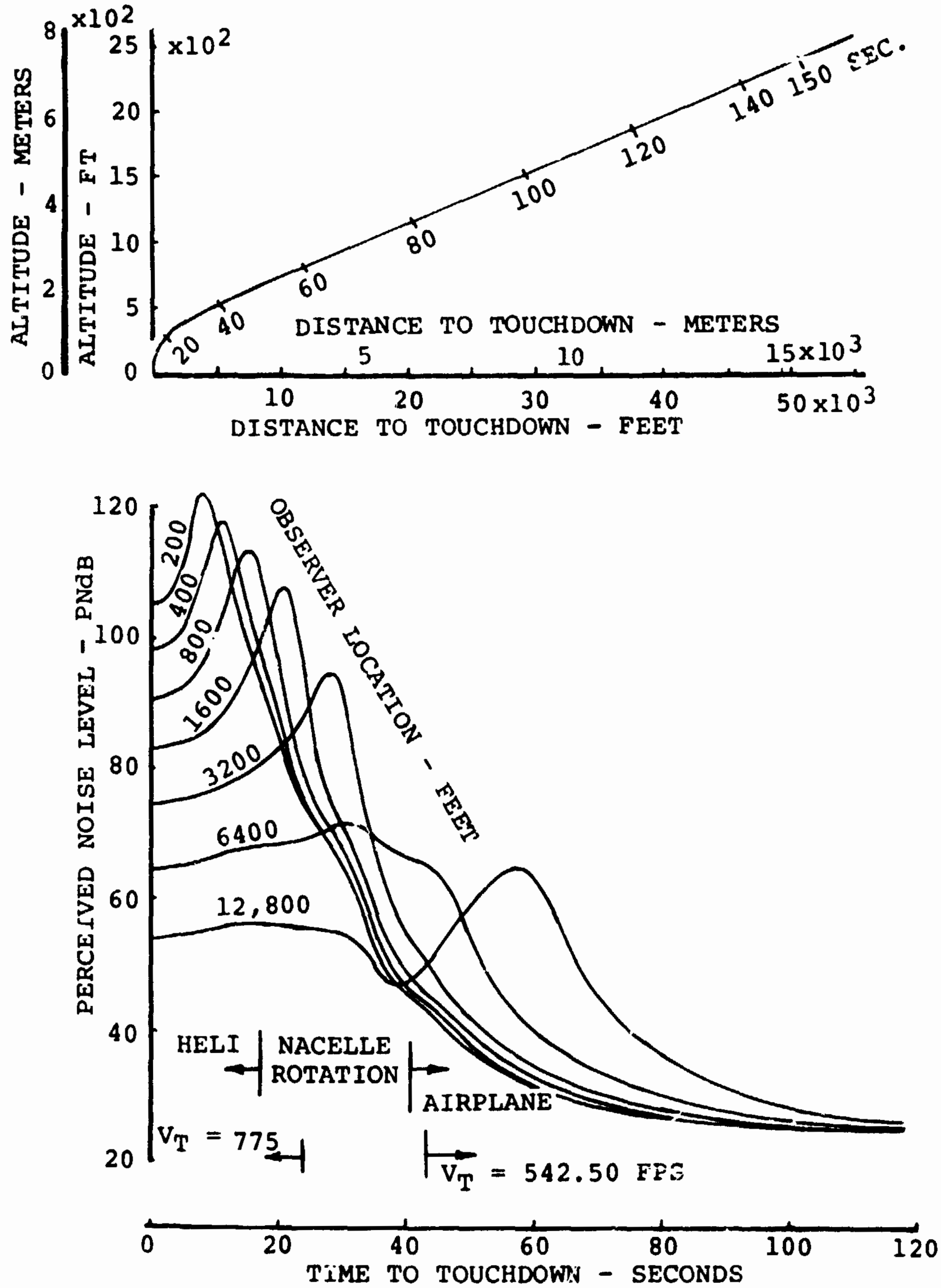


FIGURE 2.103. BASELINE TILT ROTOR DESIGN POINT STANDARD LANDING.

2.2.6 Tilt Rotor Costs

The design point tilt rotor aircraft initial costs are tabulated in Table 2.21. The fly away costs have been computed using 90 and 110 dollars per pound of airframe weight. At $90 per pound the aircraft initial cost is $5.15 million and at $110 per pound it is $5.86 million. The basic airframe costs are $3.18 million and $3.89 million respectively with dynamic system, engines and avionics costs amounting to $1.97 million.

The direct operating costs of the aircraft are also shown in Table 2.21 for utilization of 2,500 hours per year and 3,500 hours per year and for both $90 per pound and $110 per pound airframe costs.

For 2,500 hours per year and $90 per pound the direct operating cost is 2.41¢ per seat mile. This cost breaks down into 0.9¢ per seat mile for flight operations, 0.88¢ per seat mile for maintenance and 0.63¢ per seat mile depreciation.

At $110 per pound airframe cost, the direct operating cost rises to 2.54¢ per seat mile. The increase of 0.13¢ per seat mile is due to increased hull insurance costs, increased maintenance costs for airframe material and a higher depreciation cost.

With increased utilization to 3,500 hours and $90 per pound airframe cost the direct operating cost is 2.19¢ per seat mile and at $110 per pound airframe cost the direct operating

Flyaway Costs

Airframe Cost	$90.00/Lb	$110.00/Lb
Airframe	$3,179,430	$3,885,970
Dynamic System	949,920	949,920
Engines	774,416	774,416
Avionics	250,000	250,000
Total	$5,153,766	$5,860,306

Direct Operating Costs
Dollars/Seat Mile
Block Distance = 230 St. Miles

Utilization (Hrs/Yr)	2500		3500	
Airframe Cost ($/Lb)	90	110	90	110
Flying Operations				
Flight Crew	.0044	.0044	.0044	.0044
Fuel and Oil	.0033	.0033	.0033	.0033
Hull Insurance	.0013	.0015	.0009	.0011
Total Flying Operations	.0090	.0092	.0086	.0088
Direct Maintenance				
Airframe - Labor	.0013	.0013	.0013	.0013
- Material	.0011	.0014	.0011	.0014
Engines - Labor	.0006	.0006	.0006	.0006
- Material	.0008	.0008	.0008	.0008
Dynamic System - Labor	.0005	.0005	.0005	.0005
- Material	.0008	.0008	.0008	.0008
Total Direct Maintenance	.0051	.0054	.0051	.0054
Maintenance Burden	.0037	.0037	.0037	.0037
Total Maintenance	.0038	.0091	.0088	.0091
Depreciation	.0063	.0071	.0045	.0051
Total Direct Costs	.0241	.0254	.0219	.0230

TABLE 2.21. DESIGN POINT TILT ROTOR INITIAL AND DIRECT OPERATING COSTS.

TR-100(98.2)
EXTENDED RANGE VERSION

Flyaway Costs

Airframe Cost	$90.00/Lb	$110.00/Lb
Airframe	$3,195,630	$3,905,770
Dynamic System	949,920	949,920
Engines	744,416	744.416
Avionics	250,000	250,000
Total	$5,169,966	$5,880,106

Direct Operating Costs
Dollars/Seat Mile
Block Distance = 230 St. Miles

Utilization (Hrs/Yr)	2500		3500	
Airframe Cost ($/Lb)	90	110	90	110
Flying Operations				
Flight Crew	.0044	.0044	.0044	.0044
Fuel and Oil	.0033	.0033	.0033	.0033
Hull Insurance	.0013	.0015	.0009	.0011
Total Flying Operations	.0090	.0092	.0086	.0088
Direct Maintenance				
Airframe - Labor	.0014	.0014	.0014	.0014
- Material	.0012	.0014	.0012	.0014
Engines - Labor	.0006	.0006	.0006	.0006
- Material	.0008	.0008	.0008	.0008
Dynamic System - Labor	.0005	.0005	.0005	.0005
- Material	.0008	.0008	.0008	.0008
Total Direct Maintenance	.0053	.0055	.0053	.0055
Maintenance Burden	.0038	.0038	.0038	.0038
Total Maintenance	.0091	.0093	.0091	.0093
Depreciation	.0064	.0072	.0046	.0051
Total Direct Costs	.0245	.0257	.0223	.0232

TABLE 2.22. DESIGN POINT TILT ROTOR (EXTENDED RANGE VERSION) INITIAL AND DIRECT OPERATING COSTS.

BASELINE AIRCRAFT PERFORMANCE

TILT ROTOR/100 PASSENGER/98.2 PNdB

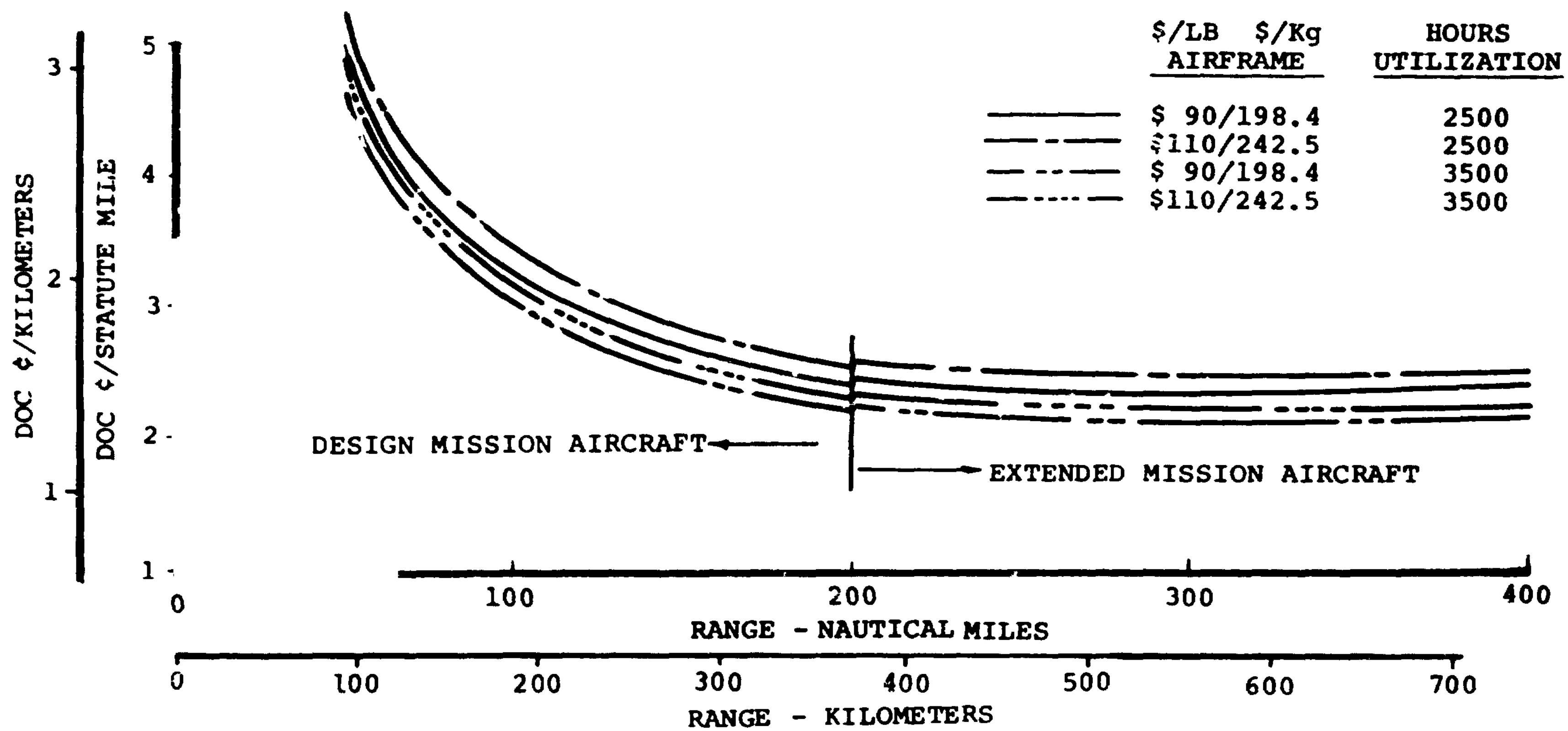


FIGURE 2.104. EFFECT OF OPERATING RANGE ON DIRECT OPERATING COST.

cost is 2.30¢ per seat mile. These reductions in DOC are due to reduced insurance and depreciation costs per seat mile since these costs are spread over more passenger miles per year at the higher level of utilization.

Table 2.22 shows similar data for a modified aircraft with increased fuel tankage to provide a 400 nautical mile range aircraft.

The aircraft fly away costs rise to $5.17 million at $90 per pounds and $5.88 million at $110 per pound due to increased aircraft weight. As shown in Figure 2.58 this aircraft can carry 99 passengers over the design mission and the direct operating cost data shown in Table 2.22 reflect 100 available seats.

Direct operating costs per seat mile and seat kilometer as a function of block distance are shown in Figure 2.104 for the specified combinations of aircraft utilization and airframe costs. Figure 2.104 also illustrates the impact of extending the design range of the TR-100 (98.2) to 460 statute miles. The increase in costs at the design point range (230 statute miles) is the result of the loss of one available seat due to the increased empty weight for the installation of larger fuel tanks. Although not shown in Figure 2.104 it should be noted that the larger fuel tanks will result in a small increase (less than 1%) in seat mile costs at ranges less than 230 statute miles due to increases in airframe maintenance and depreciation costs. In the extended range version of the TR-100 (98.2) seat mile costs show a continuing decline between 230 and 345 statute miles because the loss of eight available seats due to additional fuel requirements is offset by increasing block speed. Between 345 and 460 statute miles the delta block speeds become insufficient to offset the loss of an additional eight seats, and the seat mile costs begin to rise.

3.0 EXTERNAL NOISE CRITERIA TRADEOFF DESIGNS

One of the objectives of the design studies was to examine the effect of external noise criteria on the design of the two configurations. This is extremely pertinent since external noise and community acceptance may become governing parameters if operations with V/STOL aircraft are to achieve the advantages of potential block time savings for the short haul traveller. Such time savings will require operation from high population density urban and suburban areas as well as major airports. To evaluate design sensitivity to a noise criteria two additional aircraft of each configuration have been sized with perceived noise levels at a 500 foot sideline distance in hover which are 5 PNdB more and less noisy than the baseline aircraft.

3.1 TANDEM HELICOPTER - SELECTION OF NOISE CRITERIA DESIGNS

The primary design parameters which dictate the rotor rotational and broad band noise are tip speed and blade area or solidity. Figures 3.1, 3.2 and 3.3 show the effect of these design parameters on the aircraft gross weight, direct operating cost and sideline noise level. In these graphs solidity is plotted in terms of the ratio σ parametric aircraft/σ baseline aircraft, not absolute solidity.

The effect of decreasing solidity and increasing the tip speed reduces the aircraft design gross weight and increases the sideline perceived noise level and vice-versa. Decreasing

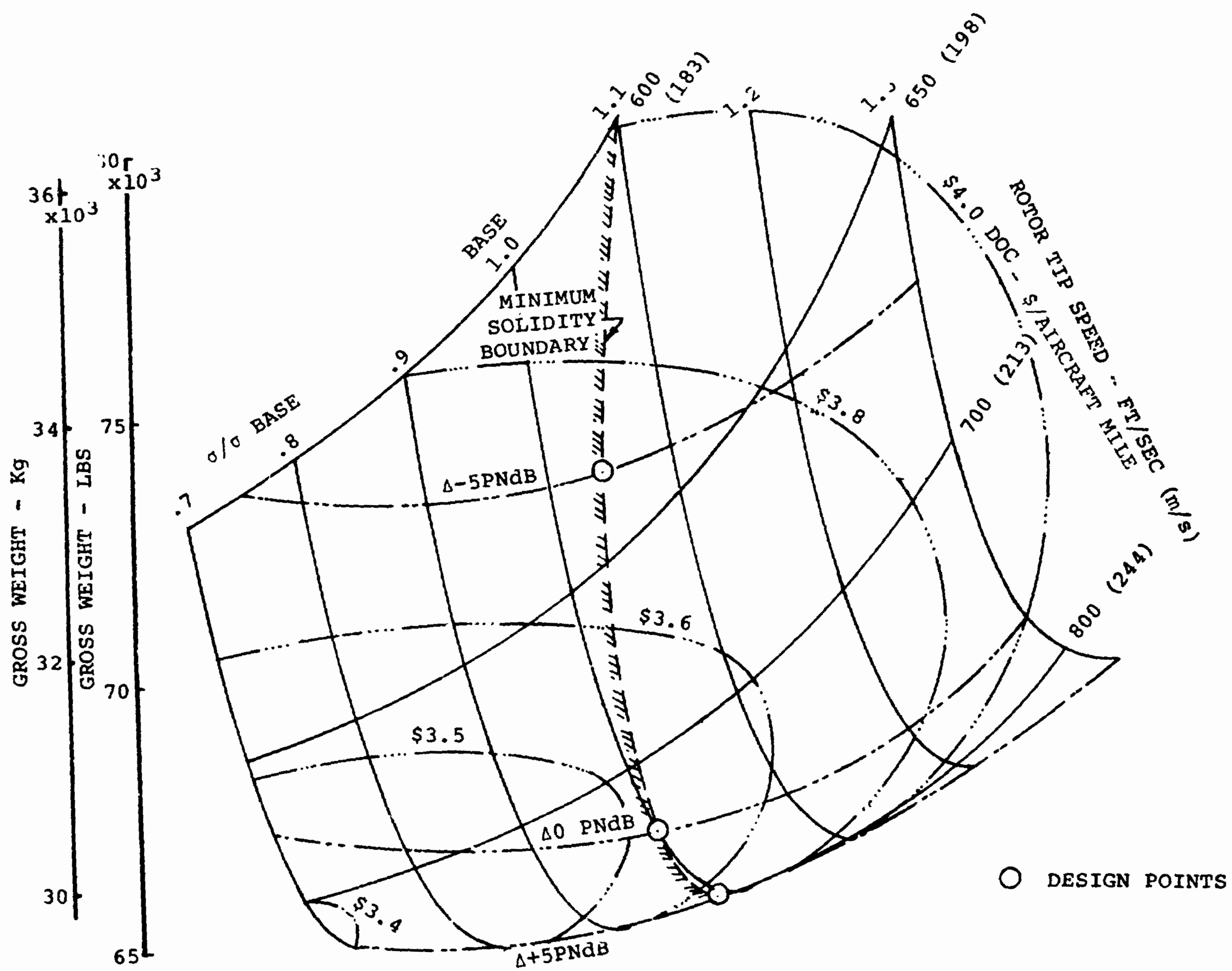


FIGURE 3.1. TANDEM HELICOPTER NOISE DERIVATIVE AIRCRAFT SELECTION CHART - GROSS WEIGHT.

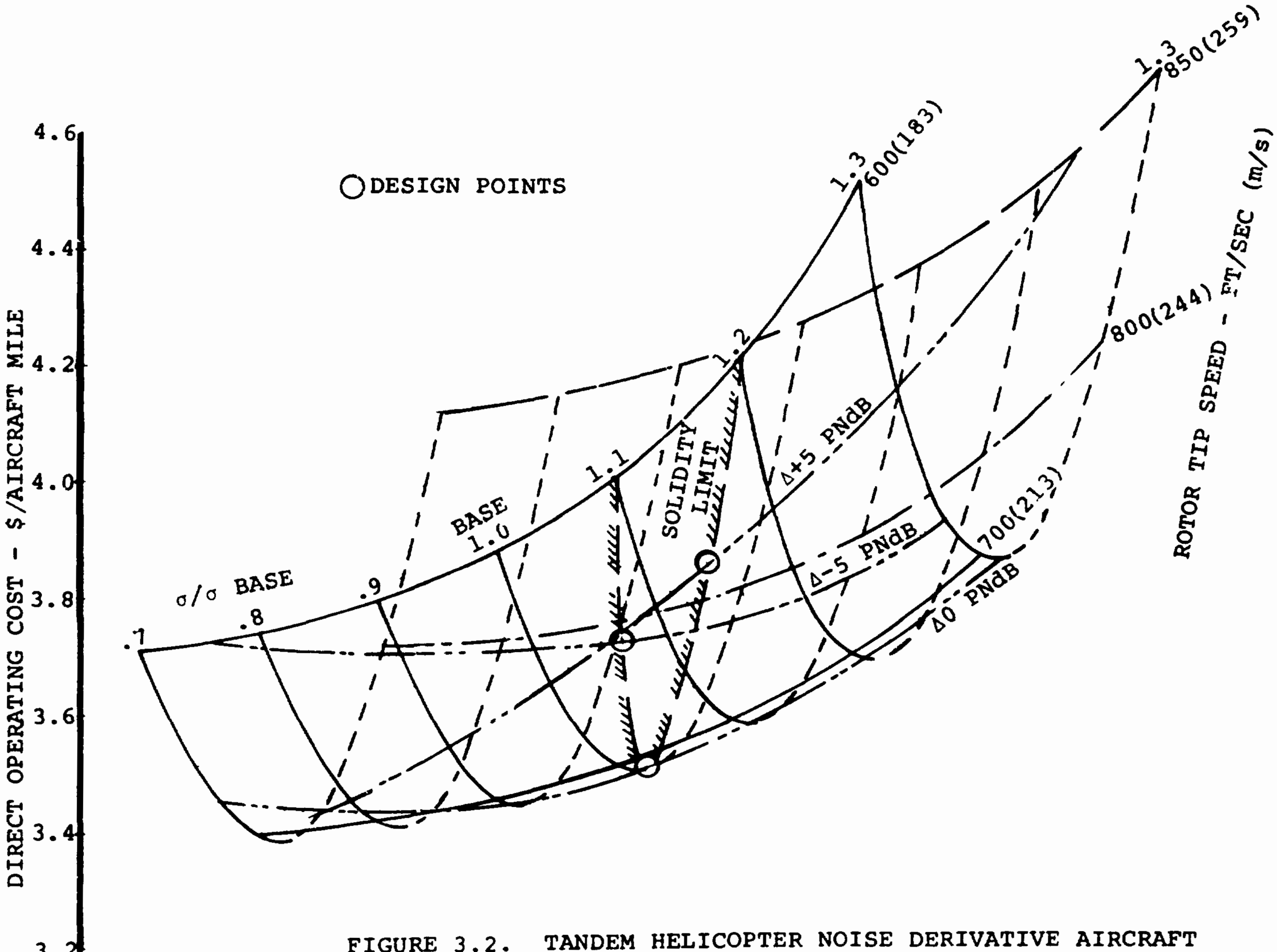


FIGURE 3.2. TANDEM HELICOPTER NOISE DERIVATIVE AIRCRAFT DESIGN SELECTION CHART - DIRECT OPERATING COST.

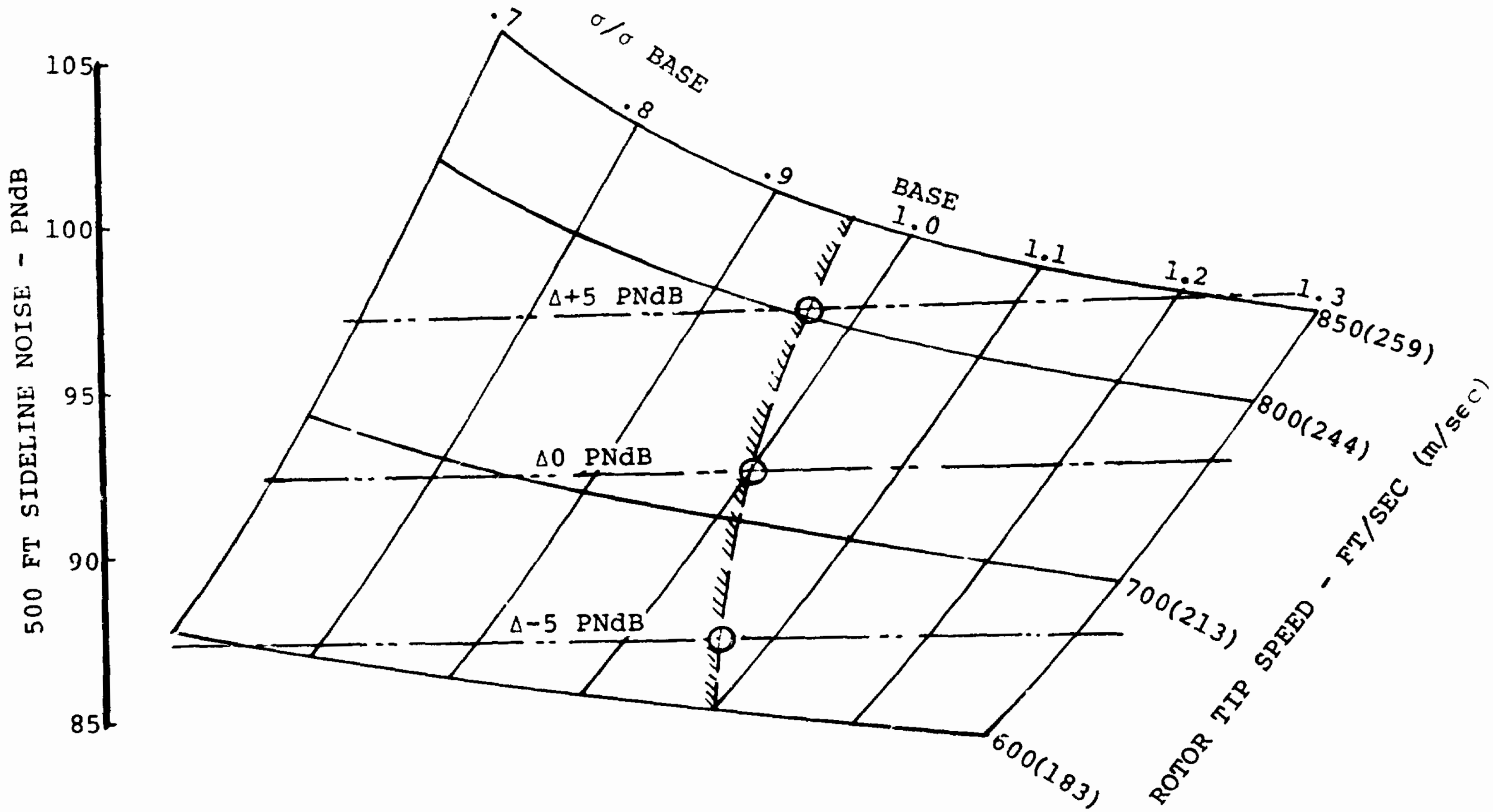


FIGURE 3.3. TANDEM HELICOPTER NOISE DERIVATIVE AIRCRAFT DESIGN SELECTION CHART - ACOUSTICS.

solidity also provided decreased direct operating costs. Departure from the baseline optimum tip speed of 720 ft/sec leads to increased direct operating costs.

The noise derivative designs were selected by striking ± 5 PNdB lines on the sideline noise plot and transporting these limits to the other plots. A minimum solidity requirement was established to maintain freedom from stall flutter with allowance for a 1.25 maneuver load factor.

The intersections of these lines define the ± 5 PNdB aircraft, the detailed characteristics of which are discussed in this section of the report.

3.1.1 Tandem Helicopter Design TH-100 (97.3)

This aircraft has a calculated sideline perceived noise level of 97.3 PNdB at 500 feet in hover, 5 PNdB more noisy than the baseline tandem helicopter.

Configuration and Layout

The characteristics of the TH-100 (97.3) tandem helicopter design are given in Table 3.1, and three view drawings shown in Figure 3.4.

The primary changes in the configuration result from an increase in tip speed to 810 feet per second and a decrease in rotor solidity to 0.07. The aircraft gross weight reduced to 65,843 pounds and the rotor diameter is reduced to 68 feet 2 inches. The pylon sweep is dictated by the decision to have zero rotor blade overlap. With a smaller rotor diameter less aft pylon sweep is required than was the case for baseline aircraft and this results in a 9 inch reduction in overall length.

The cabin and cockpit layout is exactly the same as the design point aircraft and meets the same requirements for 100 passengers. The design differences are in the rotor and installed power and transmissions. The installed power decreased to 4590 HP per engine (3 engines).

The solidity of .07 still meets the criterion of 1.25 g's maneuver capability with no stall flutter which was selected for the basic aircraft. The reduction in solidity is possible because of the higher tip speed.

	S. I. UNITS		U. S. UNITS	
WEIGHTS				
DESIGN GROSS WEIGHT	29,866	Kg	65,843	Lbs
WEIGHT EMPTY	17,305	Kg	38,152	Lbs
FUEL WEIGHT	3,495	Kg	7,705	Lbs
NUMBER OF PASSENGERS	100		100	
ROTORS				
DISC LOADING	43.94	Kg/m^2	9	Lbs/Ft^2
DIAMETER	20.8	m	68.2	Ft
SOLIDITY	.07		.07	
NUMBER OF BLADES	4		4	
TWIST	12	Degs	12	Degs
TIP SPEED	247	m/s	810	Ft/Sec
POWER				
NUMBER OF ENGINES	3		3	
RATED POWER/ENGINE	3.423×10^6	Watts	4590	SHP
FUSELAGE				
LENGTH	28.2	m	92.5	Ft
WIDTH	4.48	m	14.7	Ft
CABIN LENGTH	15.03	m	49.3	Ft
PERFORMANCE				
NRP CRUISE SPEED	67.8	m/s	141	KTAS
CRUISE ALTITUDE	1524	m	5000	Ft
BLOCK TIME	1.53	Hr	1.53	Hr
NOISE				
SIDELINE NOISE - 500 FT/HOVER	97.2	PNdB	97.2	PNdB

TABLE 3.1. +5 PNdB DESIGN POINT HELICOPTER TABLE OF CHARACTERISTICS.

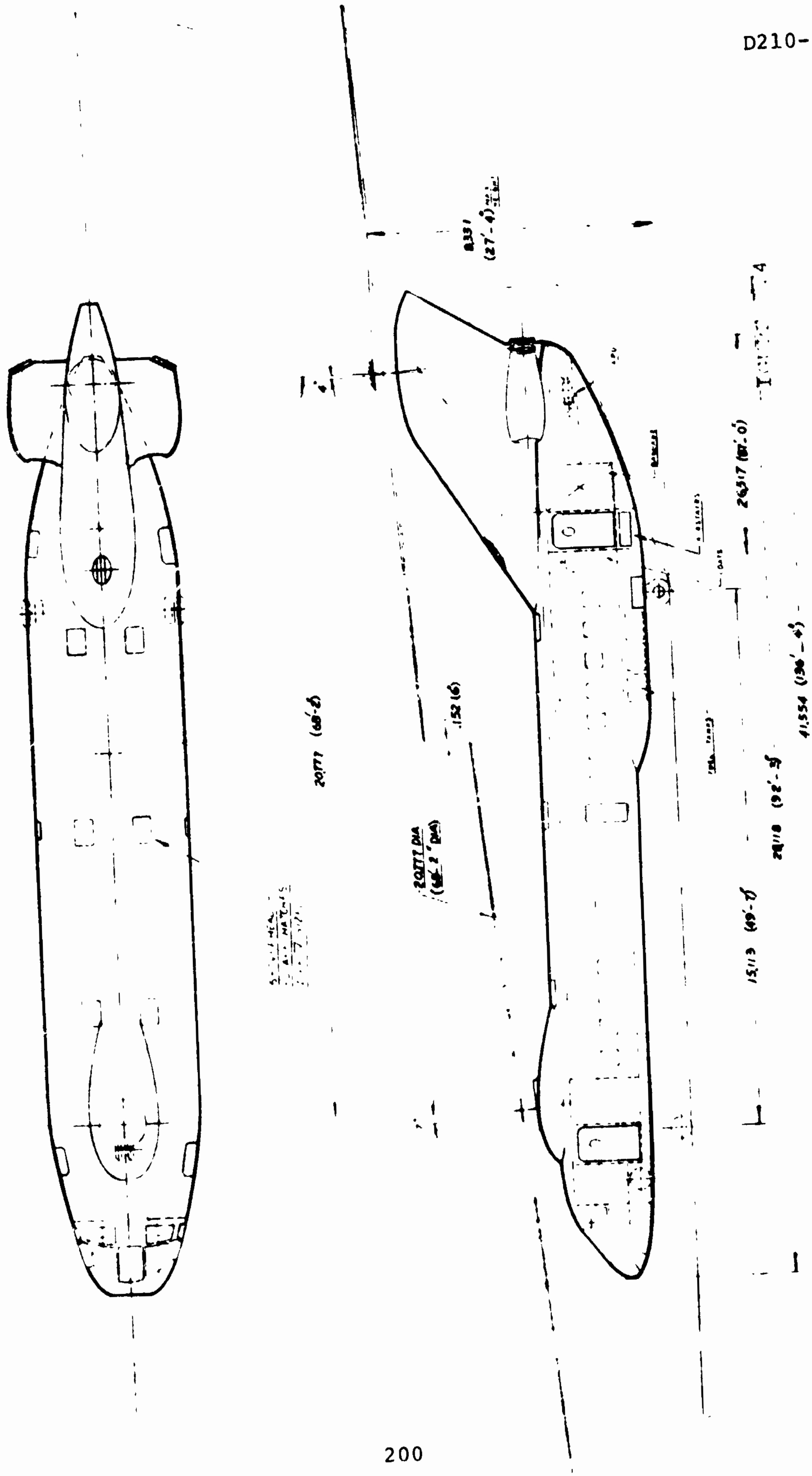


FIGURE 3.4. COMMERCIAL VTOL TRANSPORT, 100 PASSENGER - TANDEM HELICOPTER +5 PNdB DESIGN.

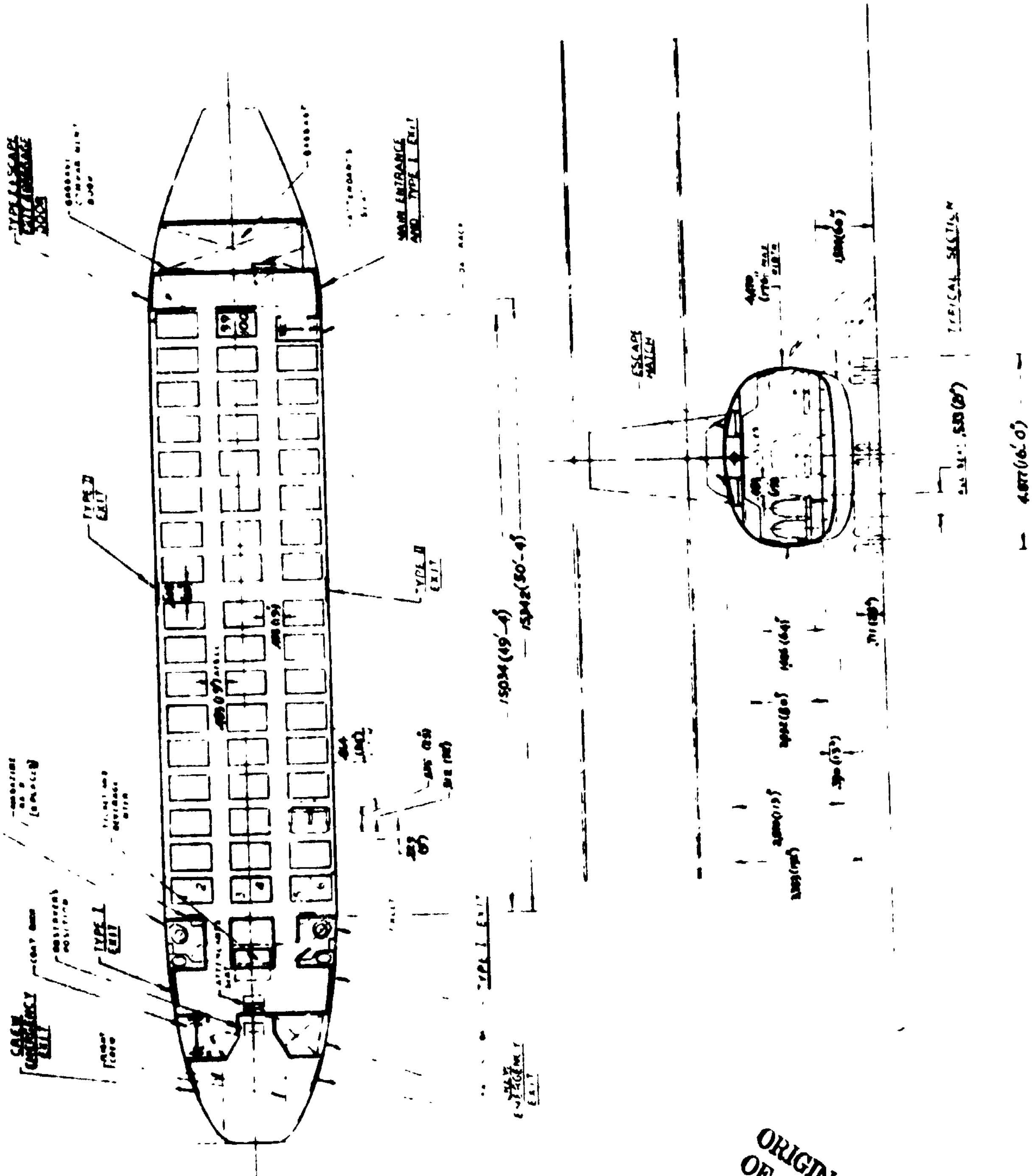


FIGURE 3.4. COMMERCIAL VTOL TRANSPORT, 100 PASSENGER - TANDEM HELICOPTER +5 PNdB DESIGN (CONTINUED).

Tandem Helicopter TH-100 (97.3) Weights

The design gross weight of the +5 PNdB aircraft is 29865.7 Kg (65,843 pounds) and the weight empty is 17,304 Kg (38,149 pounds). The aircraft weights have been established in accordance with the same groundrules as applied to the baseline aircraft. A detailed breakdown of the aircraft component and system weights is presented in Table 3.2.

The major differences in weight are in the rotor and drive systems. The reduction in diameter and solidity of the rotors reduce the rotor system weight to 2745.1 Kg (6052 pounds). The reduction in diameter also allows the distance between rotor centers to be reduced which coupled with a lighter overall gross weight reduces the bending moments in the fuselage structure and allows a reduction in body weight compared with the baseline aircraft.

The propulsion system weight is reduced by virtue of the lower installed power. The weight of the rotor flight controls is also reduced since the rotor size and inertias are smaller. The fuel required to fly the design mission is reduced since the engines are operating at a higher fraction of maximum power in cruise flight.

The aircraft center of gravity locations and principle moments of inertia are given in Table 3.3.

WEIGHT SUMMARY - PRELIMINARY DESIGN

MIL-STD-1374

	KILOGRAMS	POUNDS
WING		
ROTOR	2745.1	6052
TAIL		
SURFACES		
ROTOR		
BODY	2840.4	6262
BASIC		
SECONDARY		
ALIGHTING GEAR GROUP	1194.8	2634
ENGINE SECTION	222.7	491
PROPULSION GROUP	4211.6	9285
ENGINE INST'L	949.4	2093
EXHAUST SYSTEM *		
COOLING		
CONTROLS *		
STARTING *		
PROPELLER INST'L	*78.5	*173
LUBRICATING *		
FUEL	241.3	532
DRIVE	2942.4	6487
FLIGHT CONTROLS	718.5	1584
AUX. POWER PLANT	288.5	636
INSTRUMENTS	[illegible]	423
HYDR. & PNEUMATIC	306.4	680
ELECTRICAL GROUP	378.3	834
AVIONICS GROUP	293.9	648
ARMAMENT GROUP		
FURN. & EQUIP. GROUP	3206.9	7070
ACCOM. FOR PERSON.		
MISC. EQUIPMENT		
FURNISHINGS		
EMERG. EQUIPMENT		
AIR CONDITIONING	521.6	1150
ANTI-ICING GROUP	181.4	400
LOAD AND HANDLING GP.		
WEIGHT EMPTY	17304.0	38149
CREW	299.4	660
TRAPPED LIQUIDS	52.2	115
ENGINE OIL	59.9	132
CREW ACCOMMODATIONS	68.0	150
EMERGENCY EQUIPMENT	7.3	16
PASSENGER ACCOMMO.	415.5	916
PASSENGERS (100)	8164.6	18000
FUEL	3494.9	7705
GROSS WEIGHT	29865.7	65843

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TABLE 3.2 +5 PNBR HELICOPTER WEIGHT SUMMARY

	WEIGHT EMPTY	GROSS WEIGHT
WEIGHT	17,304 Kg (38,149 Lbs)	29,865.7 Kg (65,843 Lbs)
CENTER OF GRAVITY*		
Fuselage Station	15.26 M (600.6 Inches)	14.53 M (571.9 Inches)
Water Line	3.60 M (141.6 Inches)	2.83 M (111.6 Inches)
MOMENT OF INERTIA		
I_{xx} (Roll)	13,068.3 Kg M^2 (9,631.0 Slugs Ft^2)	14,506.2 Kg M^2 10,690.7 Slugs Ft^2)
I_{yy} (Pitch)	1,438,857.3 Kg M^2 (1,060,400.4 Slugs Ft^2)	1,597,175.6 Kg M^2 (1,177,076.8 Slugs Ft^2)
I_{zz} (Yaw)	1,389,510.0 Kg M^2 (1,024,032.7 Slugs Ft^2)	1,542,398.3 Kg M^2 (1,136,707.4 Slugs Ft^2)

*FUSELAGE STATION O IS AT NOSE OF BODY - CENTER LINE OF FORWARD ROTOR 5.0 METERS ABOVE WATER LINE.

TABLE 3.3 . WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA - +5 PNdB HELICOPTER.

Tandem Helicopter - TH-100 (97.3) - Vehicle Performance

The +5 PNdB tandem helicopter has been sized for 100 passengers over the 200 nautical mile range with the same mission definition as the baseline aircraft. Tables 3.4 and 3.5 shows a summary of the mission performance and detailed calculation results are included in Volume II.

The aircraft takeoff gross weight is 29,860.8 Kg (65,843 pounds) with a fuel load of 3,494.3 Kg (7,705 pounds). The taxi, takeoff and initial air maneuver requires 31 Kg (113 pounds)of fuel. In Tables 3.4 and 3.5 the initial air maneuver is included with the takeoff. The aircraft then climbs to altitude (5,000 feet) for a range benefit of 9.1 Km (4.9 nautical miles) using 126 Kg (231 pounds) of fuel to climb.

The mission cruise is done at normal rated power at 5,000 feet at an average airspeed of 145 knots. The cruise segment fuel is 2,747 Kg (5,053 pounds) of fuel for a range of 367.3 Kg (198.2 nautical miles).

The remaining 0.8 nautical miles is accomplished in the descent to 2,000 feet at a rate of descent of 15.6 m/s (3,078 feet per minute).

The remainder of the mission is an air maneuver at 2,000 feet, descent, landing and taxi. The mission fuel used is 3199 Kg (5,598 pounds) with 1,204 Kg (2,107 pounds) reserve fuel.

	TIME (HOURS)	DISTANCE (NMI)	WEIGHT (LBS)	FUEL (LBS)	V (KNOTS)	R/C (FT/MIN)
	0	0	65,843		0	0
TAXI				11		
	.017	0	65,832			
TAKEOFF				102	0	0
	.042	0	65,730			
CLIMB				231	81	1400
	.101	4.9	65,499			
CRUISE				5,053	145	0
	1.434	198.20	60,446			
DESCENT				18	115	-3078
	1.451	200	60,428			
AIR MANEUVER				67	80	0
	1.476	200	60,361			
DESCENT				35	70	-1000
	1.492	202	60,326			
LANDING				61	0	0
	1.509	202	60,265			
TAXI				8	0	0
	1.526	202	60,253			
RESERVE				2,107		0
	2.217	250	58,038			0

TABLE 3.4. MISSION SUMMARY +5 PNdB DESIGN POINT HELICOPTER (U.S. UNITS).

	TIME (HOURS)	DISTANCE (Km)	WEIGHT (Kg)	FUEL (Kg)	V (KNOTS)	R/C (m/s)
	0	0	35,792			
TAXI				6	0	0
	.017	0	35,786			
TAKEOFF				55	0	0
	.042	0	35,730			
CLIMB				126	81	7.1
	.101	9.1	35,605			
CRUISE				2,747	145	0
	1.434	367.3	32,858			
DESCENT				10	115	-15.6
	1.451	370.6	32,848			
AIR MANEUVER				36	80	0
	1.476	370.6	32,812			
DESCENT				19	70	-5.1
	1.492	374.3	32,793			
LANDING				33	0	0
	1.509	374.3	32,759			
TAXI				6	0	0
	1.526	374.3	32,753			
RESERVE				1,204		0
	2.217	463.3	31,549			

TABLE 3.5. MISSION SUMMARY +5 PNdB DESIGN POINT HELICOPTER (S. I. UNITS).

Hover Performance

The hover performance of the helicopter is shown in Figures 3.5 and 3.6 in terms of sea level gross weight lifting capability as a function of ambient temperature.

Data are shown for all engines operating (AEO) Figure 3.5, and one engine inoperative, both inground effect (IGE) and out of ground effect (OGE). The power levels are given on both figures. For the all engines operating data the three engine takeoff power is shown, for the OEI data two engines operating at 9% increased power are shown.

The takeoff gross weights were derived using a force to weight ratio (F/W) of 1.05 for all engines operating and F/W of 1.03 for OEI conditions in compliance with the study guidelines.

The aircraft sizing conditions were OEI at sea level 90°F and is indicated in Figure 3.6 giving a design gross weight of 29,866 Kg (65,843 pounds) out of ground effect.

Transmission rating of the vehicle allows full takeoff power from all three engines to be used at sea level standard. This condition is indicated in Figure 3.5.

Hover gross weight capability for altitudes up to 20,000 feet is shown in Figures 3.7 and 3.8. Data are presented for IGE and OGE conditions on standard day and standard day plus 31°F (17.2°C), ambient conditions at sea level standard day plus 31°F is 90°F.

Figure 3.8, the OEI data, indicates the design point sizing condition. On a standard day the vehicle is capable of OGE

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/97.2 PNdB

ALL ENGINES OPERATING

+5 PNdB

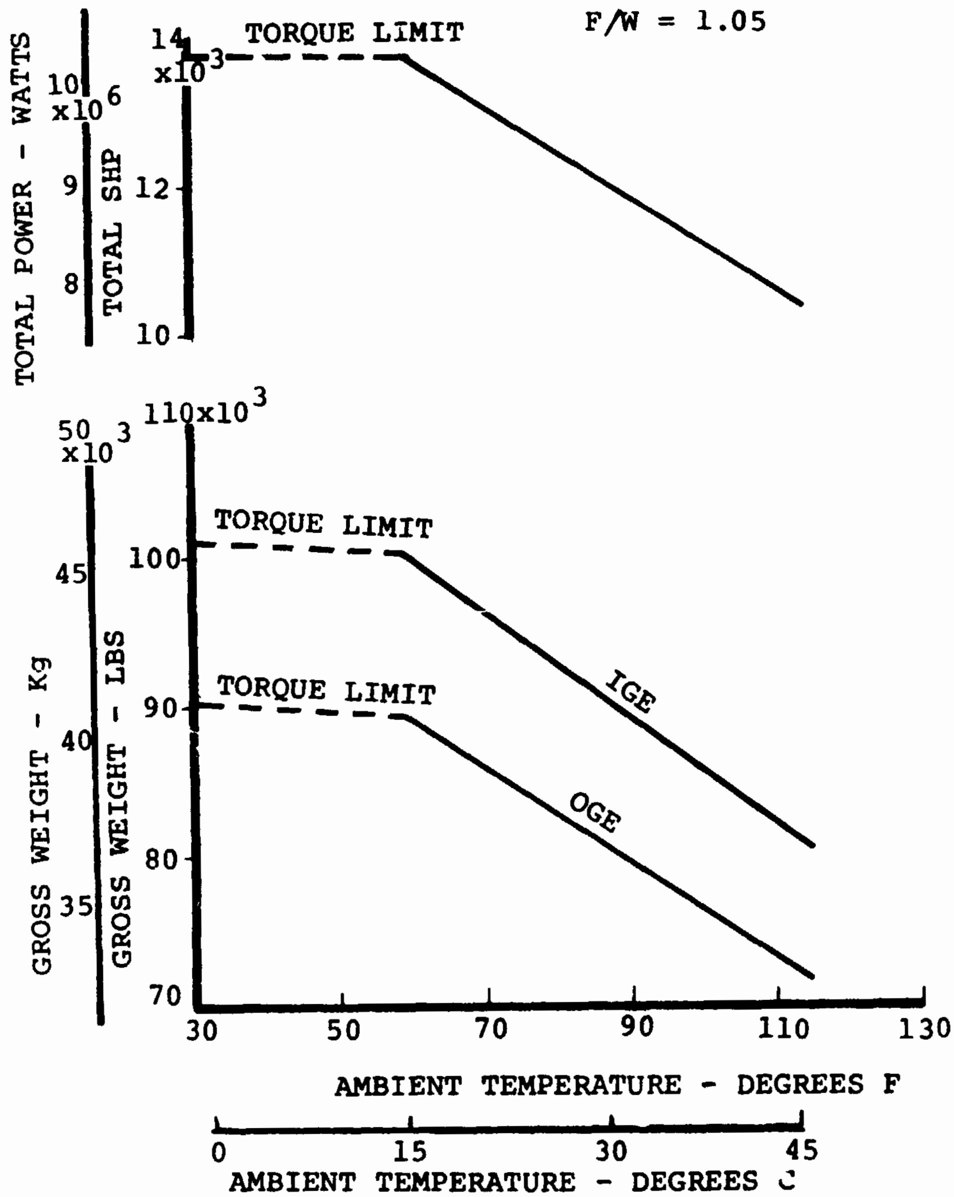


FIGURE 3.5. EFFECT OF AMBIENT TEMPERATURE ON SEA LEVEL HOVER PERFORMANCE OF +5 PNdB HELICOPTER

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/97.2 PNdB

ONE ENGINE INOPERATIVE

+5PNdB

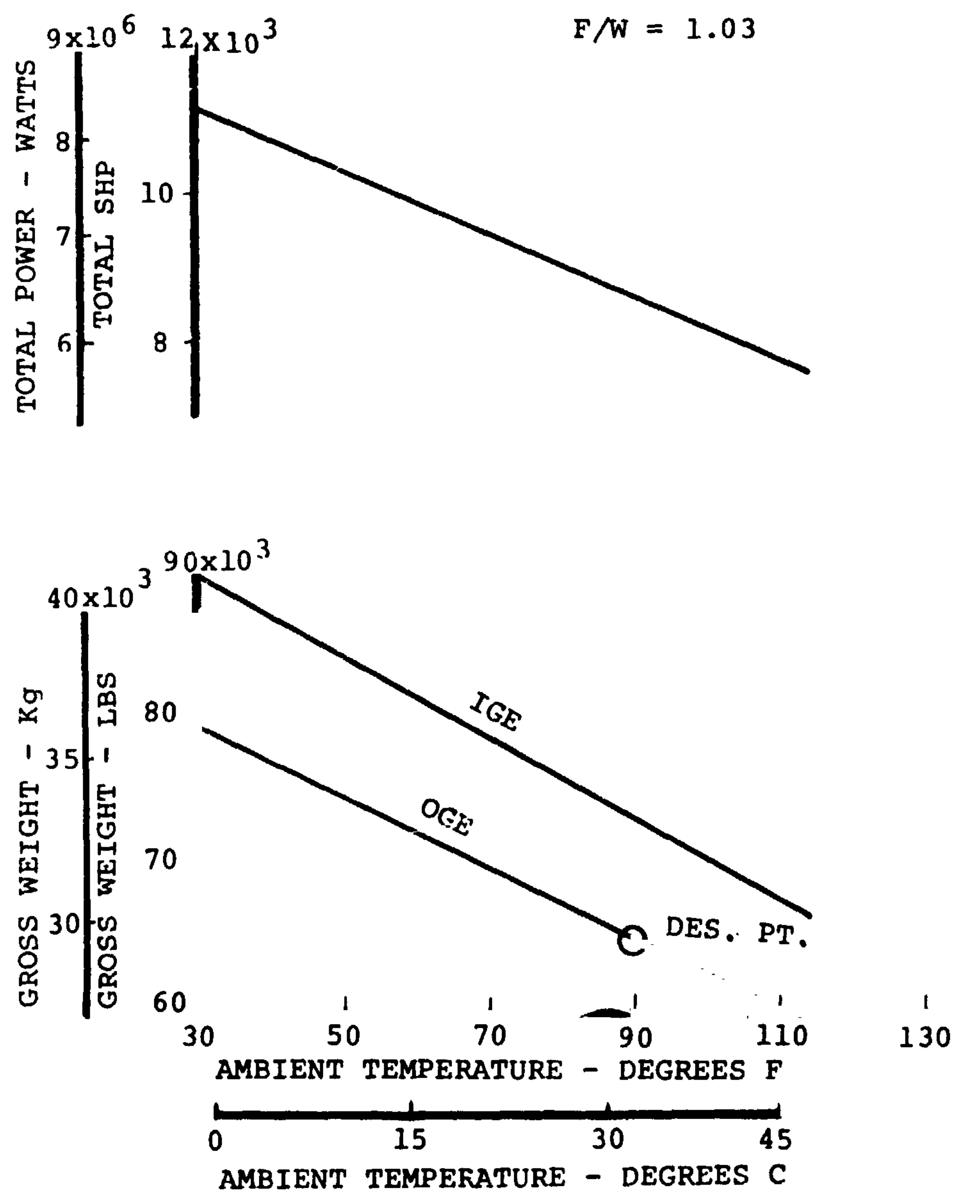


FIGURE 3.6 EFFECT OF AMBIENT TEMPERATURE ON SEA LEVEL HOVER PERFORMANCE OF +5 PNdB HELICOPTER.

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/97.2 PNdB

STANDARD DAY AND STANDARD DAY + 31°F (+17.2°C)

ALL ENGINES OPERATING
+5 PNdB

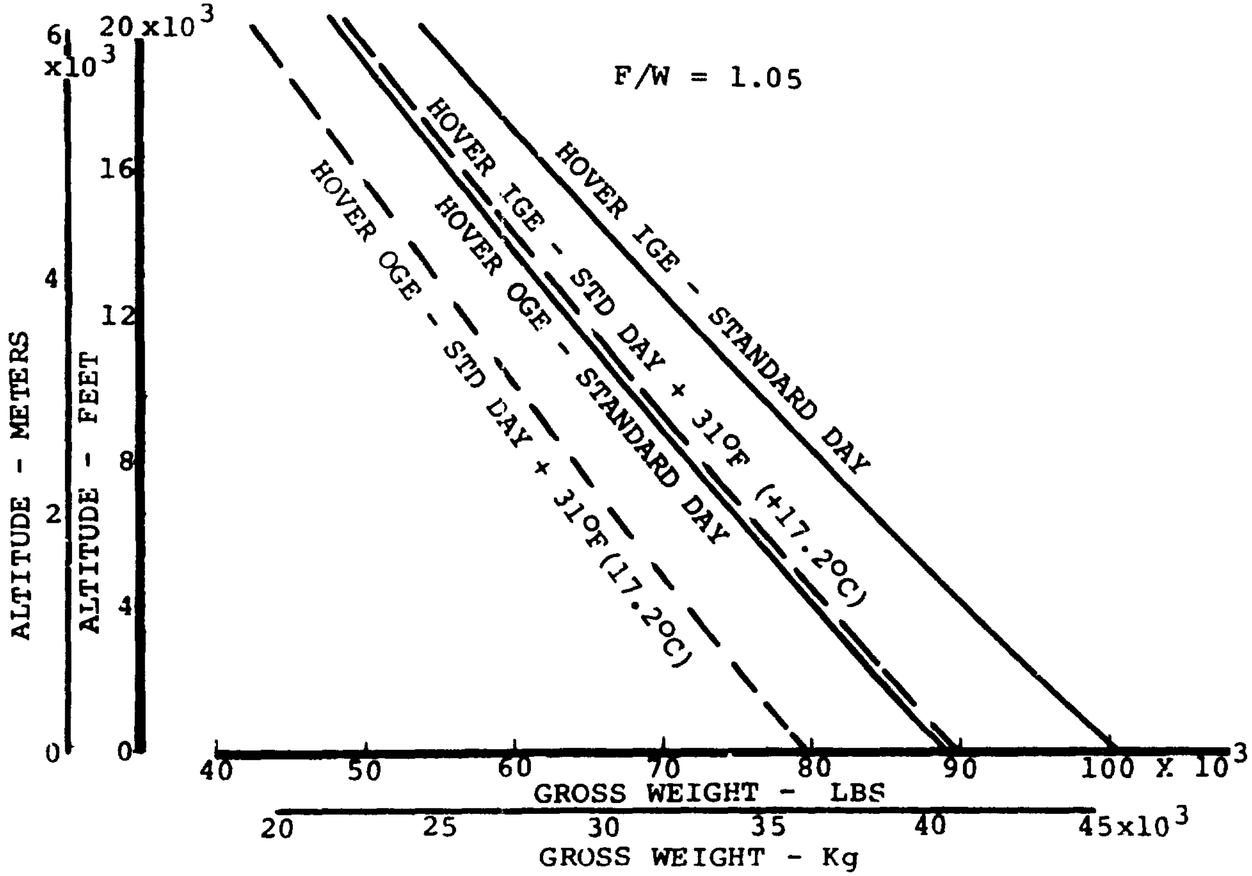


FIGURE 3.7. EFFECT OF ALTITUDE ON HOVER GROSS WEIGHT CAPABILITY. - +5 PNdB HELICOPTER

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/97.2 PNdB

STANDARD DAY AND STANDARD DAY + 31°F (+17.2°C)

ONE ENGINE INOPERATIVE

+5 PNdB

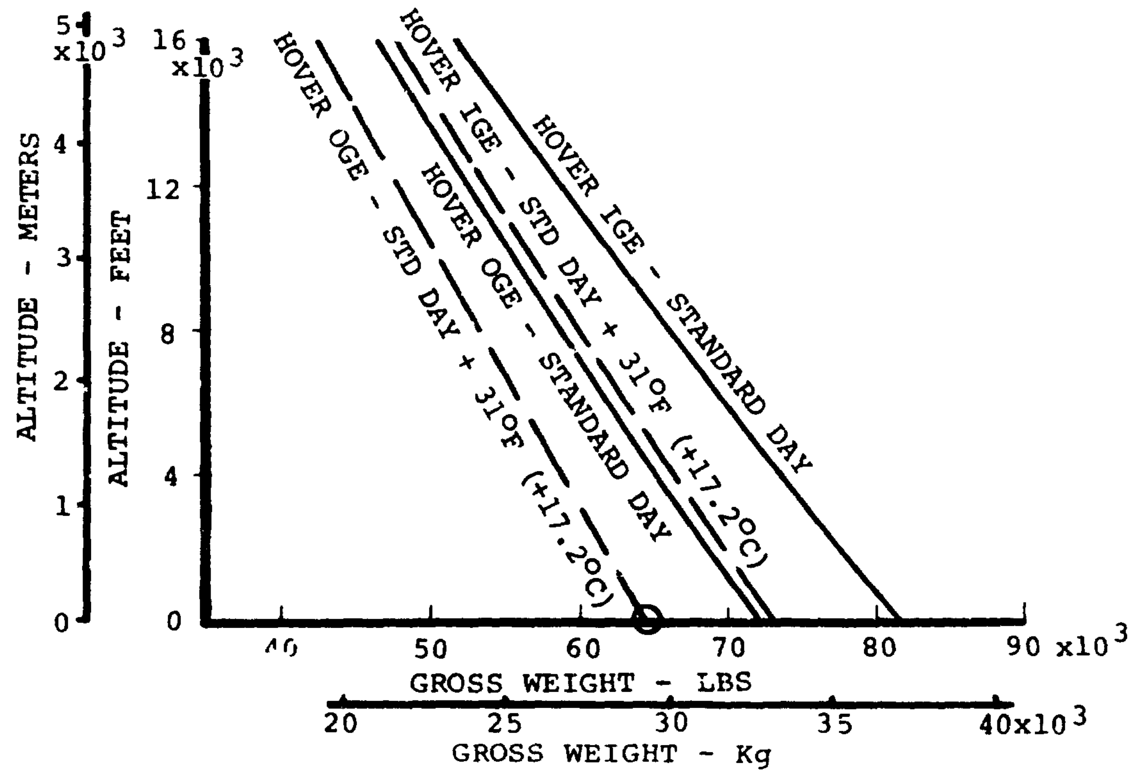


FIGURE 3.8. EFFECT OF ALTITUDE ON HOVER GROSS WEIGHT CAPABILITY.

takeoffs at altitudes up to 5,000 feet with full passenger load of 100 passengers.

All engine operating performances given in Figure 3.7 indicate that OGE takeoffs can be made up to 11,000 feet on a standard day plus 31°F.

Performance in Forward Flight TH-100 (97.3)

The power required in trimmed level flight is shown in Figure 3.9 at sea level for three aircraft weights. The power available at normal rated power (NRP) with all engines operating and one engine inoperative are superimposed. At zero forward speed the aircraft can hover at less than normal rated power with all engines operating. The intersections of the power required and power available lines give the maximum cruise speed limits for various weights. At design gross weight the aircraft cruises at 146 knots with all engines operating. With one engine inoperative a cruise speed of 130 knots can be maintained at design gross weight.

These data are also shown for 5,000 feet altitude, standard day in Figure 3.10. Even at this altitude the aircraft can maintain hover at less than NRP at design gross weight. At 5,000 feet altitude the maximum cruise speed at normal rated power and design gross weight is 141 knots and 165 knots at operating weight empty. With one engine inoperative the aircraft can maintain 118 knots and can fly as slow as 50 knots at NRP and design gross weight.

The aircraft maximum cruise speeds are shown as a function of altitude in Figure 3.11.

Climb Performance

The aircraft rate of climb data are shown in Figure 3.12. At design gross weight the rate of climb capability all engines operating is 3,200 feet per minute falling off to 2,450 feet

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/97.2 PNdB

STANDARD DAY CRUISE RPM

DGW = 65,843 LBS/29,966 Kg
MIDWT = 52,991 LBS/24,036 Kg
OWE = 40,139 LBS/18,209 Kg

+5 PNdB

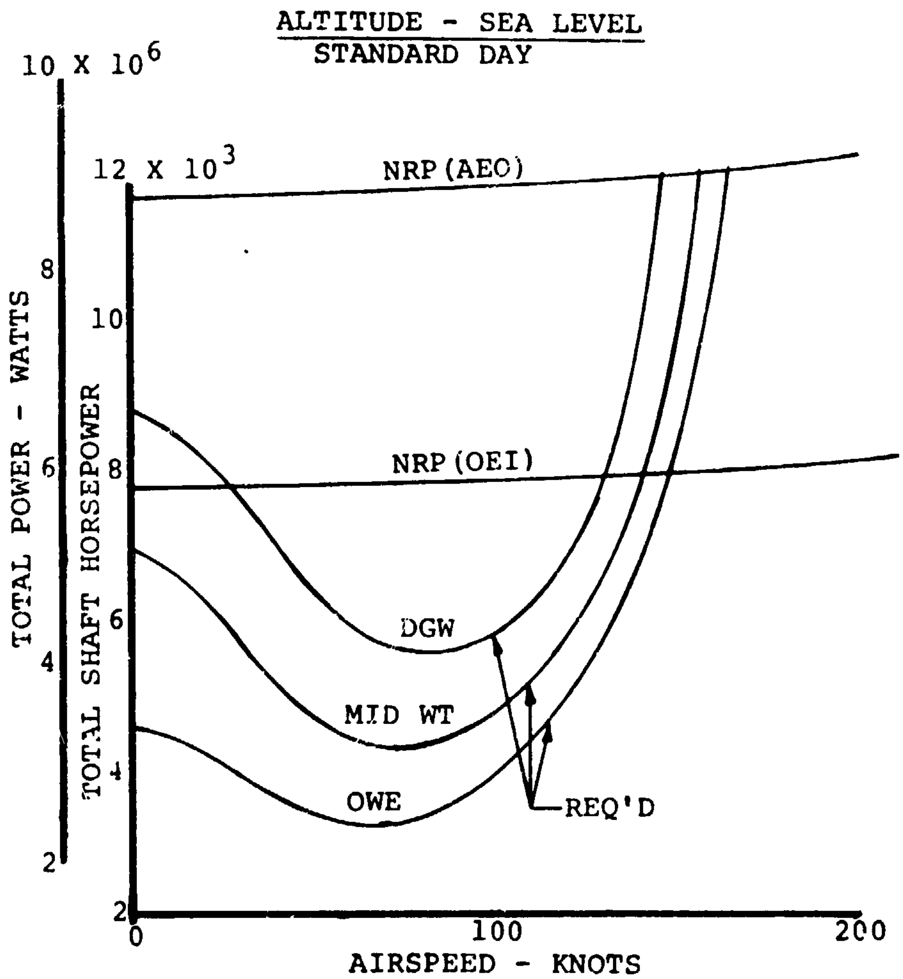


FIGURE 3.9. CRUISE PERFORMANCE - POWER REQUIRED/AVAILABLE. +5 PNdB HELICOPTER

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/97.2 PNdB

STANDARD DAY CRUISE RPM

DGW = 65,843 LBS/29,866 Kg
MIDWT = 52,991 LBS/24,036 Kg
OWE = 40,139 LBS/18,209 Kg

+5 PNdB

ALTITUDE - 5000 FEET (1524 m)
STANDARD DAY

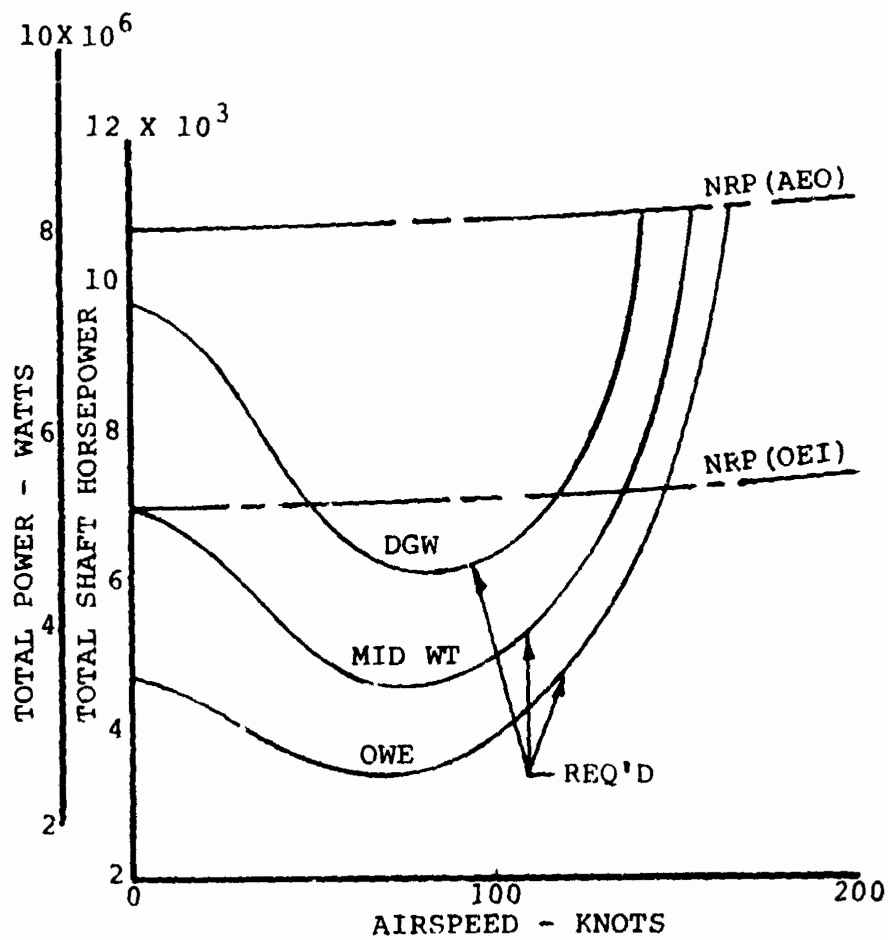


FIGURE 3.10. CRUISE PERFORMANCE - POWER REQUIRED/AVAILABLE.

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/97.2 PNdB

STANDARD DAY - NORMAL RATED POWER - AEO & OEI -
CRUISE & HOVER RPM

DGW = 65,843 LBS/29,866 Kg
OWE = 40,139 LBS/18,207 Kg

+5 PNdB

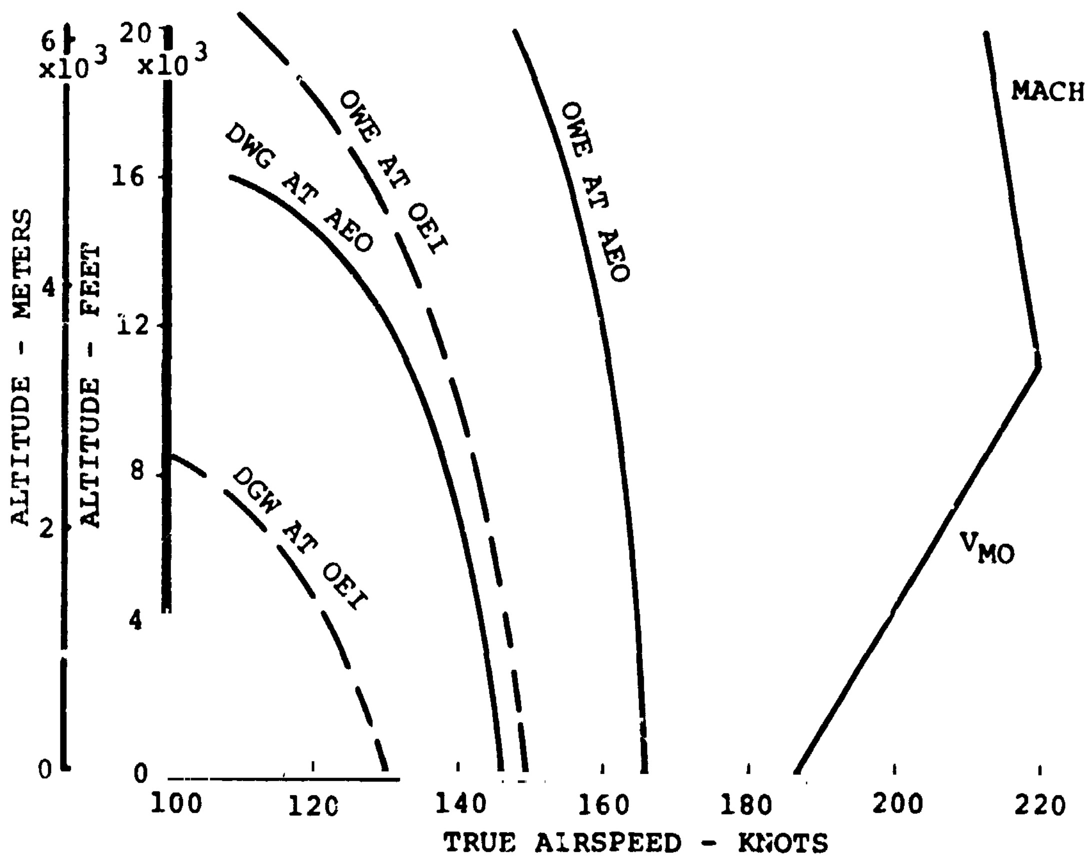


FIGURE 3.11. LEVEL FLIGHT CRUISE SPEED ENVELOPE
+5 PNdB HELICOPTER

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/97.2 PNdB

CLIMB CAPABILITY - TAKEOFF RPM - MIL POWER

DGW = 65,843 LBS/29,866 Kg
OWE = 40,139 LBS/18,207 Kg

+5 PNdB

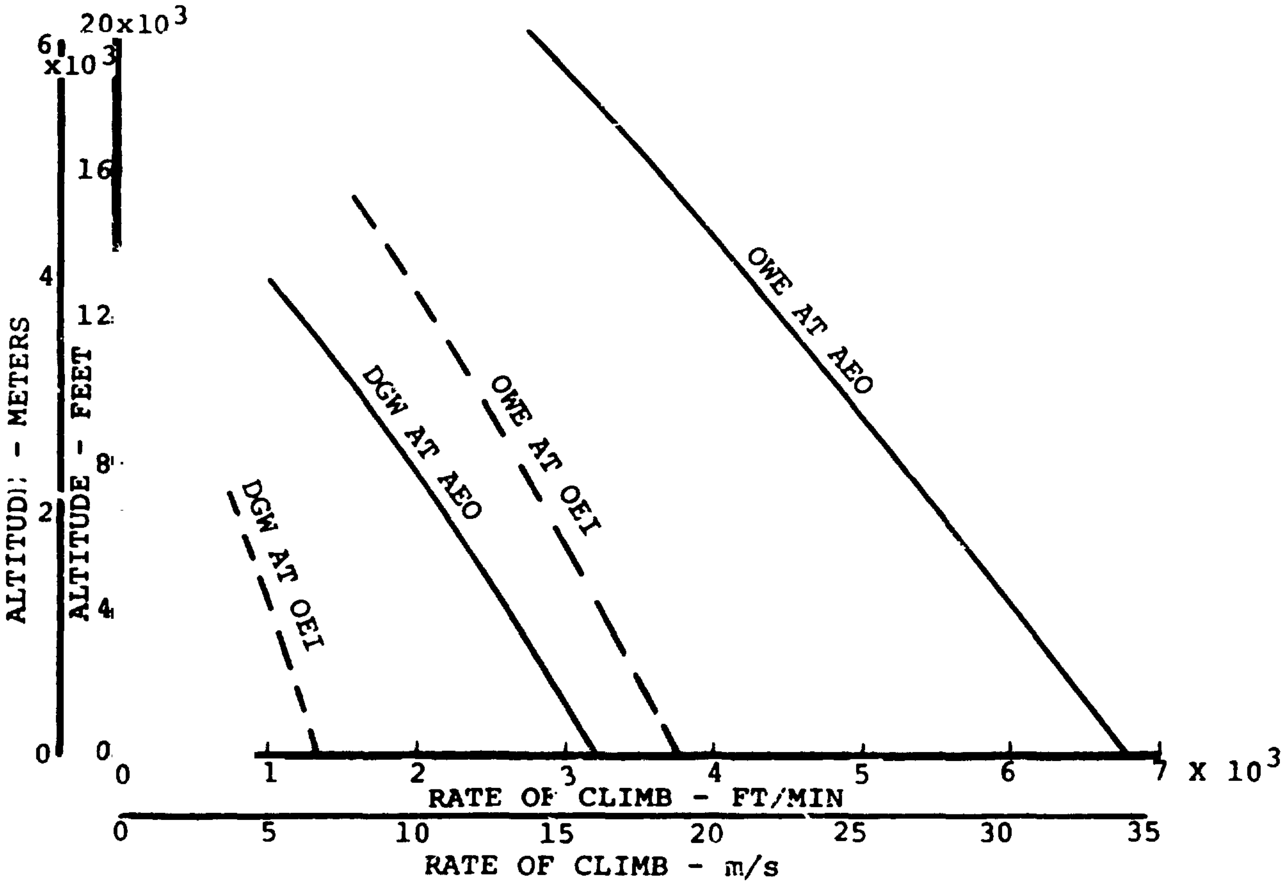


FIGURE 3.12. +5 PNdB DESIGN POINT HELICOPTER - CLIMB CAPABILITY.

per minute at design cruise altitude (5,000 feet). With one engine inoperative a rate of climb of 1,350 feet per minute can be achieved at sea level and 950 feet per minute at 5,000 feet altitude.

The rate of climb is sensitive to aircraft weight and at operating weight empty a maximum rate of climb of 6,770 feet per minute can be achieved at sea level.

Range

The specific range performance is given at both sea level and 5,000 feet altitude for all engines operating in Figure 3.13 and for one engine inoperative in Figure 3.14. At sea level a maximum specific range of .038 nautical miles per pound of fuel is obtained at design gross weight at an airspeed of 133 knots. At NRP the specific range for the same weight and altitude reduces to .037 nautical miles per pound of fuel. At 5,000 feet the NRP specific range is marginally increased to .0375 nautical miles per pound of fuel at design gross weight. The maximum specific range is 0.049 at 5,000 feet altitude and minimum weight at an airspeed of 128 knots.

With one engine inoperative the maximum specific ranges improve because of the increased power setting on the remaining two engines with an attendant reduction in specific fuel consumption.

The payload range performance of the aircraft with all engines operating is specified by the mission definition, shown in Figure 3.15. With basic mission fuel and no payload the range capability increases to 243 nautical miles. The extended range condition shown on Figure 3.15 is the same aircraft with additional tankage for a 400 nautical mile range. The additional tank weight reduces the payload by the equivalent of 2 passengers at 200 nautical miles and the aircraft can operate to 400 nautical miles with 70 passengers, while maintaining the basic mission reserve fuel.

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/97.2 PNdB

STANDARD DAY CRUISE RPM
DGW = 65,843 LBS/29,866 Kg
MIDWT = 52,991 LBS/24,036 Kg
OWE = 40,139 LBS/18,207 Kg

+5 PNdB

ALL ENGINES OPERATING

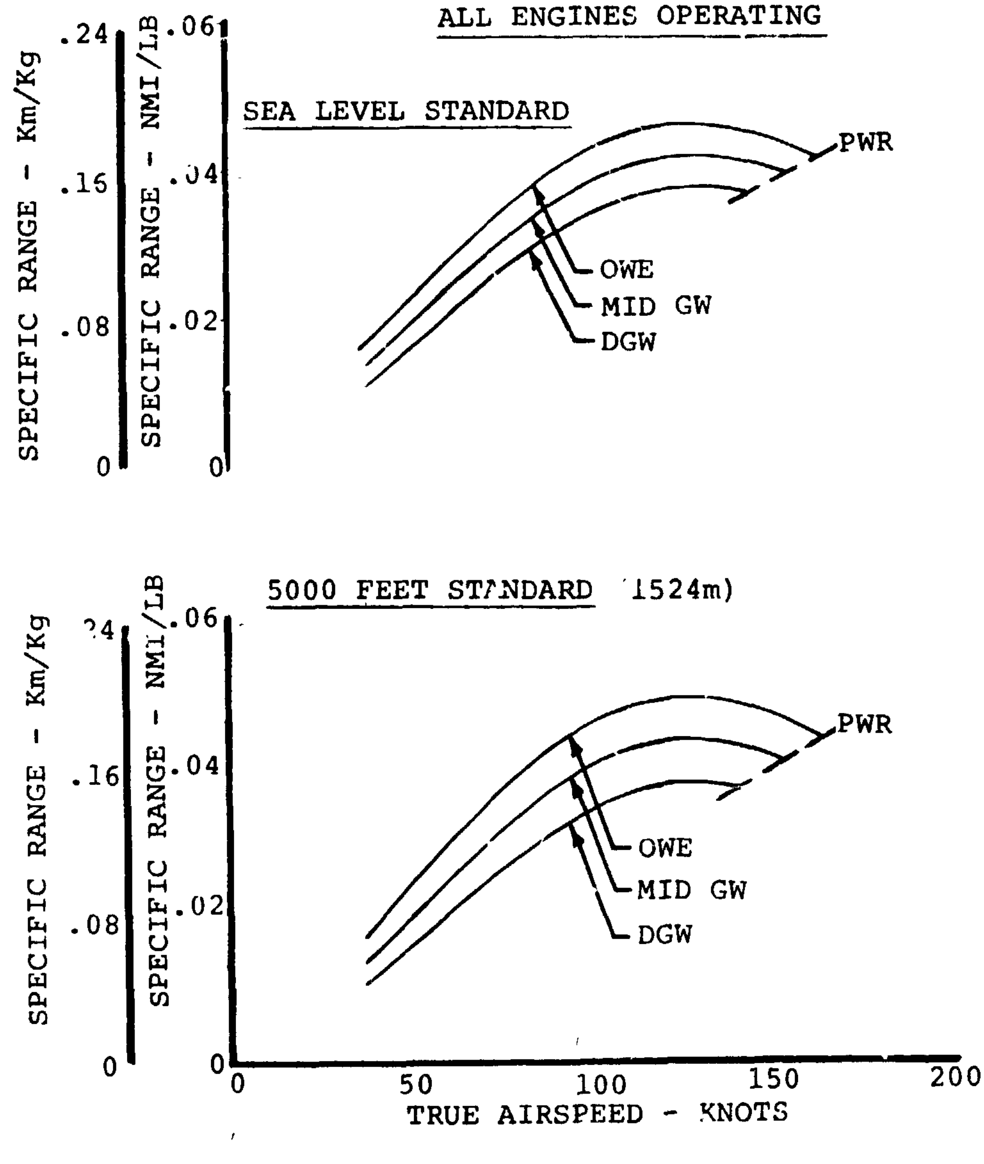


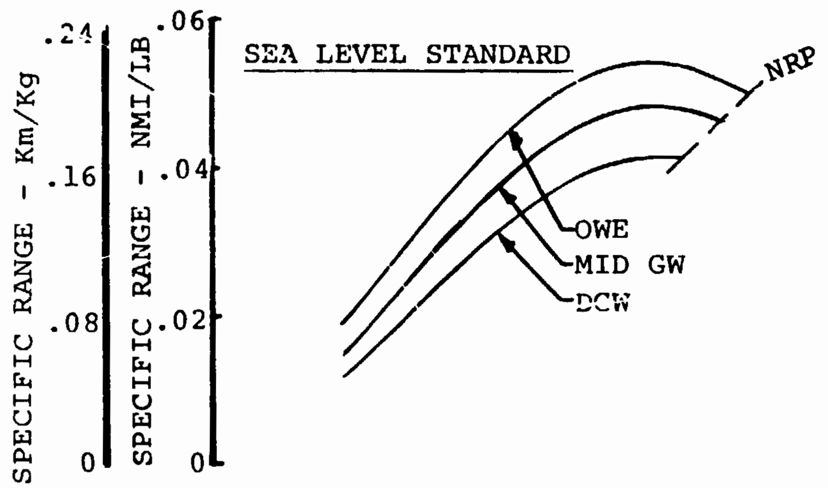
FIGURE 3.13 CRUISE PERFORMANCE - SPECIFIC RANGE. +5 PNdB HELICOPTER

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/97.2 PNdB
DGW = 65,843 LBS/29,866 Kg
MIDWT = 52,991 LBS/24,036 Kg
OWE = 40,139 LBS/18,207 Kg

ONE ENGINE INOPERATIVE

+5 PNdB



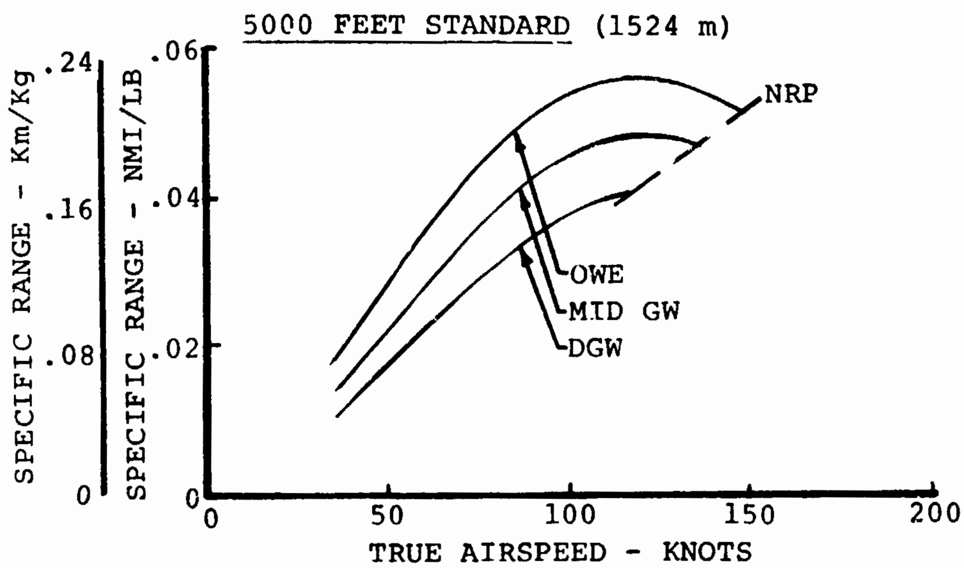


FIGURE 3.14. CRUISE PERFORMANCE - SPECIFIC RANGE - +5PNdB HELICOPTER.

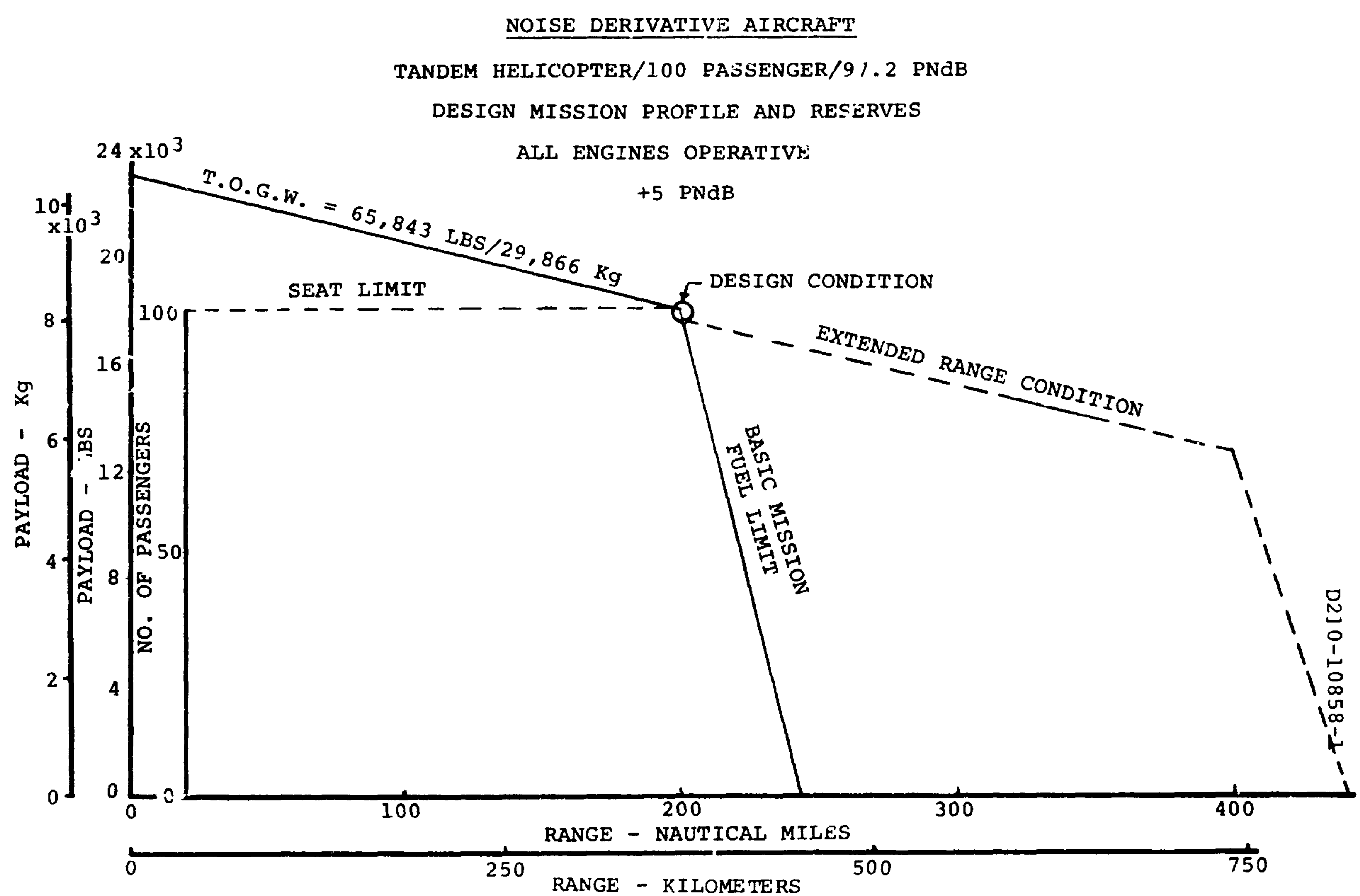


FIGURE 3.15. PAYLOAD RANGE CAPABILITY - CRUISE AT NRP (AEO) +5 PNdB HELICOPTER.

With one engine inoperative the range performance, Figure 3.16, of the basic TH-100 (97.3) design improves to 230 nautical miles by virtue of improved specific range as shown in Figures 3.13 and 3.14.

STABILITY AND CONTROL TH-100 (97.3)

The guideline requirement to have positive static stability is achieved in this case by delta 3 in the forward rotor head as was the case for the baseline aircraft. Figure 3.17 shows M_α for the unaugmented aircraft with a negative M_α at speeds in excess of 130 knots at all weights.

The lateral-directional stability is positive at all speeds (i.e., + ve N_β) Figure 3.17.

Table 3.6 summaries the derivatives for this aircraft and compares them against the baseline tandem helicopter design. The control derivatives are slightly lower for this aircraft than is the case for the baseline aircraft. This means that to achieve the same control powers slightly larger control ranges (i.e., degrees of blade angle at the rotor [illegible]d) would be required. Alternately a slight reduction in control power could be accepted since in no case are the control powers available less than the guideline requirements.

There is no significant difference between the damping derivatives for the TH-100 (97.3) design and those for the baseline aircraft indicating that the control response data will not be significantly affected for the unaugmented aircraft. The difference in the damping derivatives are small in comparison

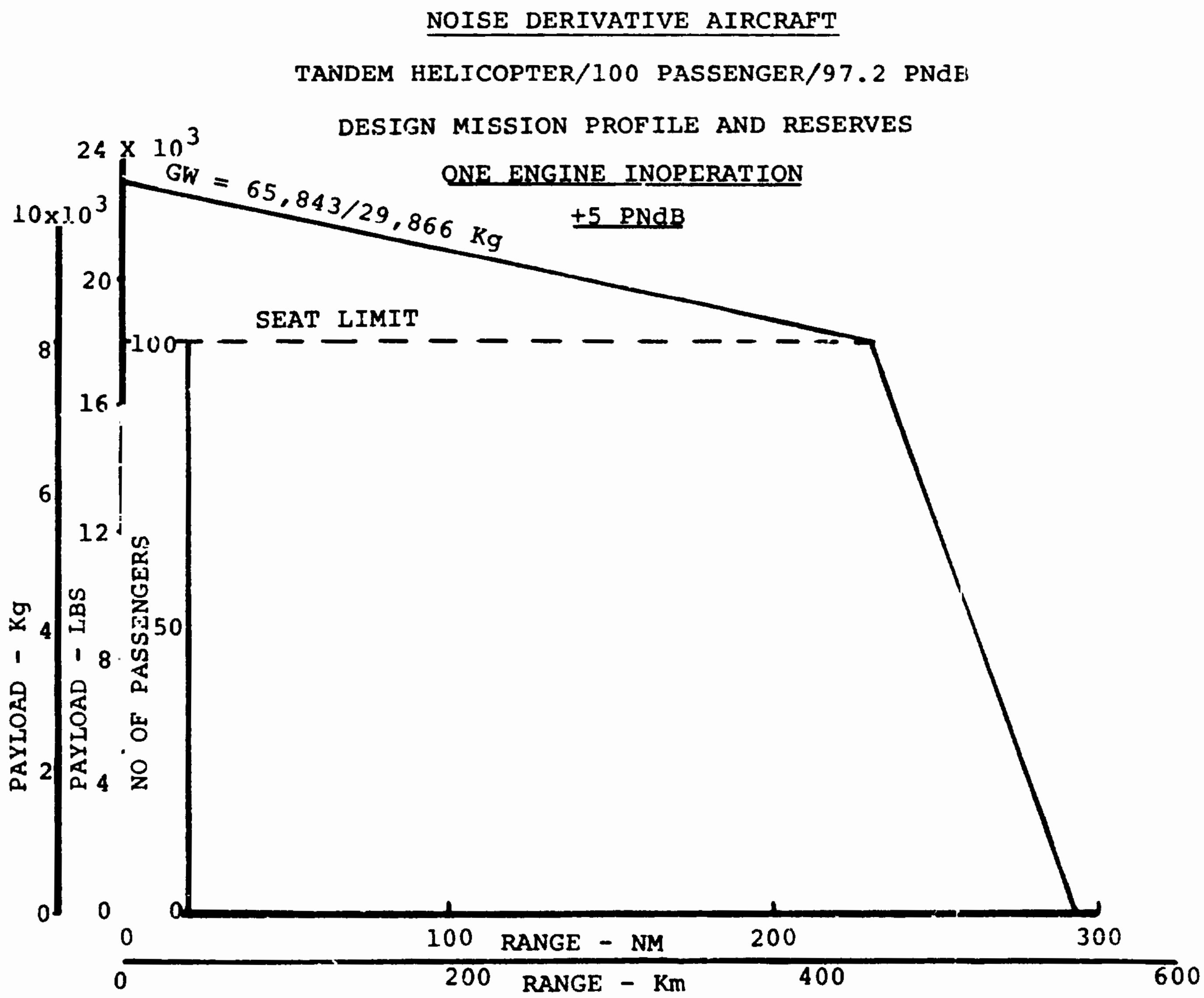


FIGURE 3.16. PAYLOAD RANGE CAPABILITY - CRUISE AT NRP (OEI) +5 PNdB HELICOPTER.

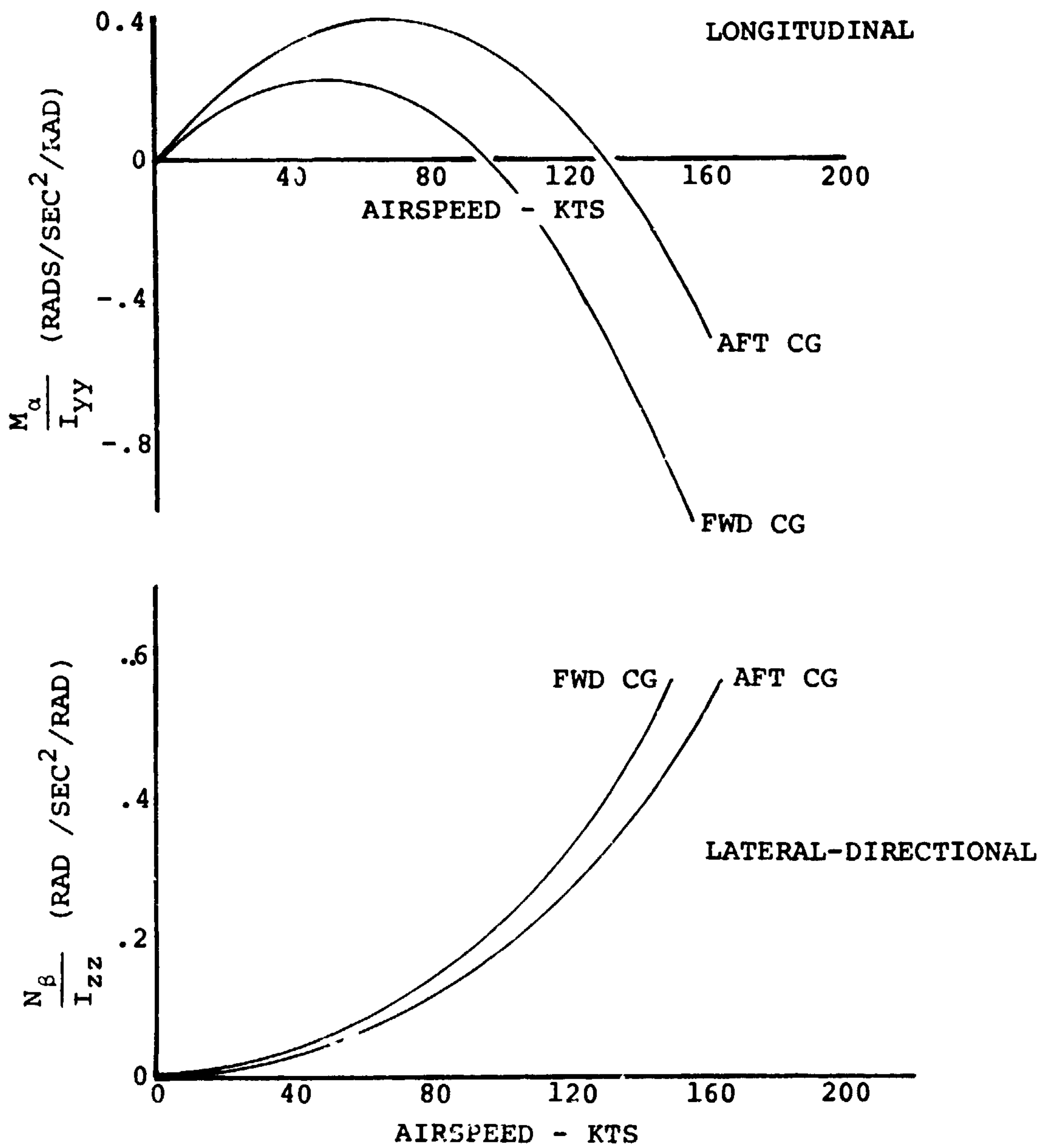


FIGURE 3.17. STATIC STABILITY - TH-100 (97.3)..

AIRCRAFT	V	0	40	80	120	150	
BASELINE W=67,175 *	L_{δ_S}	.433	.410	.407	.405	.406	TH-100 (92.3)
	M_{δ_B}	.183	.181	.213	.241	.257	
	N_{δ_R}	.170	.155	.151	.149	.153	
	Z_{δ_C}	-7.33	-6.65	-7.67	-9.03	-9.65	
	CONTROL DERIVATIVES						
	L_p	-.70	-.80	-.90	-.84	-.70	
	M_q	-.70	-.92	-1.19	-1.27	-1.34	
	N_r	-.07	-.05	-.04	-.05	-.07	
	Z_w	-.22	-3.2	-.43	-.50	-.52	
	DAMPING DERIVATIVES						
	M_α FWD	0	.15	.03	-.49	-1.12	
	AFT	0	.28	.29	-.05	-.49	
	N_β FWD	0	.04	.15	.33	.52	
	AFT	0	.03	.12	.27	.42	
	STABILITY DERIVATIVES						
NOISY W=65,845	L_{δ_S}	.403	.384	.384	.386	.390	①
	M_{δ_B}	.180	.175	.203	.224	.232	②
	N_{δ_R}	.164	.149	.145	.144	-.148	③
	Z_{δ_C}	-6.93	-6.25	-7.17	-8.40	-8.92	④
	L_p	-.73	-.83	-.92	-.86	-.72	TH-100 (97.3) ⑤
	M_q	-.67	-.83	-1.04	-1.07	-1.09	
	N_r	-.06	-.04	-.04	-.05	-.08	
	Z_w	-.22	-.32	-.41	-.47	-.46	⑥
	M_α FWD	0	.21	.15	-.32	-.87	⑦
	AFT	0	.33	.37	-.16	-.29	
	N_β FWD	0	.04	.14	.31	.54	
	AFT	0	.03	.11	.26	.44	

① ROLL - RADS/SEC^2/INCH

② PITCH - RADS/SEC^2/INCH

③ YAW - RADS/SEC^2/INCH

④ VERT - FT/SEC^2/INCH

⑤ RATE DERIVATIVES - RADS/SEC^2/RAD/SEC

⑥ FT/SEC^2/FT/SEC

⑦ STABILITY DERIVATIVES

TABLE 3.6 . STABILITY & CONTROL DERIVATIVES FOR +5 PNdB ALTERNATE CONFIGURATION AT DESIGN GROSS WEIGHT.

to the damping provided by the automatic flight control system gains of

$$\Delta M_q = 2.0 \text{ to } 2.5 \text{ rads/sec}^2\text{/rad/sec}$$

$$\Delta M\alpha = 3.0 \text{ to } 5.0 \text{ rads/sec}^2\text{/rad}$$

With the system on the pilot would not recognize the difference between this aircraft and the baseline tandem helicopter.

Tandem Helicopter TH-100 (97.3) - Noise

The 500 foot sideline noise level of this aircraft is 5 PNdB noisier than the baseline tandem helicopter at 97.3 PNdB. The sound pressure level spectrum and NOY weighting for the hover case are shown in Figure 3.18. The major component of noise is the rotor broad band noise as was the case for the baseline tandem helicopter. The engine inlets are assumed to be treated with noise attenuation linings in the same manner as the baseline aircraft. The primary contribution to the increased percieved noise level is due to the increased tip speed. The PNL contours for a typical takeoff are shown in Figure 3.19 and the takeoff trajectory and time histories of percieved noise along the flight path are shown in Figure 3.20.

Similar data are given for the landing case in Figures 3.21 and 3.22.

TH-100 (97.3) Costs

Direct operating costs per seat mile and seat kilometer as a function of block distance are shown in Figure 3.23 for the specified combinations of aircraft utilization and airframe

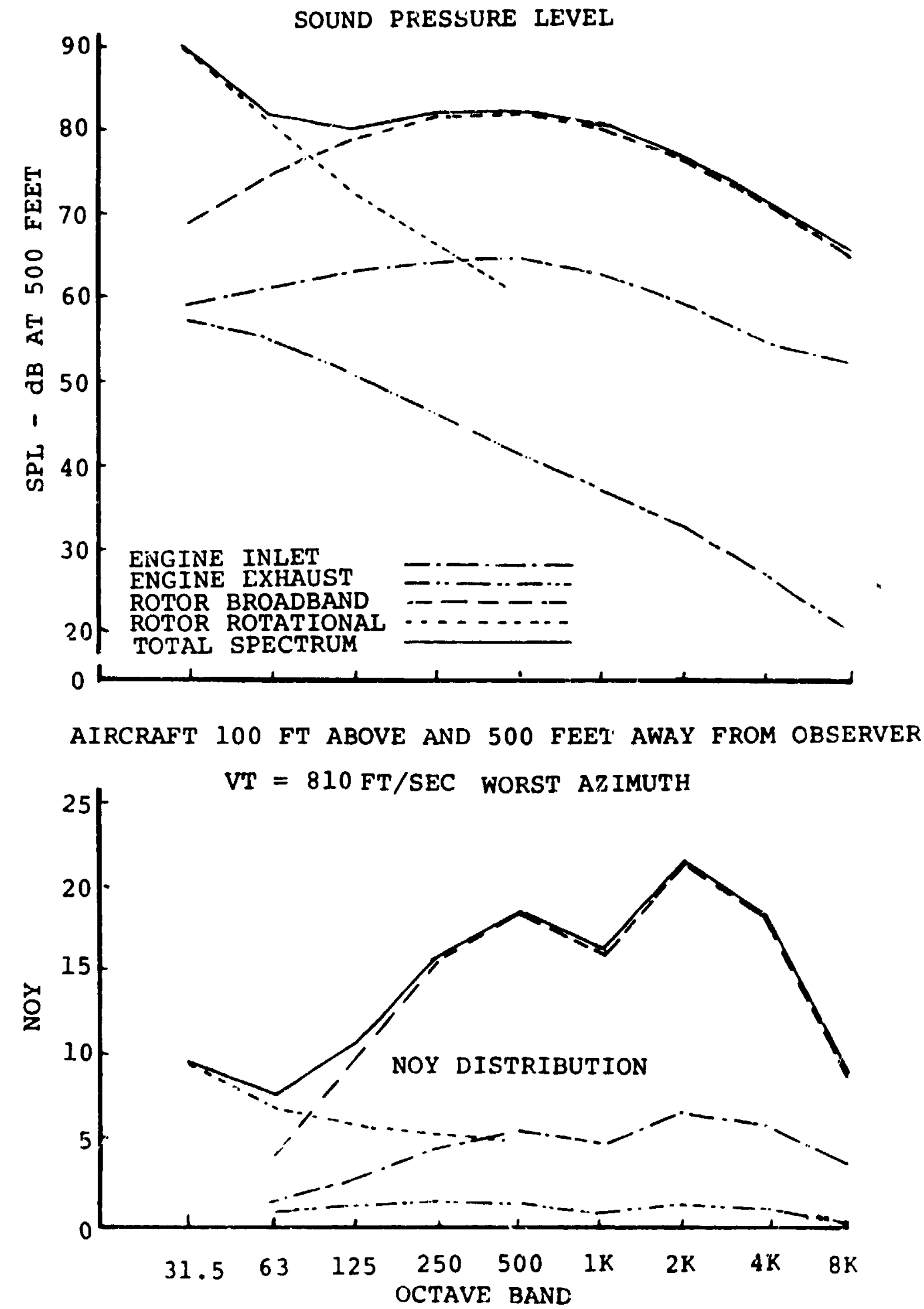


FIGURE 3.18. +5 PNdB HELICOPTER HOVER NOISE SPECTRUM AND NOY DISTRIBUTION. PNdB = 97.3

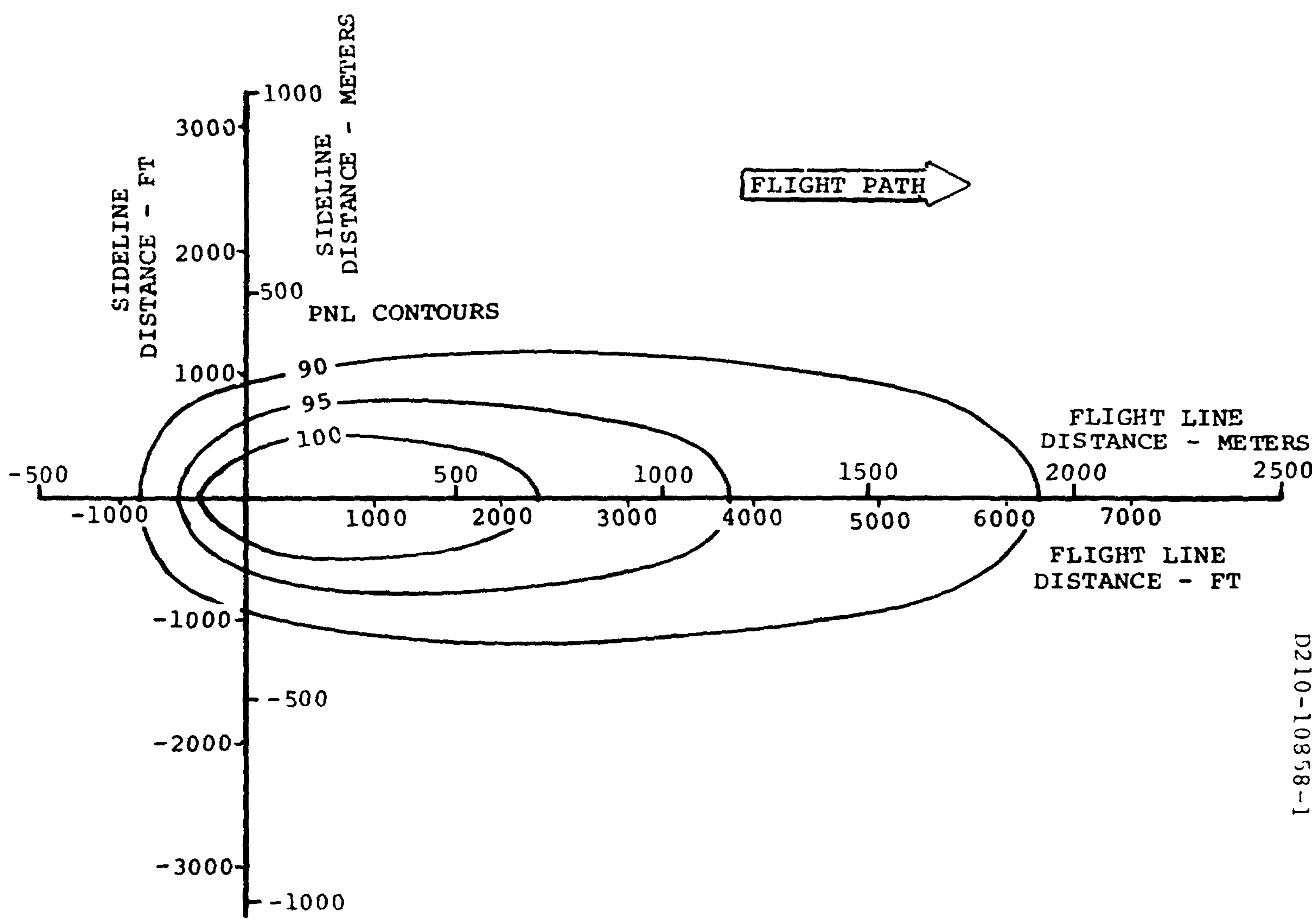


FIGURE 3.19. +5 PNdB HELICOPTER DESIGN POINT - STANDARD TAKEOFF. PNL CONTOURS.

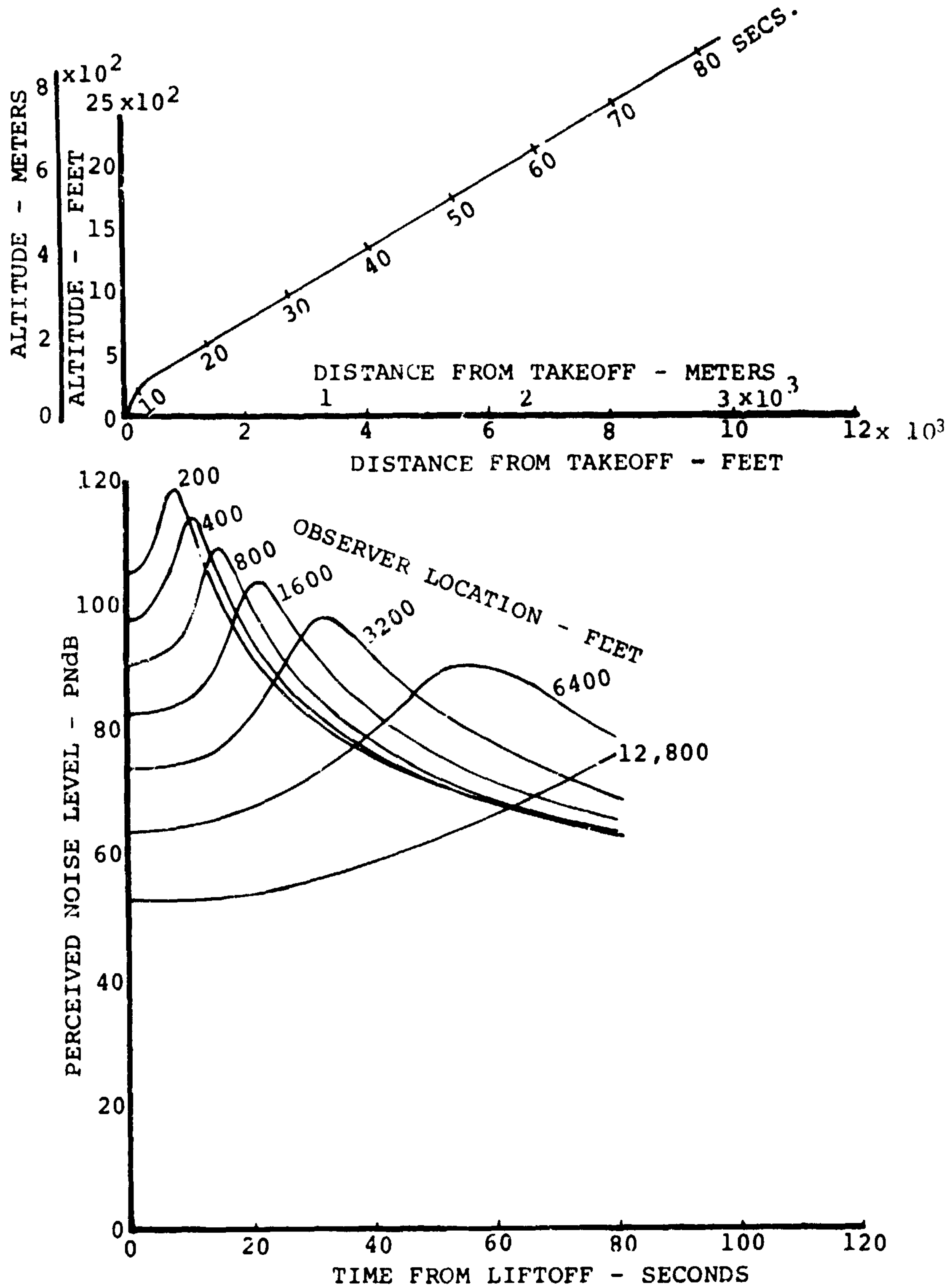


FIGURE 3.20. +5 PNdB HELICOPTER - STANDARD TAKEOFF - PERCEIVED NOISE.

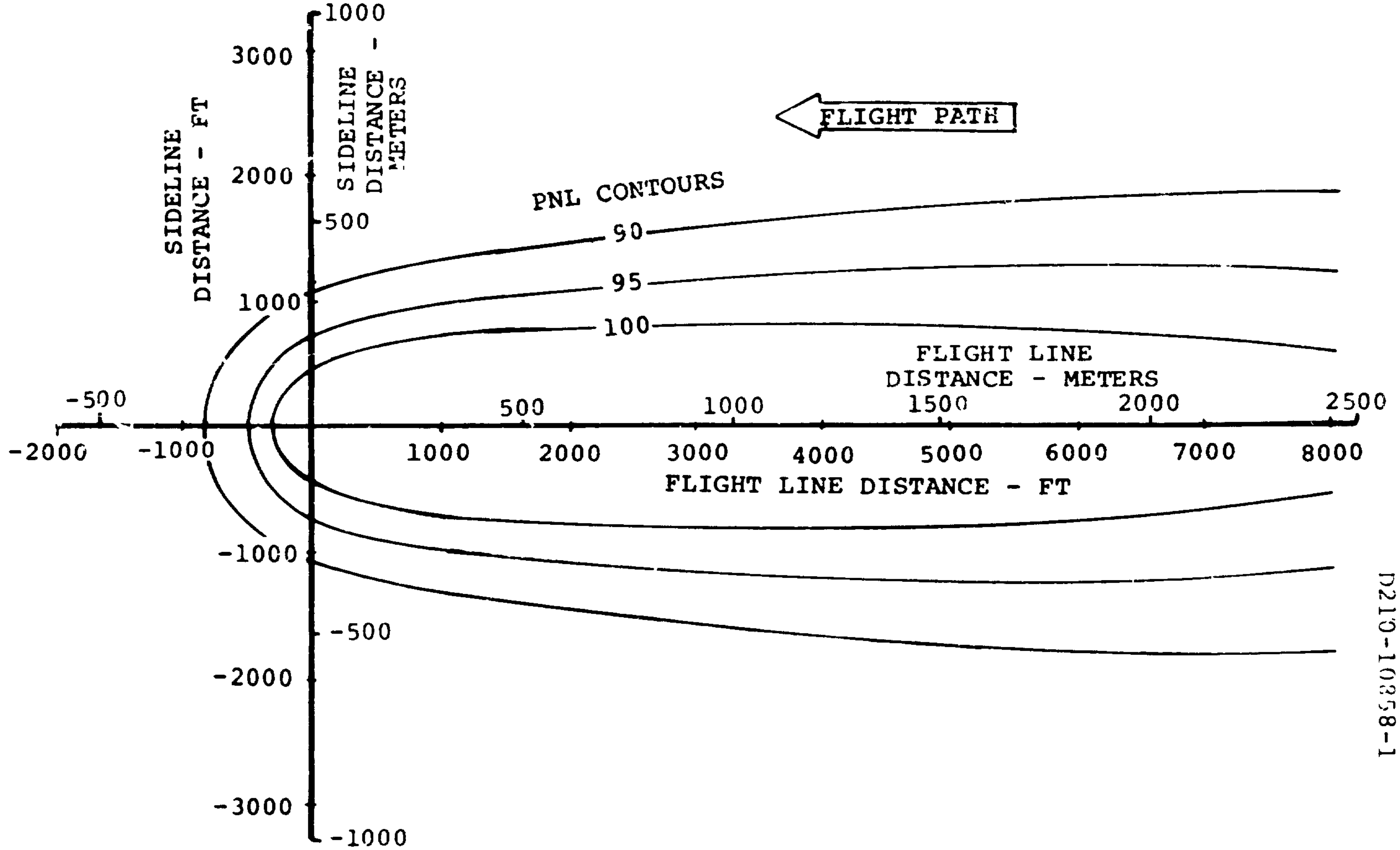


FIGURE 3.21. +5 PNdB HELICOPTER STANDARD LANDING. PNL CONTOURS.

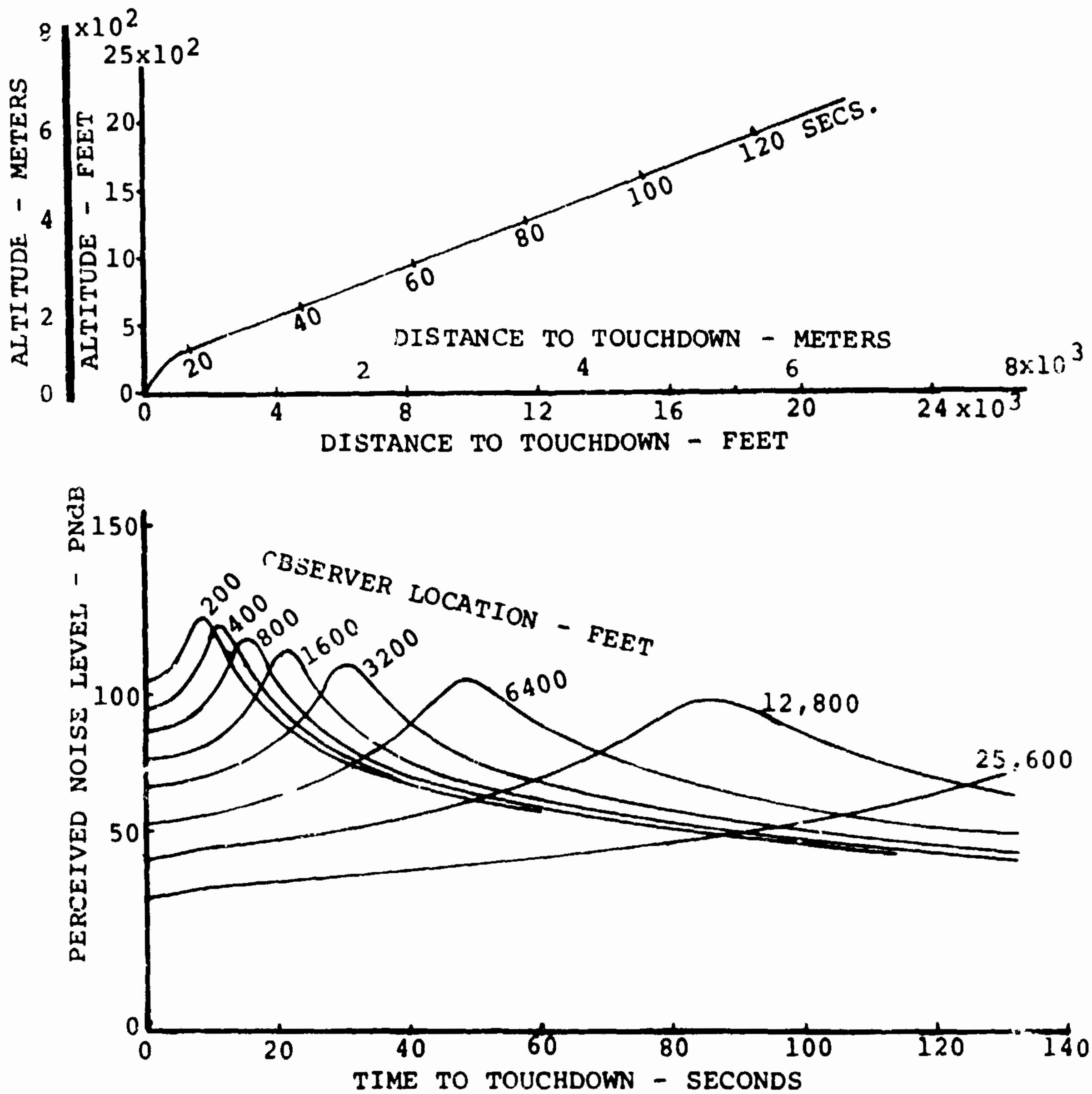


FIGURE 3.22. +5 PNdB HELICOPTER DESIGN POINT LANDING. PERCEIVED NOISE .

TANDEM HELICOPTER +5 PNdB

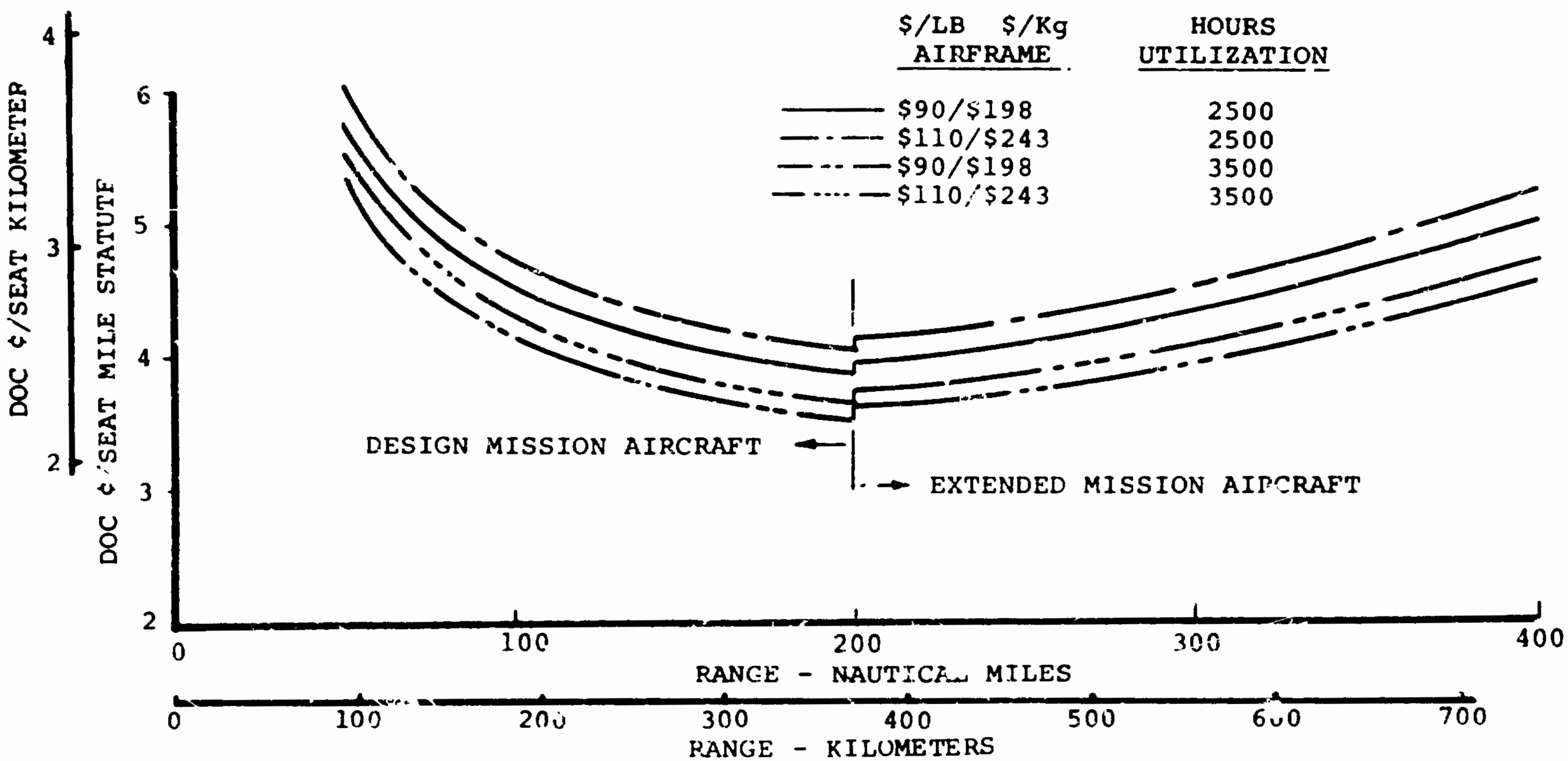


FIGURE 3.23. EFFECT OF OPERATING RANGE ON DIRECT OPERATING COST. +5 PNdB AIRCRAFT.

costs. Figure 3.23 also illustrates the impact of extending the design range of the TH-100 (97.3) to 460 statute miles. The increase in costs at the design point range (230 statute miles) is the result of the loss of 2 available seats due to the increased empty weight for the installation of larger fuel tanks. Although not shown in Figure 3.23 it should be noted that the larger fuel tanks will result in a small increase (less than 1%) in seat mile costs at ranges less than 230 statute miles due to increases in airframe maintenance and depreciation costs. In the extended range version of the TH-100 (97.3) seat mile costs show a continuing increase beyond 230 statute miles because of the loss of available seats due to additional fuel requirements at the longer block distances. Only 70 seats are available at 460 statute miles.

Table 3.7 shows the flyaway costs for the basic TH-100 (97.3) at $90.00 and $110.00 per pound of airframe. A breakout of the direct operating cost factors for the TH-100 (97.3) at 230 statute miles is shown in Table 3.7. Flyaway and direct operating cost breakouts for the extended range version of the TH-100 (97.3) are shown in Table 3.8.

3.1.2 Tandem Helicopter Design TH-100 (87.1)

The TH-100 (87.1) design has a calculated sideline perceived noise level of 87.1 PNdB, 5 PNdB less noisy than the baseline tandem rotor helicopter.

Flyaway Costs

Airframe Costs	$90.00/Lb	$110.00/Lb
Airframe	$2,144,700	$2,621,300
Dynamic System	958,080	958,080
Engines	629,220	629,220
Avionics	250,000	250,000
Total	$3,982,000	$4,458,600

Direct Operating Costs
Dollars/Seat Mile
Block Distance = 230 St. Miles

Utilization (Hrs/Yr)	2500		3500	
Airframe Cost ($/Lb)	90	110	90	110
Flying Operations				
Flight Crew	.0092	.0092	.0092	.0092
Fuel and Oil	.0050	.0050	.0050	.0050
Hull Insurance	.0021	.0024	.0015	.0017
Total Flying Operations	.0163	.0166	.0157	.0159
Direct Maintenance				
Airframe - Labor	.0014	.0014	.0014	.0014
- Material	.0010	.0012	.0010	.0012
Engines - Labor	.0008	.0008	.0008	.0008
- Material	.0010	.0010	.0010	.0010
Dynamic System - Labor	.0011	.0011	.0011	.0011
- Material	.0017	.0017	.0017	.0017
Total Direct Maintenance	.0070	.0072	.0070	.0072
Maintenance Burden	.0050	.0050	.0050	.0050
Total Maintenance	.0120	.0122	.0120	.0122
Depreciation	.0103	.0114	.0073	.0081
Total Direct Costs	.0386	.0402	.0350	.0362

TABLE 3.7. INITIAL AND DIRECT OPERATING COSTS - +5 PNdB HELICOPTER.

TH-100(97.3)
EXTENDED RANGE VERSION

Airframe Costs	Flyaway Costs $90.00/Lb	$110.00/Lb
Airframe	$2,175,300	$2,658,700
Dynamic System	958,080	958,080
Engines	629,220	629,220
Avionics	250,000	250,000
Total	$4,012,600	$4,496,000

Direct Operating Costs
Dollars/Seat Mile
Block Distance = 230 St. Miles

Utilization (Hrs/Yr)	2500		3500	
Airframe Cost ($/Lb)	90	110	90	110
Flying Operations				
Flight Crew	.0094	.0094	.0094	.0094
Fuel and Oil	.0051	.0051	.0051	.0051
Hull Insurance	.0022	.0024	.0016	.0017
Total Flying Operations	.0167	.0169	.0161	.0162
Direct Maintenance				
Airframe - Labor	.0014	.0014	.0014	.0014
- Material	.0010	.0013	.0010	.0013
Engines - Labor	.0008	.0008	.0008	.0008
- Material	.0010	.0010	.0010	.0010
Dynamic System - Labor	.0012	.0012	.0012	.0012
- Material	.0018	.0018	.0018	.0018
Total Direct Maintenance	.0072	.0075	.0072	.0075
Maintenance Burden	.0051	.0051	.0051	.0051
Total Maintenance	.0123	.0126	.0123	.0126
Depreciation	.0105	.0117	.0075	.0084
Total Direct Costs	.0395	.0412	.0359	.0372

TABLE 3.8. INITIAL AND DIRECT OPERATING COSTS - +5 PNdB HELICOPTER - (EXTENDED RANGE VERSION).

Configuration and Layout

This aircraft has the same fuselage, cabin and cockpit arrangement as the baseline tandem helicopter aircraft. The major differences are in the rotor and drive system which result from reduced rotor tip speed and increased solidity required to reduce the external noise by 5 PNdB.

The rotor tip speed has been reduced to 640 feet per second requiring an increased rotor solidity to maintain 1.25g maneuver capability in cruise. The associated increase in aircraft weight required an increased rotor diameter to 22.1 M (72.5 feet) to maintain the design disc loading of 43.94 Kg/m^2 (9.0 pounds per $feet^2$).

The drive system configuration is the same as for the baseline aircraft except that the power and torques required are increased. The installed maximum power/engine has increased to 4.294×10^6 watts (5,759 SHP).

The major characteristics of this aircraft are shown in Table 3.9 and the threeview drawing in Figure 3.24.

The increased rotor diameter required the aft pylon of the aircraft to be swept more than the baseline aircraft to maintain zero overlap. This results in an increased overall length of the aircraft.

Weights

The design gross weight of the TH-100 (87.1) design is 33,668.6 Kg (74,227 pounds), an increase of 3198.7 Kg (7052 pounds) over the baseline aircraft. The aircraft

	S. I. UNITS		U. S. UNITS	
WEIGHTS				
DESIGN GROSS WEIGHT	33,669	Kg	74,227	Lbs
WEIGHT EMPTY	21,107	Kg	46,533	Lbs
FUEL WEIGHT	3,496	Kg	7,708	Lbs
NO. OF PASSENGERS	100		100	
ROTORS				
DISC LOADING	43.94	Kg/m^2	9	Lbs/Ft^2
DIAMETER	22.1	m	72.5	
SOLIDITY	.159		.159	
NUMBER OF BLADES	4		4	
TWIST	12	Degs	12	Degs
TIP SPEED	195	m/sec	640	Ft/Sec
POWER				
NUMBER OF ENGINES	3		3	
RATED POWER/ENGINE	4.294×10^6	Watts	5759	SHP
FUSELAGE				
LENGTH	28.3	m	93	Ft
WIDTH	4.48	m	14.7	Ft
CABIN LENGTH	15.33	m	49.3	Ft
PERFORMANCE				
NRP CRUISE SPEED	67.8	m/s	181	KTAS
CRUISE ALTITUDE	1524	m	5000	Ft
BLOCK TIME	1.24	Hr	1.24	Hr
NOISE				
SIDELINE NOISE - 500 FT/HOVER	87.1	PNdB	87.1	PNdB

TABLE 3.9. -5 PNdB DESIGN POINT HELICOPTER TABLE OF CHARACTERISTICS.

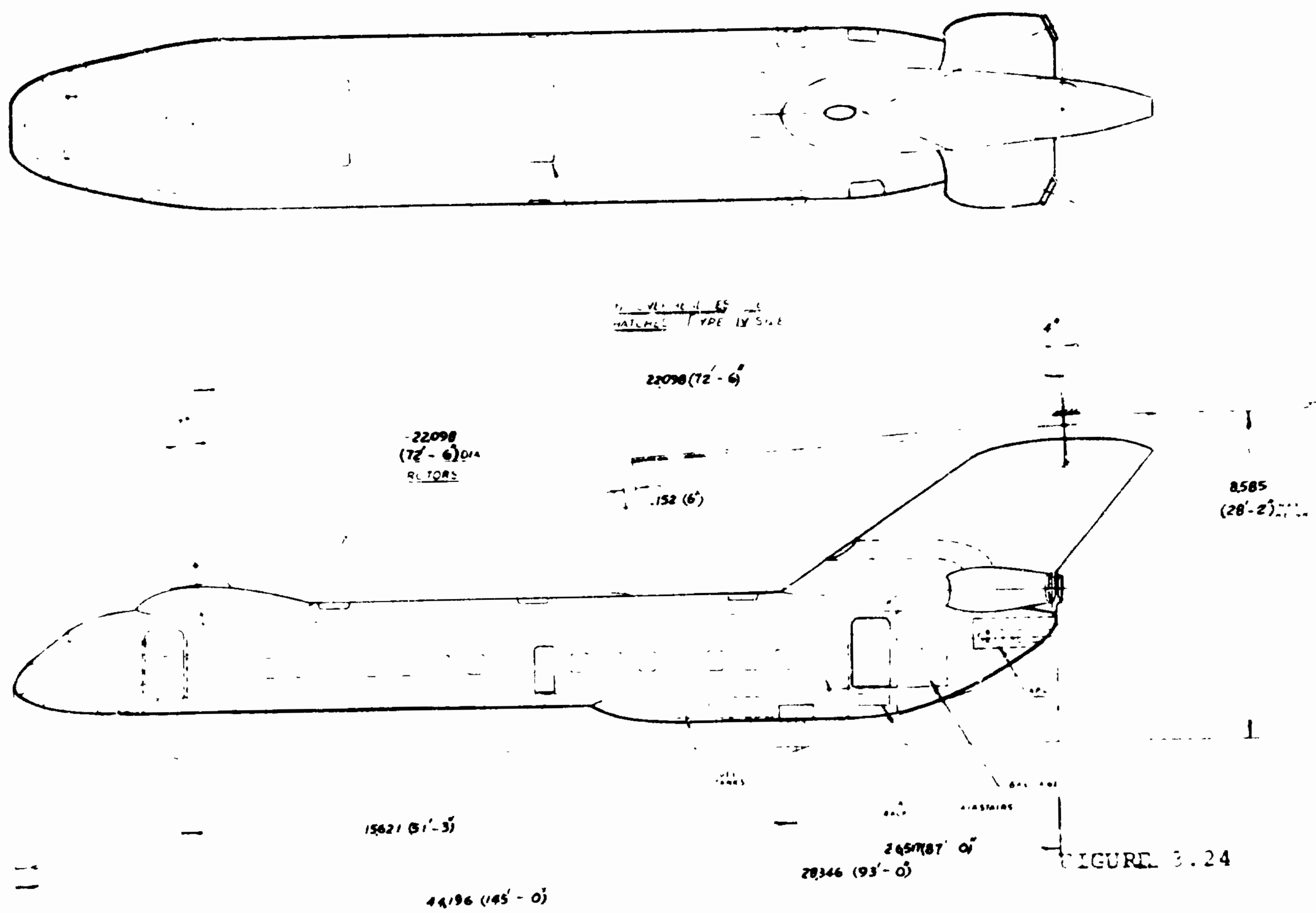


FIGURE 3.24. 1985 COMMERCIAL VTOL TRANSPORT, 100 PASSENGER - TANDEM HELICOPTER -5 PNdB DESIGN.

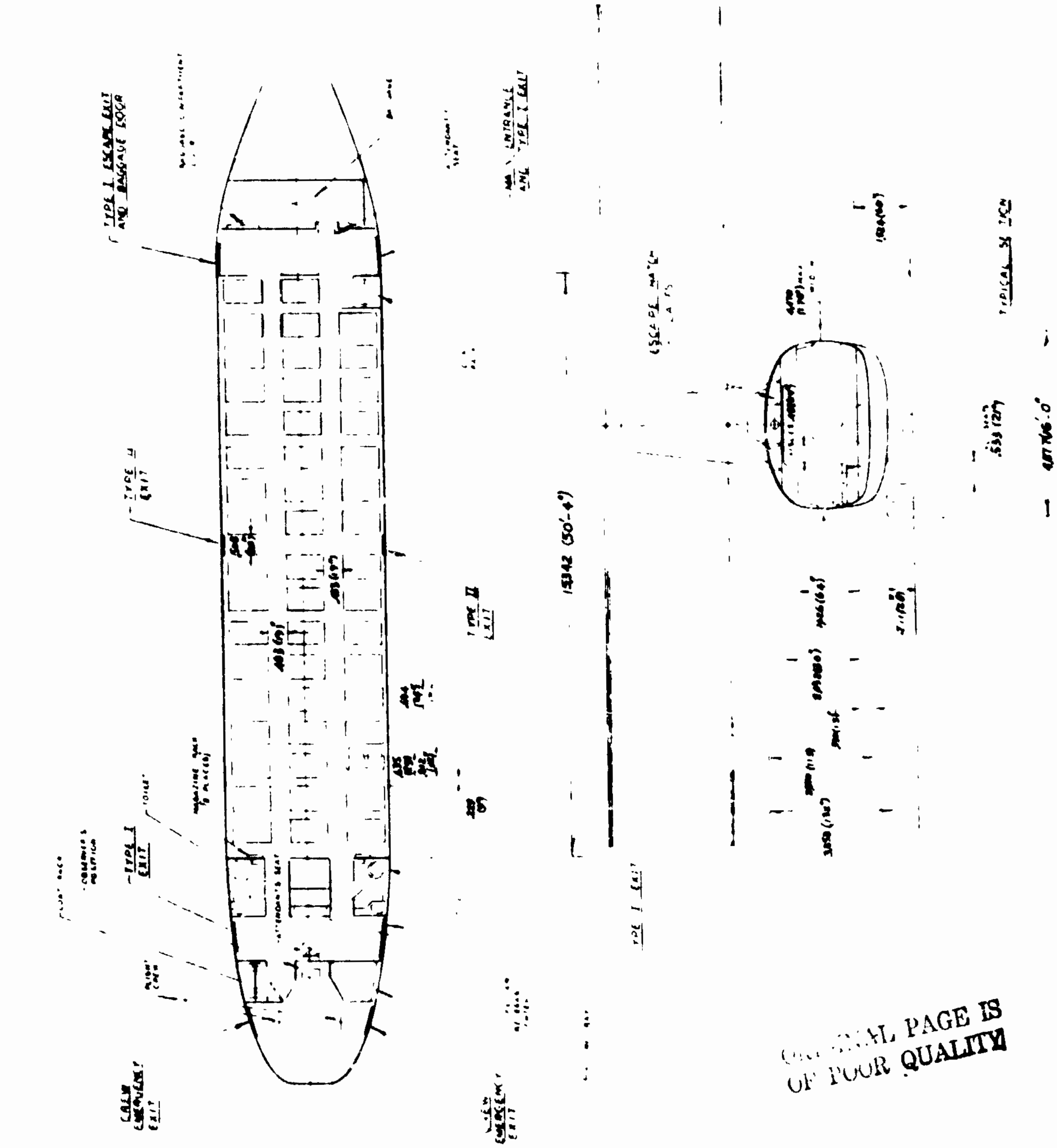


FIGURE 3.24. 1985 COMMERCIAL VTOL TRANSPORT, 100 PASSENGER - TANDEM HELICOPTER -5 PNdB DESIGN (CONTINUED).

weight breakdown is given in Table 3.10. This increase in weight is caused by the larger rotor diameter and solidity which increases the rotor system weight to 3,729.9 Kg (8,223 pounds). The body weight increases as a result of the increase in the distance between the rotor hubs since the bending moments carried by the fuselage structure increase requiring a higher structural strength and weight. The landing gear weight is governed by the change in aircraft weight and grows with the aircraft to 1,346.3 Kg (2,968 pounds). The engine section weights are increased owing to the increased engine size. The increased installed horsepower and weight requires a larger transmission reflected by the increased drive system weight.

The increased flight controls weight is a function of the increase in rotor size and weight since larger upper controls are required.

The fixed weight items such as passenger accommodations, emergency equipment are the same as the baseline aircraft.

These weight changes result in an empty weight of 21,105.5 Kg (46,530 pounds). The mission fuel load, including reserves is 3,496.2 Kg (7,708 pounds).

The principle inertias and CG locations are given in Table 3.11.

<u>Tandem Helicopter - TH-100 (87.1) - Vehicle Performance</u>

This quiet tandem helicopter is designed to fly the same mission as the baseline aircraft and a summary of its mission

BOEING VERTOL COMPANY

WEIGHT SUMMARY - PRELIMINARY DESIGN

MIL-STD-1374

	KILOGRAMS	POUNDS
WING		
ROTOR	3729.9	8223
TAIL		
SURFACE		
ROTOR		
BODY	2996.9	6607
BASIC		
SECONDARY		
ALIGHTING GEAR GROUP	1346.3	2968
ENGINE SECTION	222.7	491
PROPULSION GROUP	5705.3	12578
ENGINE INSTL	1191.1	2626
EXHAUST SYSTEM *		
COOLING		
CONTROLS *		
STARTING *		
PROPELLER INSTL	* 98.4	*217
LUBRICATING *		
FUEL	241.3	532
DRIVE	4174.4	9203
FLIGHT CONTROLS	1733.6	3822
AUX POWER PLANT	288.5	636
INSTRUMENTS	191.9	423
HYDR. & PNEUMATIC	308.4	680
ELECTRICAL GROUP	378.3	834
AVIONICS GROUP	293.9	648
ARMAMENT GROUP		
FURN & EQUIP GROUP	3206.9	7070
ACCOM FOR PERSON		
MISC EQUIPMENT		
FURNISHINGS		
EMERG EQUIPMENT		
AIR CONDITIONING	521.6	1150
ANTI-ICING GROUP	181.4	400
LOAD AND HANDLING GR.		
WEIGHT EMPTY	21105.5	46530
CREW	299.4	660
TRAPPED LIQUIDS	52.2	115
ENGINE OIL	59.9	132
CREW ACCOMMODATIONS	68.0	150
EMERGENCY EQUIPMENT	7.3	16
PASSENGER ACCOMMO.	415.[illegible]	916
PASSENGERS (100)	8164	18000
FUEL	3496.2	7708
GROSS WEIGHT	33668.6	74227

	WEIGHT EMPTY	GROSS WEIGHT
WEIGHT	21,105.5 Kg (46,530 LBS)	33,668.6 Kg (74,227 LBS)
CENTER OF GRAVITY*		
FUSELAGE STATION	15.25 M (600.5 IN.)	14.53 M (572.0 IN.)
WATER LINE	3.59 M (141.4 IN.)	2.83 M (111.4 IN.)
MOMENT OF INERTIA		
I_{xx} (ROLL)	15,939.4 Kg M^2 (11,746.9 Slug Ft^2)	17,139.0 Kg M^2 (12,631.0 Slug Ft^2)
I_{yy} (PITCH)	1,754,961.7 Kg M^2 (1,293,361.1 Slug Ft^2)	1,887,055.3 Kg M^2 (1,390,710.7 Slug Ft^2)
I_{zz} (YAW)	1,694,773.0 Kg M^2 (1,249,003.6 Slug Ft^2)	1,822,336.4 Kg M^2 (1,343,014.5 Slug Ft^2)

*FUSELAGE STATION O IS AT NOSE OF BODY, CENTERLINE OF FORWARD ROTOR 5.0 METERS ABOVE WATER LINE.

TABLE 3.11 . WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA - HELICOPTER -5 PNdB.

performance is given in Tables 3.12 and 3.13.

The taxi, takeoff and initial air maneuver require 64 Kg (143 pounds) of fuel. As before, the initial air maneuver is included with the takeoff in Tables 3.12 and 3.13. The climb to altitude is done at 10.4 m/s (2,047 feet per minute) and requires 91 Kg (200 pounds) fuel for a range credit of 7.2 Km (3.9 nautical miles). The cruise altitude is 5,000 feet. The cruise segment is done at an average speed of 93.6 m/s (182 knots). The cruise segment fuel is 2,277 Kg (5,021 pounds) for a total range credit of 366.1 Km (197.6 nautical miles). The remainder of the design range is achieved in a descent to 2,000 feet altitude at an average rate of descent of 12.0 m/s (2,360 feet per minute). The final air maneuver, descent, landing and taxi complete the mission for a total fuel weight of 2,540.2 Kg (5,601 pounds). The reserve fuel is calculated based on fuel for 50 nautical miles and 20 minutes loiter. This increases the fuel weight to a total of 3,469.2 Kg (7,708 pounds).

Hover Performance

The effect of ambient temperature on hover lift capability is shown in Figure 3.25 and 3.26 for both all engines operating, and one engine inoperative both in and out of ground effect. The sizing condition with one engine inoperative (OEI) at sea level, 90°F, defines the aircraft design gross weight and installed power. The higher lift capability at lower ambient temperature or due to ground effect or all engines operating

	TIME (HOURS)	DISTANCE (NMI)	WEIGHT (LBS)	FUEL (LBS)	V (KNOTS)	R/C (FT/MIN)
	0	0	74,227			
TAXI				15	0	0
	.017	0	74,213			
TAKEOFF				128	0	0
	.042	0	74,085			
CLIMB				200	95	2047
	.083	3.9	73,885			
CRUISE				5,021	182	0
	1.143	197.6	68,864			
DESCENT				31	115	-2360
	1.164	200	68,833			
AIR MANEUVER				176	95	0
	1.189	200	68,757			
DESCENT				41		-1000
	1.206	202	68,716			
LANDING				78	0	0
	1.223	202	68,638			
TAXI				15	0	0
	1.240	202	68,623			
RESERVE				2,107	164/96	0
	1.877	250	66,516			

TABLE 3.12. MISSION [illegible] -5 PNdB DESIGN POINT HELICOPTER (U. S. UNITS).

	TIME (HOURS)	DISTANCE (Km)	WEIGHT (Kg)	FUEL (Kg)	V (KNOTS)	R/C (m/s)
	0	0	33,669			
TAXI				6	0	0
	.017	0	33,662			
TAKEOFF				58	0	0
	.04[illegible]	0	33,604			
CLIMB				91	9[illegible]	10.4
	.083	7.2	33,513			
CRUISE				2,[illegible]77	18[illegible]	0
	1.143	366.2	31,236			
DESCENT				[illegible]	[illegible]15	-12.0
	1.164	370.6	31,222			
AIR MANEUVER				34	95	0
	1.189	370.6	31,187			
DESCENT				18		-5.1
	1.206	374.3	31,169		70	
LANDING				35	0	0
	1.223	374.3	31,134			
TAXI				7	0	0
	1.240	374.3	31,127			
RESERVE				956	164/96	0
	1.877	463.3	30,171			

TABLE 3.13 MISSION SUMMARY -5 PNdB DESIGN POINT HELICOPTER (S. I. UNITS)

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/87.1 PNdB

ALL ENGINES OPERATING

-5 PNdB

F/W = 1.05

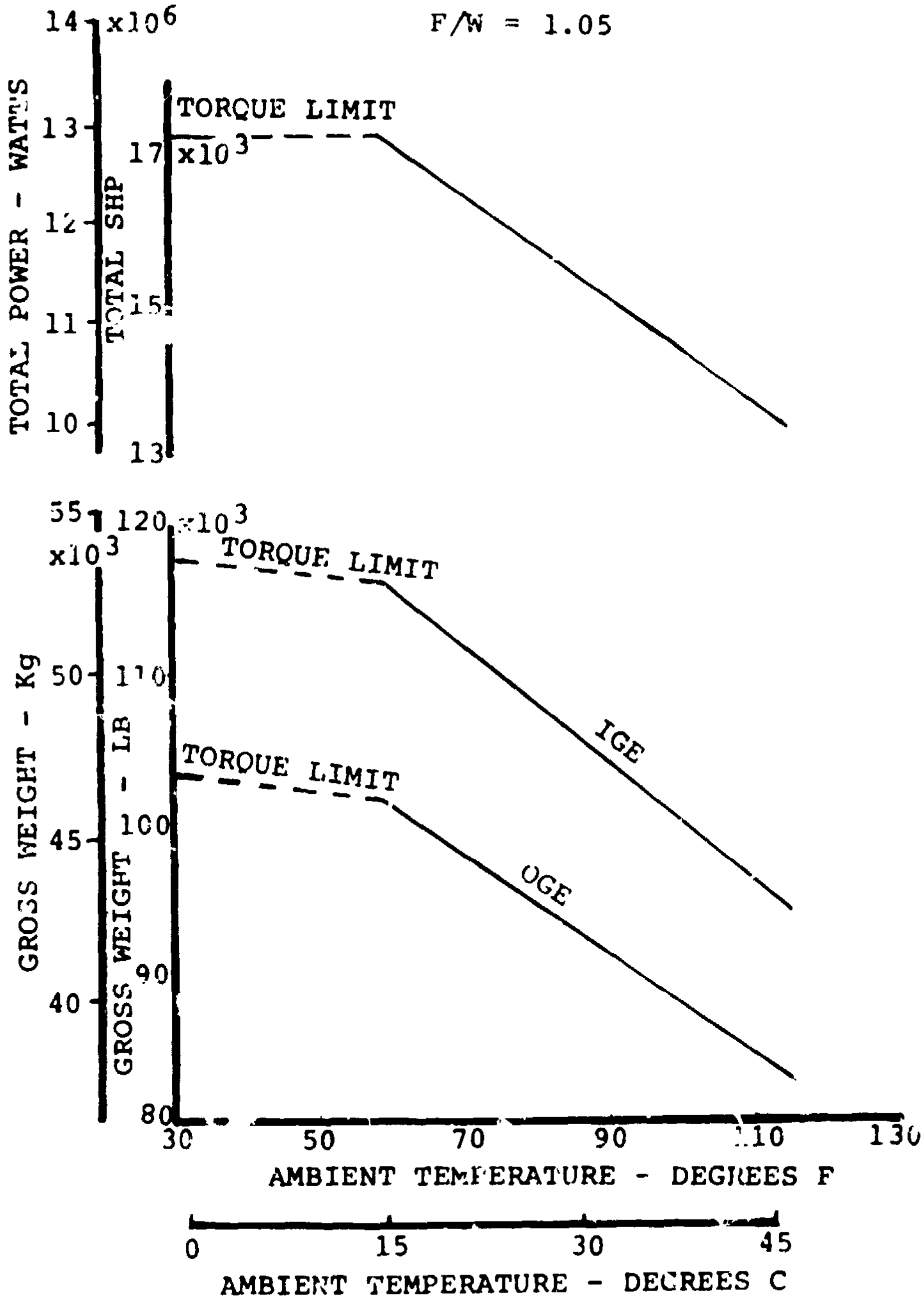


FIGURE 3.25. EFFECT OF AMBIENT TEMPERATURE ON SEA LEVEL HOVER PERFORMANCE - -5 PNdB HELICOPTER.

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/87.1 PNdB

ONE ENGINE INOPERATIVE

-5 PNdB

F/W = 1.03

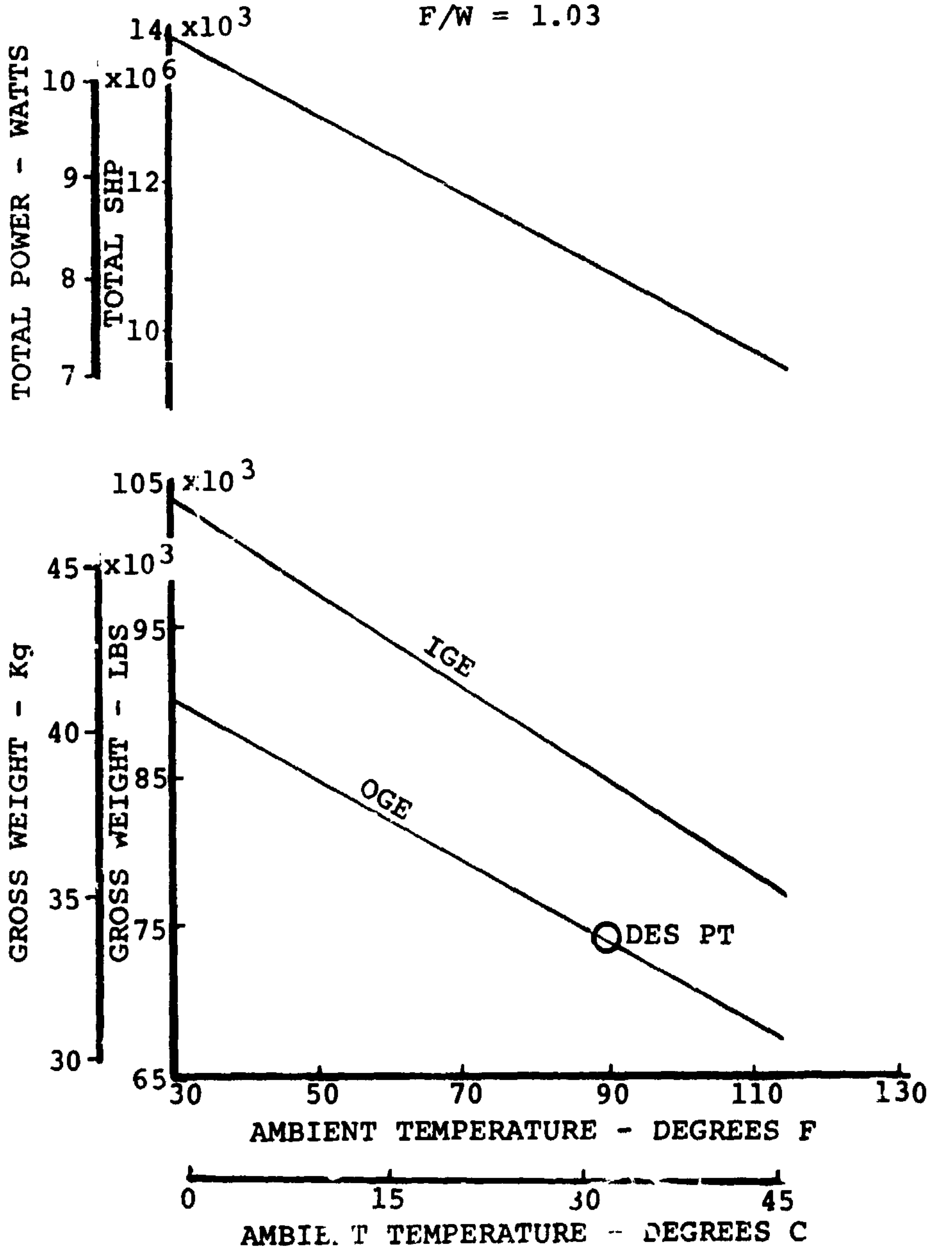


FIGURE 3.26. EFFECT OF AMBIENT TEMPERATURE ON SEA LEVEL HOVER PERFORMANCE - -5 PNdB HELICOPTER

C-4

provides additional force to weight capability in hover and takeoff. The increased takeoff weight capability can not be used as such under FAA certification groundrules since the increased weight would reduce the maneuver load factor of the aircraft.

Figures 3.27 and 3.28 show the effect of altitude on hover performance both all engines operating and one engine inoperative. With all engines operating on a standard day the fully loaded aircraft can maintain hover up to 12,500 feet altitude at takeoff power. Increasing the ambient temperature by 31°F reduces the altitude to 9,000 feet. With one engine inoperative the maximum altitude for hover on a standard day is 4,450 feet, and at standard plus 31°F the aircraft hovers at sea level OEI at design gross weight.

Performance in Forward Flight

The power required to maintain straight and level flight at both sea level and 5,000 feet altitude is given in Figures 3.29 and 3.30, for standard day conditions. At both sea level and 5,000 feet the aircraft can maintain hover at less than NRP all engines operating. As speed increases the power required decreases as shown and rises as the propulsive force requirement increases until it reaches the power available. This intersection defines the maximum cruise speed at normal rated power. At design gross weight the maximum cruise speed is 180 knots at sea level and 5,000 feet AEO. With one engine inoperative the aircraft can maintain a cruise speed of 167

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/87.1 PNdB

STANDARD DAY AND STANDARD DAY +31° F (+17.2° C)

ALL ENGINES OPERATING

-5 PNdB

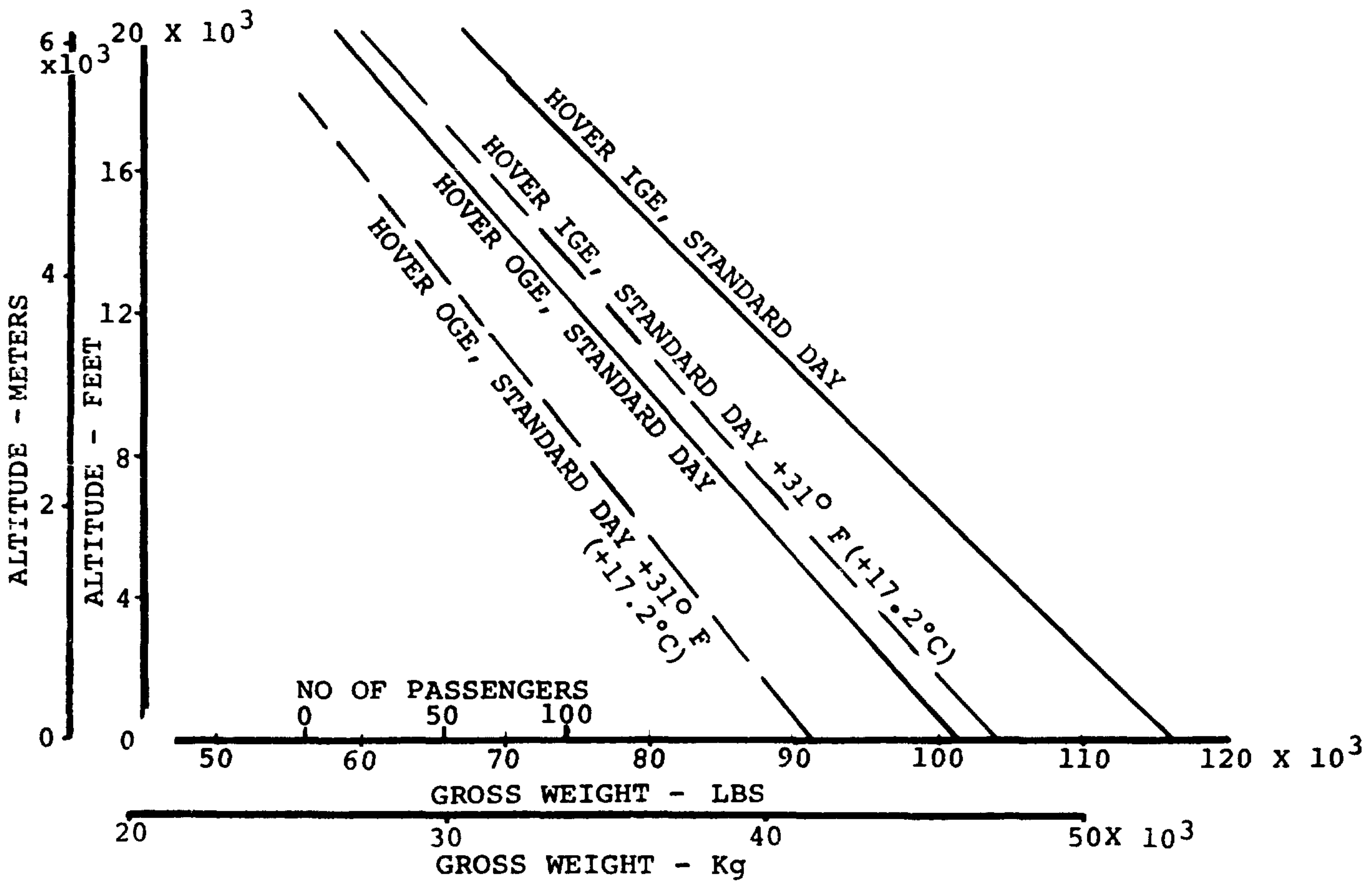


FIGURE 3.27 EFFECT OF ALTITUDE ON HOVER GROSS WEIGHT CAPABILITY.
-5 PNdB HELICOPTER

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/87.1 PNdB

STANDARD DAY AND STANDARD DAY +31° F (+17.2° C)

ONE ENGINE INOPERATIVE

-5 PNdB

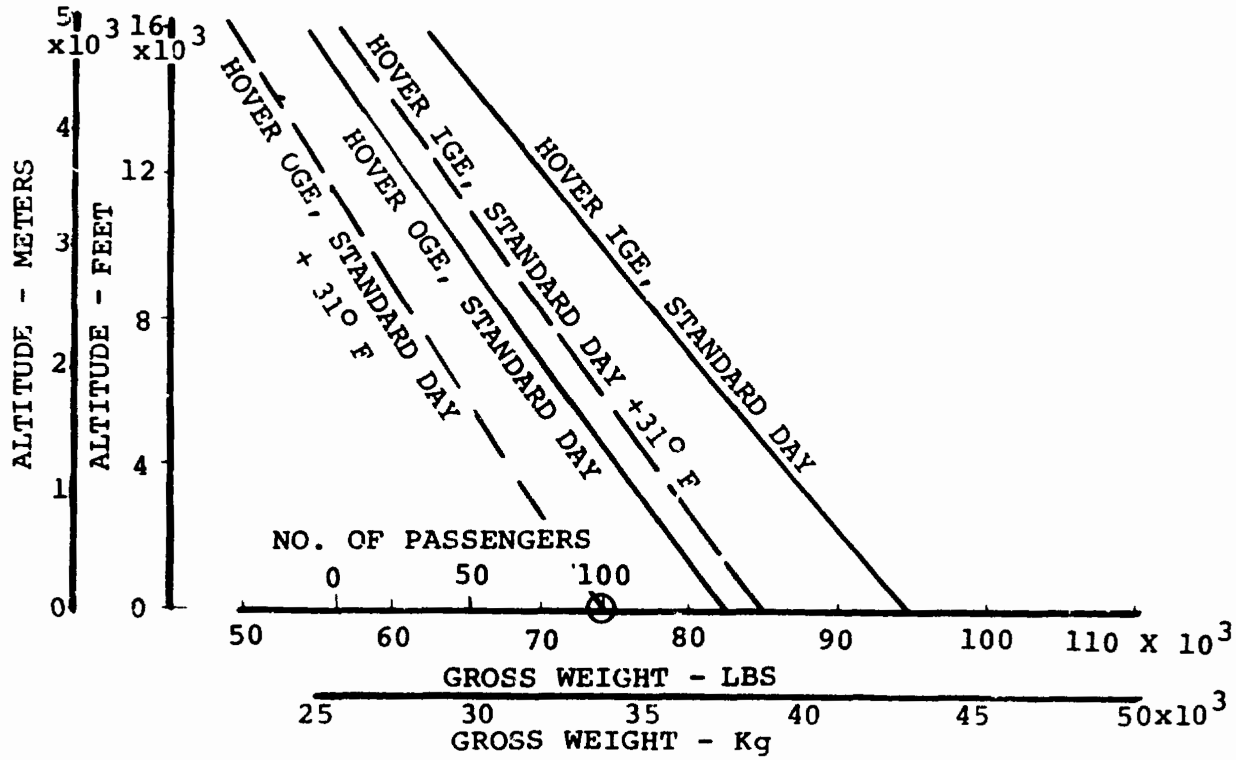


FIGURE 3.28. EFFECT OF ALTITUDE ON HOVER GROSS WEIGHT CAPABILITY. -5 PNdB HELICOPTER (OEI)

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/87.1 PNdB

STANDARD DAY CRUISE RPM
DGW = 74,227 LBS/33,669 Kg
MIDWT = 61,374 LBS/27,839 Kg
OWE = 48,520 LBS/22,008 Kg
-5 PNdB

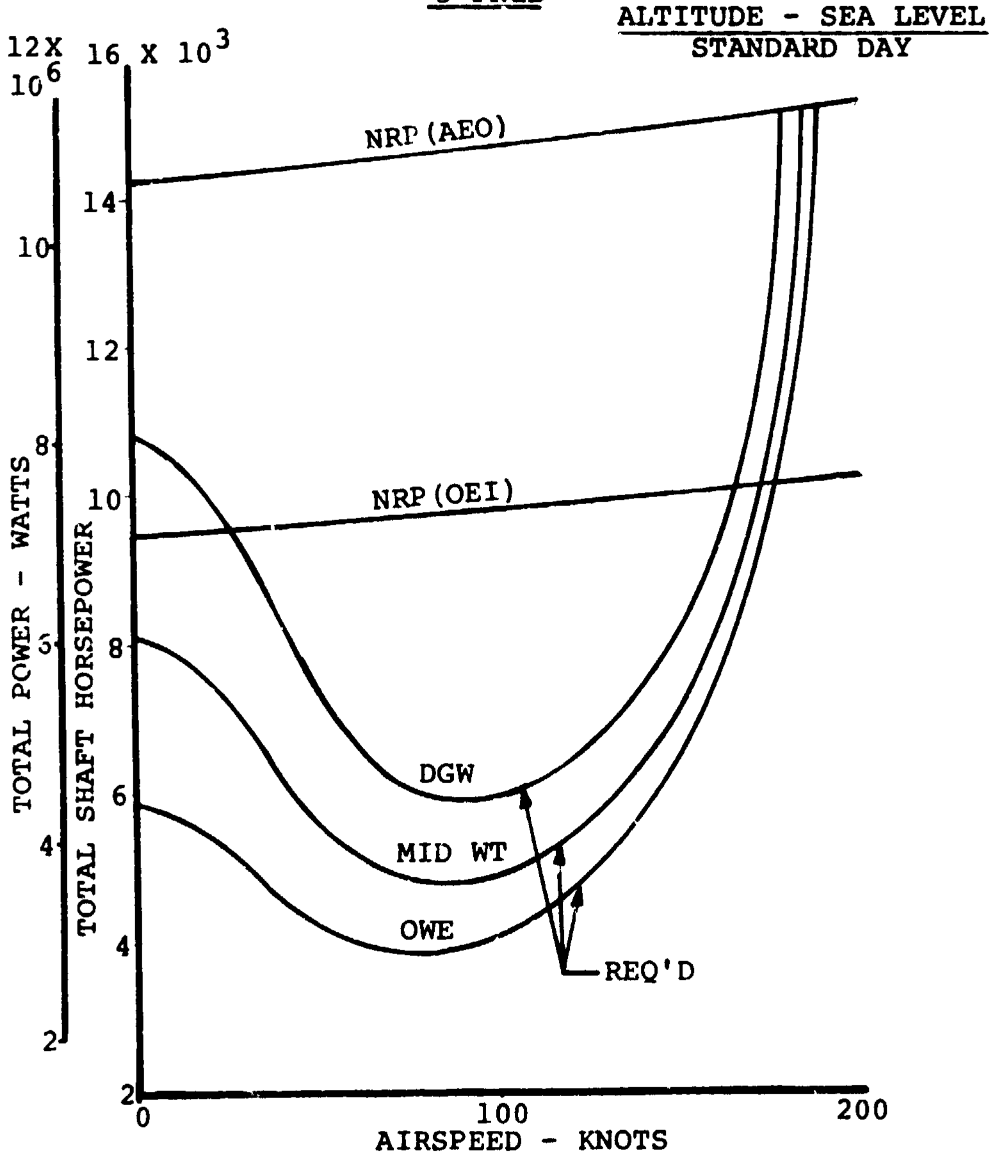


FIGURE 3.29. CRUISE PERFORMANCE - POWER REQUIRED/AVAILABLE.
-5 PNdB HELICOPTER

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/87.1 PNdB

STANDARD DAY CRUISE RPM

DGW = 74,227 LBS/33,669 Kg
MIDWT= 61,374 LBS/27,839 Kg
OWE = 48,520 LBS/22,008 Kg

-5 PNdB

ALTITUDE - 5000 FEET (1524 m)

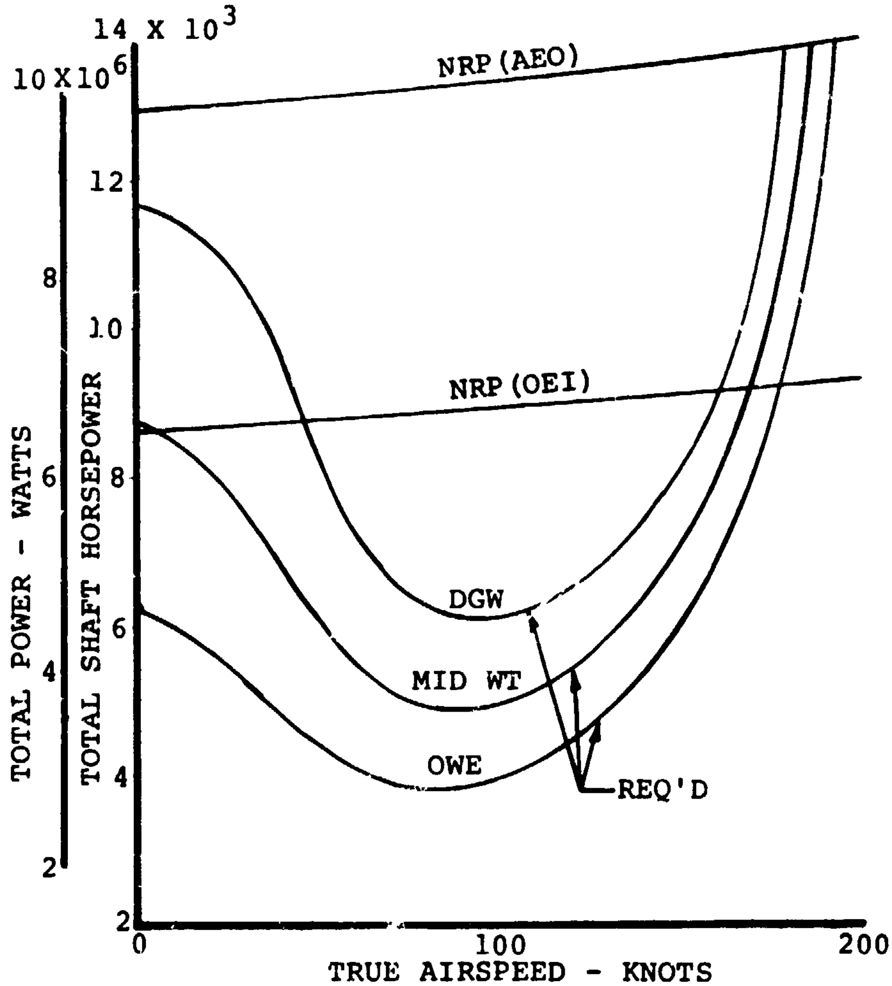


FIGURE 3.30. CRUISE PERFORMANCE - POWER REQUIRED/AVAILABLE.
-5 PNdB HELICOPTER

knots at sea level and 161 knots at 5,000 feet altitude. At lower weights the cruise speeds increase as shown in Figures 3.29 and 3.30.

The maximum cruise speeds are plotted as a function of altitude in Figure 3.31.

Rate of Climb

At design gross weight, all engines operating, the aircraft has a maximum rate of climb of 3850 feet per minute at sea level as shown in Figure 3.32. At design cruise altitude (5,000 feet) a rate of climb of 3,330 feet per minute can be achieved. With one engine inoperative a rate of climb of 1,930 feet per minute can be achieved at sea level dropping to 1,440 feet per minute at 5,000 feet altitude. The maximum rate of climb is obtained at operating weight empty at sea level. With all engines operating a rate of climb of 7,110 feet per minute can be achieved at this condition.

Specific Range

The specific range performance, all engines operating, is shown in Figures 3.33 and 3.34 at sea level and 5,000 feet altitude. The maximum specific range achieved at design gross weight is .039 nautical miles per pound of fuel at sea level, and .042 nautical miles per pound of fuel at 5,000 feet. At normal rated power these values decrease to 0.0365 and 0.038 nautical miles per pound of fuel respectively.

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/87.1 PNdB

STANDARD DAY
ALL ENGINES OPERATING
& ONE ENGINE INOPERATIVE

CRUISE RPM
NORMAL RATED POWER

DGW = 74.227 LBS/33,669 Kg
OWE = 48,520 LBS/22,008 Kg

-5 PNdB

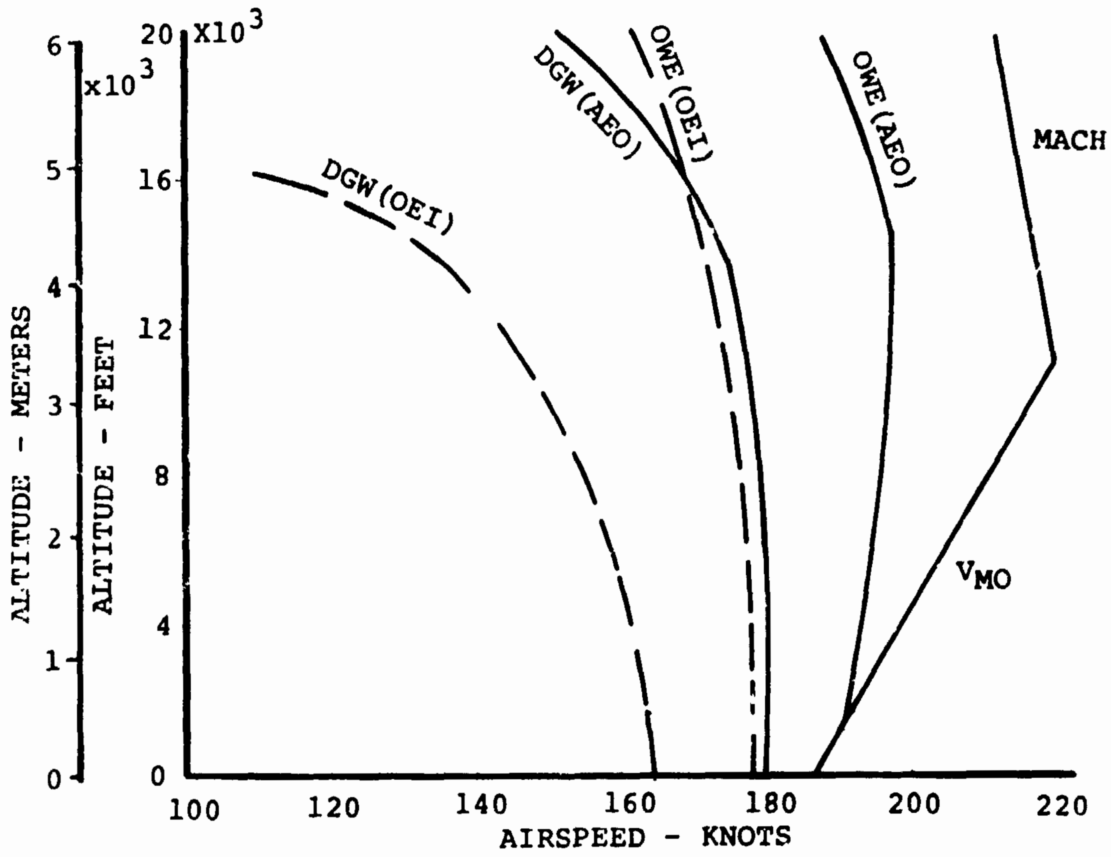


FIGURE 3.31. LEVEL FLIGHT CRUISE SPEED ENVELOPE.

-5 PNdB HELICOPTER

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/87.1 PNdB

CLIMB CAPABILITY — TAKEOFF RPM
STANDARD DAY — MIL POWER
ALL ENGINES OPERATING — ONE ENGINE INOPERATIVE
DGW = 74,229 LBS/33,669 Kg
OWE = 48,520 LBS/22,008 Kg

-5 PNdB

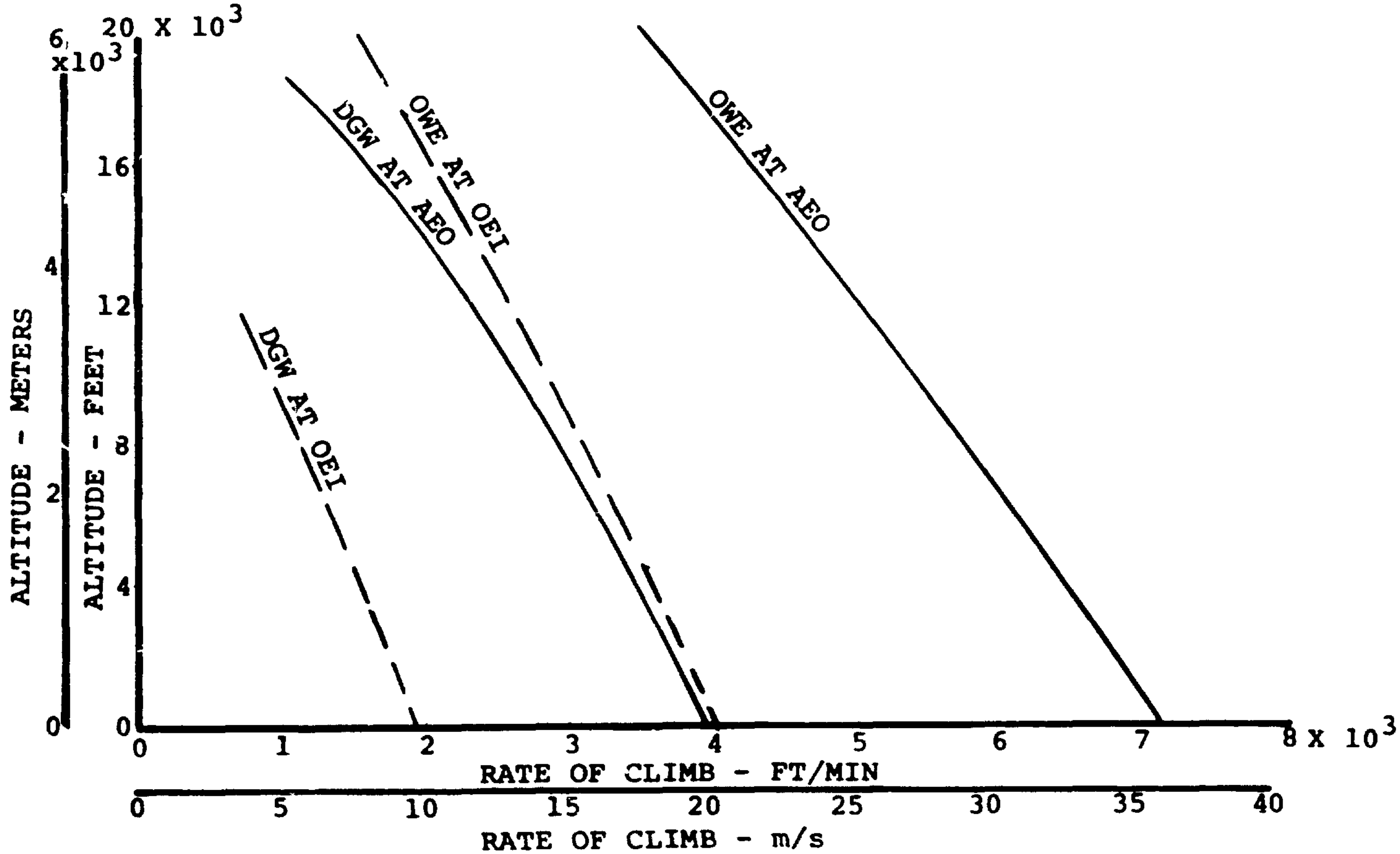


FIGURE 3.32 . -5PNdB DESIGN POINT HELICOPTER- CLIMB CAPABILITY.

NOISE DERIVATIVE AIRCRAFT

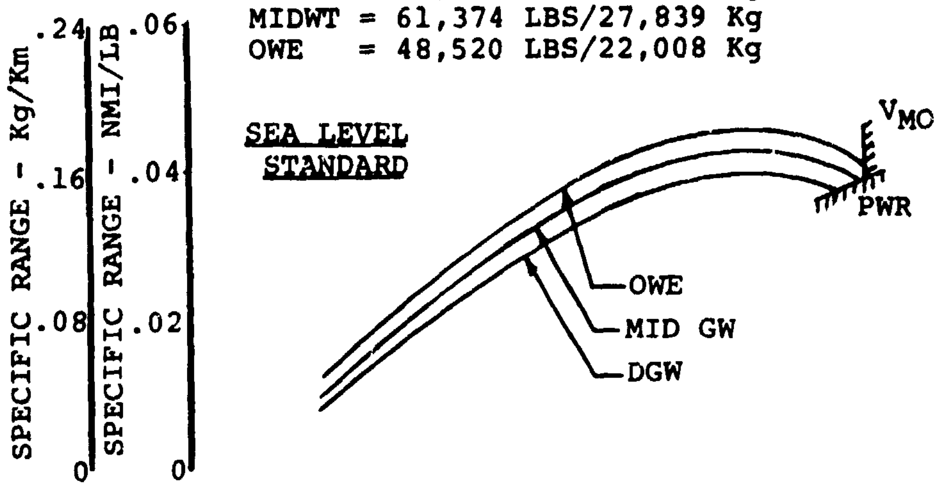
TANDEM HELICOPTER/100 PASSENGER/87.1 PNdB

CRUISE RPM STANDARD DAY

ALL ENGINES OPERATING

-5 PNdB

DGW = 74,227 LBS/33,699 Kg
MIDWT = 61,374 LBS/27,839 Kg
OWE = 48,520 LBS/22,008 Kg



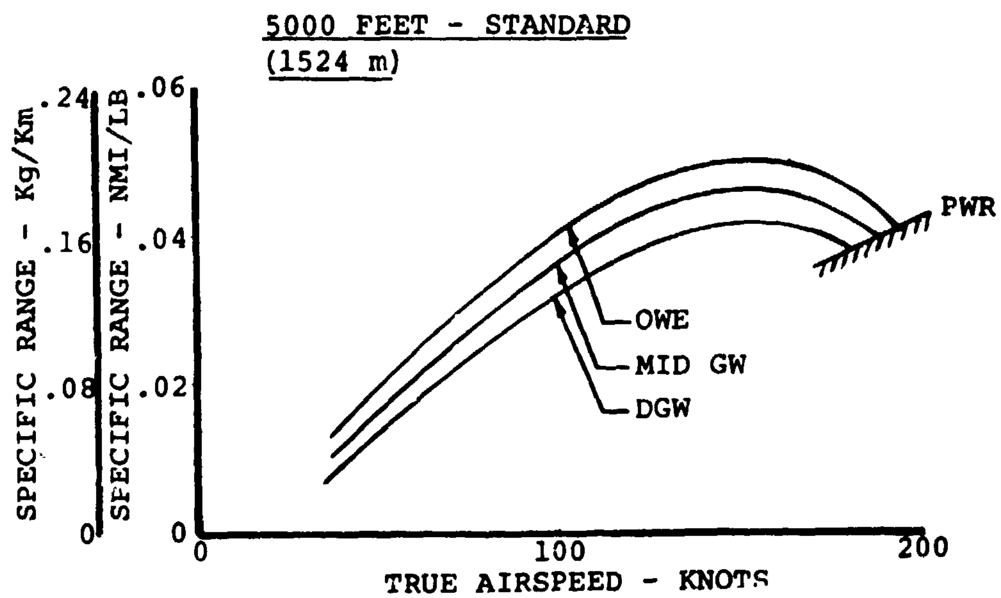


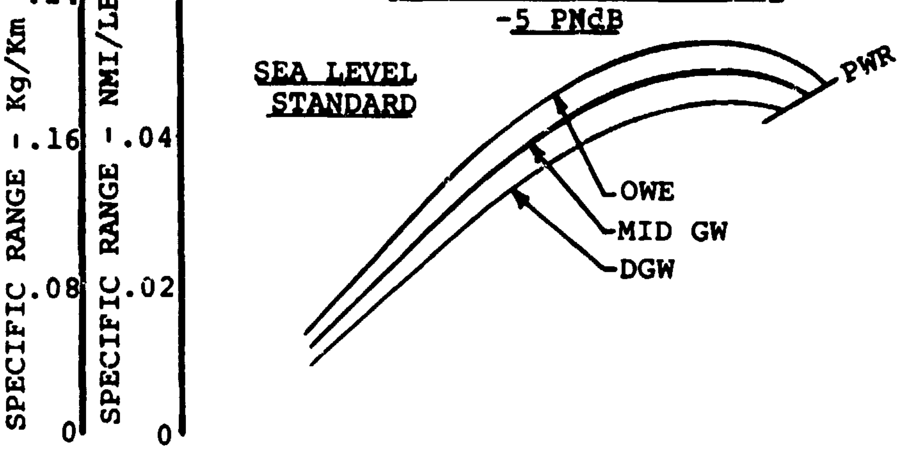
FIGURE 3.33. CRUISE PERFORMANCE - SPECIFIC RANGE. -5 PNdB HELICOPTER - AEO.

NOISE DERIVATIVE AIRCRAFT

TANDEM HELICOPTER/100 PASSENGER/87.1 PNdB

STANDARD DAY CRUISE RPM
DGW = 74,227 LBS/33,699 Kg
MIDWT = 61,374 LBS/27,839 Kg
OWE = 48,520 LBS/22,008 Kg

ONE ENGINE INOPERATIVE
-5 PNdB



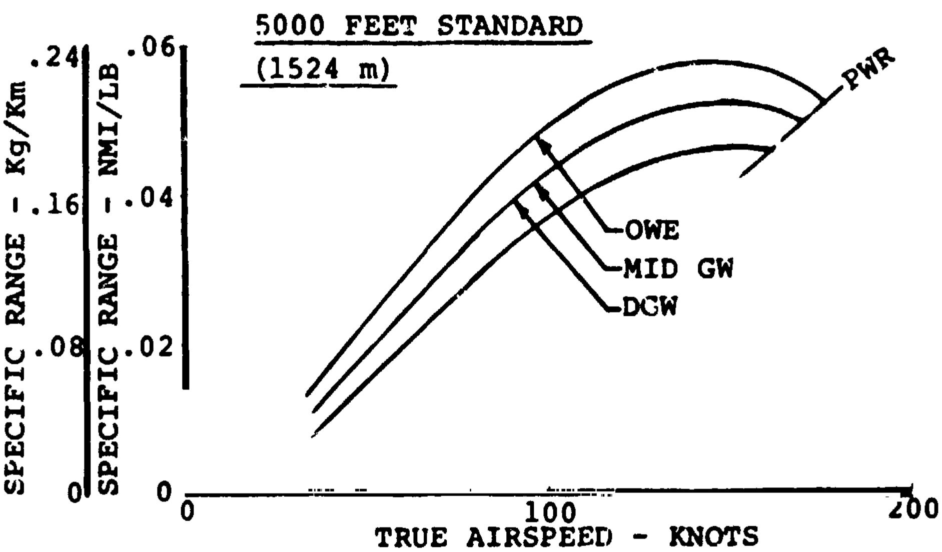


FIGURE 3.34 CRUISE PERFORMANCE - SPECIFIC RANGE.
-5 PNdB HELICOPTER - OEI.

With one engine inoperative the specific range improves because of the higher power setting on the remaining two engines resulting in lower SFC. At design gross weight a maximum specific range of 0.0445 nautical miles per pound is obtained at sea level and 0.0465 nautical miles per pound at 5,000 feet altitude.

Payload Range

The payload range performance is defined by the sizing mission and is shown to provide 200 nautical miles range in Figure 3.35. The basic mission fuel limit gives a range of 225 nautical miles for zero payload with no change in mission reserve fuel.

Additional tankage could be provided at a weight penalty equivalent of two passengers which would allow operation 400 nautical miles with 70 passengers on board.

The improvement in specific range OEI increases the pa ad - range capability as shown in Figure 3.36. The basic TH-100 (87.1) design can operate fully loaded out to 246 nautical miles with one engine shut down.

Stability and Control - TH-100 (87.3)

The longitudinal angle of attack stability is shown in Figure 3.37 for the unaugmented aircraft. As required by the guideline criteria M_α is negative (i.e., positive angle of attack stability in cruise). This is achieved by 26.3 degrees delta three in the forward rotor head. The pitch stability is slightly higher than the baseline aircraft in

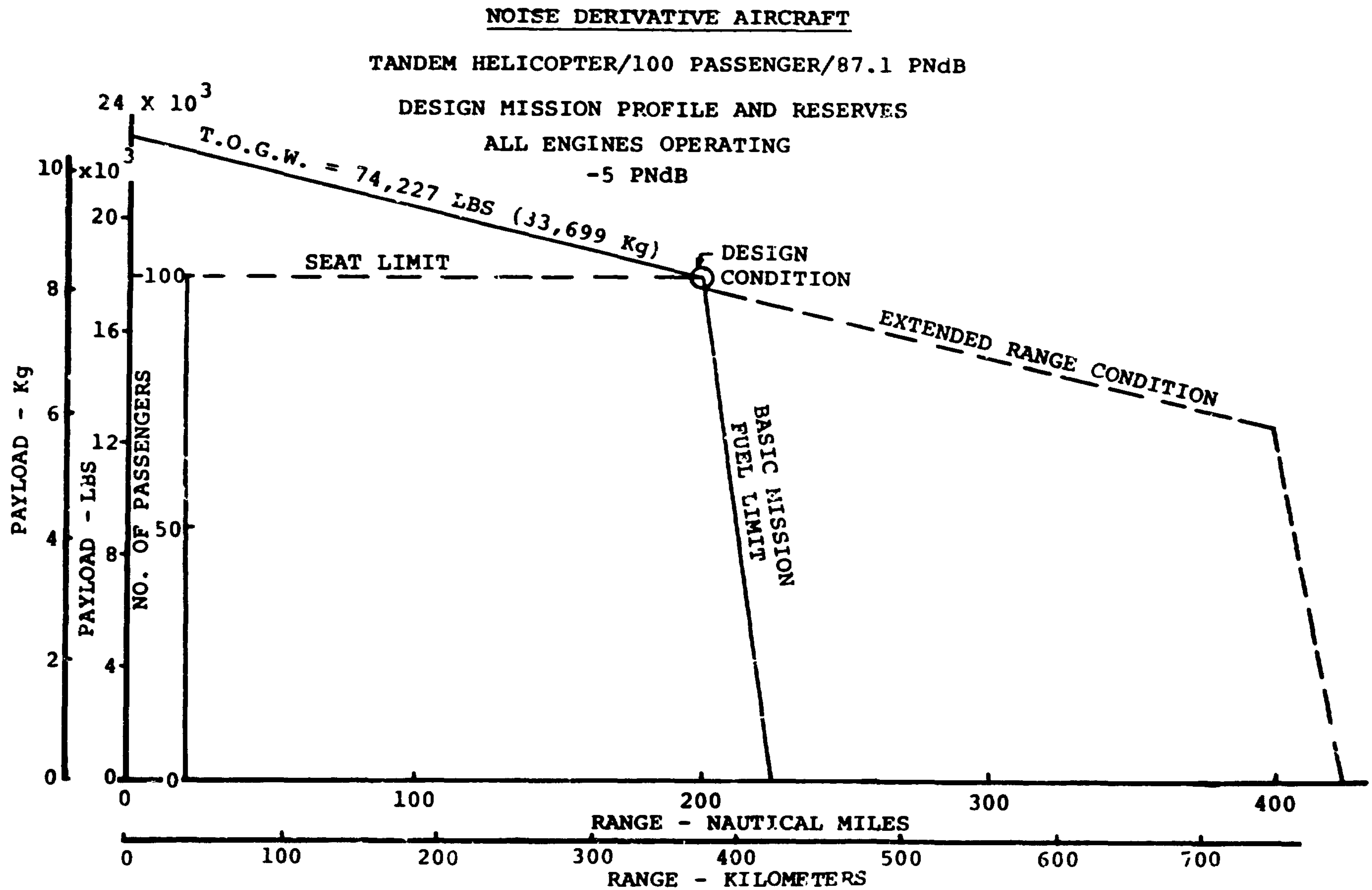


FIGURE 3.35. PAYLOAD RANGE CAPABILITY - CRUISE AT NRP - -5 PNdB HELICOPTER - AEO.

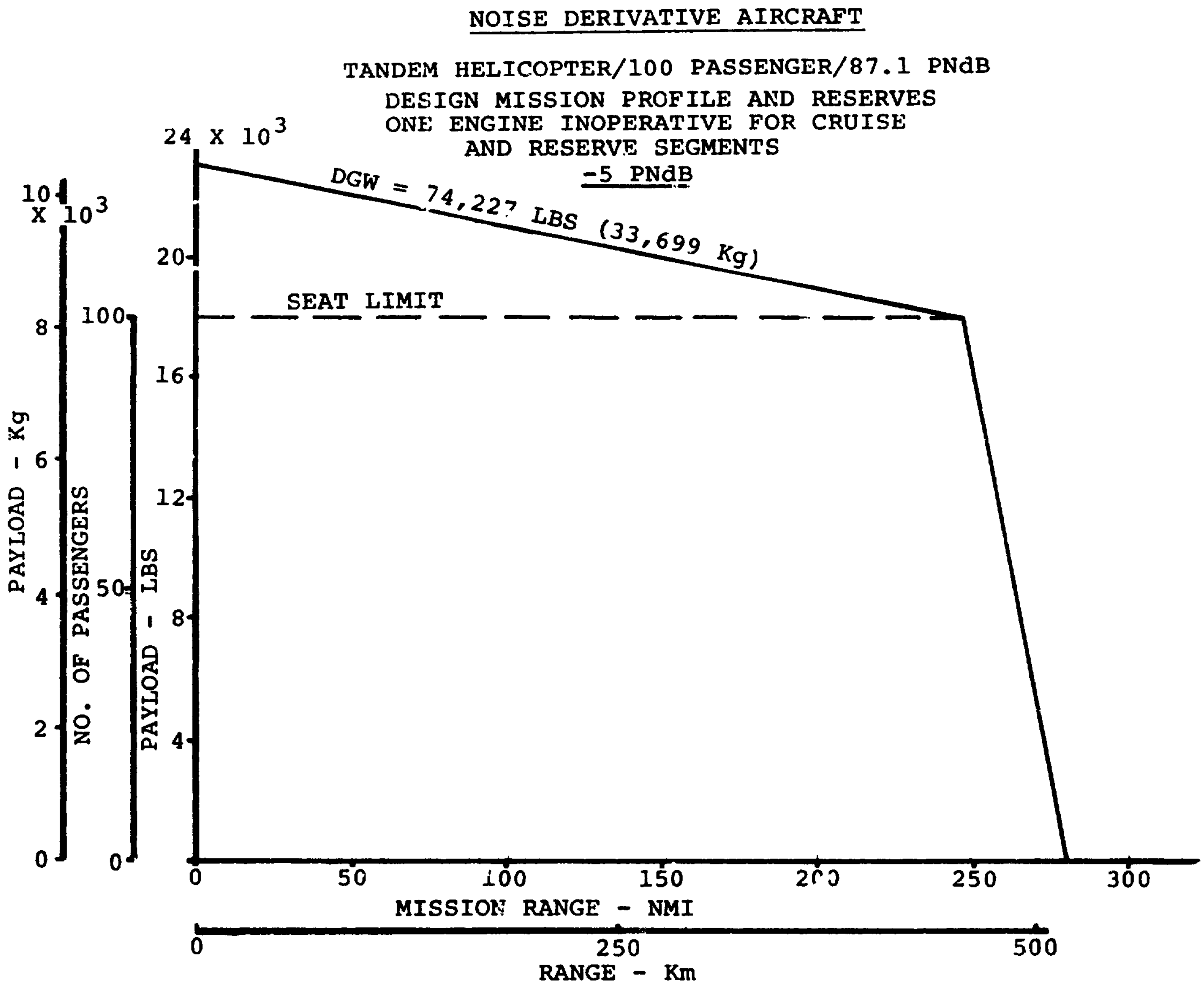


FIGURE 3.36. PAYLOAD RANGE CAPABILITY - CRUISE AT NRP - -5 PNdB HELICOPTER - OEI.

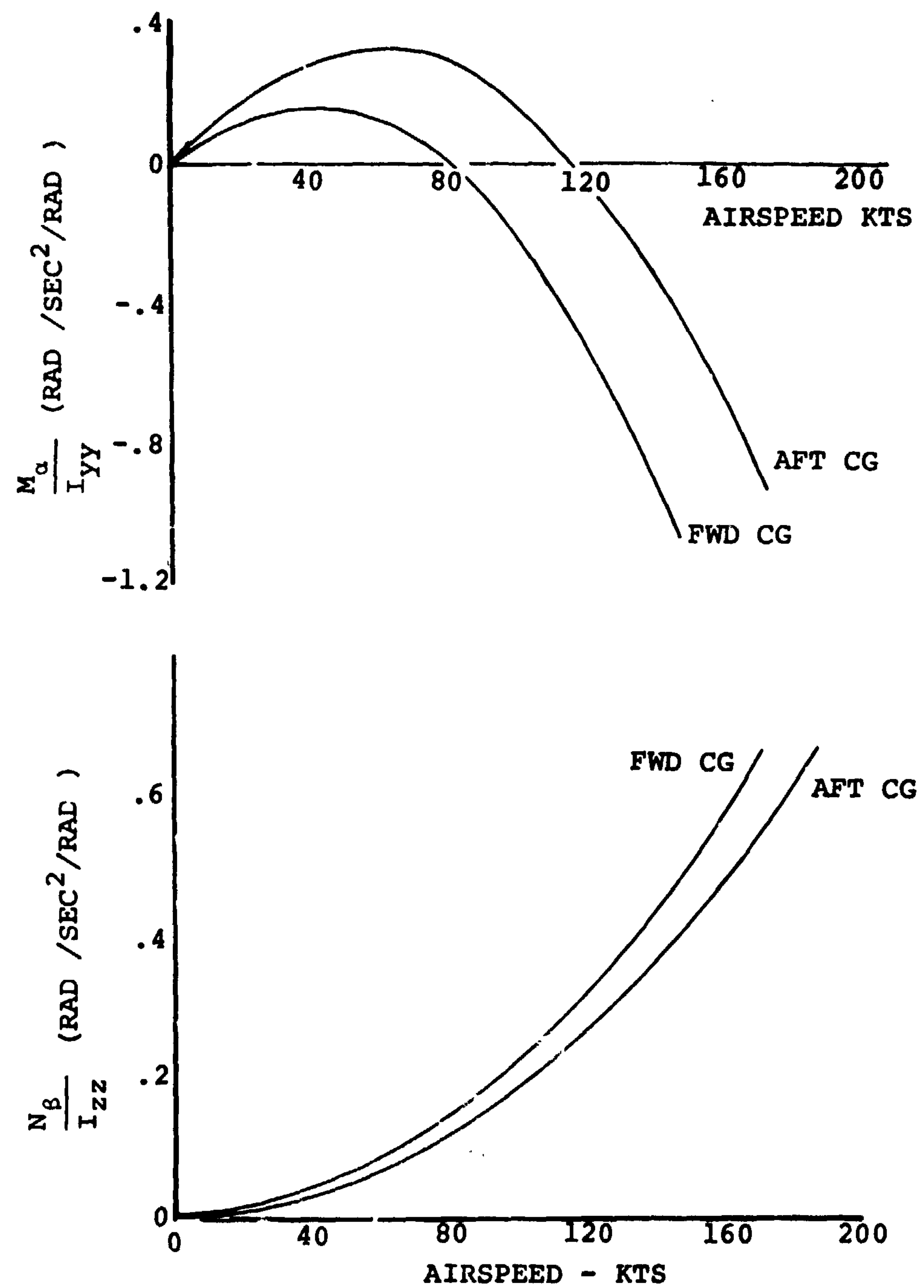


FIGURE 3.37. STATIC STABILITY - TANDEM HELICOPTER -5 PNdB (87.1PNdB)

cruise.

The lateral-directional stability derivative N_{β} is positive (i.e., stable) over the entire speed range.

The rotor control derivatives are tabulated against airspeed in Table 3.14 and compared with the baseline aircraft. The control derivatives are higher than the baseline aircraft in all cases. This means that smaller control ranges could be used to obtain the same control powers as the baseline tandem rotor helicopter or conversely larger control powers would be available with the same control authorities.

The damping derivatives of the quiet helicopter are not significantly different from the baseline aircraft except that higher damping is available at high speed. The control response of this aircraft unaugmented will be slightly worse than the baseline, however, the differences in the damping derivatives are insignificant compared with the normal automatic flight control system gains, for example

ΔM_q = 2.0 to 2.5 radians per second squared per radians per second

ΔM_{α} = 3.0 to 5.0 radians per second squared per radian.

The augmented aircraft would not be noticeably different from the baseline tandem helicopter.

<u>Tandem Helicopter - TH-100 (87.1) - Noise</u>

The external noise design criteria for this aircraft is 87.1 PNdB at 500 feet sideline distance in hover (i.e., 5 PNdB

AIRCRAFT	V	0	40	80	120	150		
BASELINE W=67,175	L_{δ_S}	.433	.410	.407	.405	.406		
	M_{δ_B}	.183	.181	.213	.241	.257		
*	N_{δ_R}	.170	.155	.151	.149	.153		
	Z_{δ_C}	-7.33	-6.65	-7.67	-9.03	-9.65		
		CONTROL DERIVATIVES						
	L_p	-.70	-.80	-.90	-.84	-.70	TH-100 (92.3)	
	M_q	-.70	-.92	-1.19	-1.27	-1.34		
	N_r	-.07	-.05	-.04	-.05	-.07		
	Z_w	-.22	-3.2	-.43	-.50	-.52		
		DAMPING DERIVATIVES						
	M_α FWD	0	.15	.03	-.49	-1.12		
	AFT	0	.28	.29	-.05	-.49		
	N_β FWD	0	.04	.15	.33	.52		
	AFT	0	.03	.12	.27	.42		
		STABILITY DERIVATIVES						
QUIET W=74,225	L_{δ_S}	.492	.462	.452	.443	.437		1
	M_{δ_B}	.189	.192	.233	.274	.307		2
	N_{δ_R}	.182	.166	.162	.159	.163		3
	Z_{δ_C}	-8.13	-7.45	-8.66	-10.28	-11.10		4
	L_p	-.64	-.74	-.85	-.79	-.66	TH-100 (87.3)	5
	M_q	-.77	-1.10	-1.48	-1.67	-1.84		
	N_r	-.09	-.07	-.05	-.05	-.06		
	Z_w	-.23	-.33	-.46	-.57	-.64		6
	M_α FWD	0	.03	-.21	-.86	-1.62		7
	AFT	0	.18	.13	-.28	-.90		
	N_β FWD	0	.04	.16	.36	.47		
	AFT	0	.03	.13	.29	.38		

1 ROLL - RADS/SEC^2/INCH

2 PITCH - RADS/SEC^2/INCH

3 YAW - RADS/SEC^2/INCH

4 VERT - FT/SEC^2/INCH

5 RATE DERIVATIVES - RADS/SEC^2/RAD/SEC

6 FT/SEC^2/FT/SEC

7 STABILITY DERIVATIVES

TABLE 3.14 . STABILITY & CONTROL DERIVATIVES FOR -5 PNdB ALTERNATE CONFIGURATION AT DESIGN GROSS WEIGHT.

quieter than the baseline aircraft).

The octave band hover spectra are plotted in Figure 3.38 for the total aircraft as well as for each component of the overall noise level. The engine inlet is treated the same with an identical amount of suppression of the inlet noise as the baseline aircraft and the overall sound pressure level is set by the rotor broad band noise over most of the frequency range. However, the engine inlet still dominates above 4 KHz because of the low level of rotor noise. The NOY distribution is also shown which is used in the definition of the perceived noise level of 87.1 PNdB for this aircraft.

The perceived noise level footprint for a typical takeoff profile is shown in Figure 3.39. The takeoff footprint for 95 PNdB is quite small in this case indicating no perceived noise above 95 PNdB at more than 1,000 feet from the point of takeoff. The highest noise levels are observed along the flight path and the time histories of perceived noise at various distances along the flight path are also shown in Figure 3.40. The landing PNL contours, Figure 3.41, are elongated as before but much narrower than the baseline or +5 PNdB case. The time histories of perceived noise for the landing case are shown in Figure 3.42.

TH-100 (87.1) - Costs

Direct operating costs per seat mile and seat kilometer as a function of block distance are shown in Figure 3.43 for the specified combinations of aircraft utilization and airframe

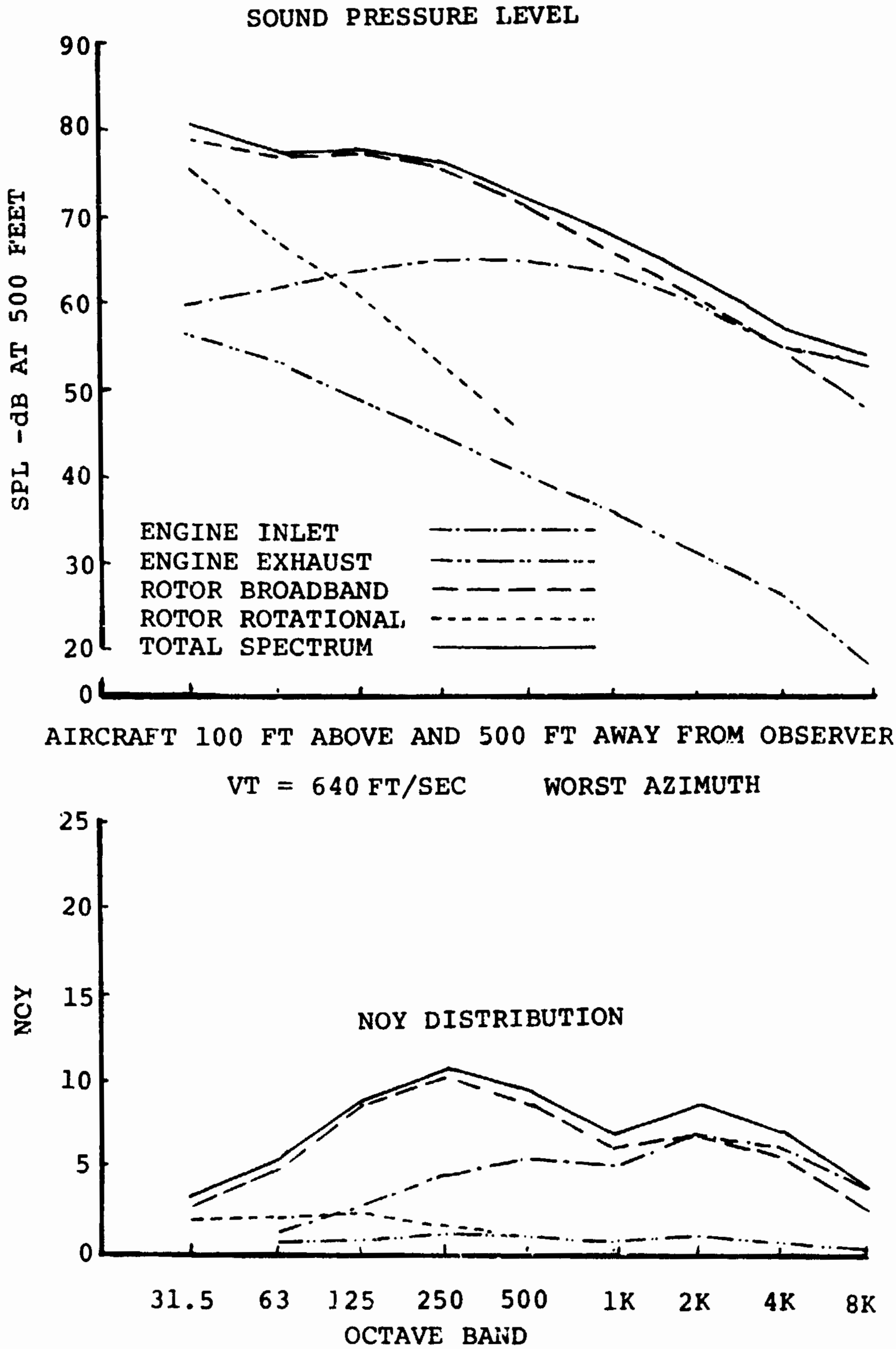


FIGURE 3.38. -5 PNdB HELICOPTER - HOVER NOISE SPECTRUM AND NOY DISTRIBUTION - 87.1 PNdB.

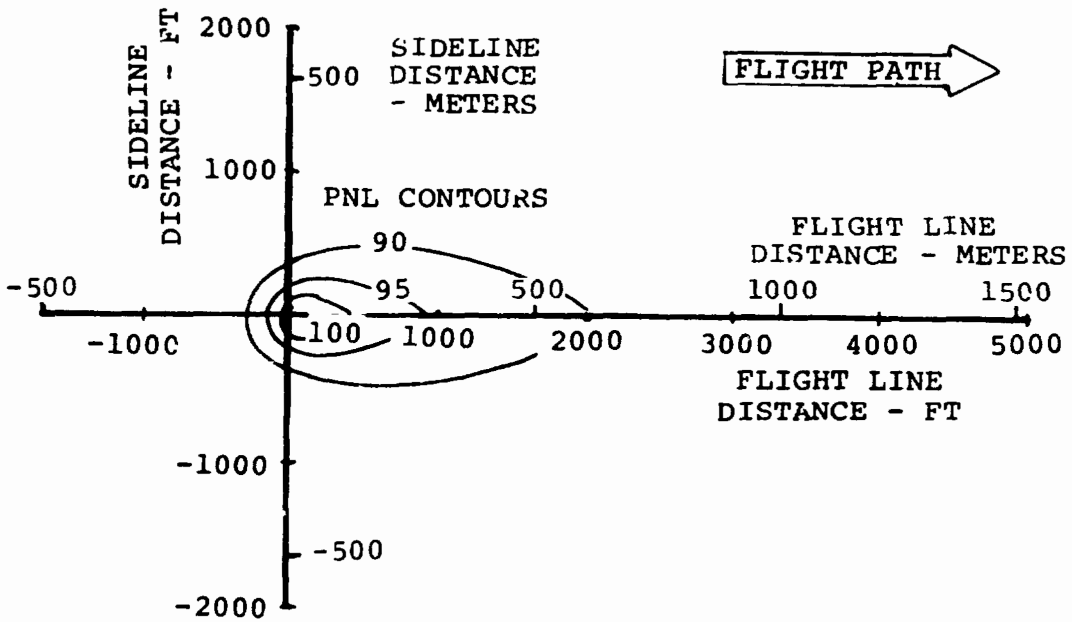


FIGURE 3.39. -5 PNdB DESIGN POINT HELICOPTER - STANDARD TAKEOFF - PNL CONTOURS.

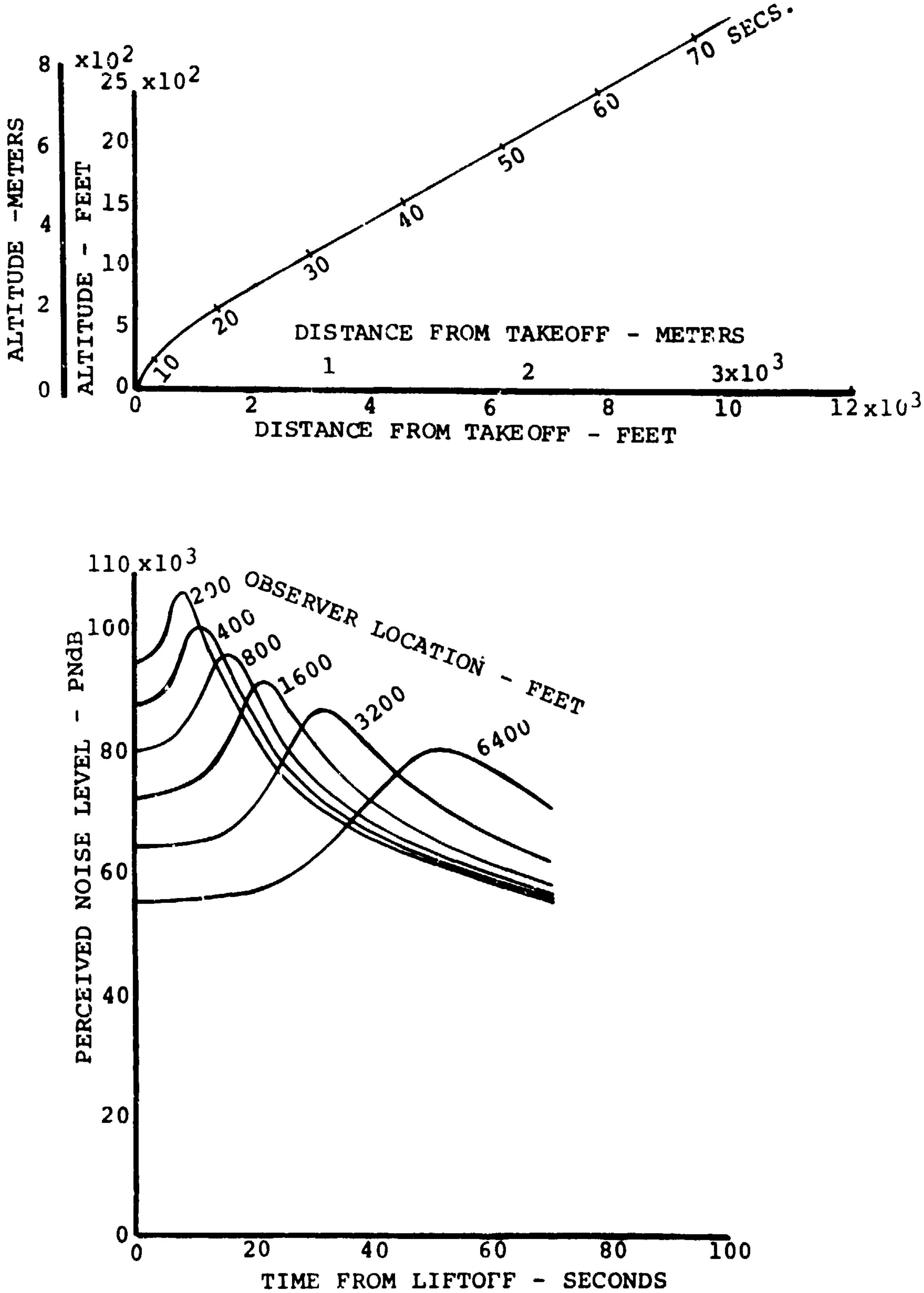


FIGURE 3.40. -5 PNdB HELICOPTER DESIGN POINT STANDARD TAKEOFF. PERCEIVED NOISE.

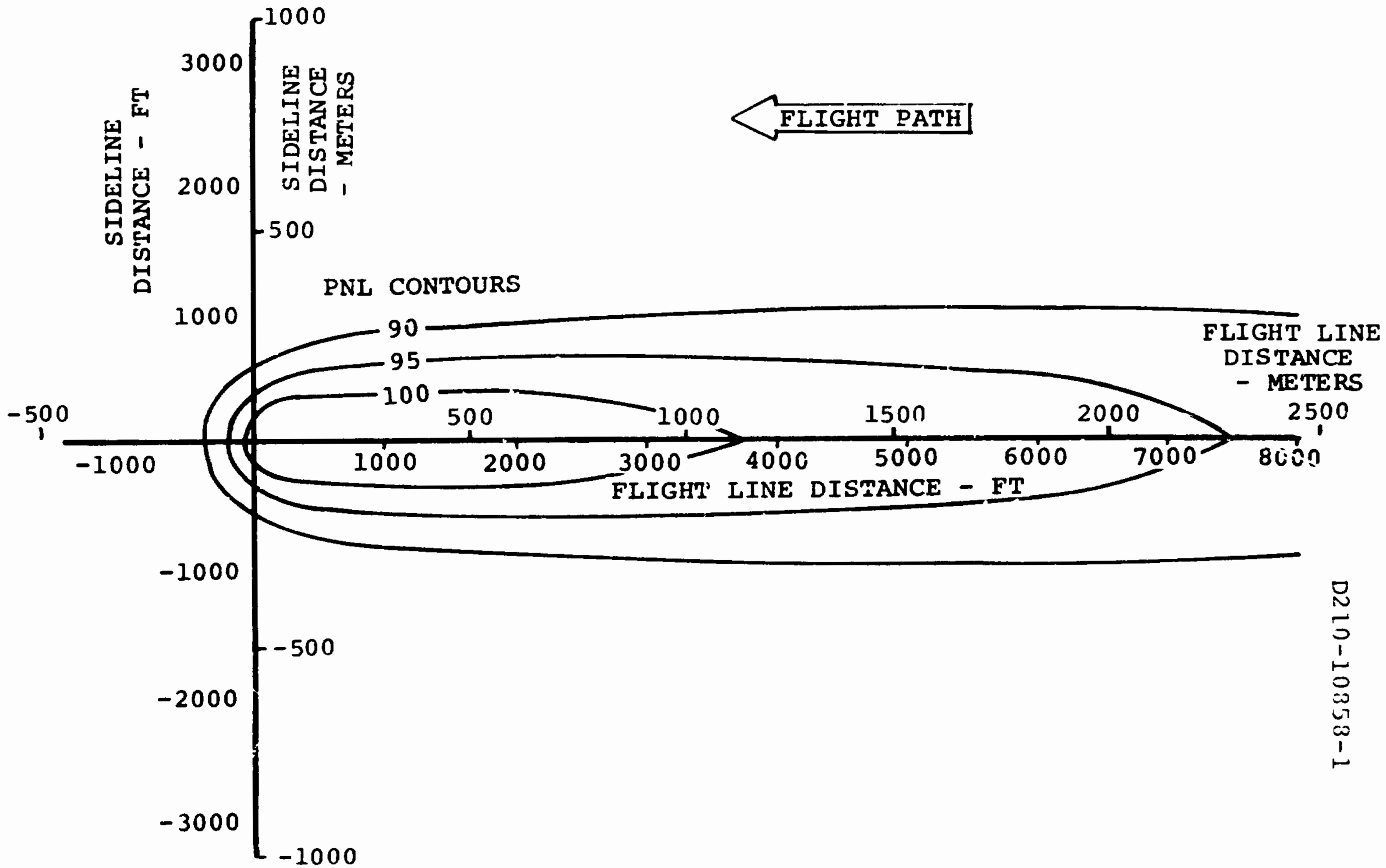


FIGURE 3.41. -5 PNdB HELICOPTER - STANDARD LANDING - PNL CONTOURS.

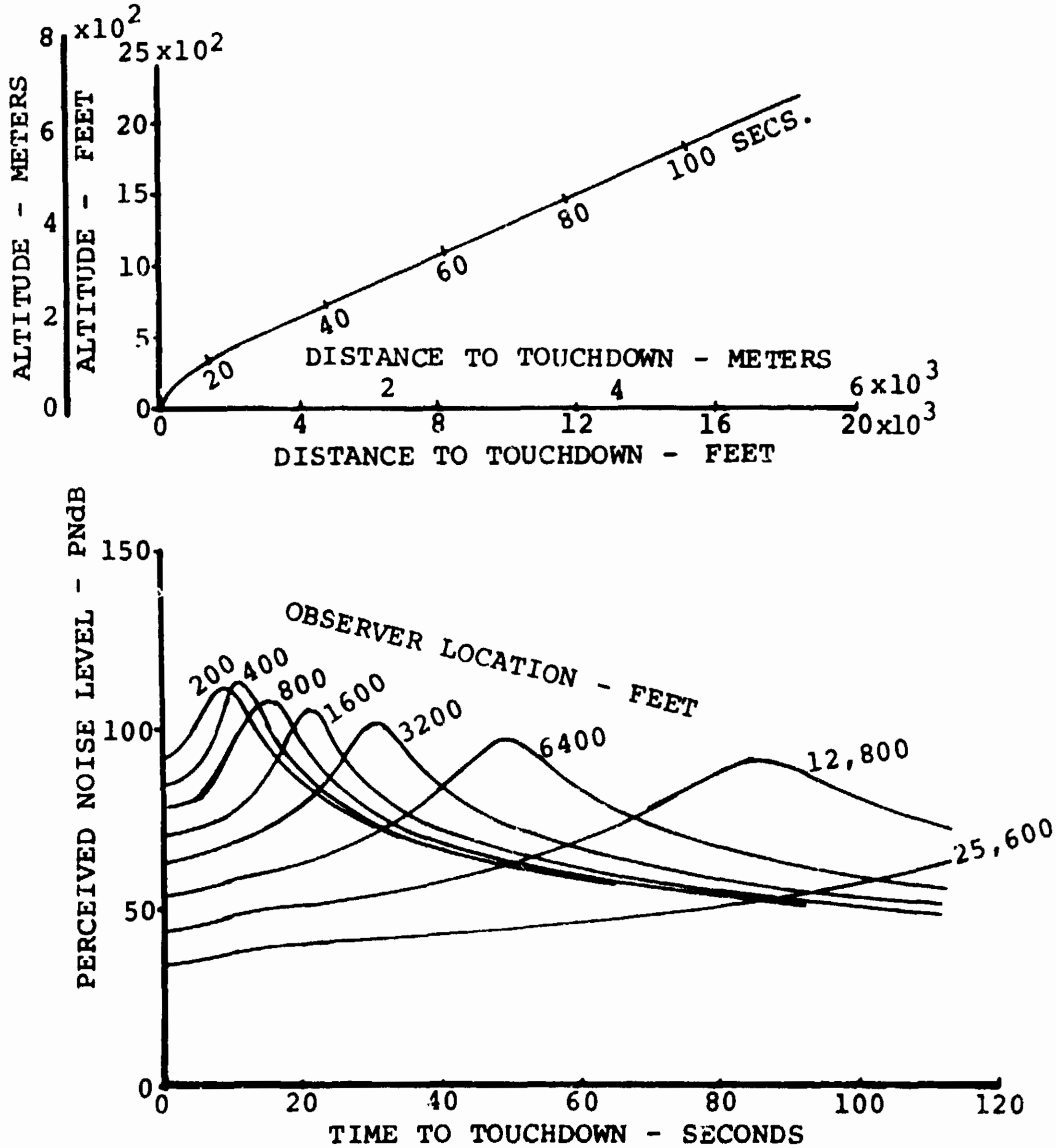


FIGURE 3.42. -5 PNdB HELICOPTER - STANDARD LANDING - PERCEIVED NOISE.

TH-100(87.1)

Flyaway Costs

Airframe Cost	$90.00/Lb	$110.00/Lb
Airframe	$2,408,670	$2,943,930
Dynamic System	1,351,200	1,351,200
Engines	751,887	751,887
Avionics	250,000	250,000
Total	$4,761,757	$5,297,017

Direct Operating Costs
Dollars/Seat Mile
Block Distance = 230 St. Miles

Utilization (Hrs/Yr)	2500		3500	
Airframe Cost ($/Lb)	90	110	90	110
Flying Operations				
Flight Crew	.0075	.0075	.0075	.0075
Fuel and Oil	.0050	.0050	.0050	.0050
Hull Insurance	.002[illegible]	.0023	.0015	.0016
Total Flying Operations	.0146	.0148	.0140	.0141
Direct Maintenance				
Airframe - Labor	.0013	.0013	.0013	.0013
- Material	.0010	.0013	.0010	.0013
Engines - Labor	.0007	.0007	.0007	.0007
- Material	.0010	.0010	.0010	.0010
Dynamic System - Labor	.0013	.0013	.0013	.0013
- Material	.0019	.0019	.0019	.0019
Total Direct Maintenance	.0072	.0075	.0072	.0075
Maintenance Burden	.0050	.0050	.0050	.0050
Total Maintenance	.0122	.0125	.0122	.0125
Depreciation	.0100	.0111	.0072	.0079
Total Direct Costs	.0368	.0384	.0334	.0345

TABLE 3.15. -5 PNdB HELICOPTER - INITIAL AND DIRECT OPERATING COSTS.

TH-100(87.1)
EXTENDED RANGE VERSION

Airframe Cost	Flyaway Costs $90.00/Lb	$110.00/Lb
Airframe	$2,440,170	$2,982,430
Dynamic System	1,351,200	1,351,200
Engines	751,887	751,887
Avionics	250,000	250,000
Total	$4,793,257	$5,335,515

Direct Operating Costs
Dollars/Seat Mile
Block Distance = 230 St. Miles

Utilization (Hrs/Yr)	2500		3500	
Airframe Cost ($/Lb)	90	110	90	110
Flying Operations				
Flight Crew	.0076	.0076	.0076	.0076
Fuel and Oil	.0051	.0051	.0051	.0051
Hull Insurance	.0021	.0023	.0015	.0017
Total Flying Operations	.0148	.0150	.0142	.0144
Direct Maintenance				
Airframe - Labor	.0014	.0014	.0014	.0014
- Material	.0011	.0013	.0011	.0013
Engines - Labor	.0007	.0007	.0007	.0007
- Material	.0011	.0011	.0011	.0011
Dynamic System - Labor	.0013	.0013	.0013	.0013
- Material	.0020	.0020	.0020	.0020
Total Direct Maintenance	.0076	.0078	.0076	.0078
Maintenance Burden	.0052	.0052	.0052	.0052
Total Maintenance	.0128	.0130	.0128	.0130
Depreciation	.0103	.0114	.0074	.0081
Total Direct Costs	.0379	.0394	.0344	.0355

TABLE 3.16 -5 PNdB - INITIAL AND DIRECT OPERATING COSTS (EXTENDED RANGE VERSION).

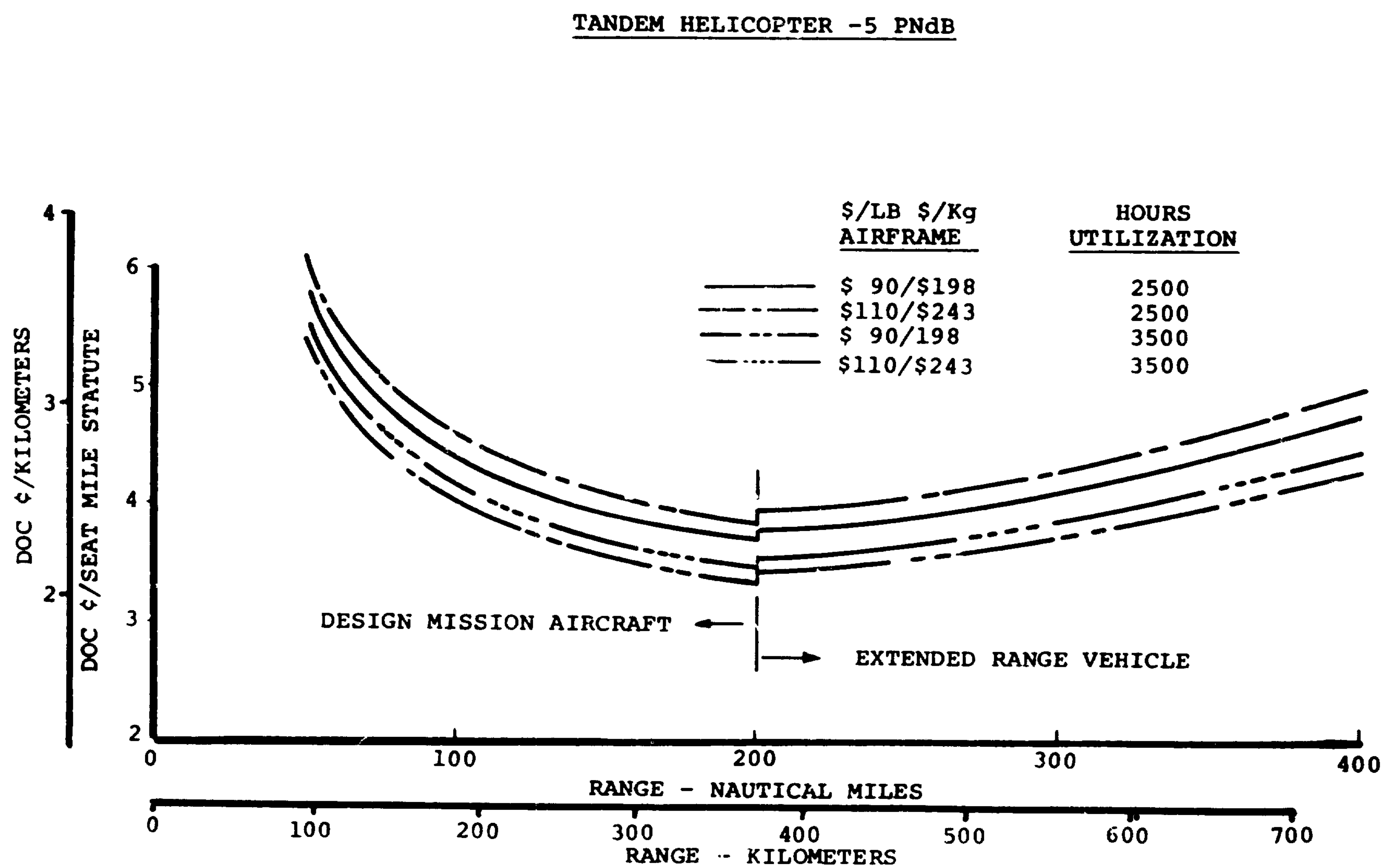


FIGURE 3.42. EFFECT OF OPERATING RANGE ON DIRECT OPERATING COST.

costs. Figure 3.43 also illustrates the impact of extending the design range of the TH-100 (87.1) to 460 statute miles. The increase in costs at the design point range (230 statute miles) is the result of the loss of 2 available seats due to the increased empty weight for the installation of larger fuel tanks. Although not shown in Figure 3.43, it should be noted that the larger fuel tanks will result in a small increase (less than 1%) in seat mile costs at ranges less than 230 statute miles due to increases in airframe maintenance and depreciation costs. In the extended range version of the TH-100 (87.1) seat mile costs show a continuing increase beyond 230 statute miles because of the loss of 30 available seats due to additional fuel requirements at the longer block distances.

Table 3.15 shows the flyaway costs for the basic TH-100 (87.1) at $90.00 and $110.00 per pound of airframe. A breakout of the direct operating cost factors for the TH-100 (87.1) at 230 statute miles is shown in Table 3.15. Flyaway and direct operating cost breakouts for the extended range version of the TH-100 (87.1) are shown in Table 3.16.

3.2 TILT ROTOR - SELECTION OF NOISE CRITERIA DESIGNS

The primary design parameters which influence the external noise of the tilt rotor are rotor tipspeed and solidity. The impact of these parameters on the aircraft gross weight, direct operating cost and sideline noise is shown in Figures 3.44, 3.45 and 3.46.

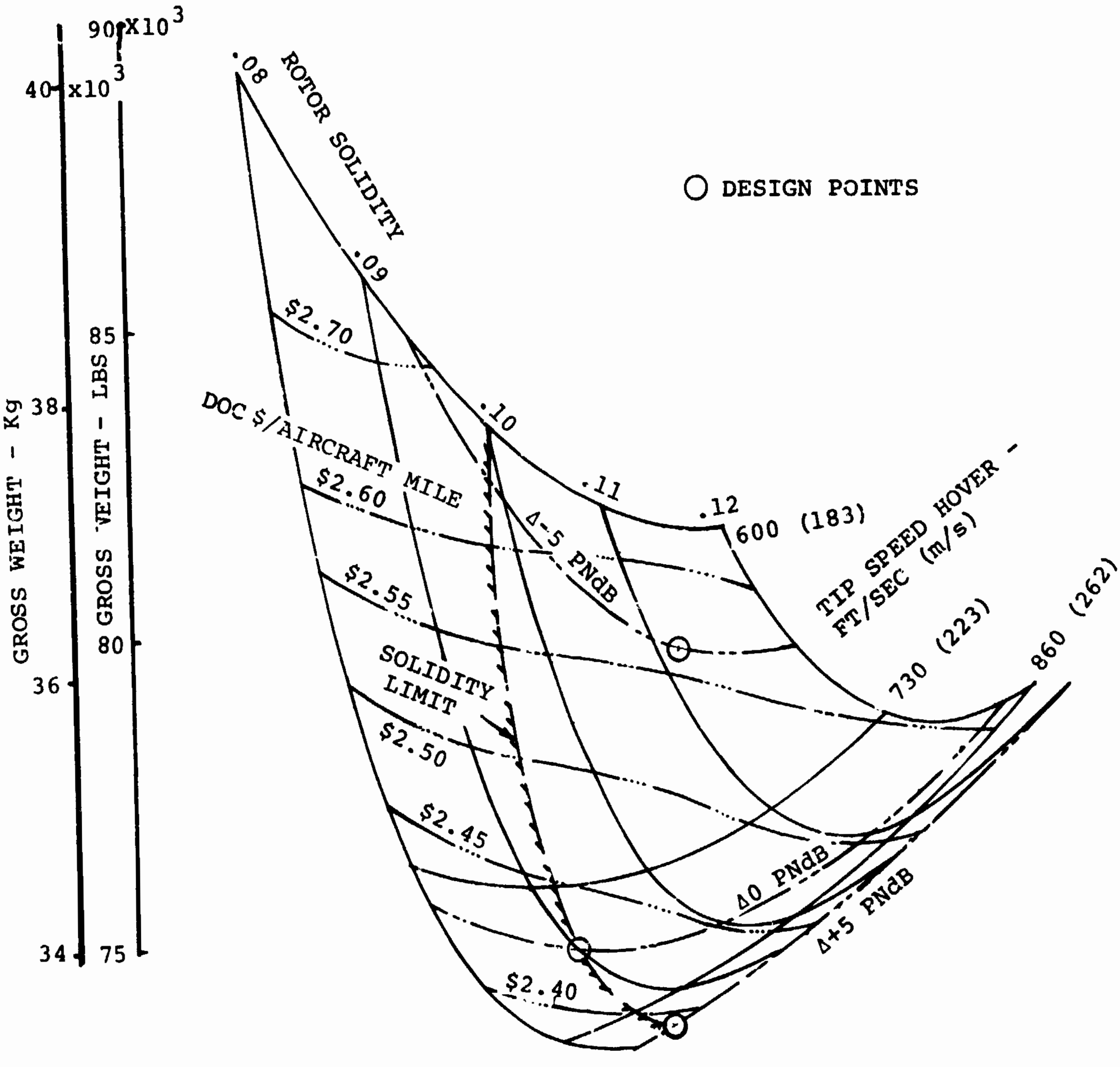


FIGURE 3.44. TILT ROTOR NOISE DERIVATIVE AIRCRAFT SELECTION CHART - GROSS WEIGHT.

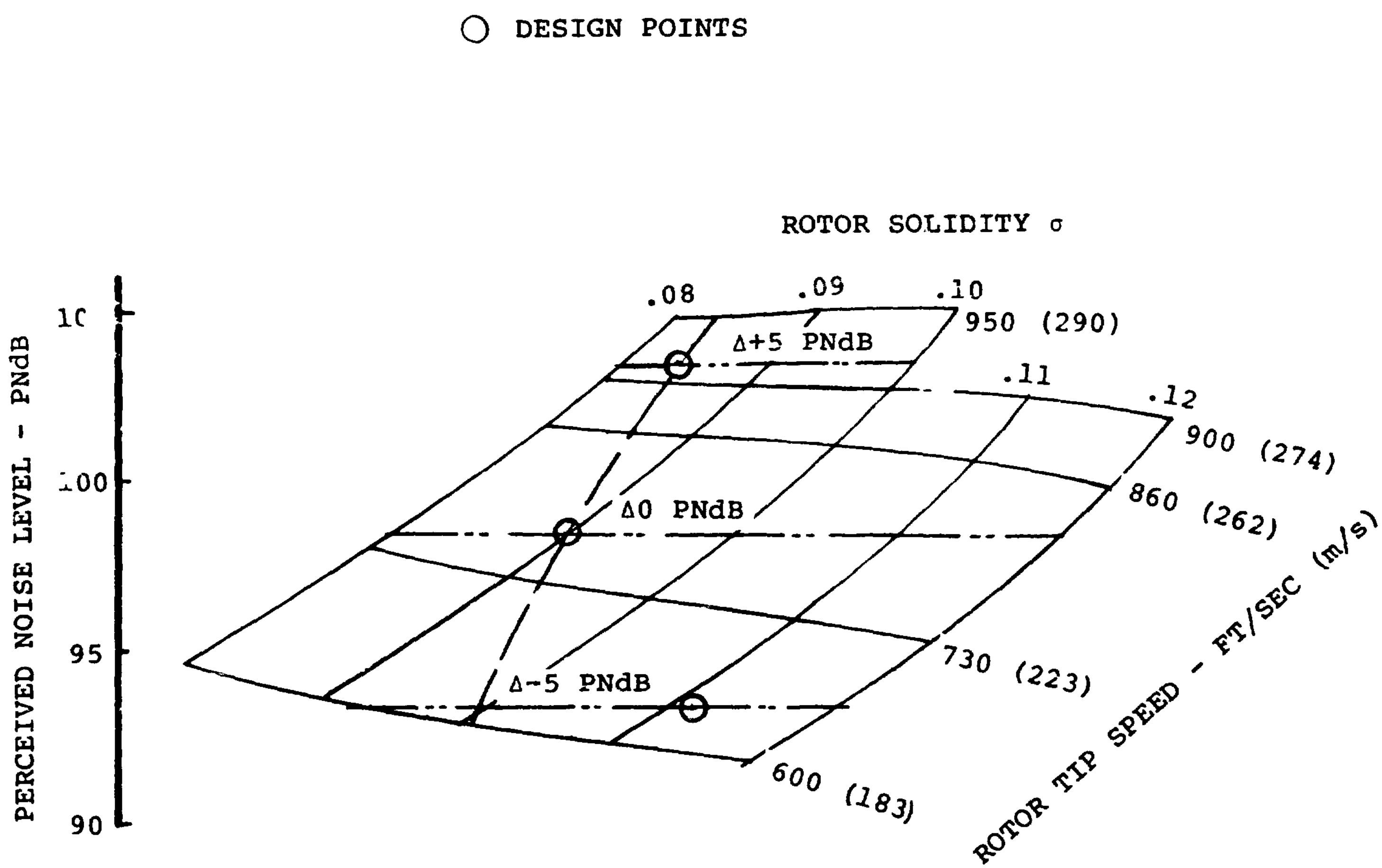


FIGURE 3.45. TILT ROTOR NOISE DERIVATIVE AIRCRAFT SELECTION CHART - ACOUSTICS.

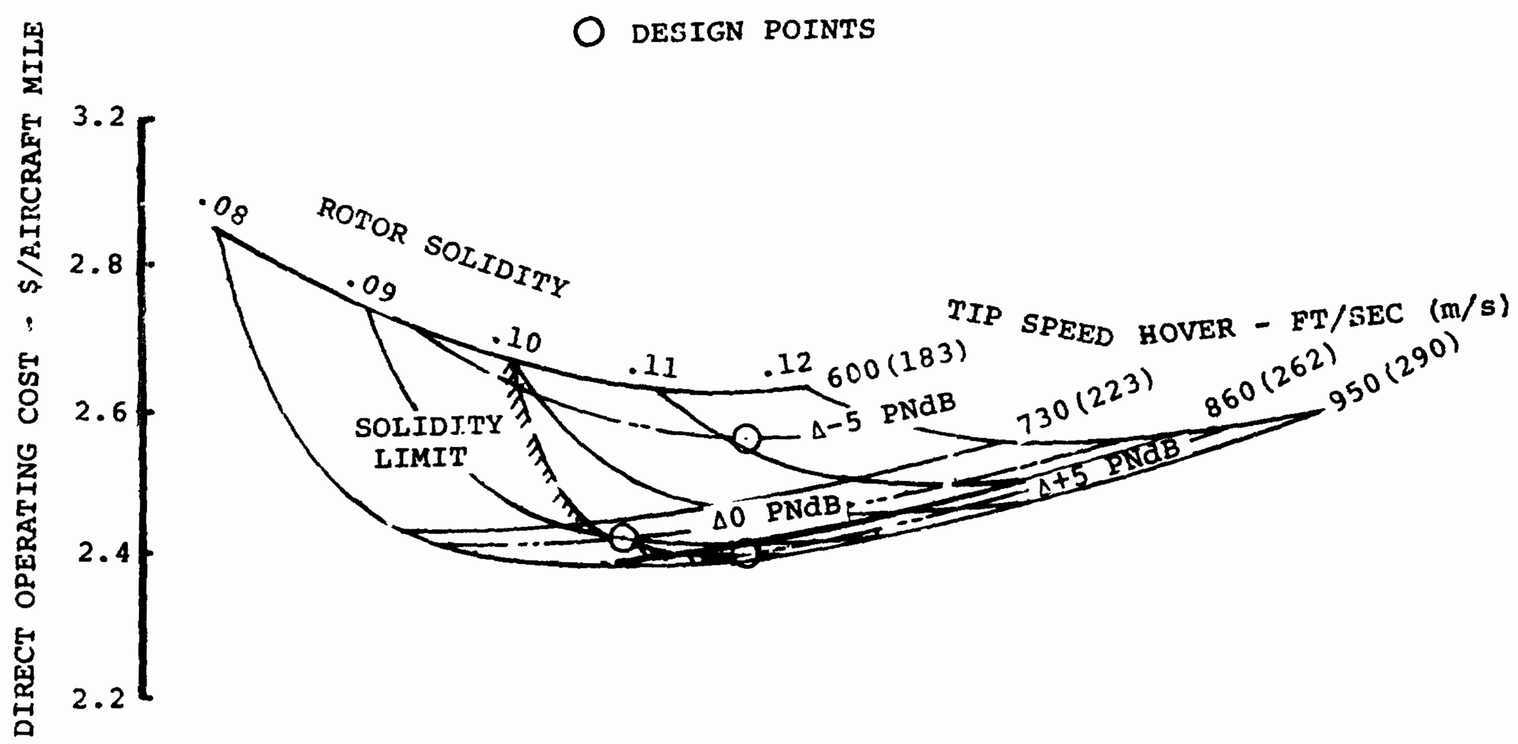


FIGURE 3.46. TILT ROTOR NOISE DERIVATIVE AIRCRAFT SELECTION CHART - DIRECT OPERATING COSTS.

The baseline aircraft sideline noise is 98.2 PNdB and therefore the noise derivative aircraft by definition will be on the 103.2 PNdB and 93.4 PNdB lines. Superimposing thes lines on the direct operating cost and gross weight plots defines the families of aircraft that meet the ± 5 PNdB criteria. The -5 PNdB aircraft line on the direct operating cost plot of Figure 3.46 shows a minimum DOC at a tipspeed of 635 feet per second. The +5 PNdB line does not show a minimum within the range of solidities shown and the aircraft selected is defined by the intersection of the +5 PNdB family and the minimum solidity for practical blade design. The aircraft selected has a tipspeed of 915 feet per second.

The two derivative aircraft designs selected in this manner are discussed in the following sections.

3.2.1 +5 PNdB Tilt Rotor - TR-100 (103.2)

This tilt rotor design is 5 PNdB noisier than the baseline tilt rotor with a perceived noise level of 103.2 PNdB at 500 feet sideline in hover.

Configuration and Layout

The basic aircraft cabin and cockpit of the aircraft is identical to the baseline aircraft. The configuration changes result from the increased tipspeed and reduced solidity. The characteristics of the +5 PNdB tilt rotor design are given in Table 3.17 and a threeview of the aircraft is shown in Figure 3.47.

	S.I. UNITS	U.S. UNITS
WEIGHTS		
DESIGN GROSS WEIGHT	33,211 Kg	73,217 Lbs
WEIGHT EMPTY	22,116 Kg	48,757 Lbs
FUEL WEIGHT	2,016 Kg	4,436 Lbs
NUMBER OF PASSENGERS	100	100
ROTORS		
DISC LOADING	73.24 Kg/m^2	15 Lbs/Ft^2
DIAMETER	17.0 m	55.7 Ft.
SOLIDITY	.081	.081
BLADE NUMBER	3	3
TWIST	36 Degs	36 Degs
TIP SPEED HOVER/CRUISE	279/195 m/s	915/641 Ft/Sec.
POWER		
NO. OF ENGINES	4	4
RATED POWER/ENGINE	2.996×10^6 Watts	4018 SHP
FUSELAGE		
LENGTH	28.19 m	92.5 Ft.
WIDTH (MAX)	4.51 m	14.8 Ft.
CABIN LENGTH	17.58 m	57.67 Ft.
WING		
AREA	68.02 m^2	732.2 $Ft.^2$
SPAN	22.10 m	72.5 Ft.
TAPER RATIO	1.0	1.0
CHORD	3.08 m	10.1 Ft.
ASPECT RATIO	7.18	7.18
AIRFOIL t/c	.21	.21
HORIZONTAL TAIL		
AREA	20.44 m^2	220 $Ft.^2$
SPAN	10.49 m	34.4 Ft.
TAIL VOLUME RATIO	1.62	1.62
ASPECT RATIO	5.37	5.37
VERTICAL TAIL		
AREA	18.39 m^2	198 $Ft.^2$
SPAN	4.94 m	16.2 Ft.
TAIL VOLUME RATIO	.159	.159
ASPECT RATIO	1.32	1.32
PERFORMANCE		
NRP CRUISE SPEED	175.1 m/s	340 Knots
CRUISE ALTITUDE	4267 m	14,000 Ft.
BLOCK TIME	.76 Hr	.76 Hr
NOISE		
SIDELINE NOISE - 500 FEET/HOVER	103.2 PNdB	103.2 PNdB

TABLE 3.17. +5 PNdB DERIVATIVE DESIGN POINT TILT ROTOR TABLE OF CHARACTERISTICS

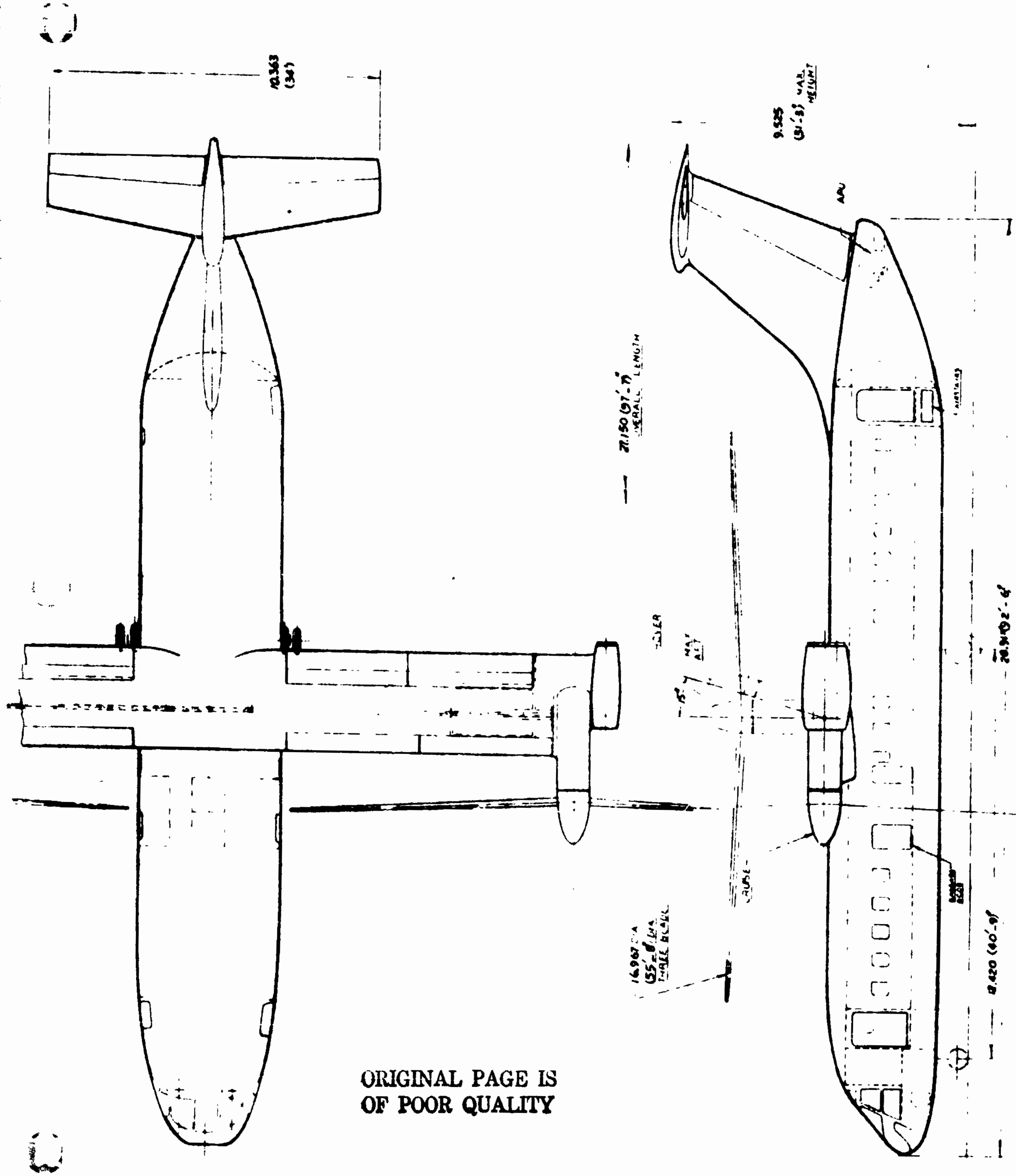


FIGURE 3.47. 1985 COMMERCIAL VTOL TRANSPORT, 100 PASSENGER - TILT ROTOR +5 PNdB DESIGN.

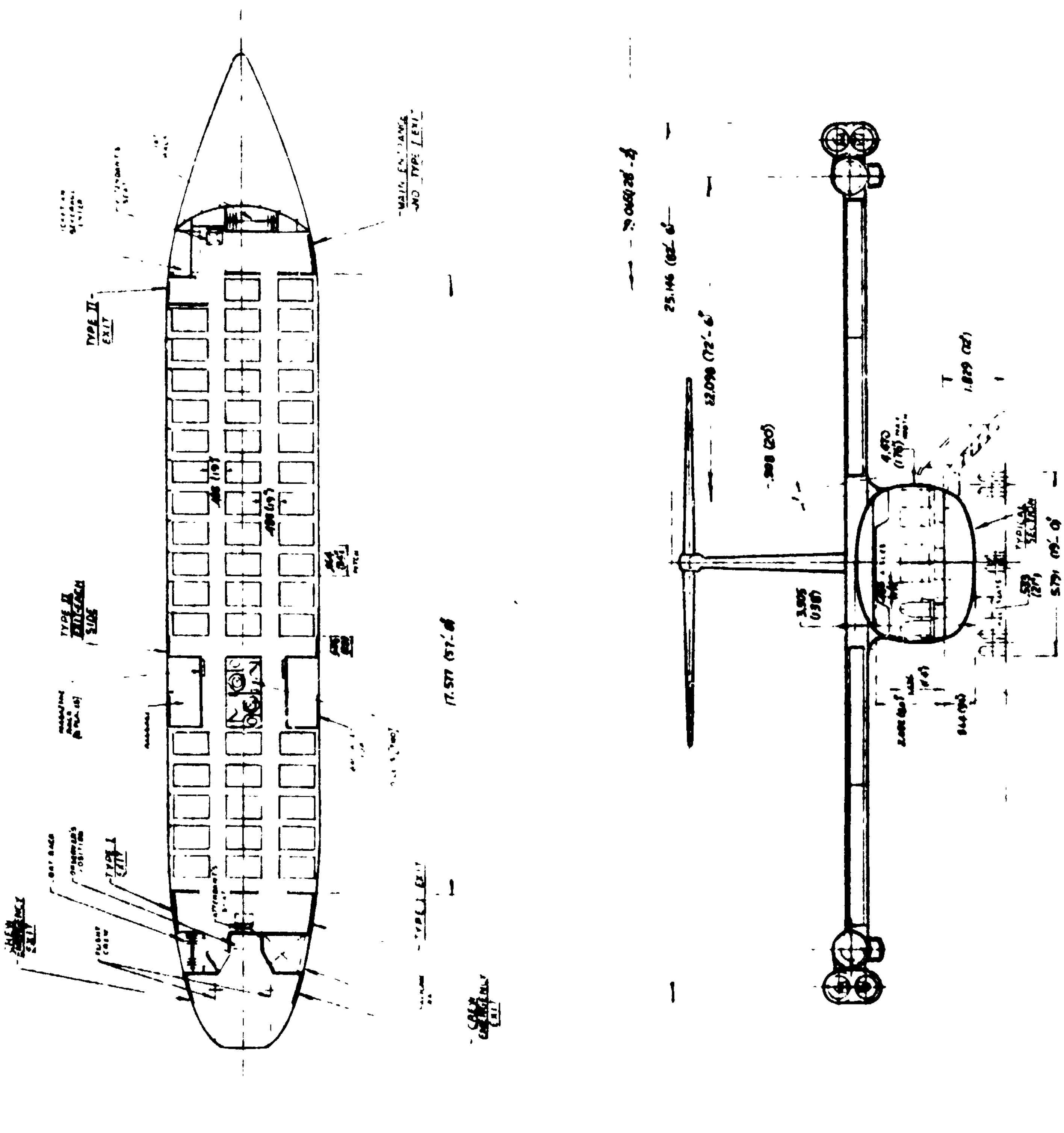


FIGURE 3.47. 1985 COMMERCIAL VTOL TRANSPORT, 100 PASSENGER - TILT ROTOR +5 PNdB DESIGN (CONTINUED).

Weights - TR-100 (103.2)

The +5 PNdB tilt rotor has a design gross weight of 33,210.5 Kg (73,217 pounds). The weight breakdown for this aircraft is shown in Table 3.18.

The increase in tipspeed results in a reduction in transmission weight to 5,791 pounds. The rotor system weight is not much less than the baseline aircraft. This is due to the effect of increased tipspeed on rotor system weight which tends to counteract the savings expected from reduced diameter and solidity. The rotor flight control weights are governed to a large extent by the rotor weights and as a result do not reduce significantly at the higher tipspeed. The lighter gross weight dictates a slightly lower installed power which shows as a small weight saving in the engine section and installation.

The body weight is essentially the same as the baseline aircraft and the cabin and cockpit accommodations, fixed equipment, etc. are identical to the baseline tilt rotor. The groundrules governing the sizing of the baseline aircraft apply to this vehicle also.

The aircraft CG positions and principle inertias are given in Table 3.19.

Titl Rotor Design - TR-100 (103.2) - Vehicle Performance

Mission Performance

The +5 PNdB tilt rotor has a slightly lower takeoff gross weight than the baseline aircraft and flies the same 200

BOEING VERTOL COMPANY

WEIGHT SUMMARY - PRELIMINARY DESIGN

MIL-STD-1374

	KILOGRAMS	POUNDS
WING	1932.7	4261
ROTOR	2323.1	5126
TAIL	618.7	1364
SURFACES	618.7	1364
ROTOR		
BODY	3849.2	8486
BASIC		
SECONDARY		
ALIGHTING GEAR GROUP	1328.6	2929
ENGINE SECTION	416.8	919
PROPULSION GROUP	4225.2	9315
ENGINE INST'L	1148.0	2531
EXHAUST SYSTEM *		
COOLING		
CONTROLS *		
STARTING *		
PROPELLER INST'L	*356.1	*785
LUBRICATING *		
FUEL	94.3	208
DRIVE	2626.7	5791
FLIGHT CONTROLS	1818.0	4008
AUX. POWER PLANT	288.5	636
INSTRUMENTS	191.9	423
HYDR. & PNEUMATIC	308.4	680
ELECTRICAL GROUP	378.3	834
AVIONICS GROUP	293.9	648
ARMAMENT GROUP		
FURN. & EQUIP. GROUP	3273.6	7217
ACCOM. FOR PERSON.		
MISC. EQUIPMENT		
FURNISHINGS		
EMERG. EQUIPMENT		
AIR CONDITIONING	612.3	1350
ANTI-ICING GROUP	254.0	560
LOAD AND HANDLING GP.		
WEIGHT EMPTY	22115.2	48756
CREW	299.4	660
TRAPPED LIQUIDS	52.2	115
ENGINE OIL	59.9	132
CREW ACCOMMODATIONS	68.0	150
EMERGENCY EQUIPMENT	23.6	52
PASSENGER ACCOMMO.	415.5	916
PASSENGERS (100)	8164.6	18000
FUEL	2012.1	4436
GROSS WEIGHT	33210.5	73217

REV

	WEIGHT EMPTY	GROSS WEIGHT
WEIGHT	22,115.2 Kg (48,756 LBS)	33,210.5 Kg (73,217 LBS)
CENTER OF GRAVITY*		
HORIZONTAL FLIGHT		
FUSELAGE STATION	12.72 M (500.9 In.)	12.77 M (502.7 In.)
WATER LINE	3.56 M (140.3 In.)	3.26 M (128.4 In.)
VERTICAL FLIGHT		
FUSELAGE STATION	13.08 M (515.1 In.)	13.12 M (516.6 In.)
WATER LINE	3.94 M (155.0 In.)	3.51 M (138.1 In.)
MOMENT OF INERTIA		
HORIZONTAL FLIGHT		
I_{xx} (ROLL)	1,164,775 Kg M^2 (858,409 Slug Ft^2)	1,265,023 Kg M^2 (932,209 Slug Ft^2)
I_{yy} (PITCH)	504,030 Kg M^2 (371,457 Slug Ft^2)	547,411 Kg M^2 (403,427 Slug Ft^2)
I_{zz} (YAW)	1,357,141 Kg M^2 (1,000,178 Slug Ft^2)	1,473,945 Kg M^2 (1,086,259 Slug Ft^2)
VERTICAL FLIGHT		
I_{xx} (ROLL)	1,224,388 Kg M^2 (902,342 Slug Ft^2)	1,329,766 Kg M^2 (980,003 Slug Ft^2)
I_{yy} (PITCH)	546,907 Kg M^2 (403,056 Slug $Feet^2$)	593,978 Kg M^2 (437,746 Slug Ft^2)
I_{zz} (YAW)	1,468,262 Kg M^2 (1,082,071 Slug Ft^2)	1,594,629 Kg M^2 (1,175,200 Slug Ft^2)

*FUSELAGE STATION 0 IS NOSE OF BODY, CENTERLINE OF ROTOR IN HORIZONTAL FLIGHT IS 4.6 METERS ABOVE WATER LINE O.

TABLE 3.19 . WEIGHT CENTER OF GRAVITY AND MOMENT OF INERTIA, +5 PNdB TILT ROTOR.

nautical miles design mission with the same payload. A summary of the aircraft design mission performance is shown in Tables 3.20 and 3.21.

The taxi, takeoff and initial air maneuver use 85 Kg (187 pounds) of fuel. The aircraft then climbs to an altitude of 4,267 miles (14,000 feet) at an average rate of climb of 16.76 m/s (3,330 feet per minute). The climb segment consumes 176 Kg (389 pounds) of fuel for a range credit of 22.7 Km (12.25 nautical miles). The aircraft then cruises at normal rated power at an average speed of 175.4 m/s (342 knots) to a range of 319 Km (172 nautical miles). The cruise segment fuel is 988 Kg (2,178 pounds). The remainder of the 370 Km (200 nautical miles) design range is achieved during the descent to 2,000 feet at an average rate of descent of 10.49 m/s (2,064 feet per minute). The fuel used during descent is 61 Kg (134 pounds).

The final air maneuver, descent landing and taxi use an additional 207 pounds of fuel to complete the design range. The reserve fuel is computed based upon fuel for an additional 50 nautical miles and a 20 minute loiter which requires a further 613 Kg (1,351 pounds) for a total fuel load of 1,935 Kg (4,266 pounds).

Noise Derivative Tilt Rotor (103.2 PNdB)

Hover Performance

The hover performance of the +5 PNdB design point tilt rotor is shown in Figures 3.48 through 3.51. The sea level hover

+5 PNdB TILT ROTOR - MISSION BREAKDOWN

	TIME (HOURS)	DISTANCE (N.MI.)	WEIGHT (LBS)	FUEL (LBS)	V (KNOTS)	R/C (FT/MIN)
	0	0	73,217			
TAXI				13	0	0
	.017	0	73,204			
TAKEOFF				174	0	0
	.050	0	73,030			
CLIMB				389	178	+3300
	.120	12.25	72,641			
CRUISE				2,178	341	0
	.587	172	70,463			
DESCENT				134	286	-2064
	.683	200	70,329			
AIR MANEUVER				58	150	0
	.708	200	70,271			
DESCENT				11	255	-2040
	.716	202	70,260			
LANDING				125	0	0
	.741	202	70,135			
TAXI				13	0	0
	.757	202	70,122			
RESERVE				1,171	155/224	0
	1.304	250	68,951			

TABLE 3.20 . +5 PNdB TILT ROTOR DESIGN MISSION PERFORMANCE (U.S. UNITS).

+5 PNdB TILT ROTOR - MISSION BREAKDOWN

	TIME (HOURS)	DISTANCE (Km)	WEIGHT (Kg)	FUEL (Kg)	V (KNOTS)	R/C (m/s)
	0	0	33,211			
TAXI				6	0	0
	.017	0	33,205			
TAKEOFF				79	0	0
	.050	0	33,126			
CLIMB				176	178	+16.76
	.120	22.7	32,950			
CRUISE				988	341	0
	.587	319	31,962			
DESCENT				61	286	-10.49
	.683	370	31,901			
AIR MANEUVER				26	150	0
	.708	370	31,875			
DESCENT				5	255	-10.36
	.716	374	31,870			
LANDING				57	0	0
	.741	374	31,813			
TAXI				6	0	0
	.757	374	31,807			
RESERVE				531	155/224	0
	1.304	463	31,276			

TABLE 3.21. +5 PNdB TILT ROTOR DESIGN MISSION PERFORMANCE (S.I. UNITS).

NOISE DERIVATIVE AIRCRAFT

TILT ROTOR/100 PASSENGER/103.2 PNdB

ALL ENGINES OPERATING

+5 PNdB

F/W = 1.05

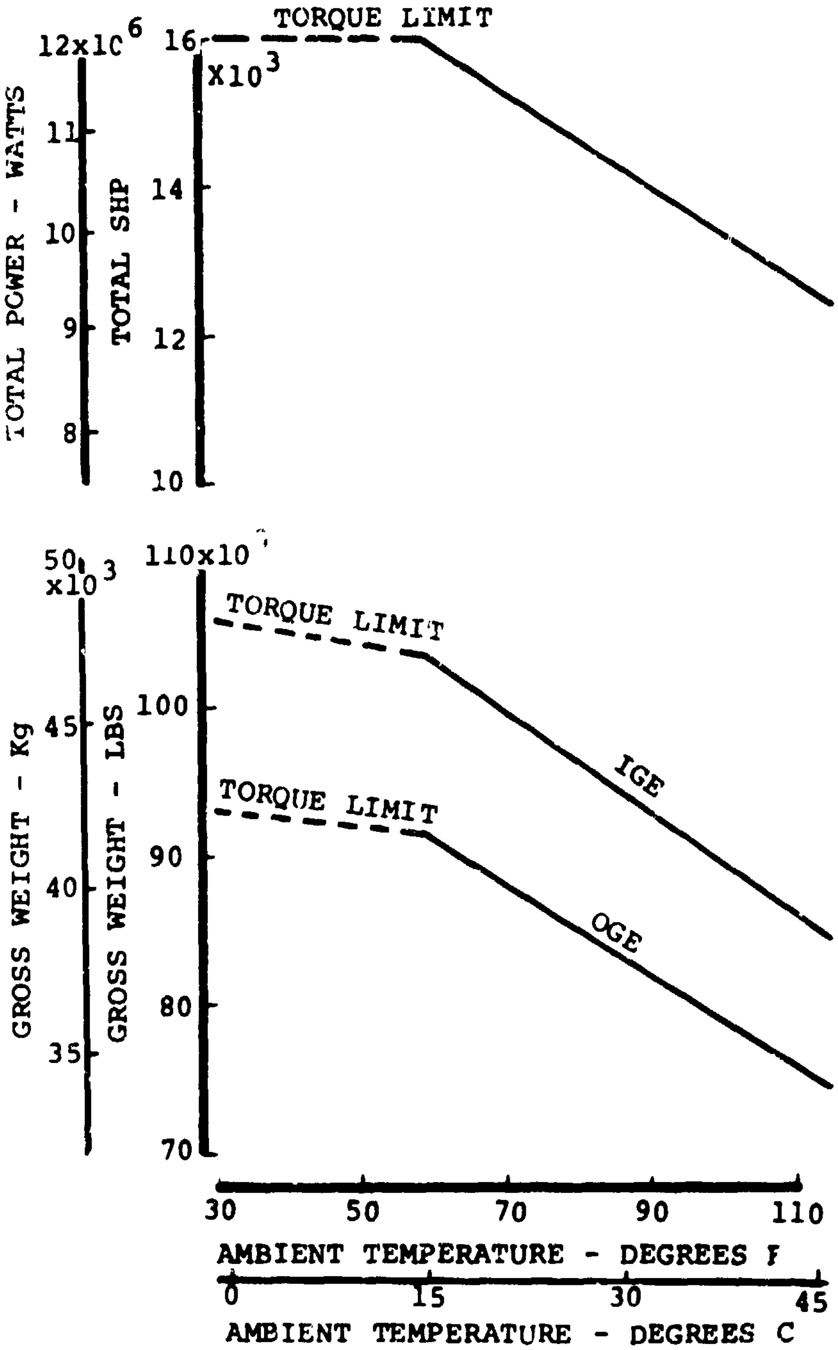


FIGURE 3.48. EFFECT OF AMBIENT TEMPERATURE ON SEA LEVEL HOVER PERFORMANCE - AEO.

NOISE DERIVATIVE AIRCRAFT

TILT ROTOR/100 PASSENGER/103.2 PNdB

ONE ENGINE INOPERATIVE

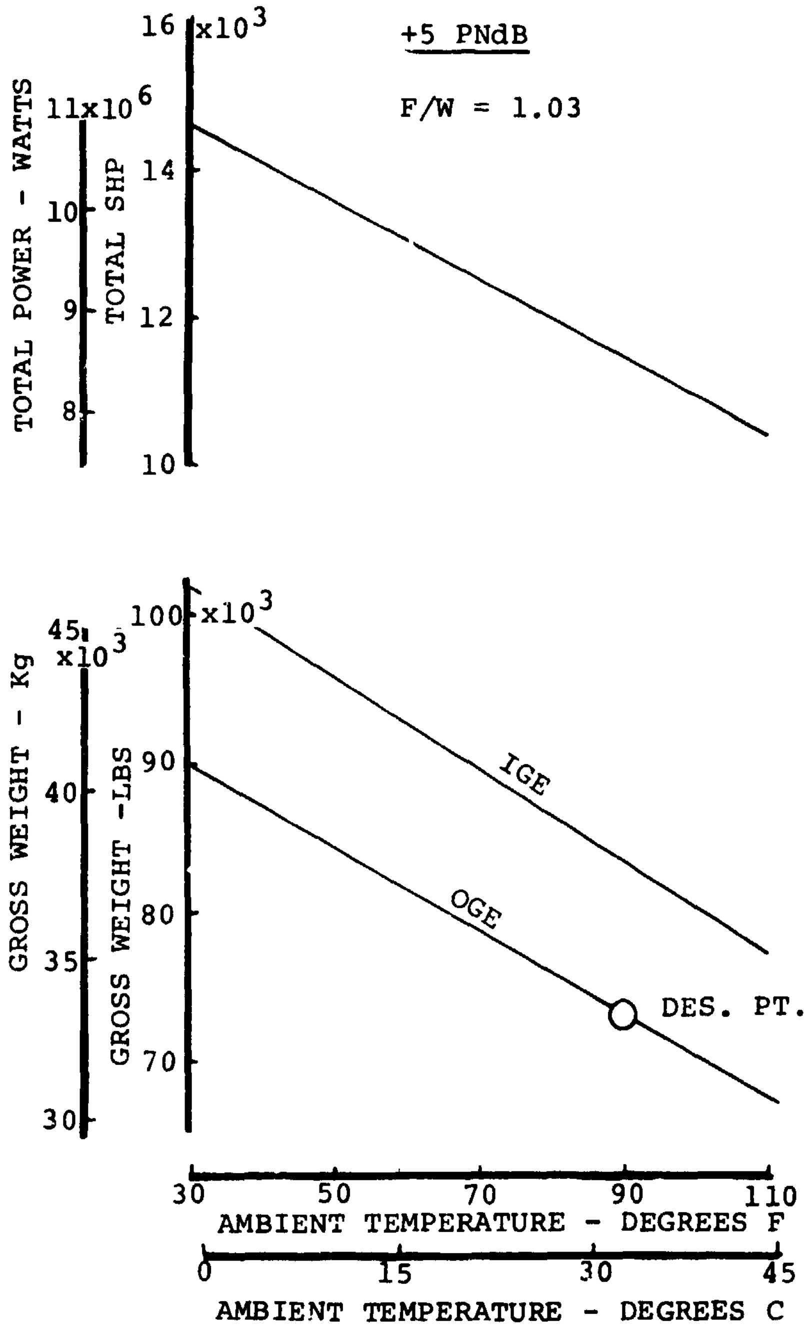


FIGURE 3.49. EFFECT OF AMBIENT TEMPERATURE ON SEA LEVEL HOVER PERFORMANCE - OEI.

NOISE DERIVATIVE AIRCRAFT

TILT ROTOR/100 PASSENGER/103.2 PNdB

STANDARD DAY AND STANDARD DAY +31° F (+17.2° C)

ALL ENGINES OPERATING

+5 PNdB

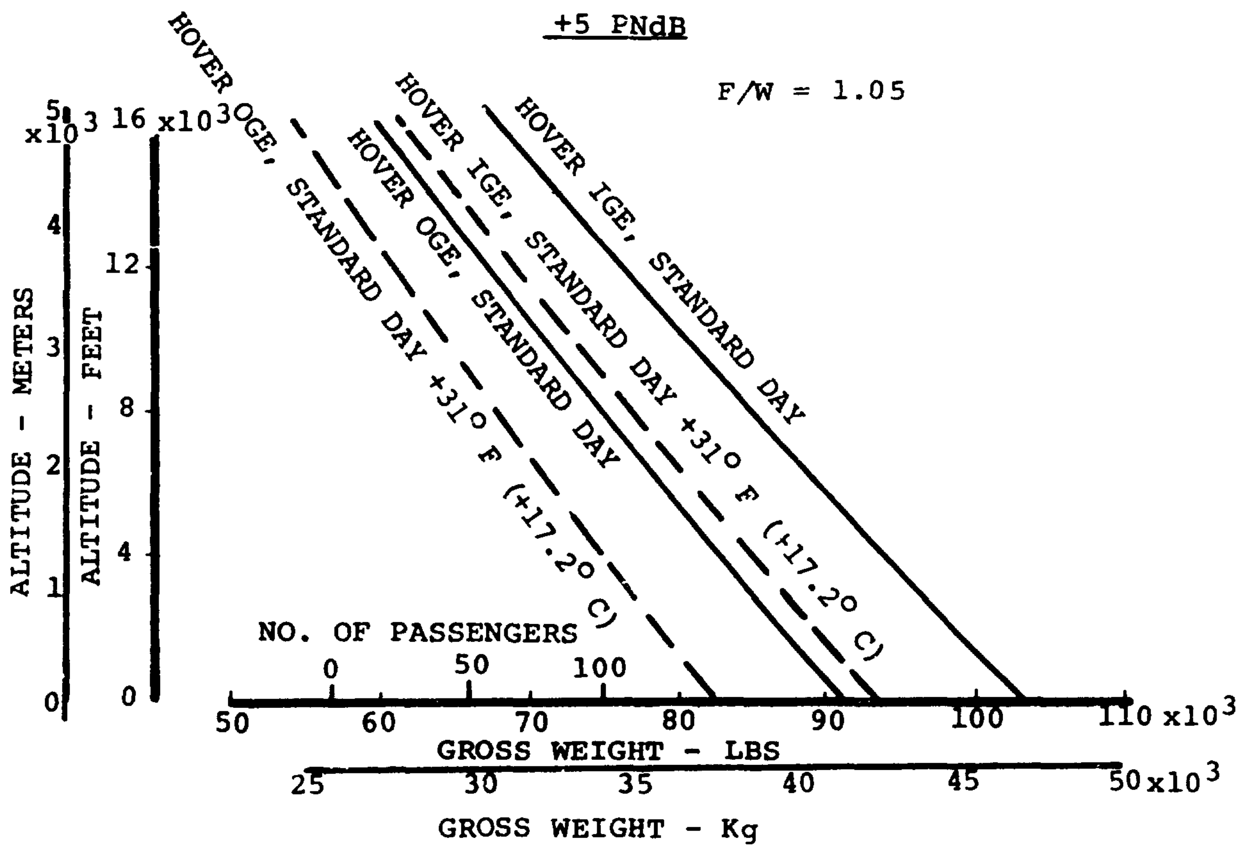


FIGURE 3.50. EFFECT OF ALTITUDE ON HOVER GROSS WEIGHT CAPABILITY. +5 PNdB TILT ROTOR - AEO.

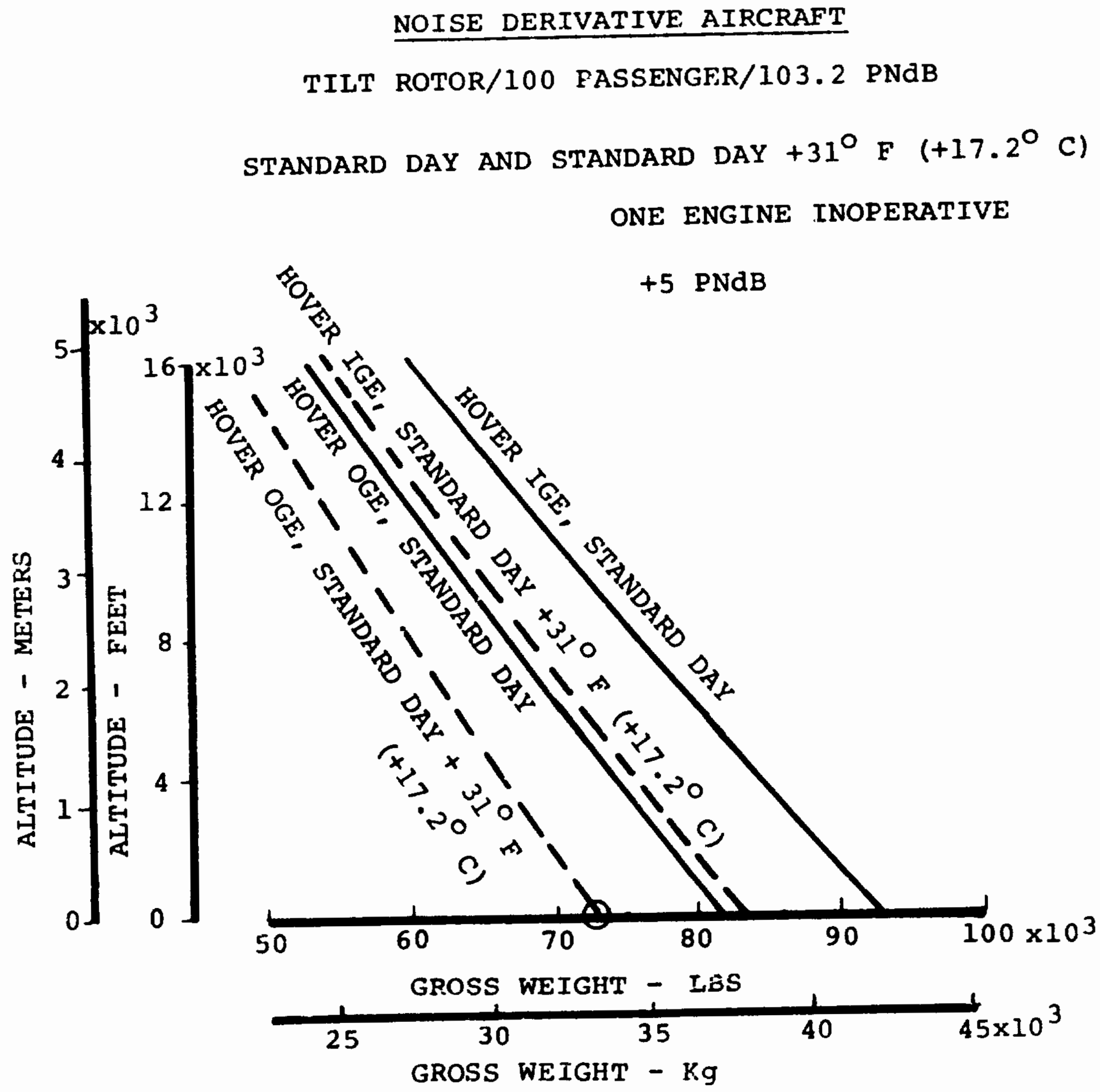


FIGURE 3.51. EFFECT OF ALTITUDE ON HOVER GROSS WEIGHT CAPABILITY.

gross weight capability as a function of ambient temperature for AEO and OEI conditions is given in Figures 3.48 and 3.49.

The sizing condition is one engine inoperative, out of ground effect at sea level, 90-degrees F and is shown in Figure 3.49. For this condition the design gross weight is 33,211 Kg (73,217 pounds). This assumes a force to weight ratio of 1.03 and a 9% increase in the rated takeoff power of the three remaining operating engines.

Both figures indicate the variation of gross weight capability for in and out of ground effect as a function of temperature. For temperatures above 90-degrees F the passenger capacity would need to be reduced, for temperatures below 90-degrees F the 100 passenger capacity would restrict the aircraft weight to the design gross weight of 33,211 Kg (73,217 pounds) and the additional weight capability can be converted to additional force to weight for maneuvers.

Hover gross weight as a function of altitude up to 16,000 feet is shown for AEO and OEI conditions in Figures 3.50 and 3.51. Both OGE and IGE data are given for standard day atmosphere and standard day plus 31-degrees F, (equivalent to 90-degrees F at sea level altitude). The fully loaded aircraft (100 passengers) can take off on a standard day up to 5,000 feet with one engine inoperative, as shown in Figure 3.51. With all engines operating this altitude is increased to 8,000 feet as shown in Figure 3.51.

Cruise Performance

The cruise performance is dictated by the power required to fly straight and level and the power available is determined by the one engine inoperative sizing criteria. These data are shown as a function of airspeed in Figure 3.52 for three aircraft weights. The intersection of the power required and power available lines at design gross weight is the transmission sizing condition. This point allows a maximum cruise speed of 341 knots (TAS) at 14,000 feet. The power available at normal rated power with one engine inoperative is also shown and indicates that a cruise speed of 291 KTAS can be maintained at design gross weight. At the lightest flying weight the maximum speeds achieved at normal rated power are 360 KTS TAS all engines operating and 317 KTS TAS one engine inoperative.

Similar data are shown in Figure 3.53 for 1,524 m (5,000 feet) altitude. The power required lines are higher at this altitude and the performance of the aircraft becomes limited by the transmission torque limit at almost the same power level as the one engine inoperative case. The transmission limit maximum airspeed at design gross weight is 305 knots and with one engine out a cruise speed of 302 knots can be maintained

NOISE DERIVATIVE AIRCRAFT

TILT ROTOR/100 PASSENGER/103.2 PNdB

DGW = 73,217 LBS (33,210 Kg)
MIDWT = 62,000 LBS (28,123 Kg)
OWE = 50,782 LBS (23,034 Kg)

+5 PNdB

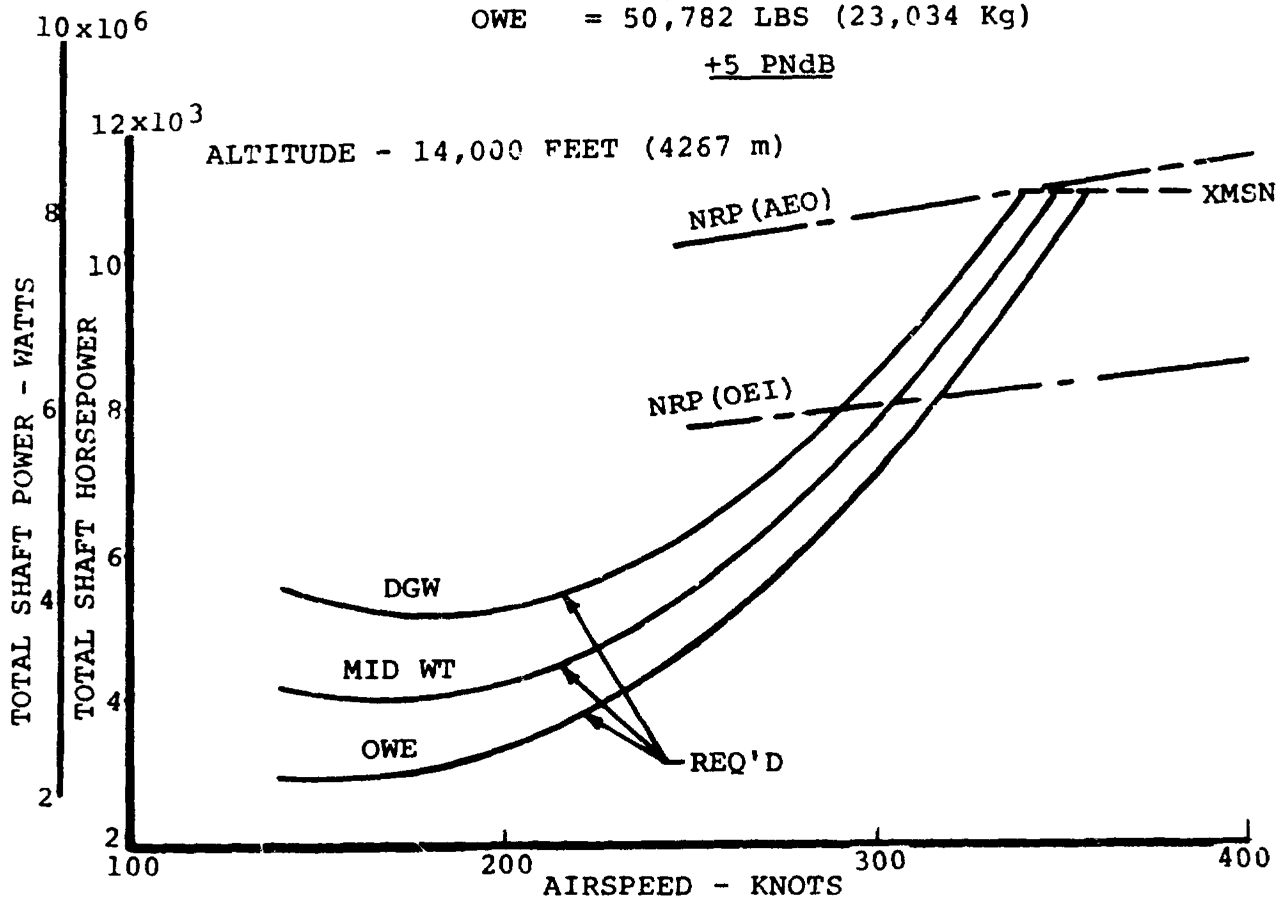


FIGURE 3.52. CRUISE PERFORMANCE - POWER REQUIRED/AVAILABLE. +5 PNdB - TILT ROTOR.

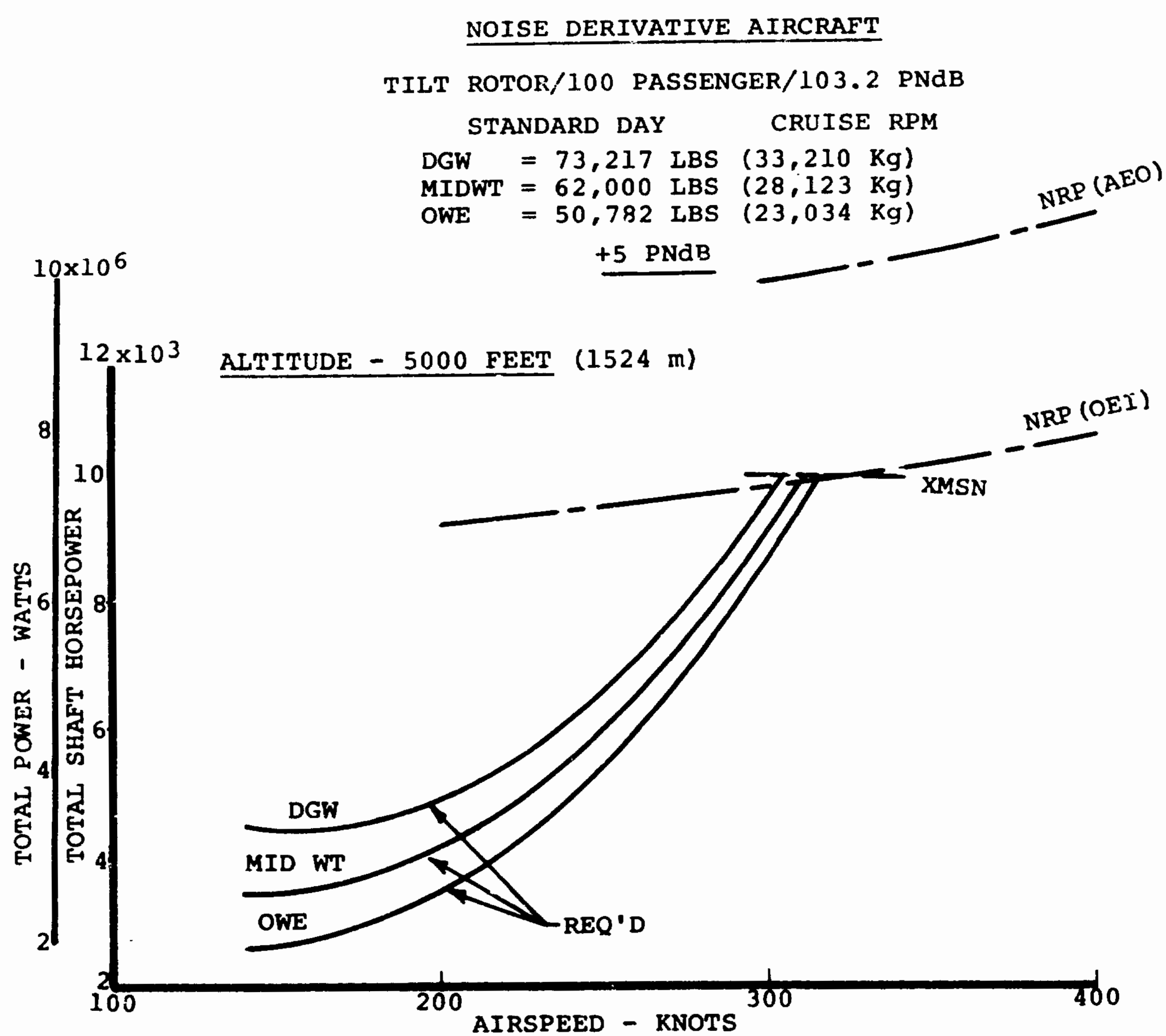


FIGURE 3.53. CRUISE PERFORMANCE - POWER REQUIRED/AVAILABLE.

at 5,000 feet. These speeds are in excess of the 250 knot EAS restriction below 10,000 feet.

The maximum speed performance is shown as a function of altitude in Figure 3.54 for both design gross weight and operating weight empty.

The aircraft maximum rate of climb data are shown in Figure 3.55. With all engines operating at design gross weight the maximum rate of climb is 23.42 m/s (4,610 feet per minute) at sea level. At design cruise altitude 4,267 m (14,000 feet) a rate of climb of 13.71 m/s (2,700 feet per minute) is available. With one engine inoperative the aircraft can achieve 17.88 m/s (3,520 feet per minute) rate of climb at sea level and 9.29 m/s (1,830 feet per minute) at 14,000 feet. The rates of climb at design gross weight do not require a fuselage attitude of greater than 20-degrees. At operating weight empty the maximum rates of climb, (both all engines operating and one engine inoperative) exceed 20 degrees fuselage attitude at the lower altitude. Two lines are given for these cases in Figure 3.55. One for maximum rate of climb and one limited to a fuselage incidence of 20 degrees. With no fuselage attitude limit a maximum rate of climb of 8,200 feet per minute can be obtained at operating weight empty and all engines operating.

Specific Range

The fuel consumption of the aircraft is given in terms of specific range in cruise at both 5,000 feet altitude and

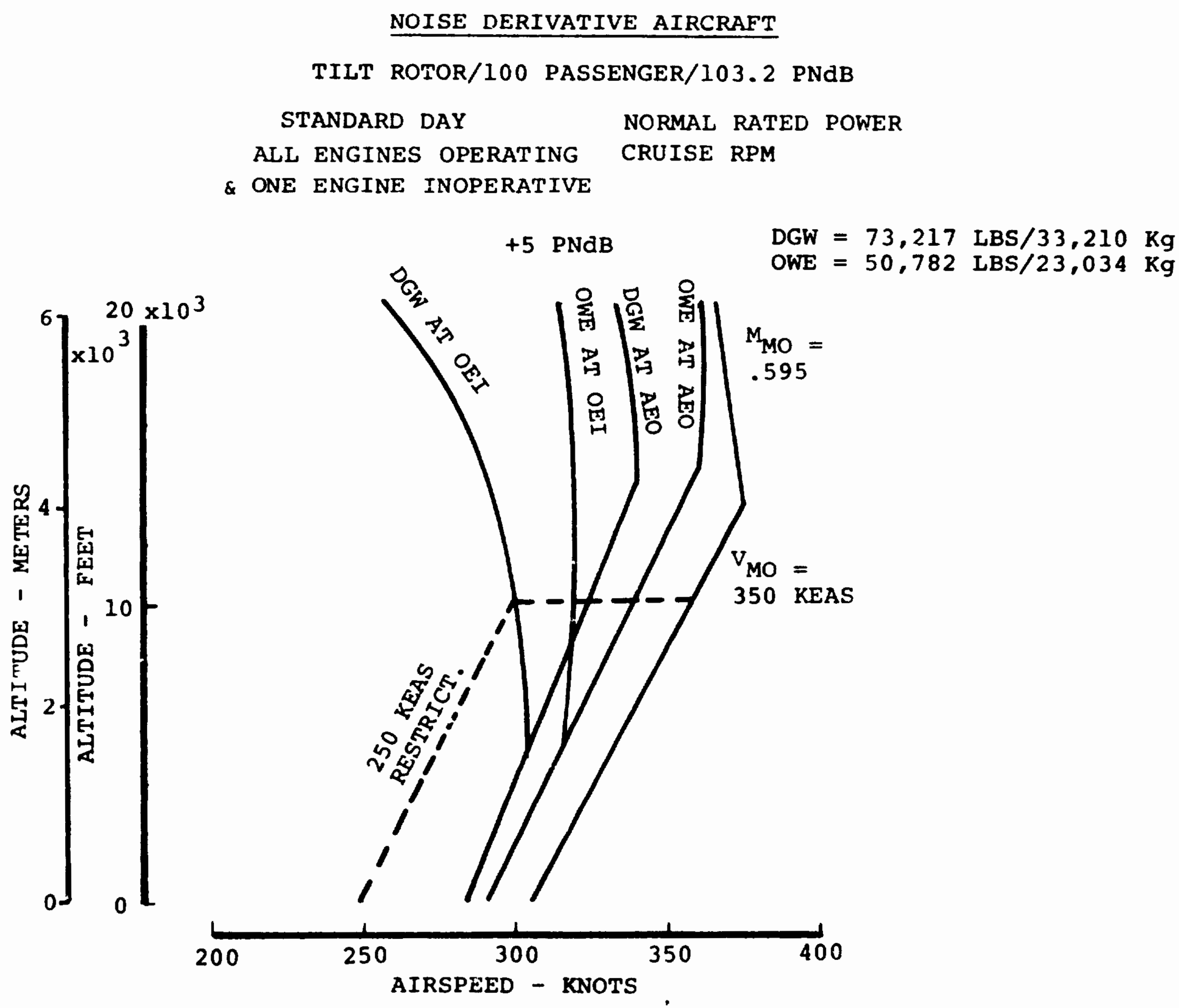


FIGURE 3.54. LEVEL FLIGHT CRUISE SPEED ENVELOPE - +5 PNdB TILT ROTOR.

NOISE DERIVATIVE AIRCRAFT

TILT ROTOR/100 PASSENGER/103.2 PNdB

CLIMB CAPABILITY — TAKEOFF RPM
STANDARD DAY — MIL POWER
+5 PNdB — AEO & OEI

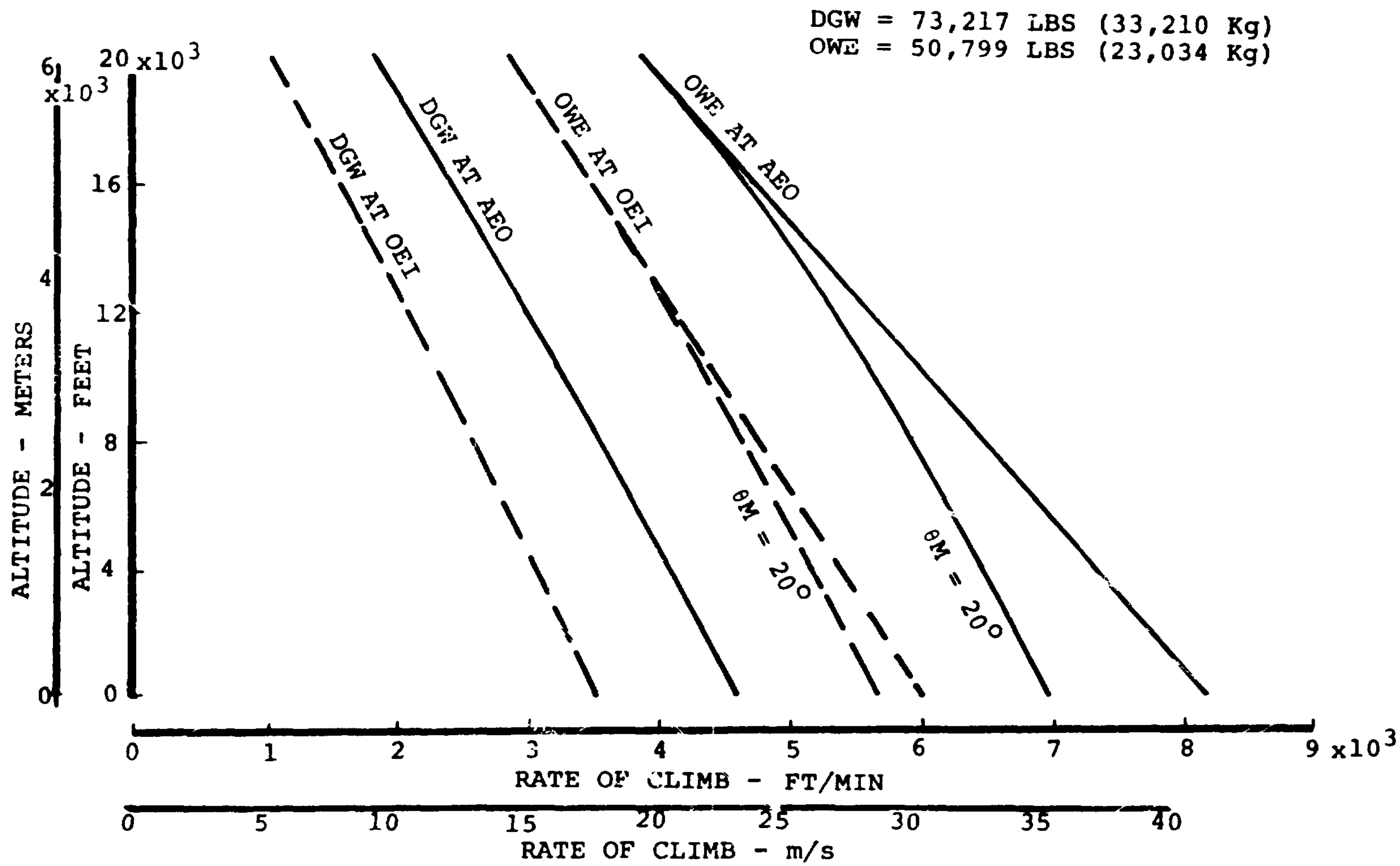


FIGURE 3.55. +5 PNdB DESIGN POINT TILT ROTOR - CLIMB CAPABILITY.

14,000 feet altitude with all engines operating in Figure 3.56. At the design cruise altitude of 4,267 m (14,000 feet) the maximum specific range is 0.088 nautical miles per pound of fuel at design gross weight increasing to 0.116 nautical miles per pound at operating weight empty. The best range speed at design gross weight is 232 knots TAS and reduces to 212 KTAS at minimum flying weight.

At 5,000 feet the best range speeds are about 14 knots slower and the specific range is reduced.

At design altitude and weight the specific range at NRP is 0.069 nautical miles per pound of fuel.

The specific range data with one engine inoperative at the same aircraft weights and altitudes are shown in Figure 3.57. The effect on the three remaining engines of operating at an increased fraction of power is to improve the specific fuel consumption and increase the specific range.

Payload Range

The design payload range is specified in the mission profile and is shown to be met in Figure 3.58. The aircraft carries a 100 passenger payload (18,000 pounds) including baggage over a 200 nautical mile range with reserves as per the guidelines.

The basic mission fuel limit defines the range at zero payload to be 238 nautical miles.

The extra tanks were included in this aircraft for a 400 nautical mile range. The additional tank weight would reduce

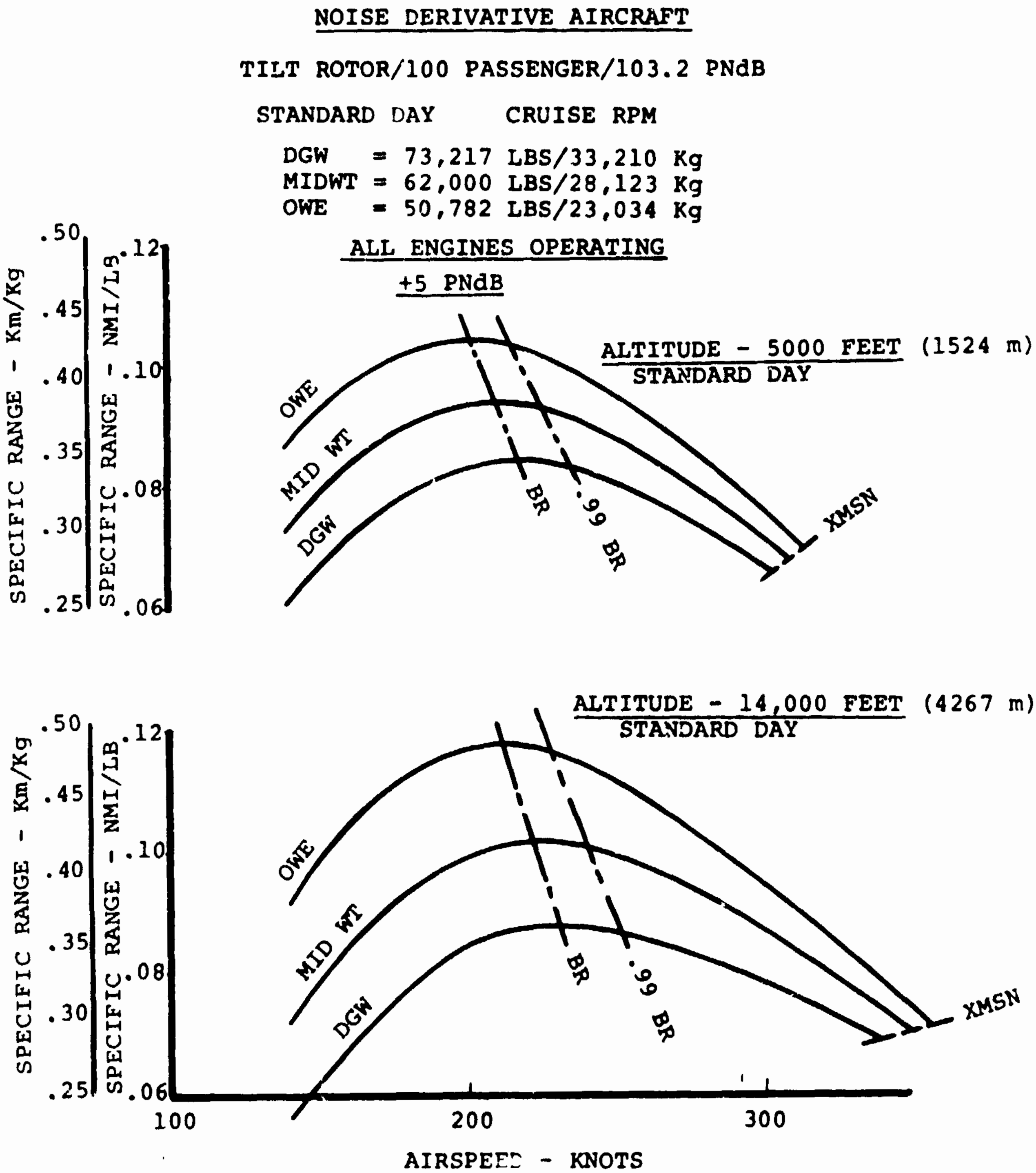


FIGURE 3.56. CRUISE PERFORMANCE - SPECIFIC RANGE - AEO - +5 PNdB TILT ROTOR.

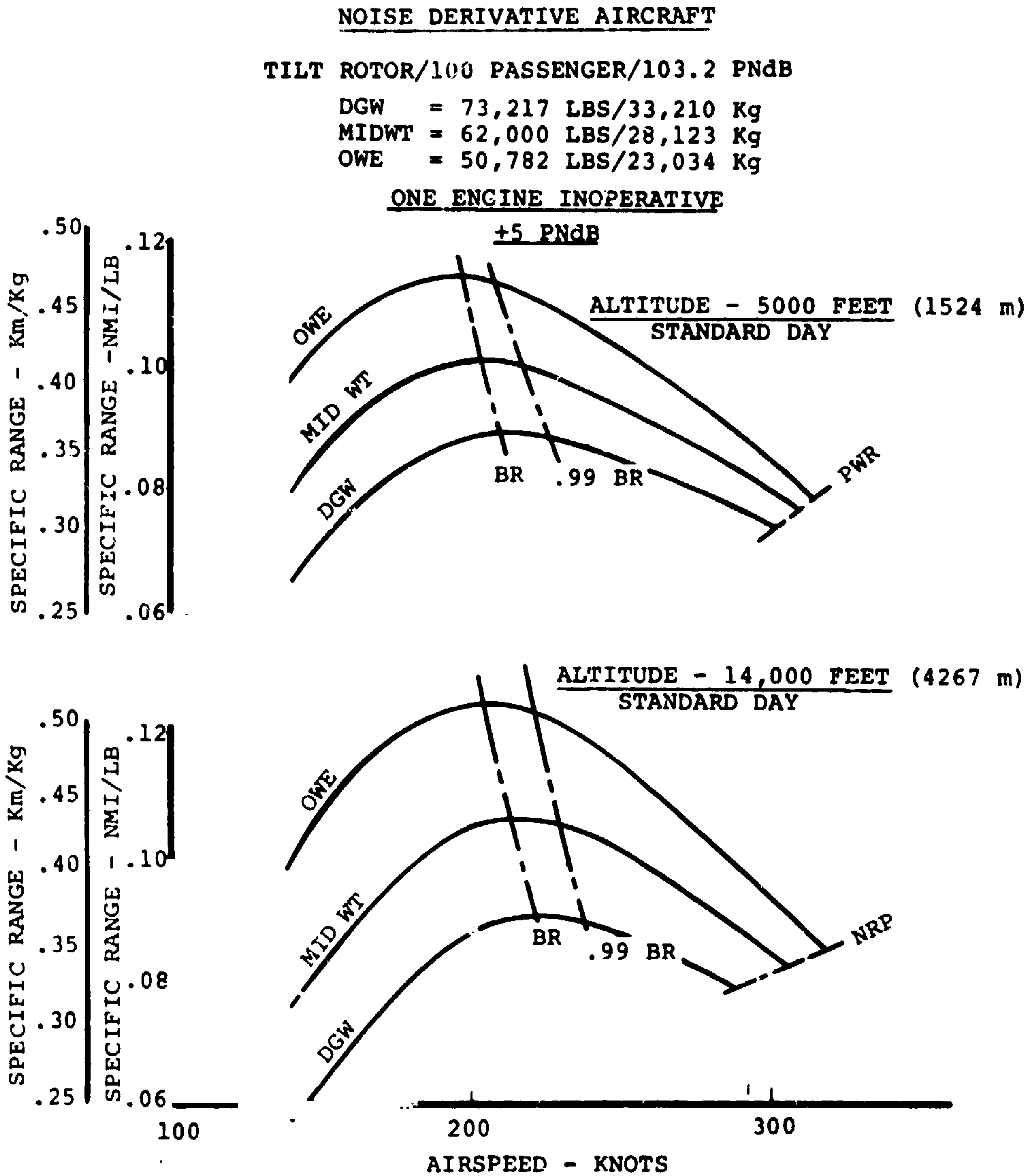


FIGURE 3.57. CRUISE PERFORMANCE - SPECIFIC RANGE - OEI - +5 PNdB TILT ROTOR.

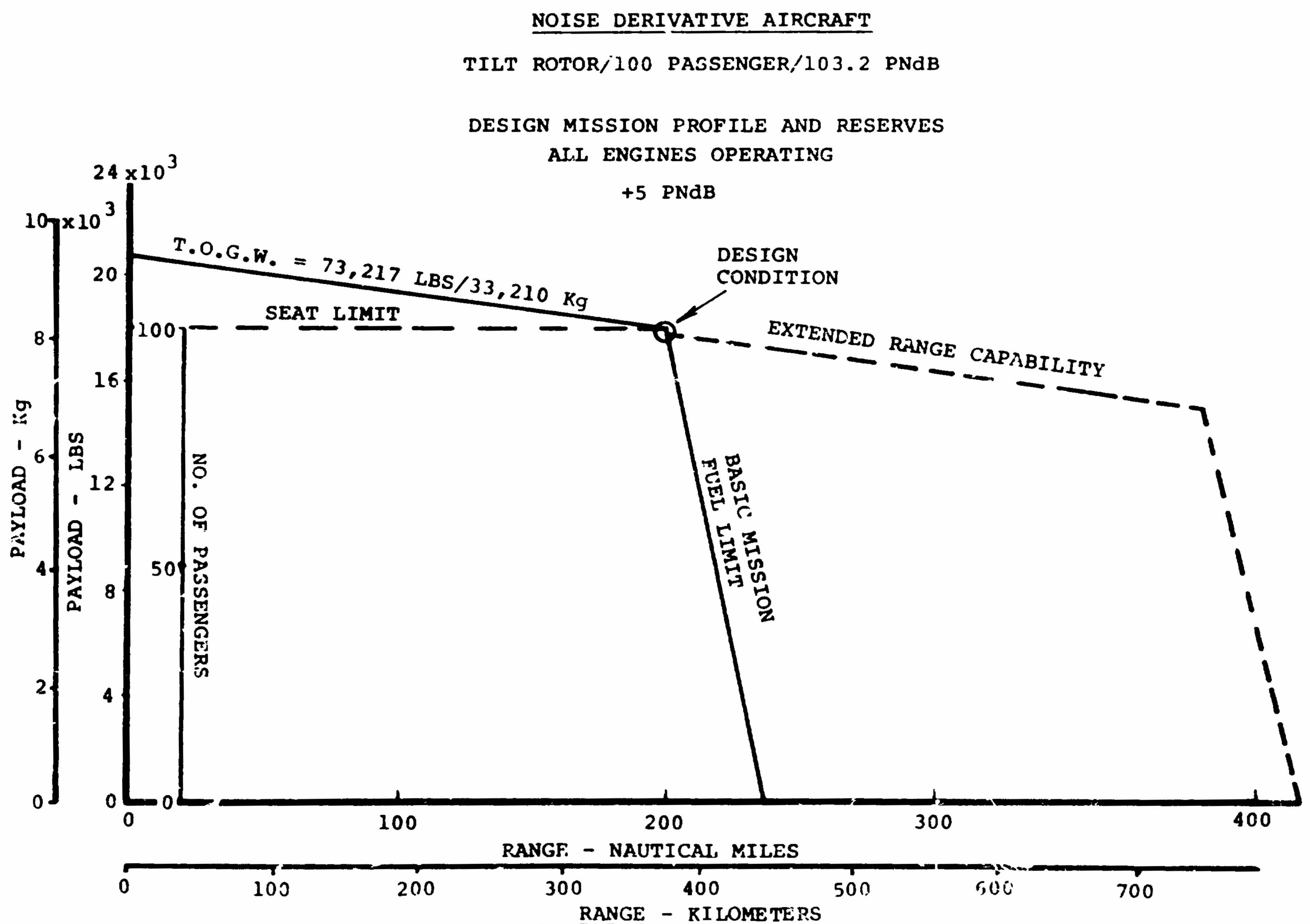


FIGURE 3.58. PAYLOAD RANGE CAPABILITY - CRUISE AT NRP AND CRUISE RPM. +5 PNdB TILT ROTOR.

the passenger load by one passenger at 200 nautical miles and the aircraft could operate with 83 passengers out to 400 nautical miles.

With one engine inoperative the payload range of the basic TR-100 (103.2) design improves due to the increase in specific range shown in Figure 3.57. The payload range data for this case is given in Figure 3.59. The fully loaded aircraft range increases to 238 nautical miles while maintaining the basic mission fuel reserves.

Stability and Control - TR-100 (103.2)

Hover Trim and Control

The cyclic pitch required to trim in hover with the nacelles at 90 degrees and at design gross weight is shown in Figure 3.60. This assumes the same hover CG range as the baseline aircraft. The cyclic required to trim the most critical condition is 2.2 degrees.

The control power in pitch is shown in Figure 3.60. The sensitivity of pitch control power to cyclic control is 0.121 radians per second squared per degree. At a maximum cyclic pitch control setting of 5.72 degrees a pitch acceleration of .685 radians per second squared is available when zero cyclic is required for trim; with the CG at its most adverse location a pitch acceleration of .42 radians per second squared can be obtained.

Yaw and roll control powers are shown in Figure 3.61. The yaw control sensitivity is slightly below the value estimated

NOISE DERIVATIVE AIRCRAFT

TILT ROTOR/100 PASSENGER/103.2 PNdB

DESIGN MISSION PROFILE AND RESERVES
OEI FOR CRUISE AND RESERVE SEGMENTS

+5 PNdB

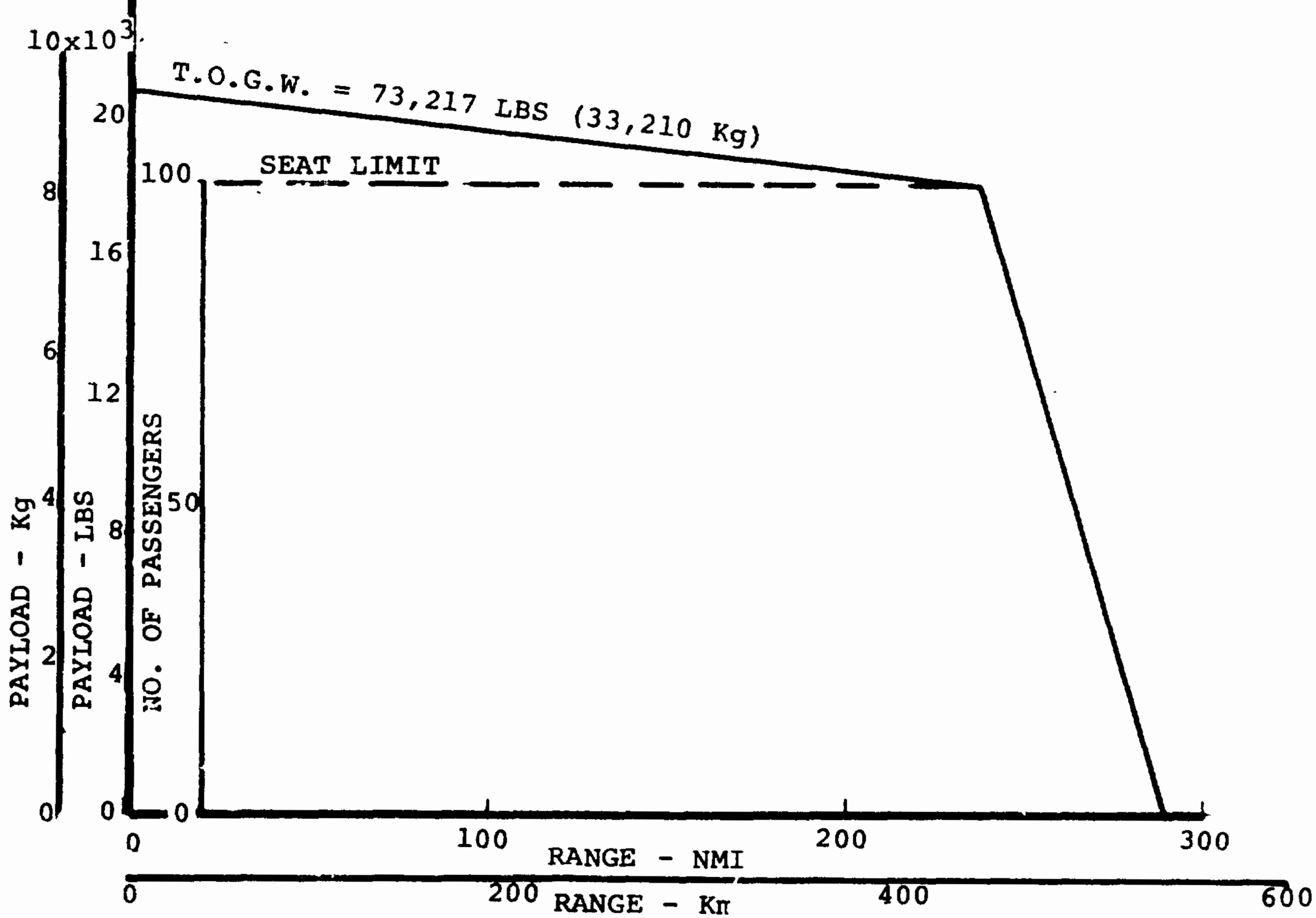


FIGURE 3.59. PAYLOAD RANGE CAPABILITY - CRUISE AT NRP AND CRUISE RPM.

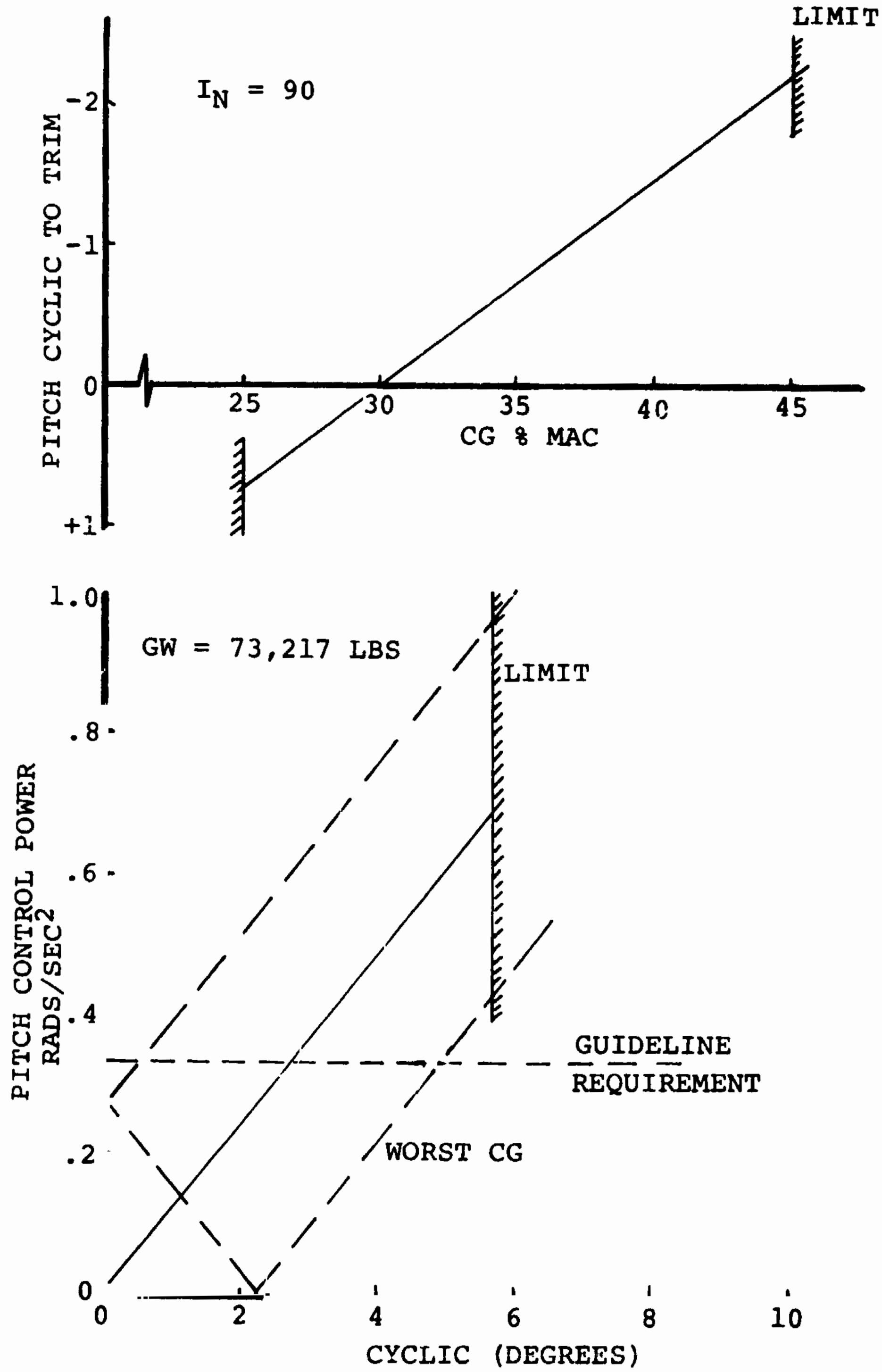


FIGURE 3.60. HOVER PITCH CONTROL +5 PNdB TILT ROTOR.

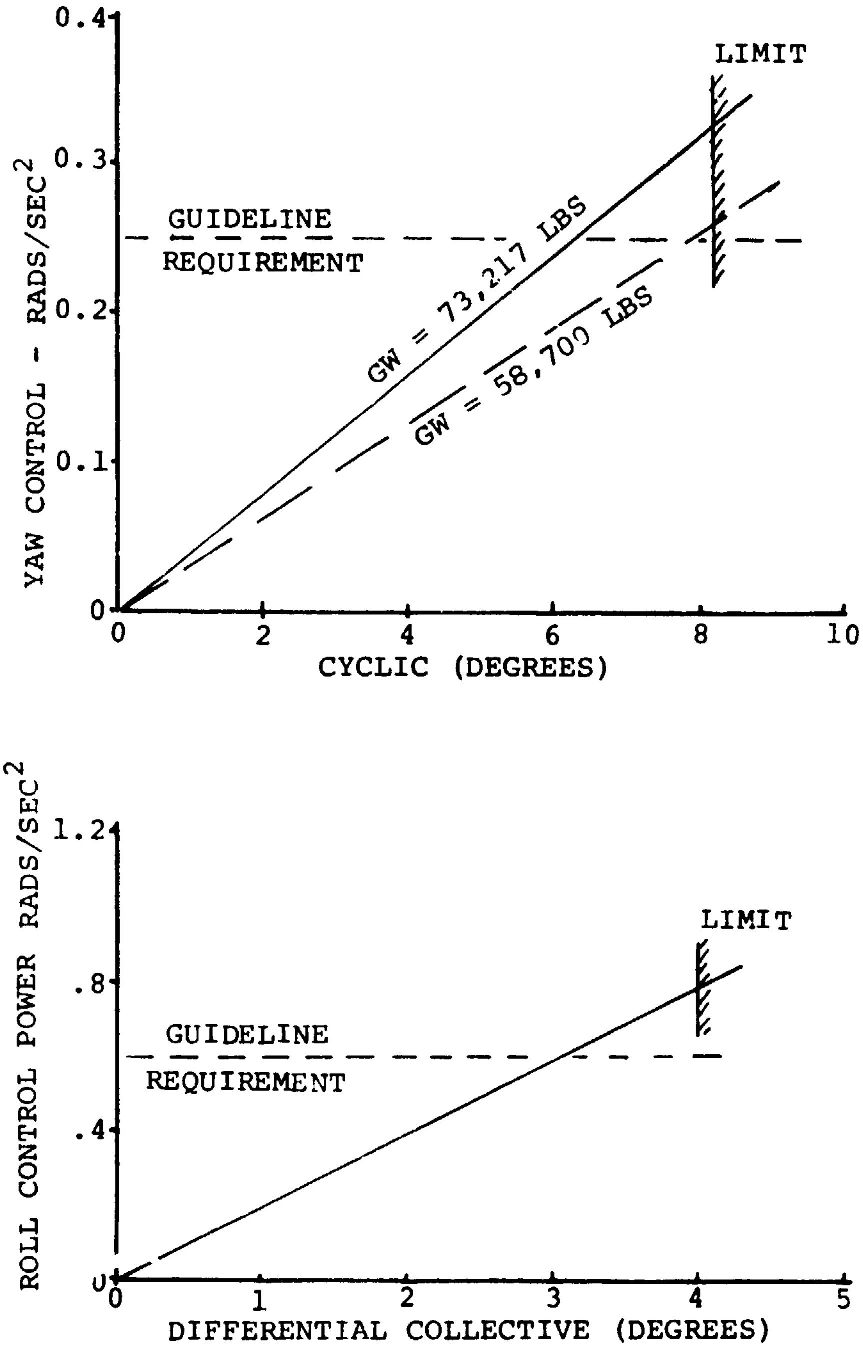


FIGURE 3.61. HOVER YAW AND ROLL CONTROL +5 PNdB T[illegible] ROTOR.

for the design point aircraft. However, maximum yaw acceleration of .335 radians per second squared is available at design gross weight. At a gross weight of 58,700 pounds the available yaw control is reduced to 0.262 radians per second squared. This reduction is caused by the variation of differential in-plane force with gross weight. This in-plane force is mostly obtained by tilting the thrust vector. Thus, there is almost a direct reduction of yawing moment with gross weight whereas yaw inertia does not reduce as fast. This results in a reduction of yaw acceleration.

The roll control per degree of differential collective is slightly less than for the baseline aircraft. This is due to reduced differential thrust as a result of lower solidity and also to a reduced span. This is offset to some extent by the reduced roll inertia, however, the net effect is to reduce the roll control power to 0.79 radians per second squared at 4 degrees differential collective. This is still higher than the 0.6 radians per second squared guideline requirement. At lighter weights roll control power is marginally increased.

The variation of neutral point with cruise speed is plotted in Figure 3.62 for various horizontal tail size. Increased RPM tends to increase nose up pitching moment of the rotor. Reduced solidity and diameter tend to counteract this effect, however, the tail volume ratio providing the required 5%

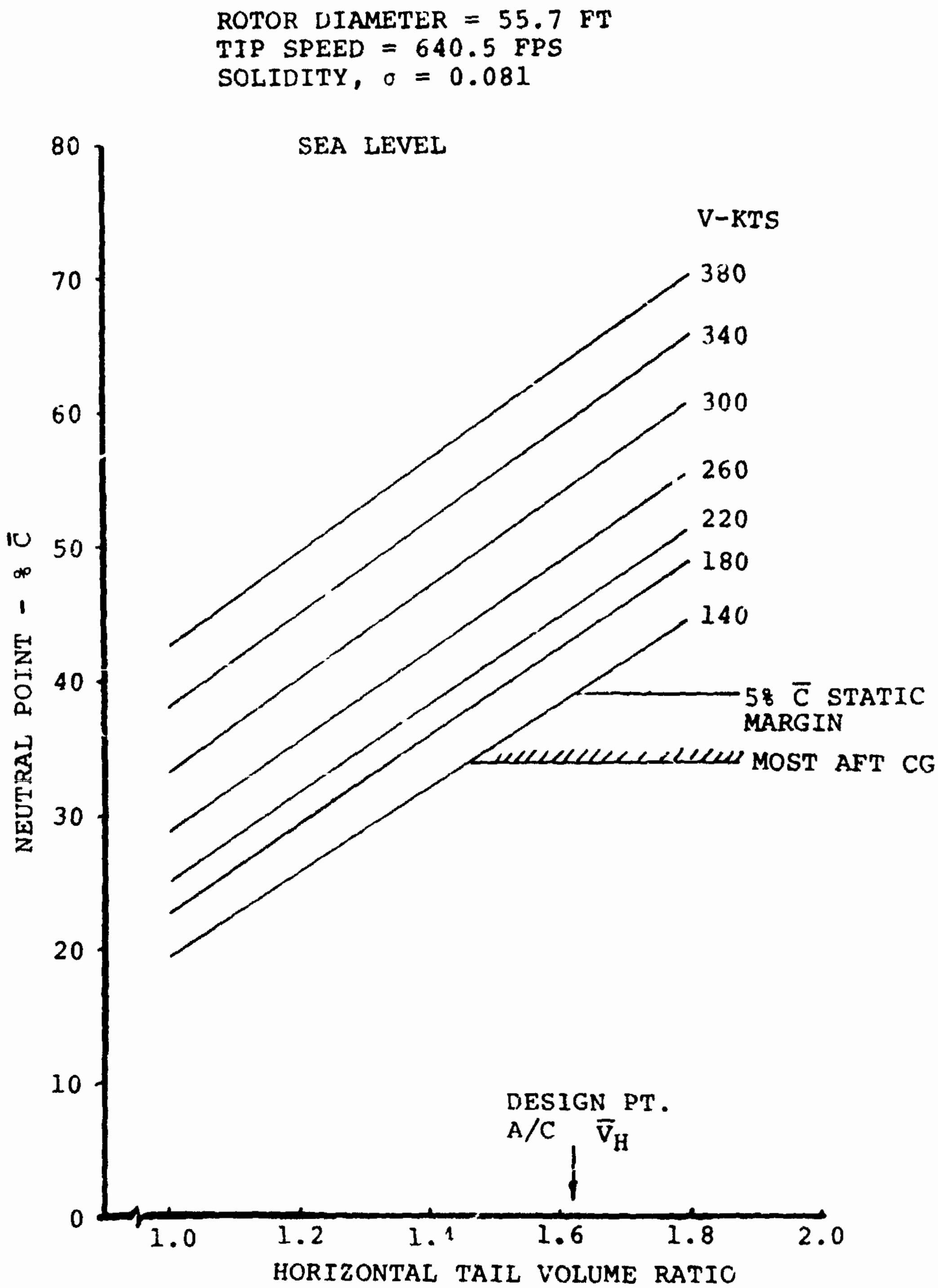


FIGURE 3.62 . +5 PNdB TILT ROTOR DESIGN LONGITUDINAL STATIC STABILITY.

static margin is still longer than that of the baseline aircraft. The +5 PNdB aircraft has a horizontal tail volume ratio of 1.62 and provides 5% static margin down to 140 knots.

The lateral directional derivatives are shown in Figure 3.63. The side force due to sideslip is larger than the design point aircraft and will require a larger roll angle to trim in flying a straight-ground track in sideslip.

The dihedral effect is positive (i.e., $-C_{\ell\beta}$) but reduces to almost zero at maximum speed 360 knots.

The directional stability is positive, i.e., positive $C_{n\beta}$ and increases with airspeed. As with the design point aircraft the dutch roll mode will be improved at high speeds as a result of increased $C_{n\beta}$ and less dihedral effect. The damping derivatives due to roll and yaw are greater than for the design point aircraft and are shown in Figures 3.64 and 3.65.

<u>Tilt Rotor Design - TR-100 (103.2) Noise</u>

The +5 PNdB noise derivative aircraft has a design hover tipspeed of 915 feet per second. This increase in tipspeed increases the rotational and broadband components of rotor noise as shown in Figure 3.66 where sound pressure levels as a function of octave band frequency are presented. The tipspeed effect is noticed at the higher frequencies when compared to similar data for the baseline aircraft in Figure 2.90. NOY values for the hover condition used to establish

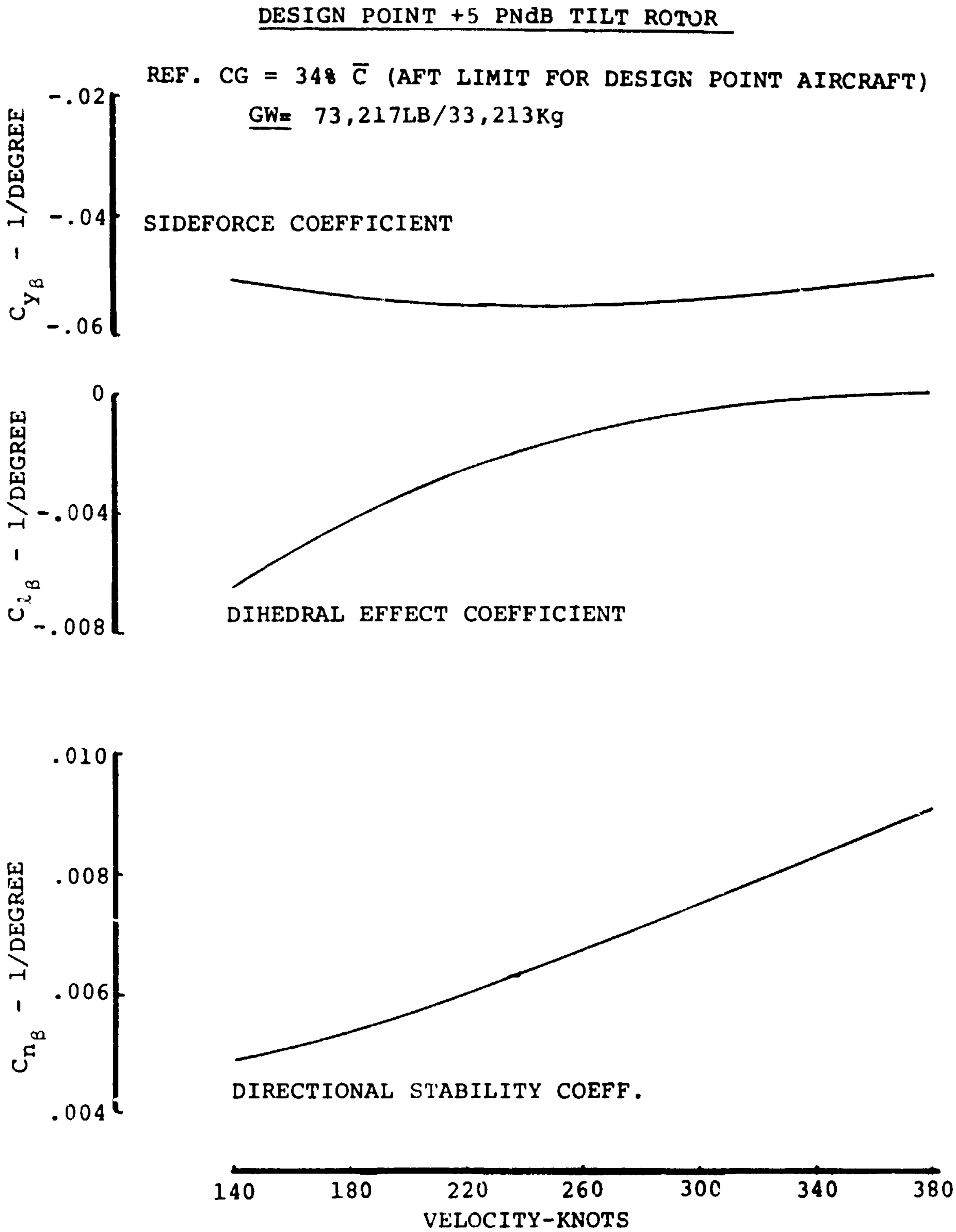


FIGURE 3.63. LATERAL-DIRECTIONAL STABILITY DERIVATIVE COEFFICIENTS DUE TO SIDESLIP - +5 PNdB AIRCRAFT.

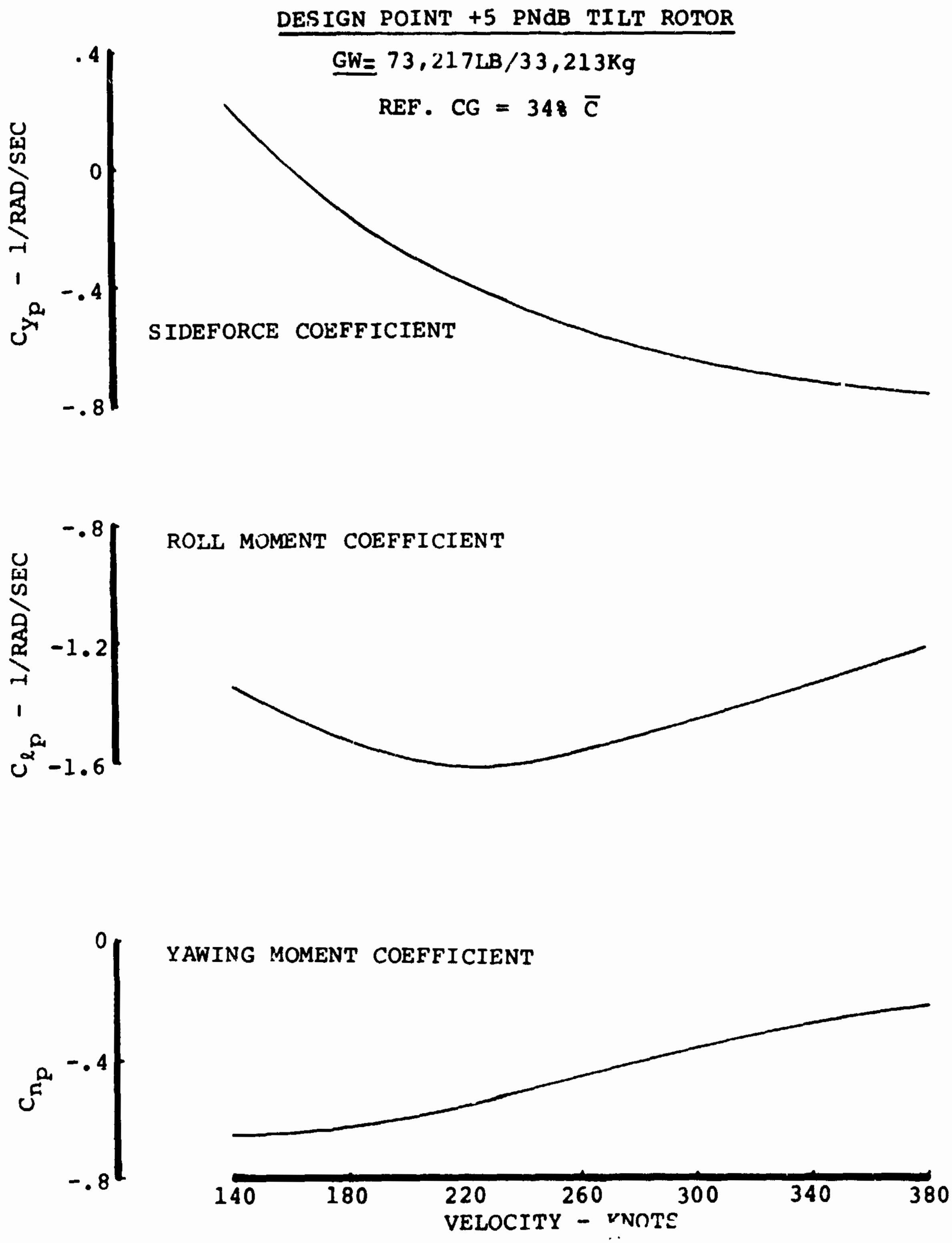


FIGURE 3.64. +5 PNdB TILT ROTOR STABILITY COEFFICIENTS DUE TO ROLL RATE.

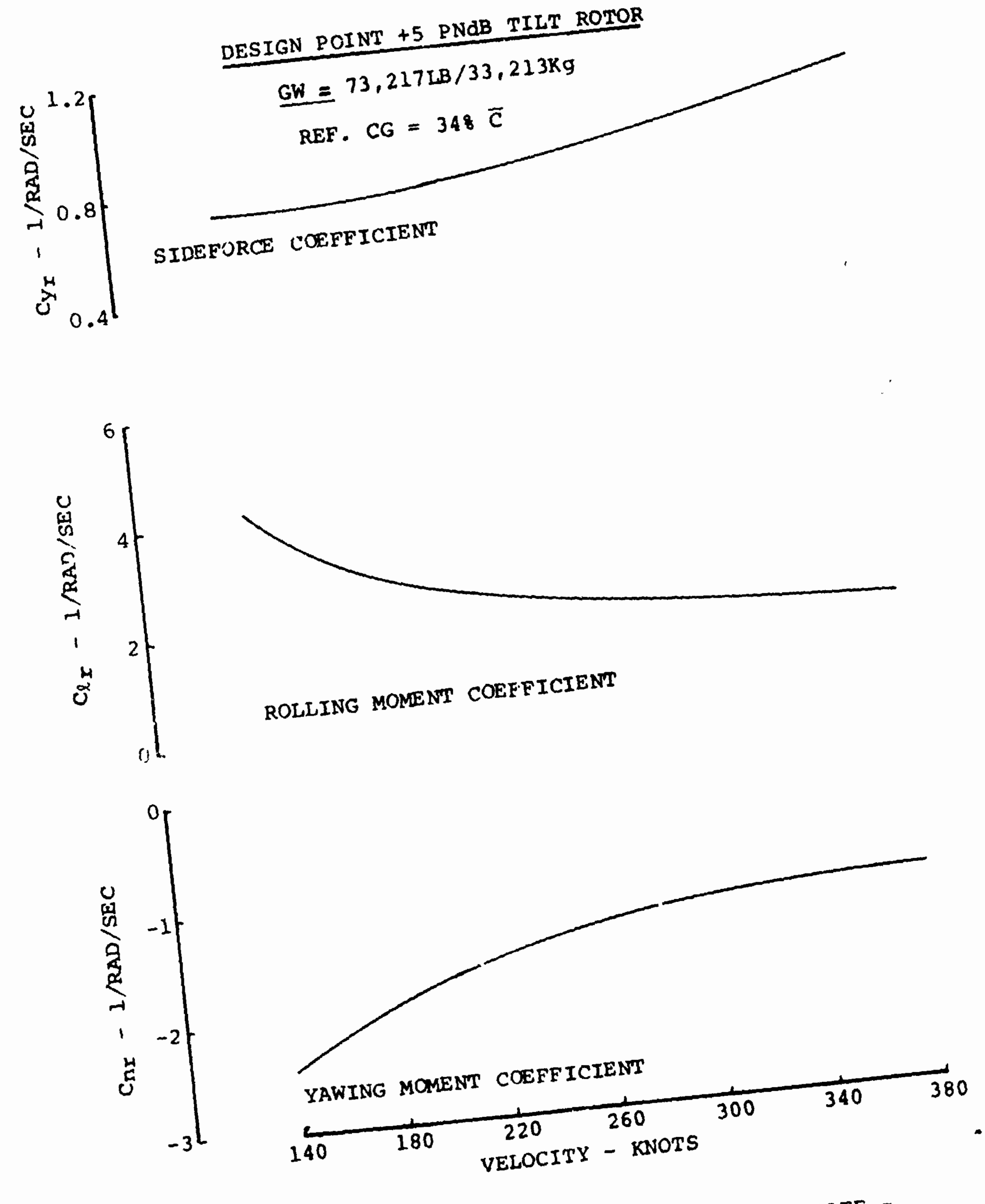


FIGURE 3.65. STABILITY COEFFICIENTS DUE TO YAW RATE - +5 PNdB TILT ROTOR.

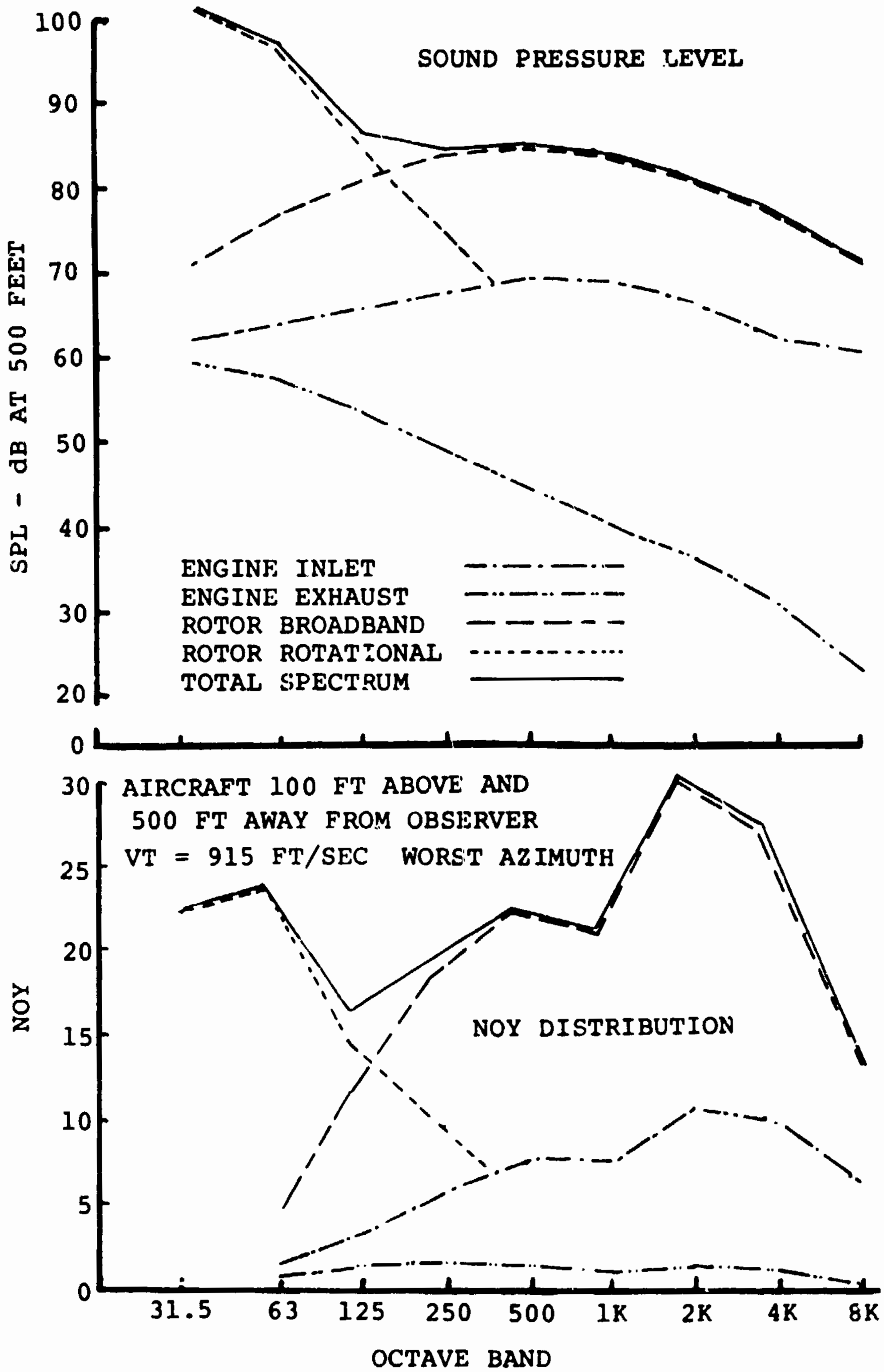


FIGURE 3.66. TILT ROTOR +5 PNdB HOVER NOISE SPECTRUM AND NOY DISTRIBUTION. PNdB = 103.2.

the 103.2 PNdB noise level are given in Figure 3.66.

Contours of constant perceived noise levels 90, 95 and 100 PNdB values for takeoff and landing are given in Figures 3.67 and 3.69. The contours are symmetrical about the flight path, with the landing contours being somewhat wider than the takeoff resulting from the increased time in the landing profile dictated by the guideline sink rate requirements.

Figures 3.68 and 3.70 show the takeoff and landing profile as a function of distance from the ground terminal. Time increments are indicated on the curve to be used with the flight path center line PNdB data given in the lower portion of the figure. The PNdB as a function of time from takeoff or touchdown give time history of an observer stationed at the indicated distance from the terminal. The peak noise values decrease at the fartherest observer position as a result of the altitude of the aircraft and the tipspeed reduction as indicated on the plot.

Tilt Rotor - TR-100 (103.2) - Costs

The initial cost of this tilt rotor design is slightly less than the design point aircraft at $5.03 million at $90.00 per pound airframe costs and $5.73 million at $110 per pound. The reduction in "fly away" cost between this tilt rotor and the design point is primarily due to dynamic systems and engine cost reductions.

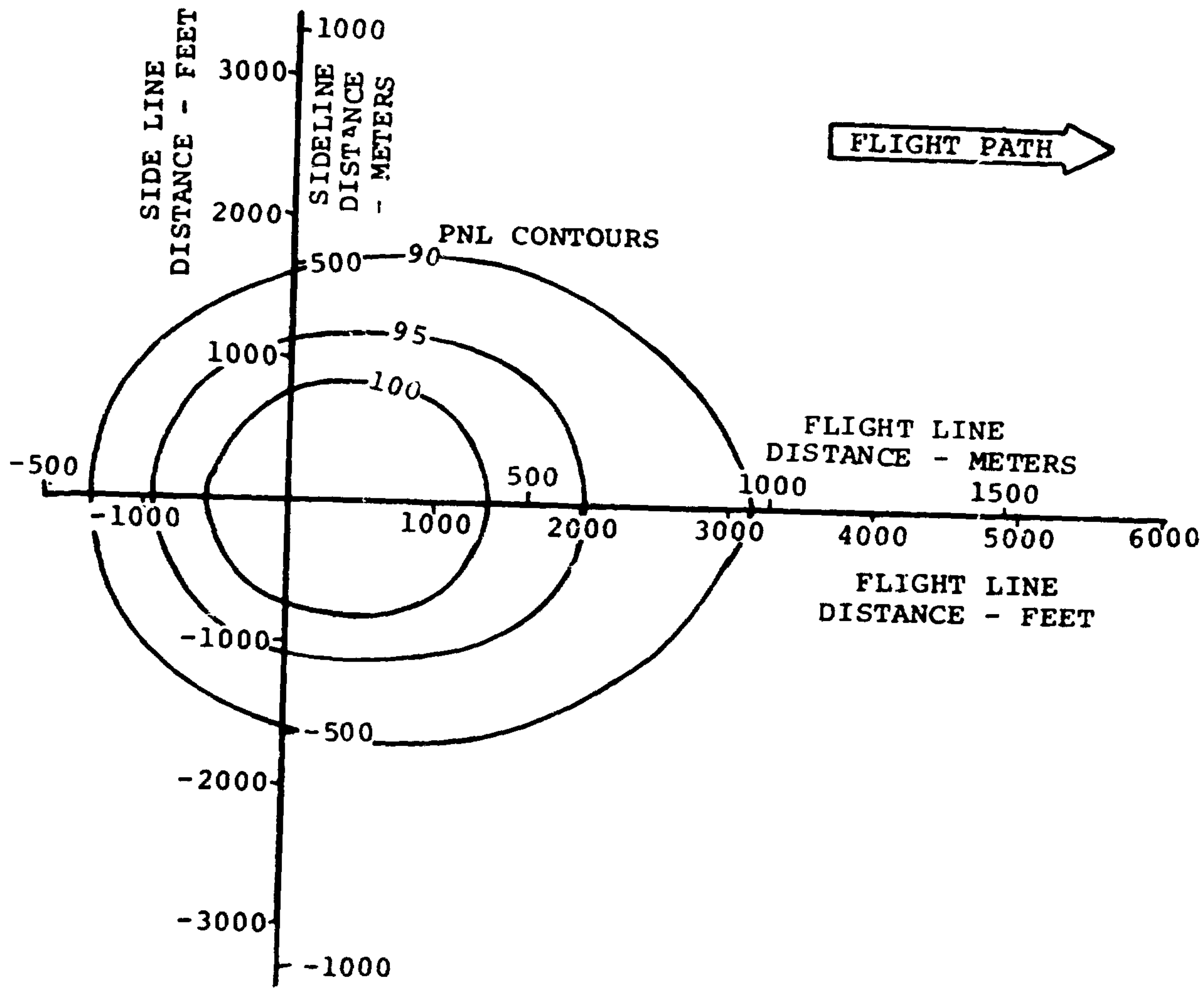


FIGURE 3.67. +5 PNdB TILT ROTOR STANDARD TAKEOFF.

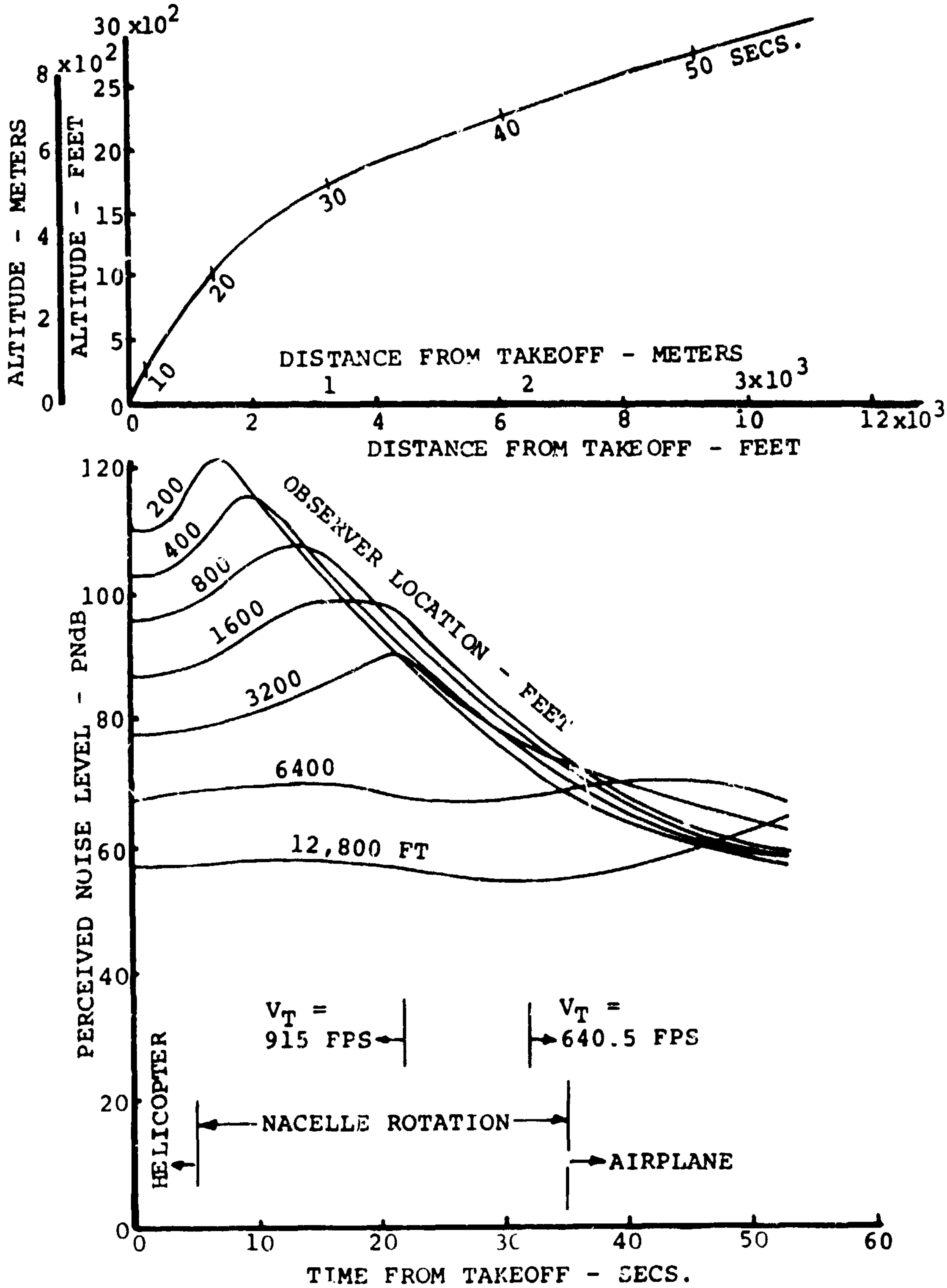


FIGURE 3.68. +5 PNdB TILT ROTOR STANDARD TAKEOFF - PERCEIVED NOISE.

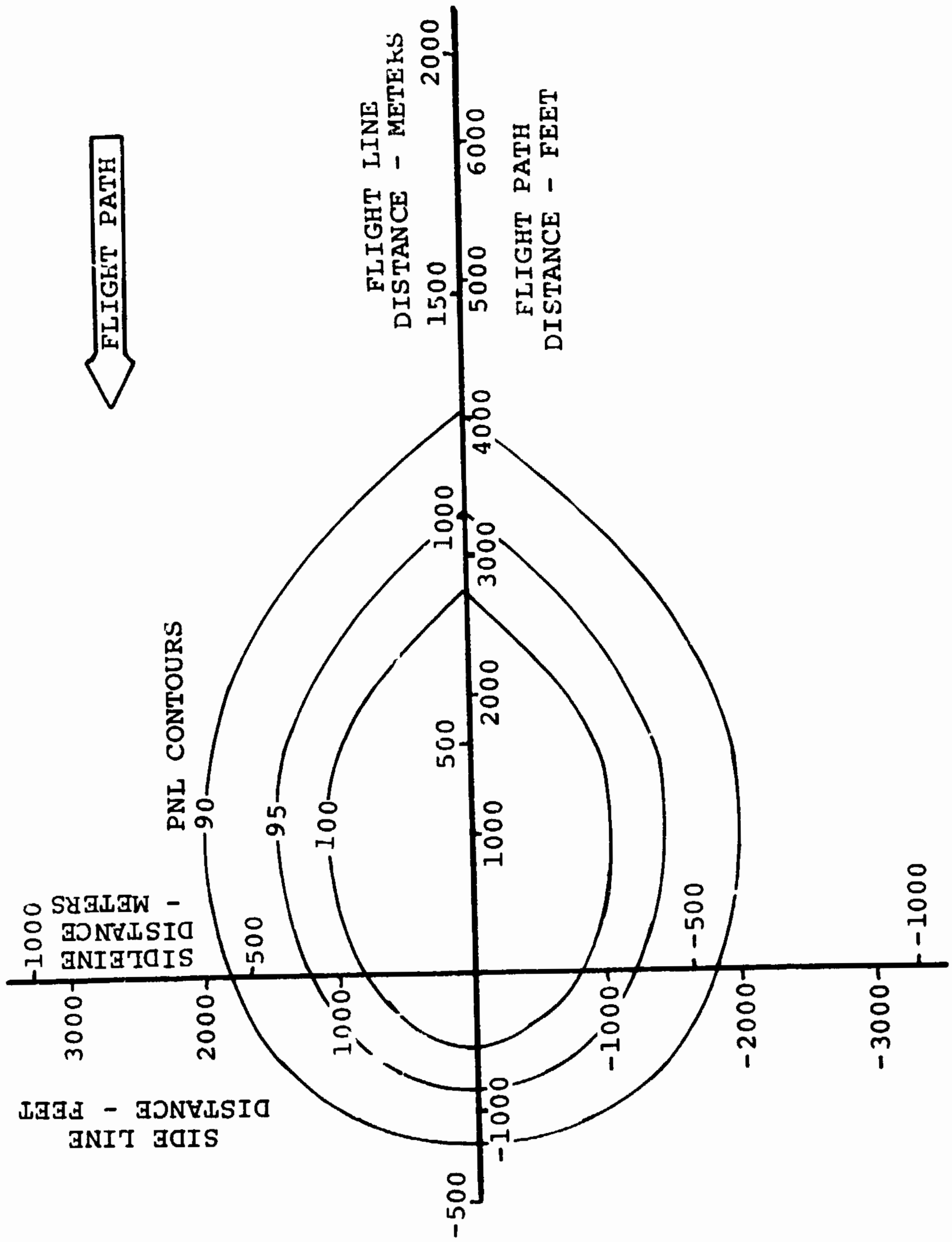


FIGURE 3.69. +5 PNdB TILT ROTOR STANDARD LANDING - PNL CONTOURS.

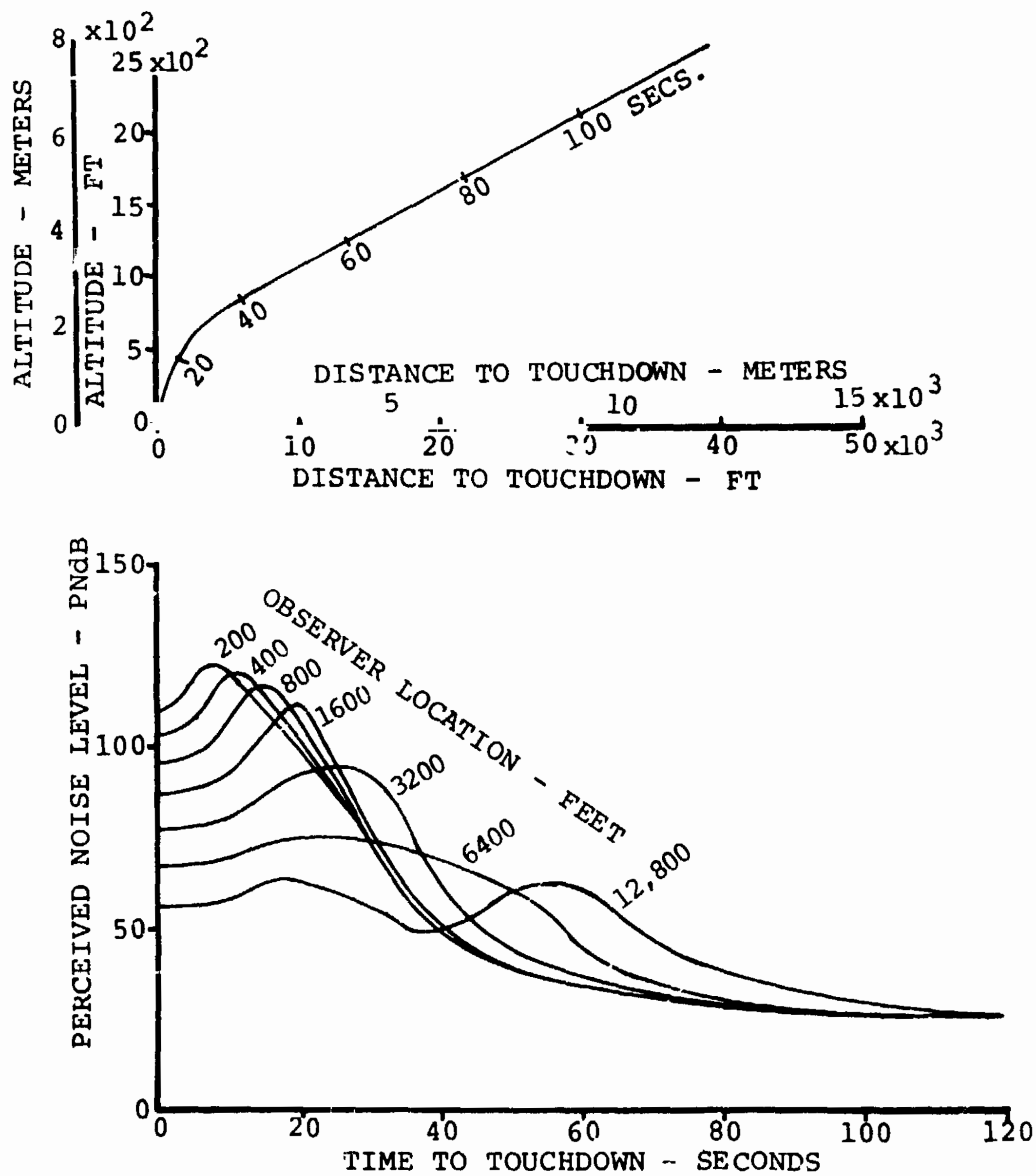


FIGURE 3.70. +5 PNdB TILT ROTOR STANDARD LANDING - PERCEIVED NOISE.

The direct operating cost of the +5 PNdB aircraft is shown in Table 3.22 and indicates almost identical operating costs to the baseline aircraft indicating that no savings were obtained by increasing the external noise design criteria.

An extended range version of the +5 PNdB was also calculated and the initial and direct operating costs at the design range of 230 statute miles is shown in Table 3.23.

The effect of range on direct operating cost is shown in Figure 3.71. The plot shows the basic TR-100 (103.2) design up to 200 nautical miles and the extended range version of the aircraft from 200 to 400 nautical miles. Despite the reduction in available seats on the extended range aircraft the costs do not start to rise again until beyond 300 nautical miles.

3.2.2 Tilt Rotor Design - TR-100 (93.4)

Reduction of noise levels from the baseline aircraft is achieved by changing the rotor design parameters since rotor broadband noise is the dominating contribution to the perceived noise level. To obtain a reduction in PNL of 5 PNdB the tipspeed is reduced to 640 feet per second, and the solidity is increased to 0.111 to obtain minimum direct operating cost for this aircraft as indicated in Figures 3.44 to 3.46.

Configuration and Layout

The -5 PNdB aircraft is shown in Figure 3.72 and a table of characteristics is given in Table 3.24. The fuselage, cabin and cockpit layout and configuration is identical to that

Flyaway Costs

Airframe Costs	$90.00/Lb	$110.00/Lb
Airframe	$3,154,950	$3,856,050
Dynamic System	873,360	873,360
Engines	755,728	755,728
Avionics	250,000	250,000
Total	$5,034,038	$5,735,138

Direct Operating Costs
Dollars/Seat Mile
Block Distance = 230 St. Miles

Utilization (Hrs/Yr)	2500		3500	
Airframe Cost ($/Lb)	90	110	90	110
Flying Operations				
Flight Crew	.0045	.0045	.0045	.0045
Fuel and Oil	.0033	.0033	.0033	.0033
Hull Insurance	.0013	.0015	.0009	.0011
Total Flying Operations	.0091	.0093	.0087	.0089
Direct Maintenance				
Airframe - Labor	.0013	.0013	.0013	.0013
- Material	.00[illegible]	.0014	.0011	.0014
Engines - Labor	.00[illegible]	.0006	.0006	.0006
- Material	.0008	.0008	.0008	.0008
Dynamic System - Labor	.0005	.0005	.0005	.0005
- Material	.0008	.0008	.0008	.0008
Total Direct Maintenance	.0051	.0054	.0051	.0054
Maintenance Burden	.0037	.0037	.0037	.0037
Total Maintenance	.0088	.0091	.0088	.0091
Depreciation	.0063	.0071	.0045	.0051
Total Direct Costs	.0242	.0255	.0220	.0231

TABLE 3.22. +5 PNdB TILT ROTOR, INITIAL AND DIRECT OPERATING COSTS.

TR-100(103.2)
EXTENDED RANGE VERSION

Flyaway Costs		
Airframe Cost	$90.00/Lb	$110.00/Lb
Airframe	$3,171,150	$3,875,850
Dynamic System	873,360	873,360
Engines	755,728	755,728
Avionics	250,000	250,000
Total	$5,050,238	$5,754,938

Direct Operating Costs
Dollars/Seat Mile
Block Distance = 230 St. Miles

Utilization (Hrs/Yr)	2500		3500	
Airframe Cost ($/Lb)	90	110	90	110
Flying Operations				
Flight Crew	.0046	.0046	.0046	.0046
Fuel and Oil	.0033	.0033	.0033	.0033
Hull Insurance	.0013	.0015	.0010	.0011
Total Flying Operations	.0092	.0094	.0089	.0090
Direct Maintenance				
Airframe - Labor	.0014	.0014	.0014	.0014
- Material	.0012	.0014	.0012	.0014
Engines - Labor	.0006	.0006	.0006	.0006
- Material	.0008	.0008	.0008	.0008
Dynamic System - Labor	.0005	.0005	.0005	.0005
- Material	.0008	.0008	.0008	.0008
Total Direct Maintenance	.0053	.0055	.0053	.0055
Maintenance Burden	.0038	.0038	.0038	.0038
Total Maintenance	.0091	.0093	.0091	.0093
Depreciation	.0064	.0072	.0046	.0052
Total Direct Costs	.0247	.0259	.0226	.0235

TABLE 3.23 INITIAL AND DIRECT OPERATIVE COSTS -
+5 PNdB TILT ROTOR (EXTENDED VERSION).

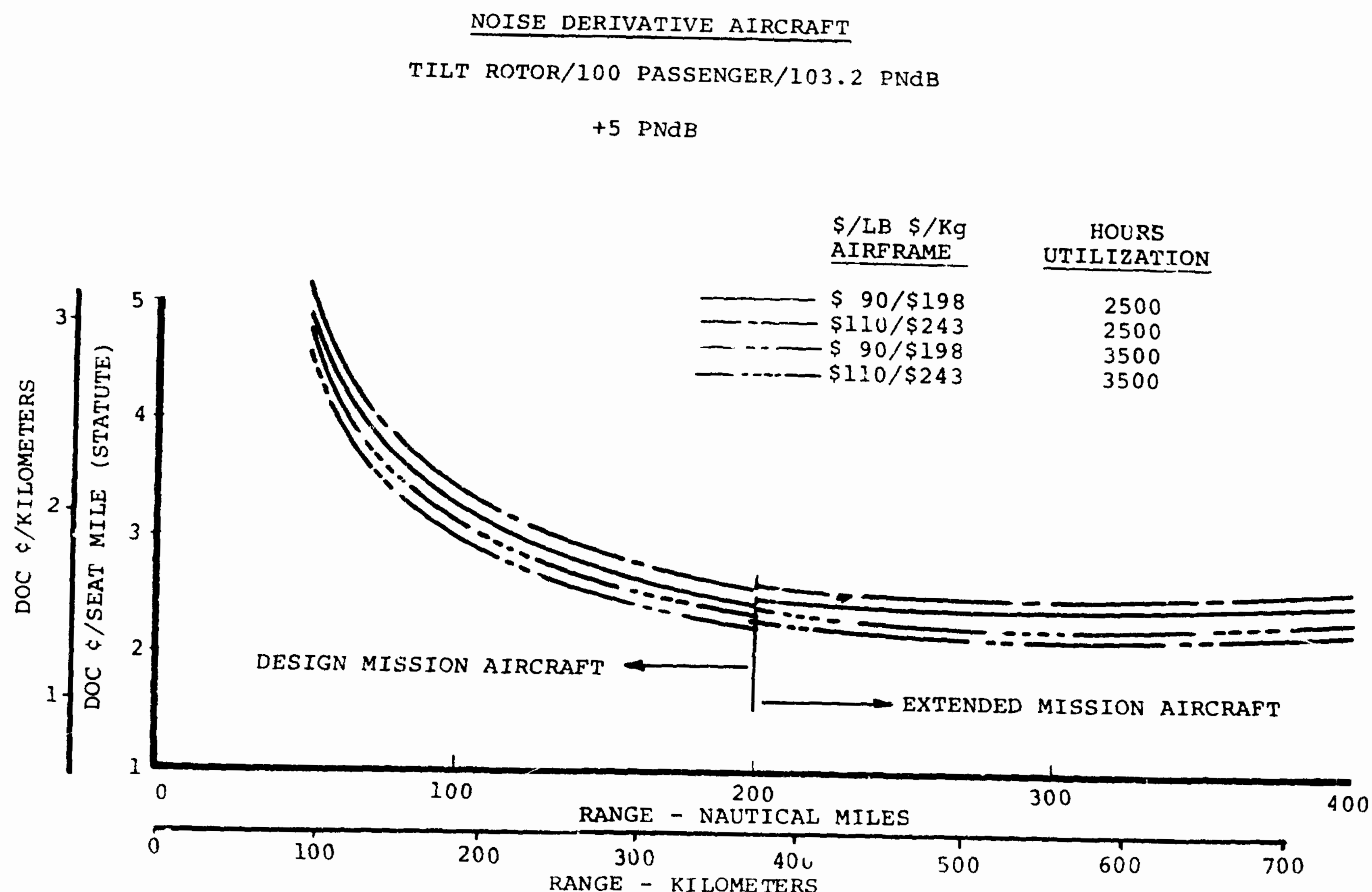


FIGURE 3.71. EFFECT OF OPERATING RANGE ON DIRECT OPERATING COSTS.

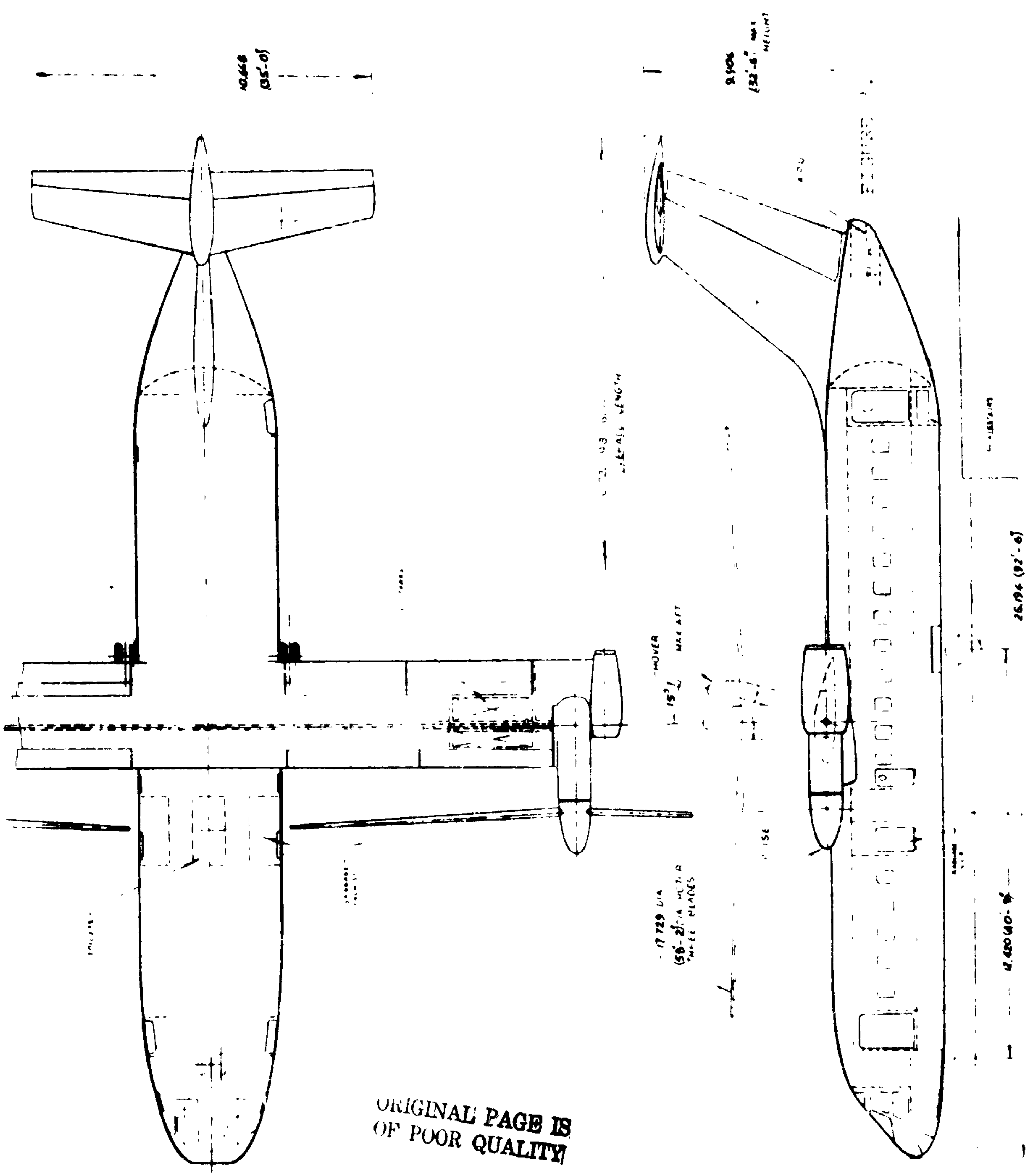


FIGURE 3.72. 1985 COMMERCIAL VTOL TRANSPORT, 100 PASSENGER - TILT ROTOR -5 JB DESIGN.

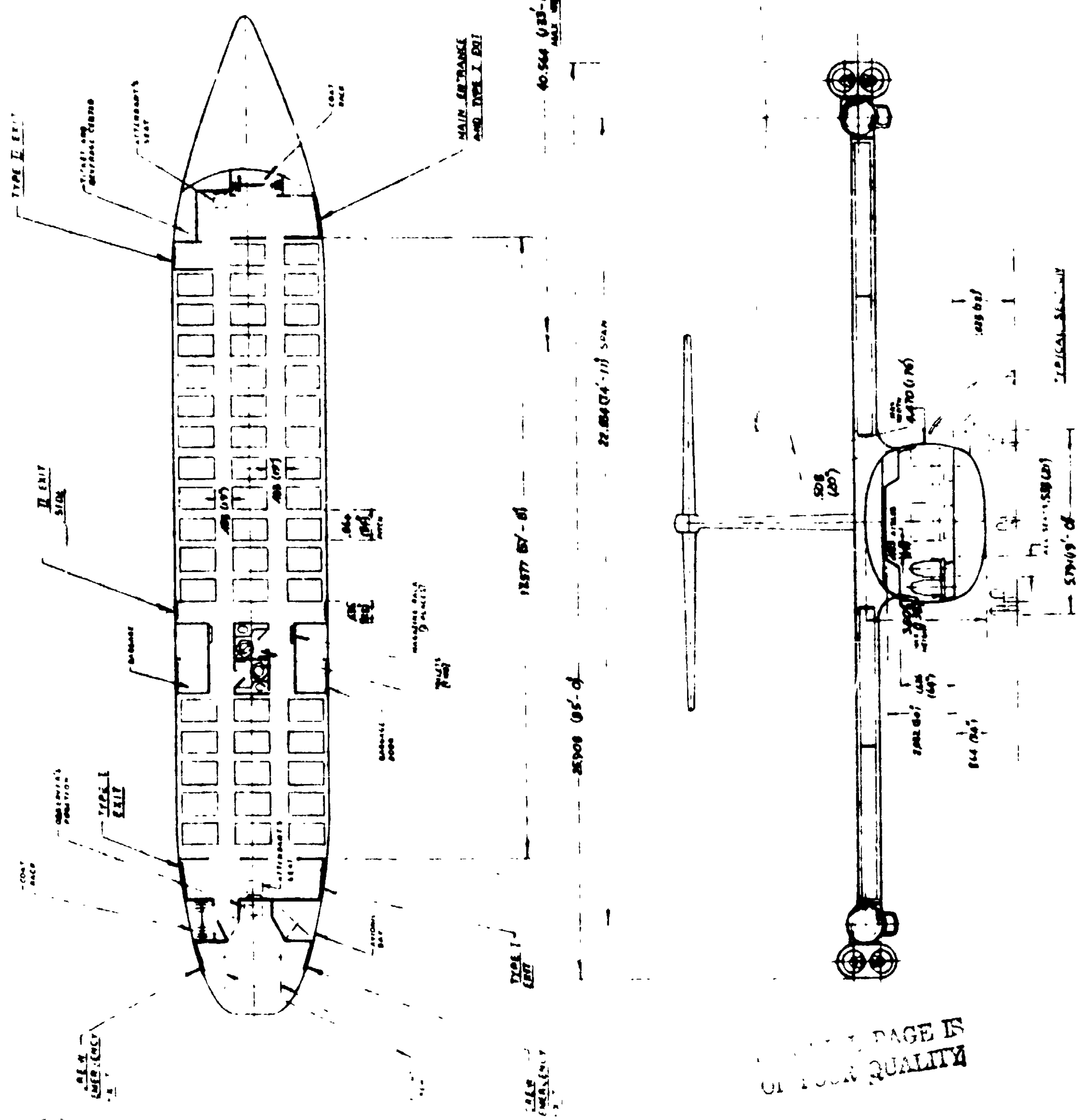


FIGURE 3.72. 1985 COMMERCIAL VTOL TRANSPORT, 100 PASSENGER - TILT ROTOR -5 PNdB DESIGN (CONTINUED).

	S. I. UNITS	U. S. UNITS
WEIGHTS		
DESIGN GROSS WEIGHT	36,143 Kg	79,682 Lbs
EMPTY WEIGHT	24,820 Kg	54,718 Lbs
FUEL WEIGHT	2,240 Kg	4,939 Lbs
NUMBER OF PASSENGERS	100	100
ROTORS		
DISC LOADING	73.24 Kg/m^2	15 Lbs/Ft^2
DIAMETER	17.74 m	58.2
SOLIDITY	.111	.111
BLADE NUMBER	3	3
TWIST	36 Degs	36 Degs
TIP SPEED HOVER/CRUISE	195/137 m/s	640/448 Ft/Sec
POWER		
NUMBER OF ENGINES	4	4
RATED POWER/ENGINE	3.631×10^6 Watts	4869 SHP
FUSELAGE		
LENGTH	28.19 m	92.5 Ft
WIDTH	4.51 m	14.8 Ft
CABIN LENGTH	17.58 m	57.67 Ft
WING		
AREA	74.03 m^2	796.8 Ft^2
SPAN	22.83 m	74.9 Ft
TAPER RATIO	1.0	1.0
CHORD	3.23 m	10.6 Ft
ASPECT RATIO	7.04	7.04
AIRFOIL t/c	.21	.21
HORIZONTAL TAIL		
AREA	20.13 m^2	216.7 Ft^2
SPAN	10.67 m	35 Ft
TAIL VOLUME RATIO	1.31	1.31
ASPECT RATIO	5.65	5.65
VERTICAL TAIL		
AREA	26.95 m^2	290.1 Ft^2
SPAN	6.14 m	20.15 Ft
TAIL VOLUME RATIO	.159	.159
ASPECT RATIO	1.4	1.4
PERFORMANCE		
NRP CRUISE SPEED	182.6 m/s	355 KTAS
CRUISE ALTITUDE	4267 m	14,000 Ft
BLOCK TIME	.730 Hours	.730 Hours
NOISE		
SIDELINE NOISE - 500 FEET/ HOVER	93.4 PNdB	93.4 PNdB

TABLE 3.24. -5 PNdB DERIVATIVE DESIGN POINT TILT ROTOR TABLE OF CHARACTERISTICS.

of the baseline tilt rotor aircraft. The design parameters changed to obtain the reduced PNL are solidity and tipspeed. The major effect of changes in these parameters is in the dynamic system of the vehicle. The rotor disc loading was held at 15 pounds per feet squared and the rotor diameter increased to 58.2 feet. The wing span is dictated by the rotor radius plus rotor fuselage clearance and also increases to 74.9 feet.

The wing loading of 100 pounds per feet square was maintained and as a result wing area increased to 796.8 feet square and the aspect ratio reduced to AR = 7.04.

The increased aircraft gross weight demands a higher installed power (4869 SHP per engine) which in combination with reduced tipspeed and therefore higher torque levels implies a larger and heavier transmission.

The change in cruise RPM reduces the nose up pitching moment effect of the rotor and results in a lower horizontal tail volume ratio (1.31).

The increased installed power and decreased RPM (i.e., increased rotor efficiency) improve the cruise performance a little to give a normal rated power speed of 355 knots at 14,000 feet altitude.

TR-100 (93.4) - Weights

The design takeoff gross weight for the -5 PNdB tilt rotor is 36,143 Kg (79,682 pounds) an increase of nearly 5,000 pounds over the baseline aircraft. This is due to the reduction in rotor tipspeed and increased rotor solidity and diameter. The weights statement is given in Table 3.25.

The reduction in tipspeed tends to reduce rotor weight, but this effect is more than offset by the increase due to solidity and diameter and the net result is a slighly heavier rotor system.

The flight control weights follow the rotor weight because the upper control design is set by rotor blade size, weight and pitch inertia. The flight controls weight is increased accordingly. The governing parameter in the drive system weights is the reduction in tipspeed which increases the torque requirements. This coupled with the larger power requirement of the -5 PNdB tilt rotor causes a substantial increase in the drive system weight. The larger power requirement also implies higher engine and installation weights. The result is a 29.5% increase in propulsion group weights over the baseline aircraft. The landing gear is taken as a percentage of empty weight and increases accordingly.

The basic fuselage weight, cabin and cockpit accommodations, etc., remain the same as those of the basic tilt rotor design.

The increase in takeoff gross weight of this aircraft requires an increase in mission fuel to 2,240.3 Kg (4,939 pounds), 11% more than the baseline aircraft. The principle inertias and CG locations for this aircraft are given in Table 3.26.

WEIGHT SUMMARY - PRELIMINARY DESIGN

	KILOGRAMS	POUNDS
WING	2018.0	4449
ROTOR	2566.0	5657
TAIL	695.8	1534
SURFACES	695.8	1534
ROTOR		
BODY	3863.7	8518
BASIC		
SECONDARY		
ALIGHTING GEAR GROUP	1445.6	3187
ENGINE SECTION	504.8	1113
PROPULSION GROUP	6145.2	13548
ENGINE INST'L	1391.6	3068
EXHAUST SYSTEM *		
COOLING		
CONTROLS *		
STARTING *		
PROPELLER INST'L *	*431.4	*951
LUBRICATING *		
FUEL	105.2	232
DRIVE	4217.0	9297
FLIGHT CONTROLS	1979.5	4364
AUX. POWER PLANT	288.5	636
INSTRUMENTS	191.9	423
HYDR. & PNEUMATIC	308.4	680
ELECTRICAL GROUP	378.3	834
AVIONICS GROUP	293.9	648
ARMAMENT GROUP		
FURN. & EQUIP. GROUP	3273.6	7217
ACCOM. FOR PERSON.		
MISC. EQUIPMENT		
FURNISHINGS		
EMERG. EQUIPMENT		
AIR CONDITIONING	612.3	1350
ANTI-ICING GROUP	254.0	560
LOAD AND HANDLING GP.		
WEIGHT EMPTY	24819.5	54718
CREW	299.4	660
TRAPPED LIQUIDS	52.2	115
ENGINE OIL	59.9	1[illegible]2
CREW ACCOMMODATIONS	68.0	150
EMERGENCY EQUIPMENT	23.6	52
PASSENGER ACCOMMO.	415.5	916
PASSENGERS (100)	8164.6	180[illegible]0
FUEL	2240.3	4939
GROSS WEIGHT	36143.0	79682

TABLE 3.25 . -5 PNdB TILT ROTOR WEIGHT SUMMARY.

	WEIGHT EMPTY	GROSS WEIGHT
WEIGHT	24,819.5 Kg (54,718 LB)	36,143.0 Kg (79,682 LB)
CENTER OF GRAVITY*		
HORIZONTAL FLIGHT		
FUSELAGE STATION	12.72 M (500.6 In.)	12.77 M (502.6 In.)
WATER LINE	3.56 M (140.6 In.)	3.27 M (128.6 In.)
VERTICAL FLIGHT		
FUSELAGE STATION	13.08 M (515.0 In.)	13.12 M (516.5 In.)
WATER LINE	3.97 M (156.2 In.)	3.54 M (139.2 In.)
MOMENT OF INERTIA		
HORIZONTAL FLIGHT		
I_{xx} (ROLL)	1,307,205 Kg M^2 (963,376 Slug Ft^2)	1,405,598 Kg M^2 (1,035,889 Slug Ft^2)
I_{yy} (PITCH)	563,663 Kg M^2 (416,879 Slug Ft^2)	608,241 Kg M^2 (448,258 Slug Ft^2)
I_{zz} (YAW)	1,523,096 Kg M (1,122,482 Slug Ft^2)	1,637,736 Kg M (1,206,969 Slug Ft^2)
VERTICAL FLIGHT		
I_{xx} (ROLL)	1,374,108 Kg M^2 (1,012,682 Slug Ft^2)	1,477,537 Kg M^2 (1,088,906 Slug Ft^2)
I_{yy} (PITCH)	613,784 Kg M^2 (452,343 Slug Ft^2)	659,983 Kg M^2 (486,390 Slug Ft^2)
I_{zz} (YAW)	1,647,804 Kg M^2 (1,214,389 Slug Ft^2)	1,771,832 Kg M^2 (1,305,794 Slug Ft^2)

*FUSELAGE STATION 0 IS NOSE OF BODY, CENTERLINE OF ROTOR IN HORIZONTAL FLIGHT IS 4.6 METERS ABOVE WATER LINE 0.

TABLE 3.26. WEIGHT, CENTER OF GRAVITY, AND MOMENT OF INERTIA -5 PNDB TILT ROTOR

Tilt Rotor Design - TR-100 (93.4) - Vehicle Performance

Mission Performance

The -5 PNdB tilt rotor is sized to fly the 200 nautical mile mission with the same reserve capability as the baseline aircraft. A summary of the mission performance is provided in Tables 3.27 and 3.28.

The taxi takeoff and initial air maneuver require 113 Kg (227 pounds) of fuel. The initial air maneuver is included with the takeoff in Tables 3.27 and 3.28. The aircraft then climbs to a cruise altitude of 14,000 feet at an average rate of climb of 27 m/s (5,316 feet per minute) for a range credit of 19 Km (10 nautical miles). The fuel used during the climb to altitude amounts to 165 Kg (364 pounds).

The cruise segment is performed at 14,000 feet at an average speed of 183 m/s (355 KTAS) for a range credit of 315 Km (170 nautical miles). The cruise fuel used is 1,162 Kg (2,561 pounds).

The descent to 2,000 feet altitude is done at an average rate of descent of 20 m/s (1,960 feet per minute) and completes the range to 371 Km (200 nautical miles). The fuel required for the descent is 79 Kg (173 pounds). An air maneuver at 2,000 feet used an additional 28 Kg (52 pounds) of fuel. The descent from 2,000 feet, landing and taxi complete the design mission for a total fuel weight of 78 Kg (181 pounds).

The additional fuel load required for 50 nautical miles and 20 minutes stand-off at an alternate landing site require a

-5 PNdB TILT ROTOR - MISSION BREAKDOWN

	TIME (HOURS)	DISTANCE (N.MI.)	WEIGHT (LBS)	FUEL (LBS)	V (KNOTS)	R/C (FT/MIN)
	0	0	79,682			
TAXI				16	0	0
	.017	0	79,666			
TAKEOFF				211	0	0
	.050	0	79,455			
CLIMB				364	190	+4300
	.104	9.98	79,091			
CRUISE				2,561	356	0
	.553	170	76,530			
DESCENT				174	282	-2000
	.655	200	76,356			
AIR MANEUVER				62	152	0
	.680	200	76,294			
DESCENT				14	255	-2000
	.689	202	76,280			
LANDING				150	0	0
	.714	202	76,130			
TAXI				17	0	0
	.730	202	76,113			
RESERVE				1,403	154/233	0
	1.268	250	74,710			

TABLE 3.27 . -5 PNdB TILT ROTOR DESIGN MISSION PERFORMANCE (U.S. UNITS).

-5 PNdB TILT ROTOR - MISSION SUMMARY

	TIME (HOURS)	DISTANCE (Km)	WEIGHT (Kg)	FUEL (Kg)	V (KNOTS)	R/C (m/s)
	0	0	36,143			
TAXI				7	0	0
	.017	0	36,136			
TAKEOFF				96	0	0
	.050	0	36,040			
CLIMB				165	190	27
	.104	19	35,875			
CRUISE				1162	356	0
	.553	315	34,713			
DESCENT				79	282	-20/-10
	.655	371	34,634			
AIR MANEUVER				28	152	0
	.680	371	34,606			
DESCENT				6	255	-10
	.689	374	34,600			
LANDING				68	0	0
	.714	374	34,532			
TAXI				8	0	0
	.730	374	34,524			
RESERVE				639	154/233	0
	1.268	463	33,885			

TABLE 3.28. MISSION SUMMARY FOR -5 PNdB NOISE DERIVATIVE TILT ROTOR - (S.I. UNITS).

reserve fuel load of 639 Kg (1,403 pounds) bringing the total mission fuel to 2,240 Kg (4,939 pounds).

The basic mission block time is 0.73 hours.

Hover Performance

The effect of ambient temperature on sea level hover performance s shown in Figures 3.73 and 3.74. The all engines operating case, Figure 3.73 shows a gross weight lift capability of 99,500 pounds at sea level, standard day and 89,000 pounds at 90 degrees F at takeoff power out of ground effect. At temperatures above 59 degrees F the aircraft is power limited and below 59 degrees F the transmission torque limits the gross lift.

With one engine inoperative, Figure 3.74, performance is power limited. This is the sizing condition which defines the takeoff gross weight as 79,682 pounds OEI at sea level, 90 degrees F.

The effect of altitude on hover performance is shown in Figures 3.75 and 3.76. With all engines operating on a hot day the aircraft can maintain hover up to 5,000 feet and 9,000 feet on a standard day. With one engine inoperative, the aircraft can hover at design gross weight at 5,000 feet on a standard day.

Cruise Performance

The power required and power available data for the -5 PNdB tilt rotor is shown in Figures 3.77 and 3.78 at 5,000 feet altitude and 14,000 feet altitude. At 5,000 feet the cruise

NOISE DERIVATIVE AIRCRAFT

TILT ROTOR/100 PASSENGER/93.4 PNdB

ALL ENGINES OPERATING

-5 PNdB

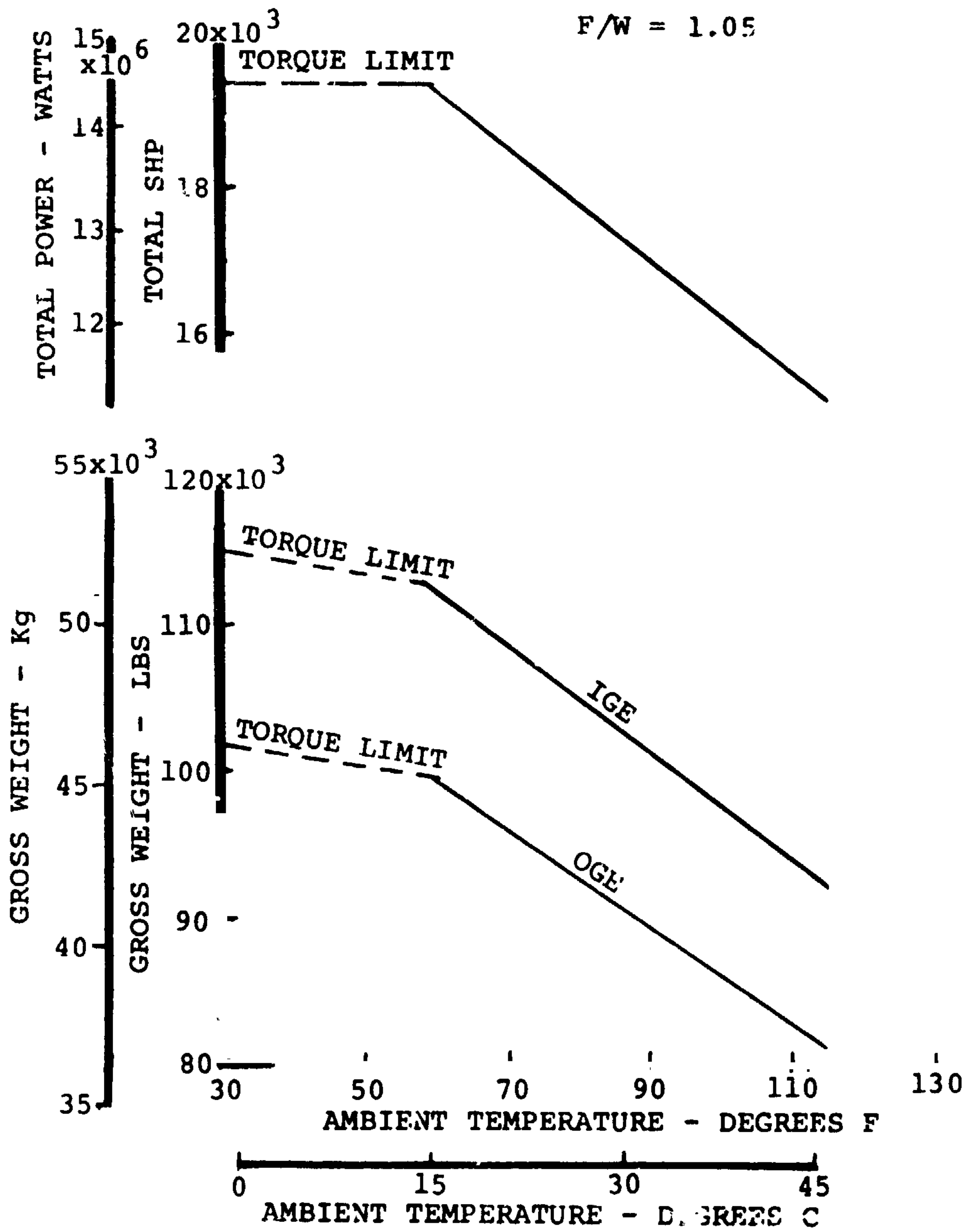


FIGURE 3.73. EFFECT OF AMBIENT TEMPERATURE ON SEA LEVEL HOVER PERFORMANCE - -5 PNdB TILT ROTOR.

NOISE DERIVATIVE AIRCRAFT

TILT ROTOR/100 PASSENGER 93.4 PNdB

ONE ENGINE INOPERATIVE

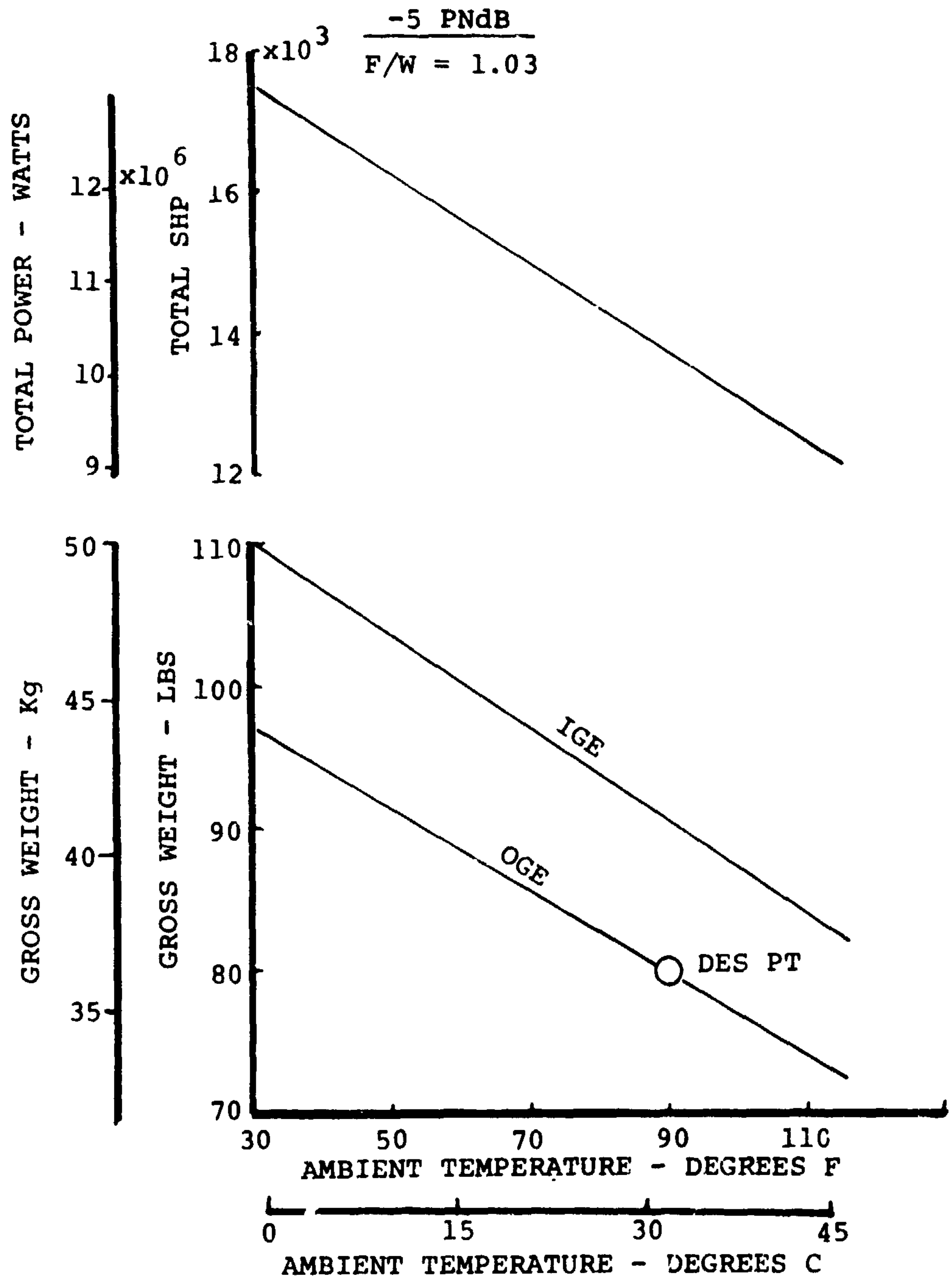


FIGURE 3.74. EFFECT OF AMBIENT TEMPERATURE ON SEA LEVEL HOVER PERFORMANCE - -5 PNdB TILT ROTOR.

NOISE DERIVATIVE AIRCRAFT

TILT ROTOR/100 PASSENGER/93.4 PNdB

STANDARD DAY AND STANDARD DAY +31° F (+17.2° C)

ALL ENGINES OPERATING

-5 PNdB

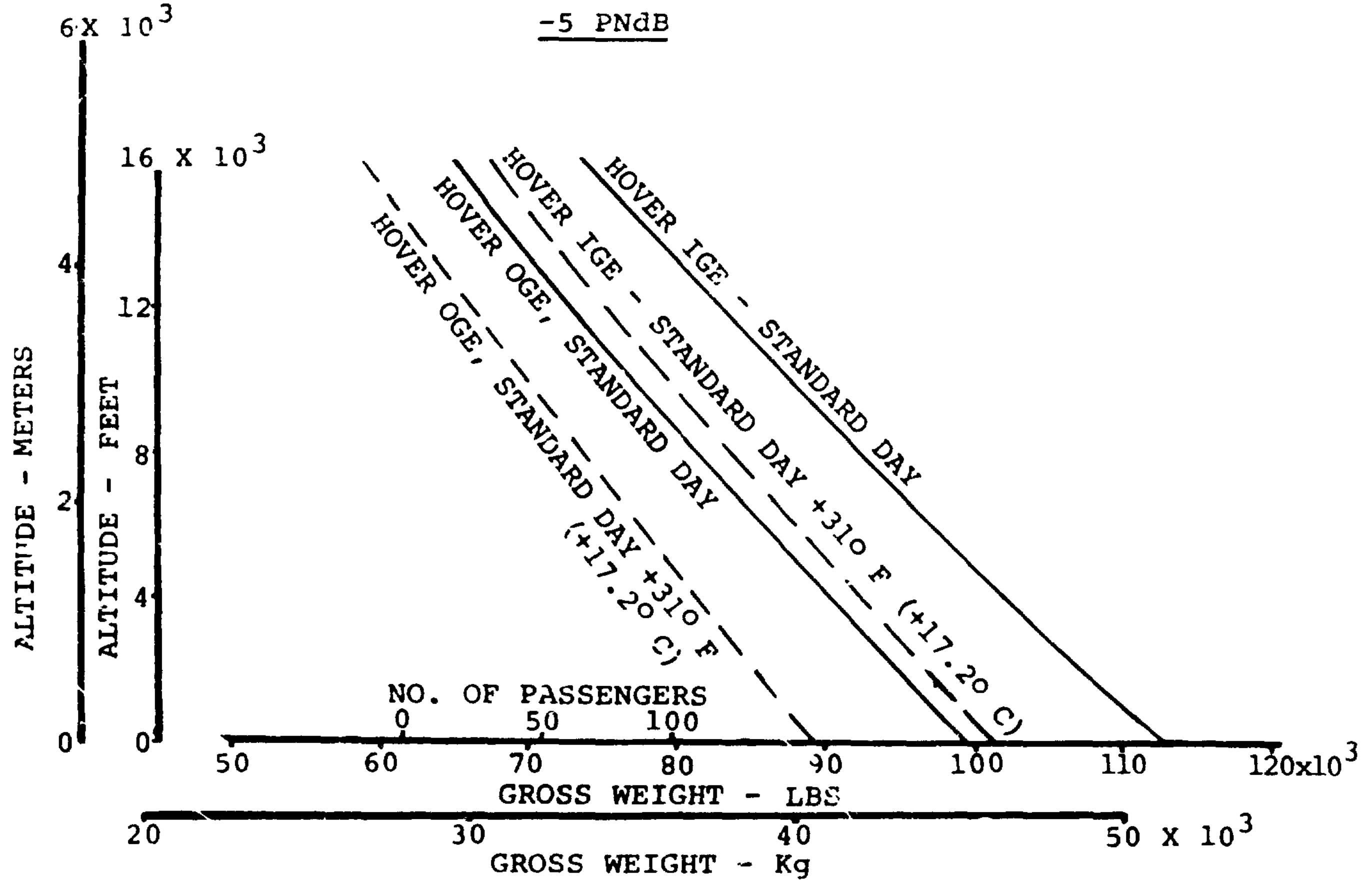


FIGURE 3.75. EFFECT OF ALTITUDE ON HOVER PERFORMANCE - -5 PNdB TILT ROTOR.

NOISE DERIVATIVE AIRCRAFT

TILT ROTOR/100 PASSENGER/93.4 PNdB

STANDARD DAY AND STANDARD DAY +31° F (+17.2° C)

ONE ENGINE INOPERATIVE

-5 PNdb

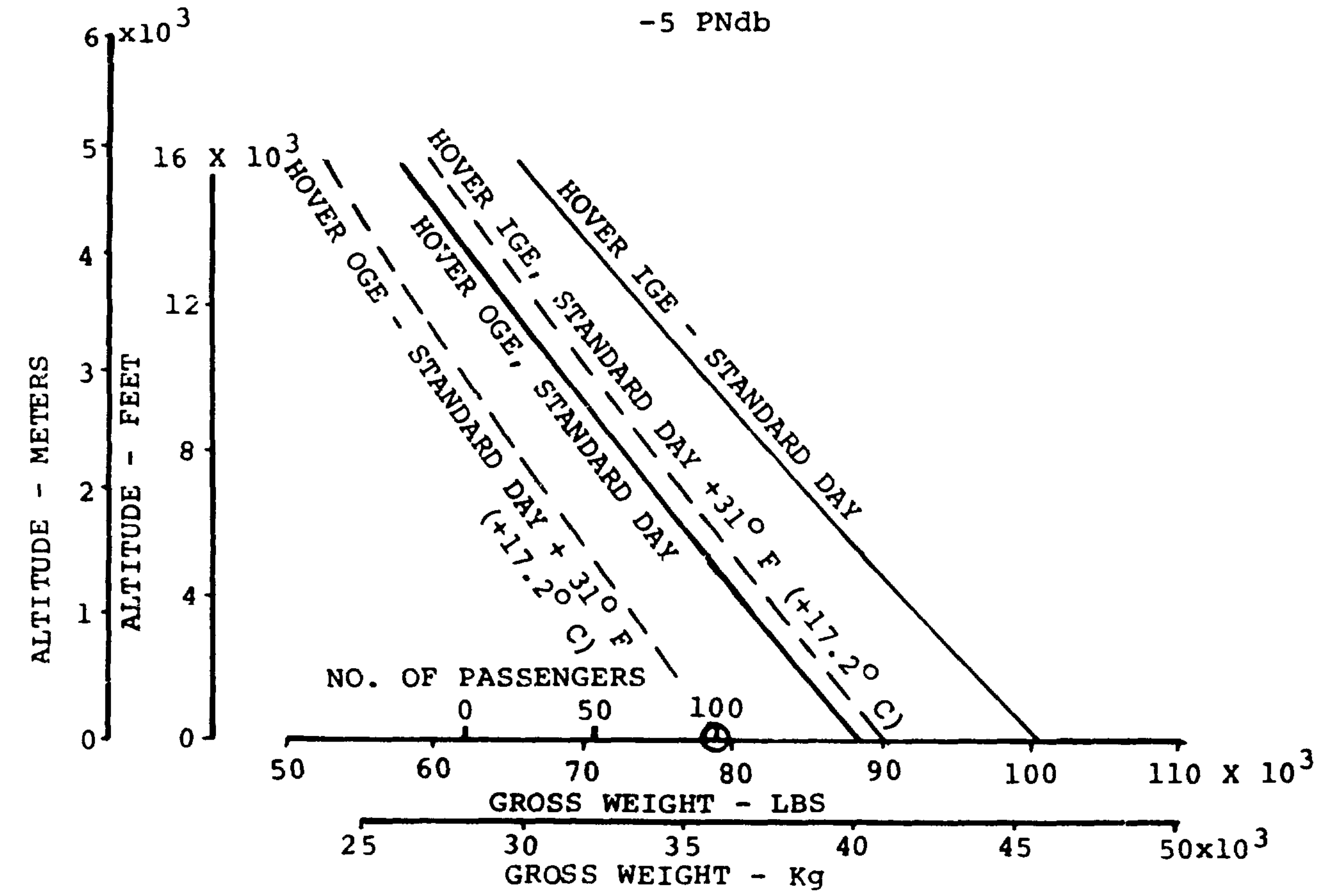


FIGURE 3.76. EFFECT OF ALTITUDE ON HOVER GROSS WEIGHT CAPABILITY.

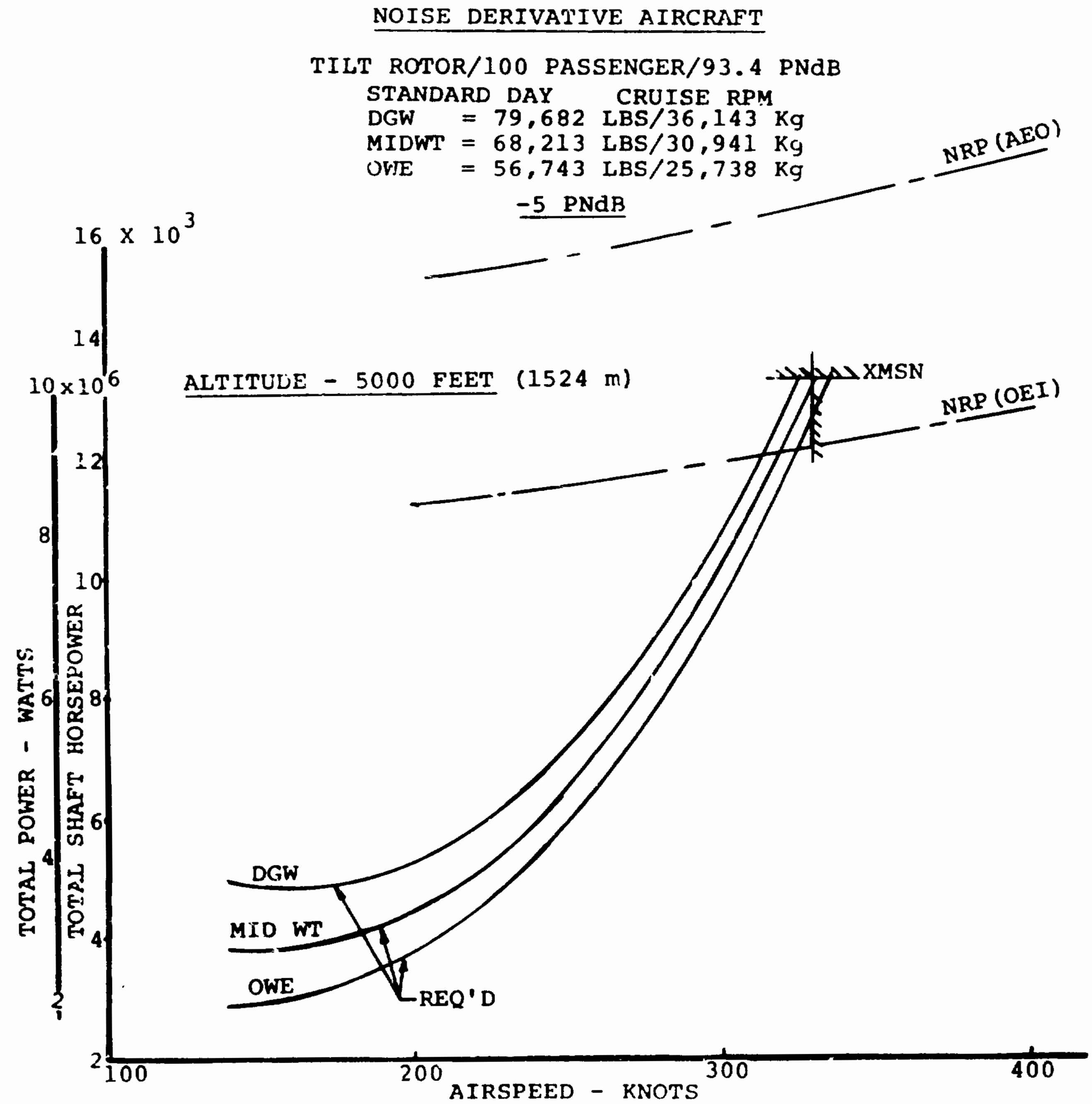


FIGURE 3.77. -5 PNdB TILT ROTOR CRUISE PERFORMANCE POWER REQUIRED/AVAILABLE.

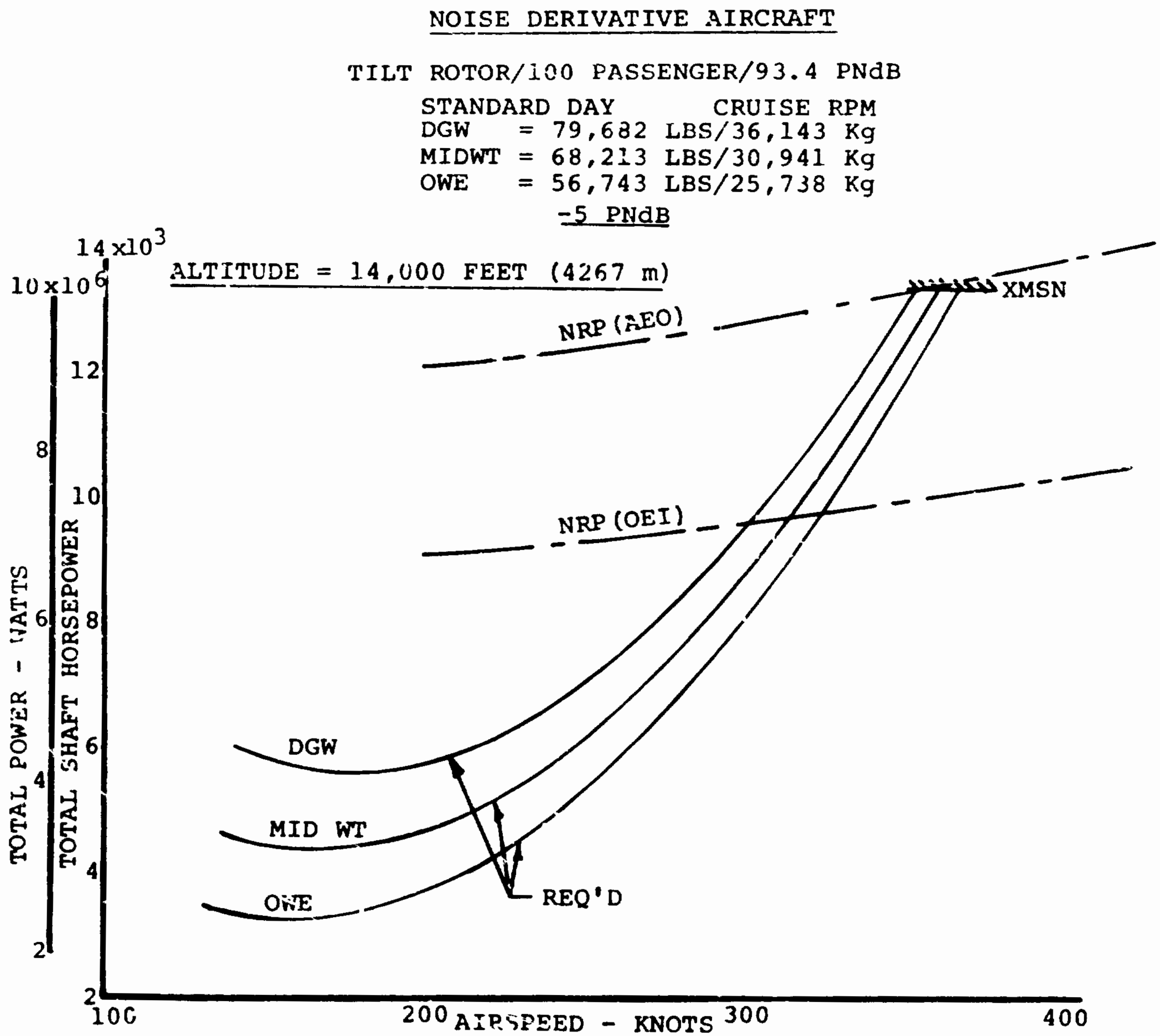


FIGURE 3.78. CRUISE PERFORMANCE - POWER REQUIRED/AVAILABLE - -5 PNdB TILT ROTOR.

performance is limited by the transmission torque limit (all engines operating). The one engine inoperative power limit at NRP is less than the transmission torque limit and defines the maximum cruise capability at 5,000 feet.

The maximum cruise speed at 14,000 feet at design gross weight is 355 Knots and is limited by both power and transmission torque limit, all engines operating. This condition was used to size the transmission limit. At lighter weights the maximum performance is transmission limited (AEO). With one engine inoperative the power available limits cruise performance as shown in Figure 3.78.

The maximum speed performance of the -5 PNdB aircraft is plotted as a function of altitude in Figure 3.79. With all engines operating, the transmission limit defines the maximum cruise speed below 14,000 feet, and the power is limiting at higher altitudes.

The aircraft is capable of speeds in excess of 300 knots at all weights OEI and AEO, however, the 250 knot EAS restriction shown on Figure 3.79 defines the operating envelope at altitudes less than 10,000 feet.

The maximum rate of climb at design gross weight is 5,250 feet per minute at sea level with all engines operating as shown in Figure 3.80. At design cruise altitude the rate of climb reduces to 3,300 feet per minute. With one engine inoperative the aircraft can still maintain 4,100 feet per minute rate of climb at sea level and 2,270 feet per minute at 14,000 feet

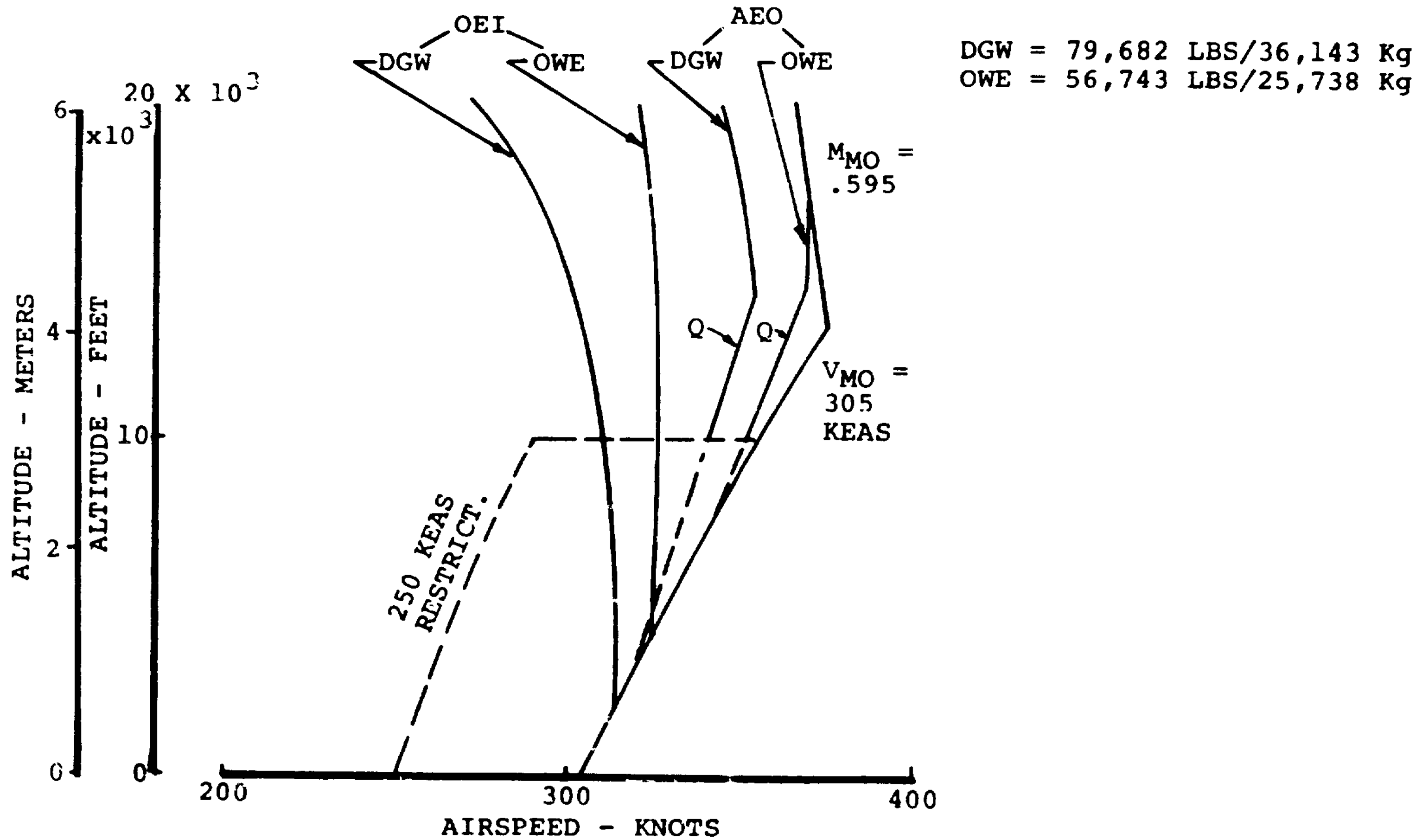


FIGURE 3.79. LEVEL FLIGHT CRUISE SPEED ENVELOPE - -5 PNdB TILT ROTOR.

NOISE DERIVATIVE AIRCRAFT

TILT ROTOR/100 PASSENGER/93.4 PNdB

CLIMB CAPABILITY TAKEOFF RPM
MIL POWER STANDARD DAY
AEO & OEI

-5 PNdB

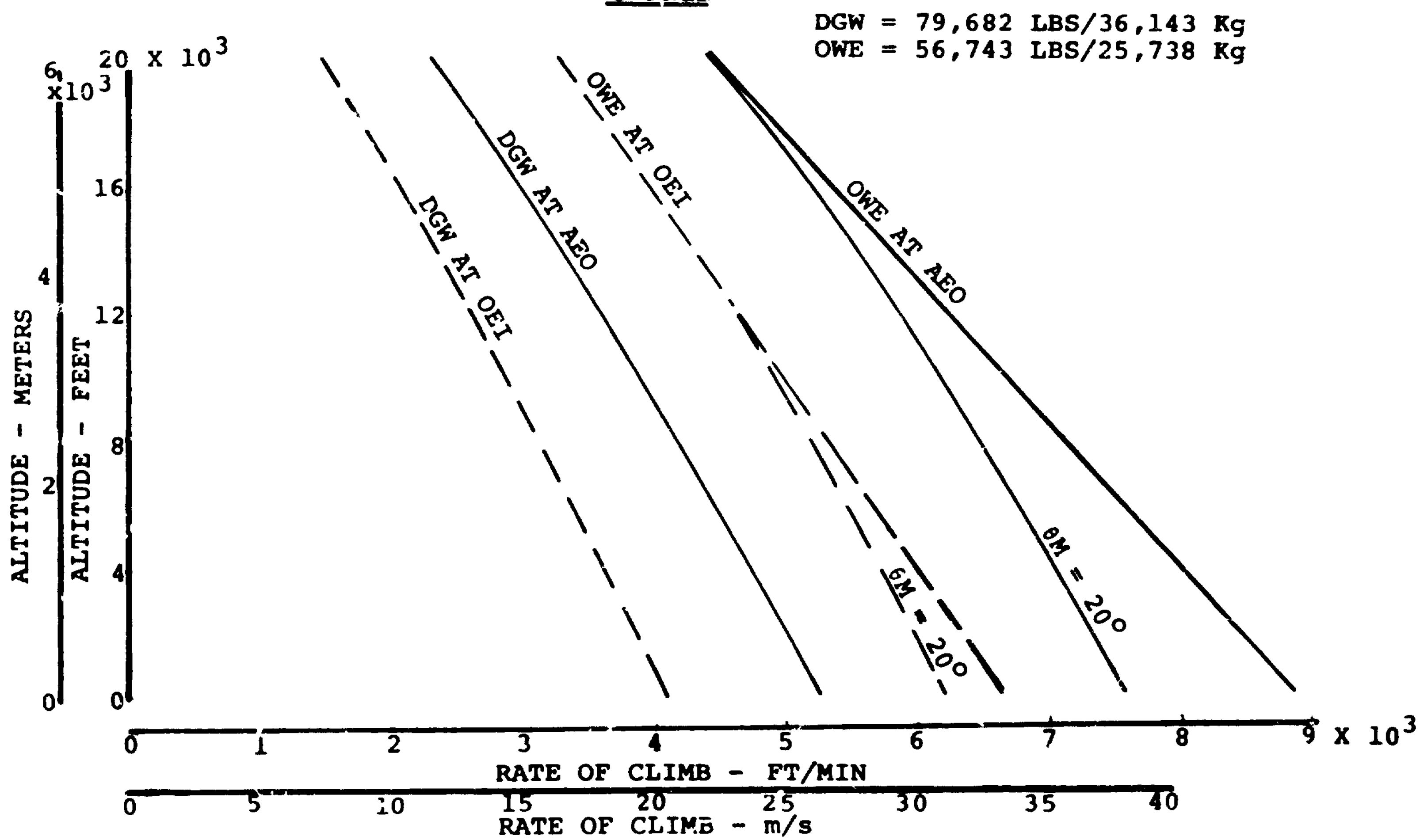


FIGURE 3.80. -5 PNdB TILT ROTOR - RATE OF CLIMB.

at design gross weight.

The aircraft fuel consumption data in cruise are shown in Figures 3.81 and 3.82 in terms of specific range. Data are given for 5,000 feet and 14,000 feet altitudes for both all engines operating, Figure 3.81, and one engine inoperative, Figure 3.82.

At 5,000 feet, all engines operating, the maximum specific range at design gross weight is 0.0667 nautical miles per pound of fuel rising to 0.082 nautical miles per pound of fuel at operating weight empty.

At design altitude these values increase to 0.081 nautical miles per pound of fuel and 0.106 nautical miles per pound of fuel respectively.

The maximum specific range performance improves with one engine inoperative, Figure 3.82, because the engine is operating at a higher power fraction which reduces the SFC.

The payload range performance of the aircraft is as defined by the mission and is shown in Figure 3.83. The basic aircraft carries 100 passengers over 200 nautical mile range exclusive of reserve fuel. The basic mission f el allows a range of 252 nautical miles with no payload. The improved specific range OEI is reflected in the payload range data shown in Figure 3.84. The fully laden aircraft has a range of 219 nautical miles exclusive of reserves.

Flying Qualities - TR-100 (93.4)

In hover, longitudinal cyclic pitch control is used to trim

NOISE DERIVATIVE AIRCRAFT

TILT ROTOR/100 PASSENGER/93.4 PNdB

STANDARD DAY CRUISE RPM

DGW = 79,628 LBS/36,143 Kg
MIDWT = 68,213 LBS/30,941 Kg
OWE = 56,743 LBS/25,738 Kg

ALL ENGINES OPERATING

-5 PNdB

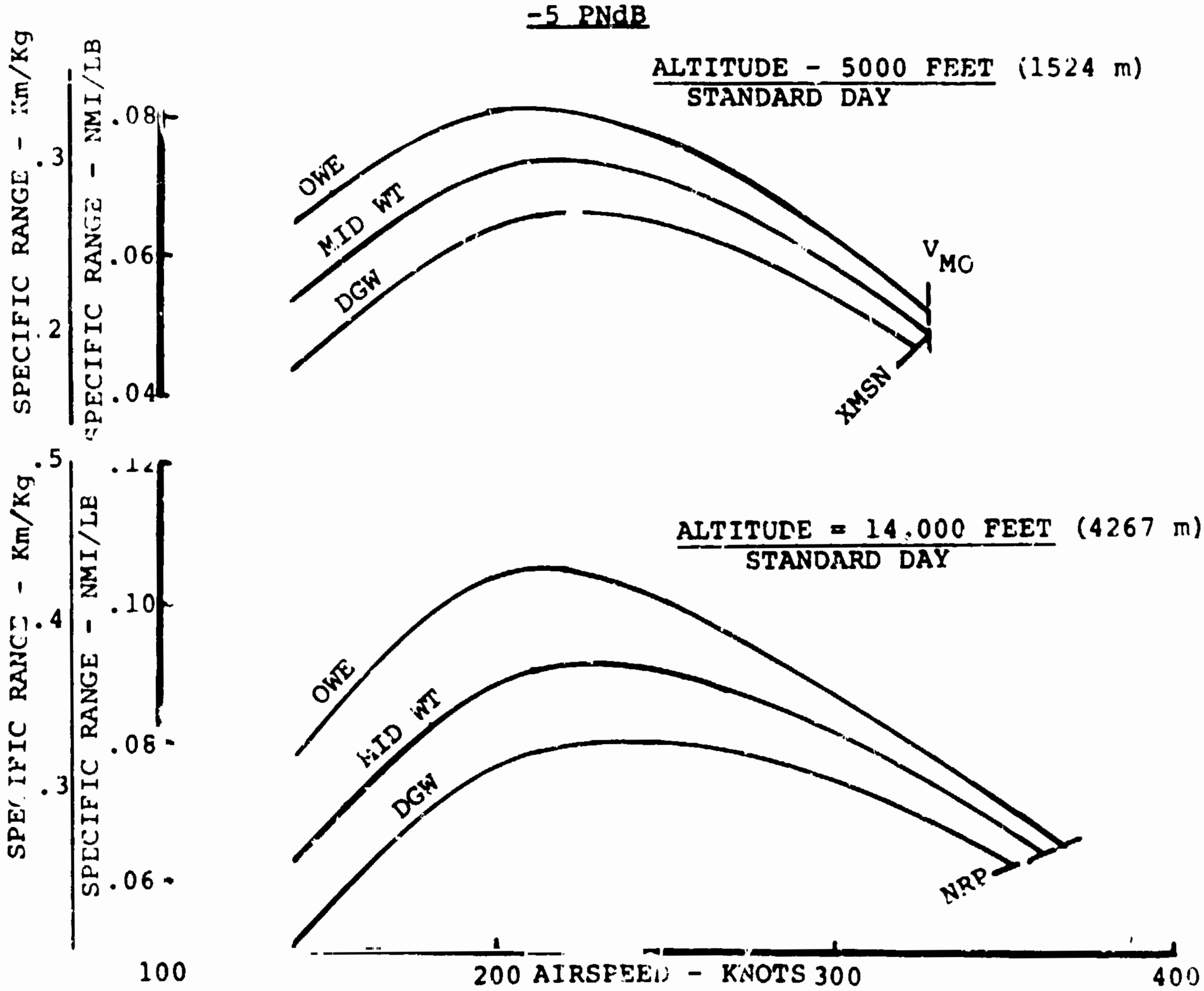


FIGURE 3.81. CRUISE PERFORMANCE - SPECIFIC RANGE - -5 PNdB TILT ROTOR - AEO.

NOISE DERIVATIVE AIRCRAFT

TILT ROTOR/100 PASSENGER/93.4 PNdB

STANDARD DAY CRUISE RPM

DGW = 79,682 LBS/36,148 Kg
MIDWT = 68,213 LBS/30,941 Kg
OWE = 56,743 LBS/25,738 Kg

ONE ENGINE INOPERATVIE

-5 PNdB

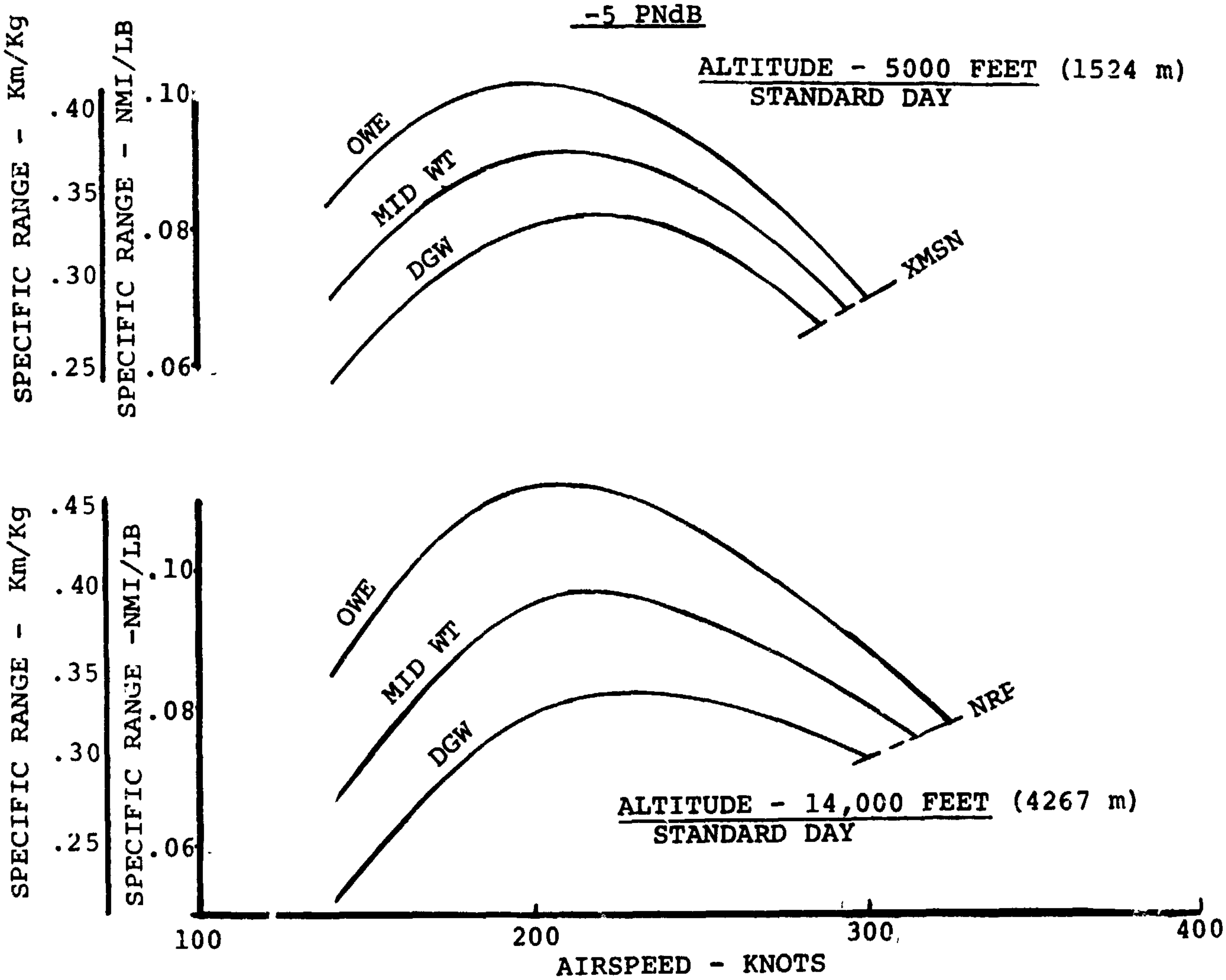


FIGURE 3.82. CRUISE PERFORMANCE - SPECIFIC RANGE - -5 PNdB TILT ROTOR - OEI.

C-5

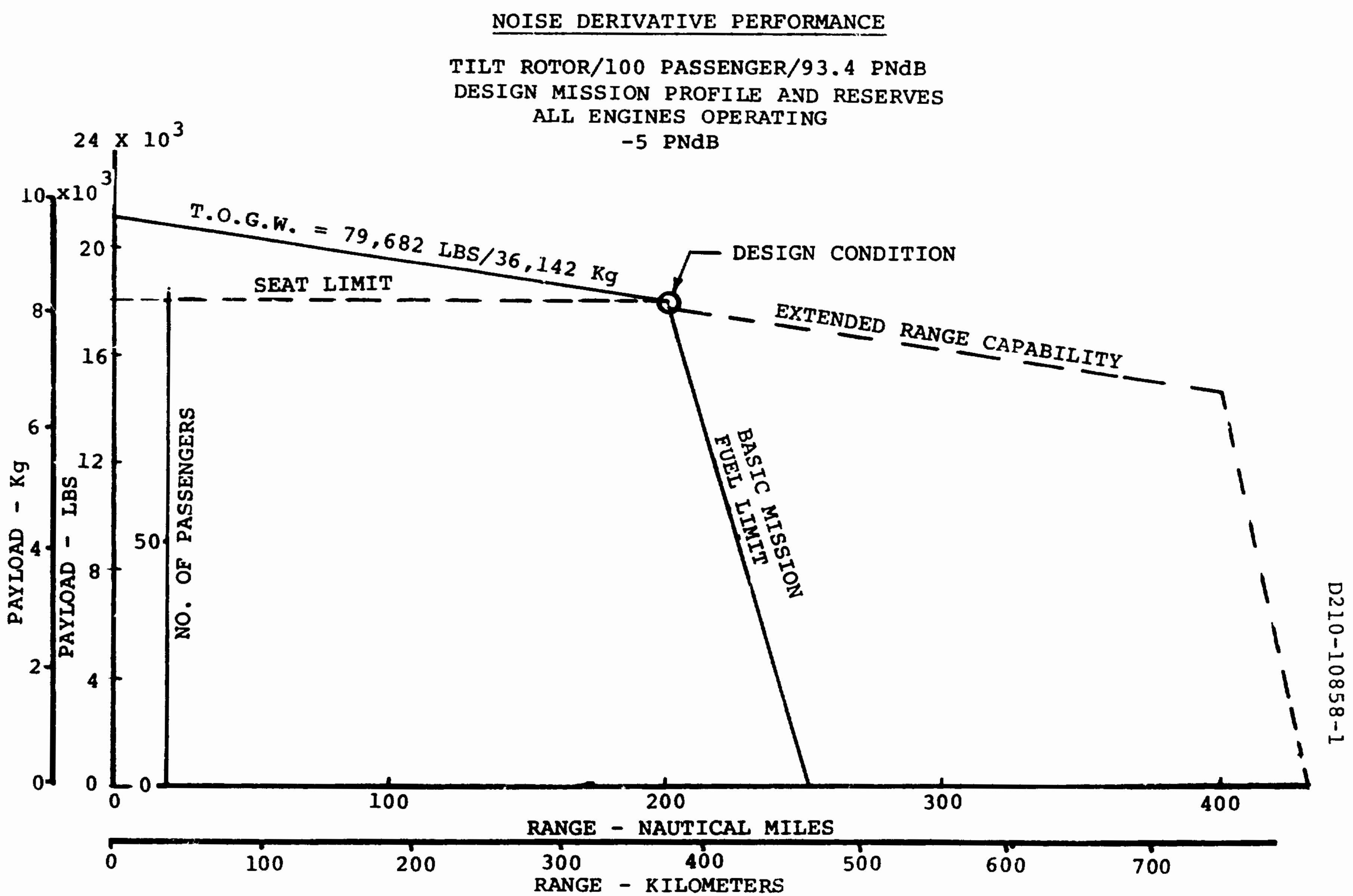


FIGURE 3.83. PAYLOAD RANGE CAPABILITY - CRUISE AT NRP AND CRUISE RPM. -5 PNdB TILT ROTOR.

NOISE DERIVATIVE AIRCRAFT

TILT ROTOR/100 PASSENGER/93.4 PNdB

DESIGN MISSION PROFILE AND RESERVES

ONE ENGINE INOPERATIVE FOR CRUISE AND RESERVE SEGMENTS

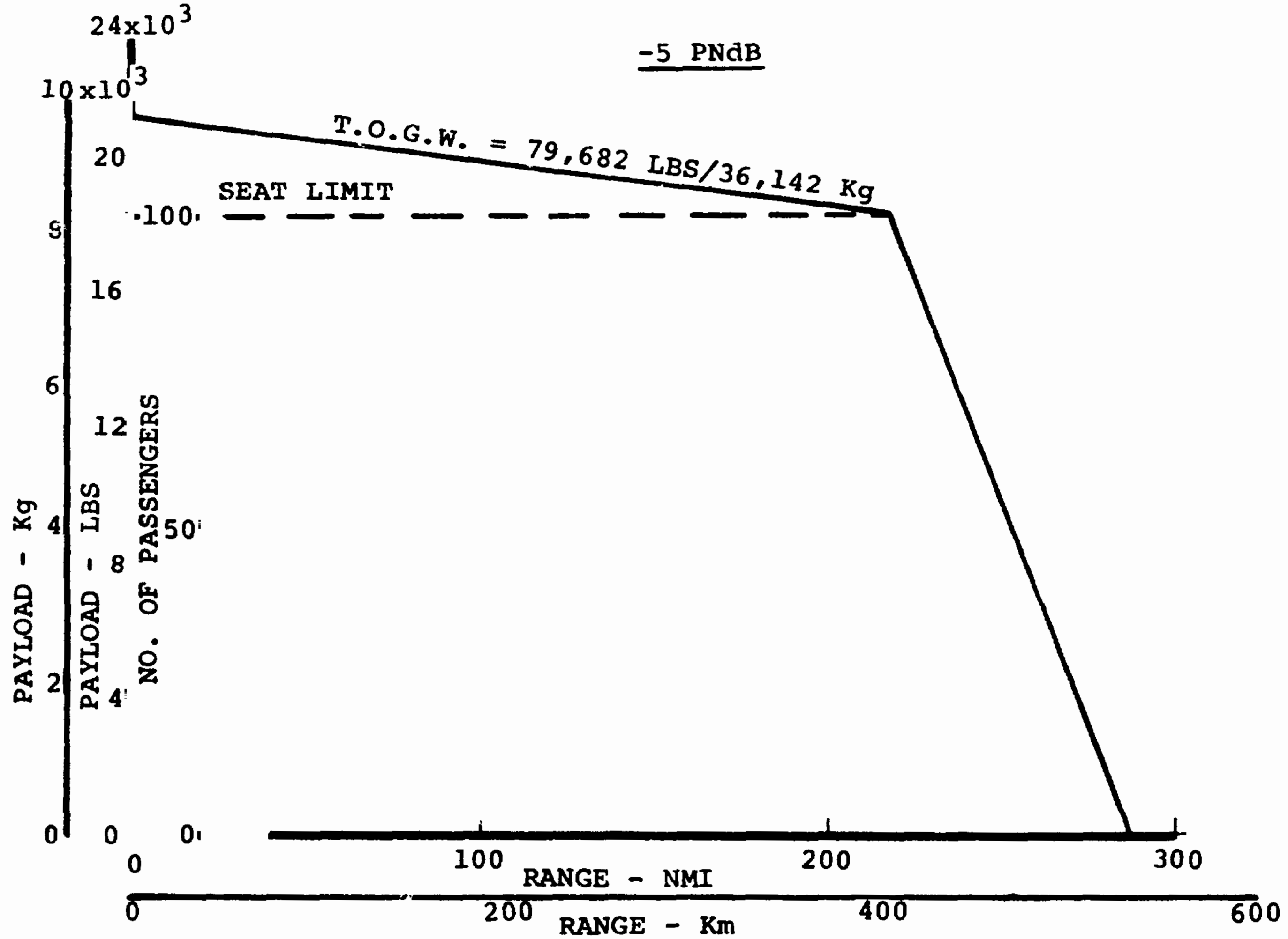


FIGURE 3.84. PAYLOAD RANGE CAPABILITY CRUISE AT NRP & CRUISE RPM. (OEI).

the weight moments due to CG excursion. The cyclic pitch to trim is given in Figure 3.85 as a function of CG position. With an adverse CG and nacelle incidence of 90 degrees the aircraft requires 2.35 degrees cyclic pitch to trim in hover.

This cyclic to trim reduces the pitch control power available as shown in Figure 3.85. However, at the most adverse CG an initial pitch acceleration of 0.41 radians per second square can be obtained. At a nominal mean CG position 0.65 radians per second square of pitch control is available.

The yaw and roll control powers in hover are shown in Figure 3.86. Yaw control is more critical at lighter weights since the reduced thrust vector produces a smaller yaw couple for the same cyclic pitch. The reduction in RPM reduces the thrust per degree of collective pitch and results in a lower roll control sensitivity for the quiet tilt rotor compared with the baseline aircraft. The guideline requirement of 0.6 radians per second square can be met with 5 degrees of differential collective pitch.

Longitudinal static stability in cruise is shown in Figure 3.87. This plot shows excursions in neutral point in terms of percent MAC for various airspeeds and tail sizes. A horizontal tail volume ratio of 1.31 is required to provide a 5% static margin at 140 knots at the 34% MAC CG limit.

The lateral-directional derivatives due to sideslip are shown in Figure 3.88. C_{n_β} is positive and stable over the cruise

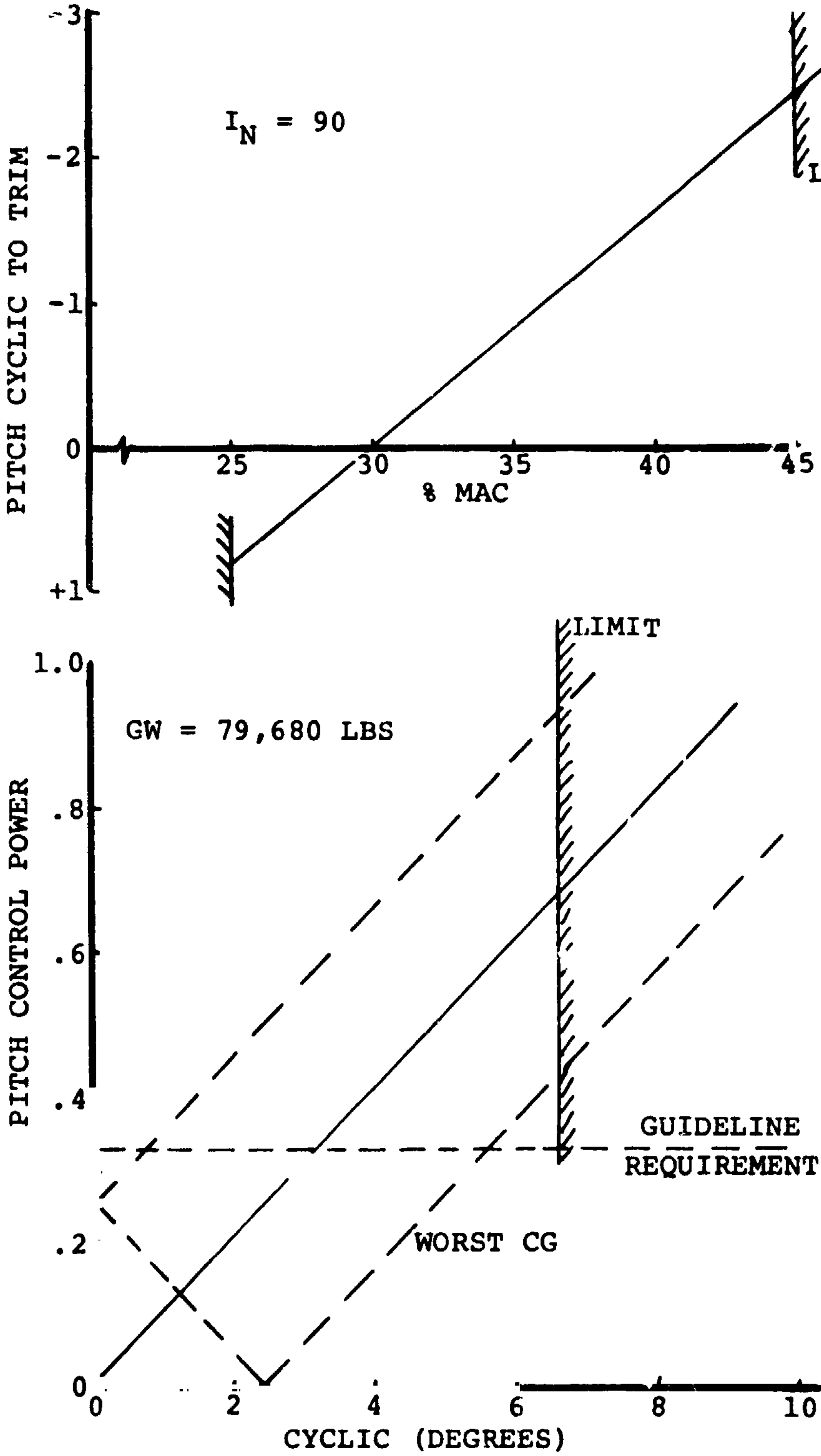


FIGURE 3.85. -5 PNdB HOVER PITCH CONTROL.

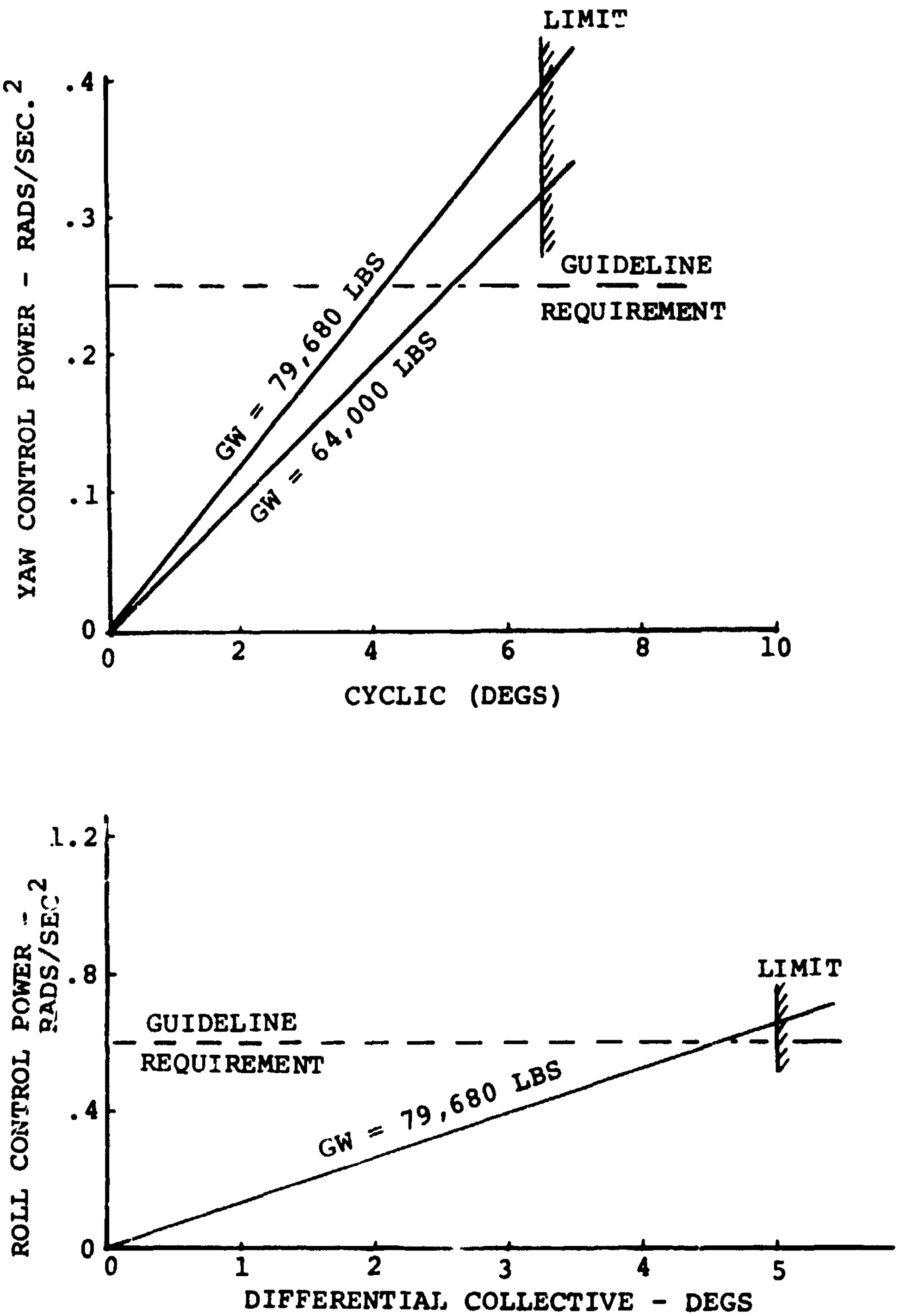


FIGURE 3.86 . HOVER CONTROL POWER - ROLL AND YAW - -5 PNdB TILT ROTOR.

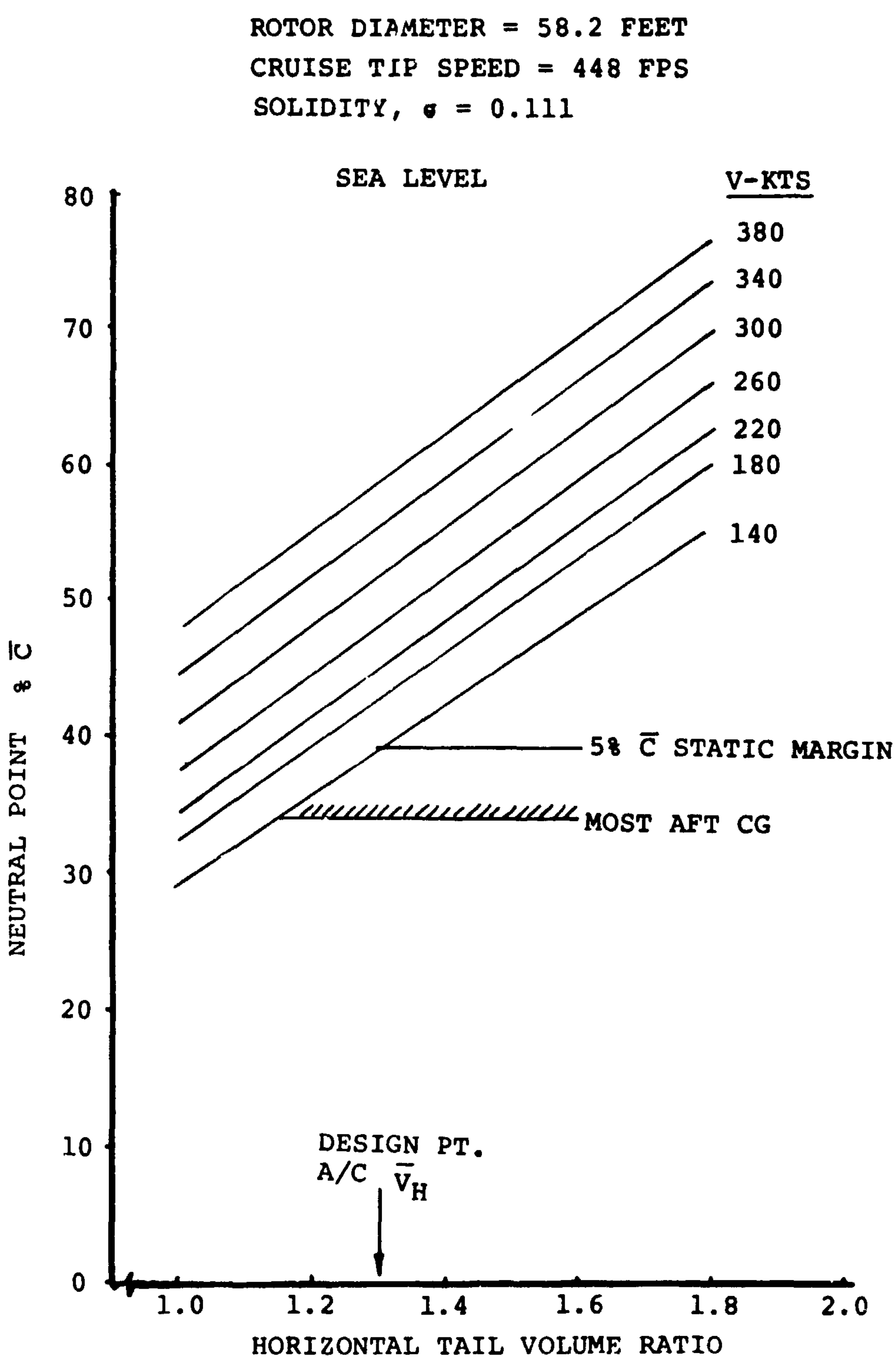


FIGURE 3.87. -5 PNdB TILT ROTOR DESIGN LONGITUDINAL STATIC STABILITY.

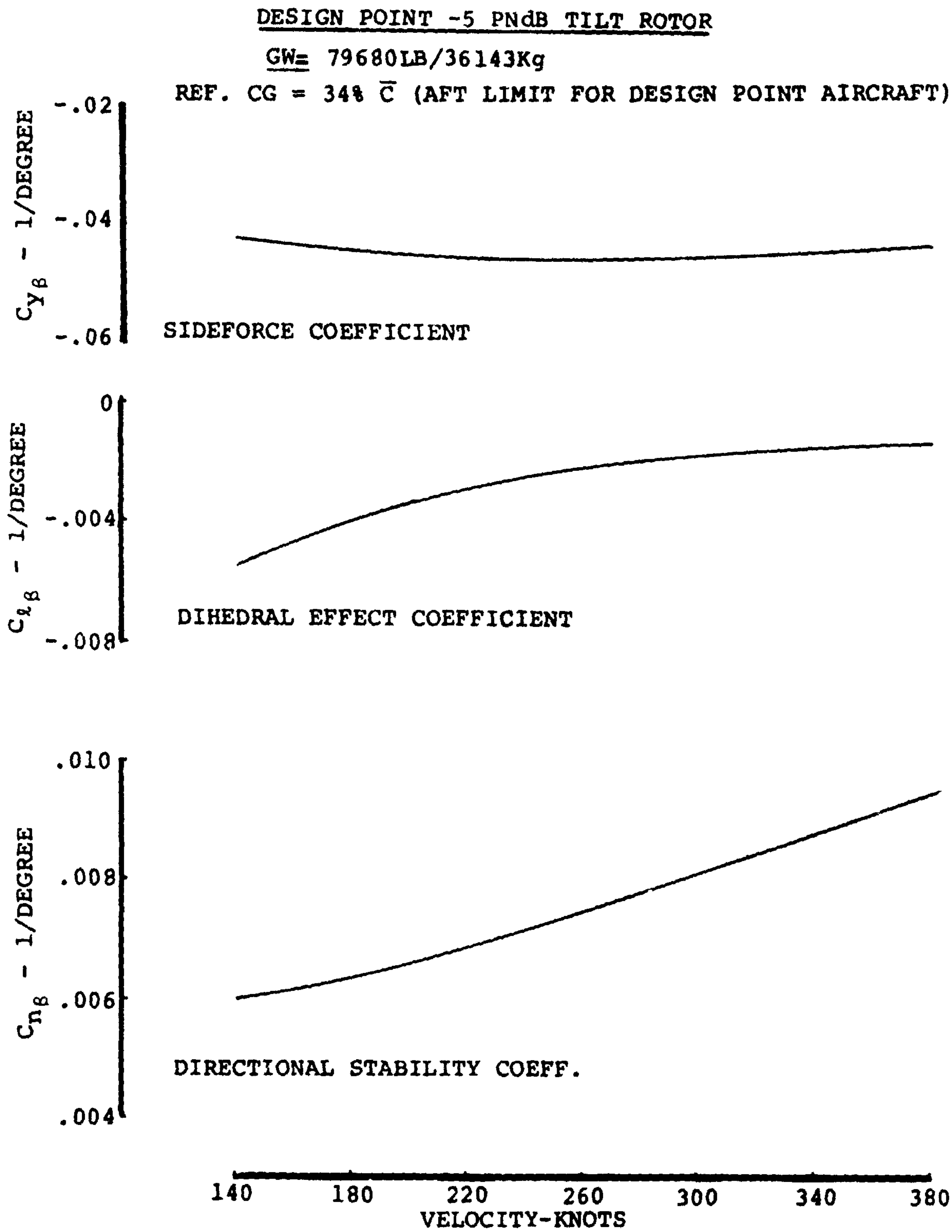


FIGURE 3.88. LATERAL-DIRECTIONAL STABILITY DERIVATIVE COEFFICIENTS DUE TO SIDESLIP - -5 PNdB TILT ROTOR.

range. The sideforce coefficient C_{y_β} is still large and would require a significant roll angle to trim a straight-ground track flight path in a sideslip as was the case for the baseline tilt rotor.

The derivatives due to roll and yaw rate are shown in Figures 3.89 and 3.90 and indicate high damping derivatives in the lateral modes.

Tilt Rotor Design - TR-100 (93.4) - Noise

The sound pressure level spectrum data for the 500 foot sideline case in hover are shown in Figure 3.91 for the -5 PNdB tilt rotor. The predominant component of noise is still the rotor broadband noise over most of the frequency range. The reduction in RPM decreased the broadband noise by comparison with the baseline vehicle and resulted in the engine inlet noise just becoming predominant at the very high frequencies.

The engine inlet noise suppression assumed in this case is the same as the baseline aircraft.

As before the NOY distribution used in computing the 93.3 PNdB value is also shown in Figure 3.91.

The perceived noise level controus for the -5 PNdB tilt rotor on takeoff are shown in Figure 3.92. The area impacted by a given noise level is much smaller in this case than the baseline aircraft. The takeoff trajectory and perceived noise levels along the flight path are given in Figure 3.93.

Similar data are shown for the landing case in Figures 3.94 and 3.95.

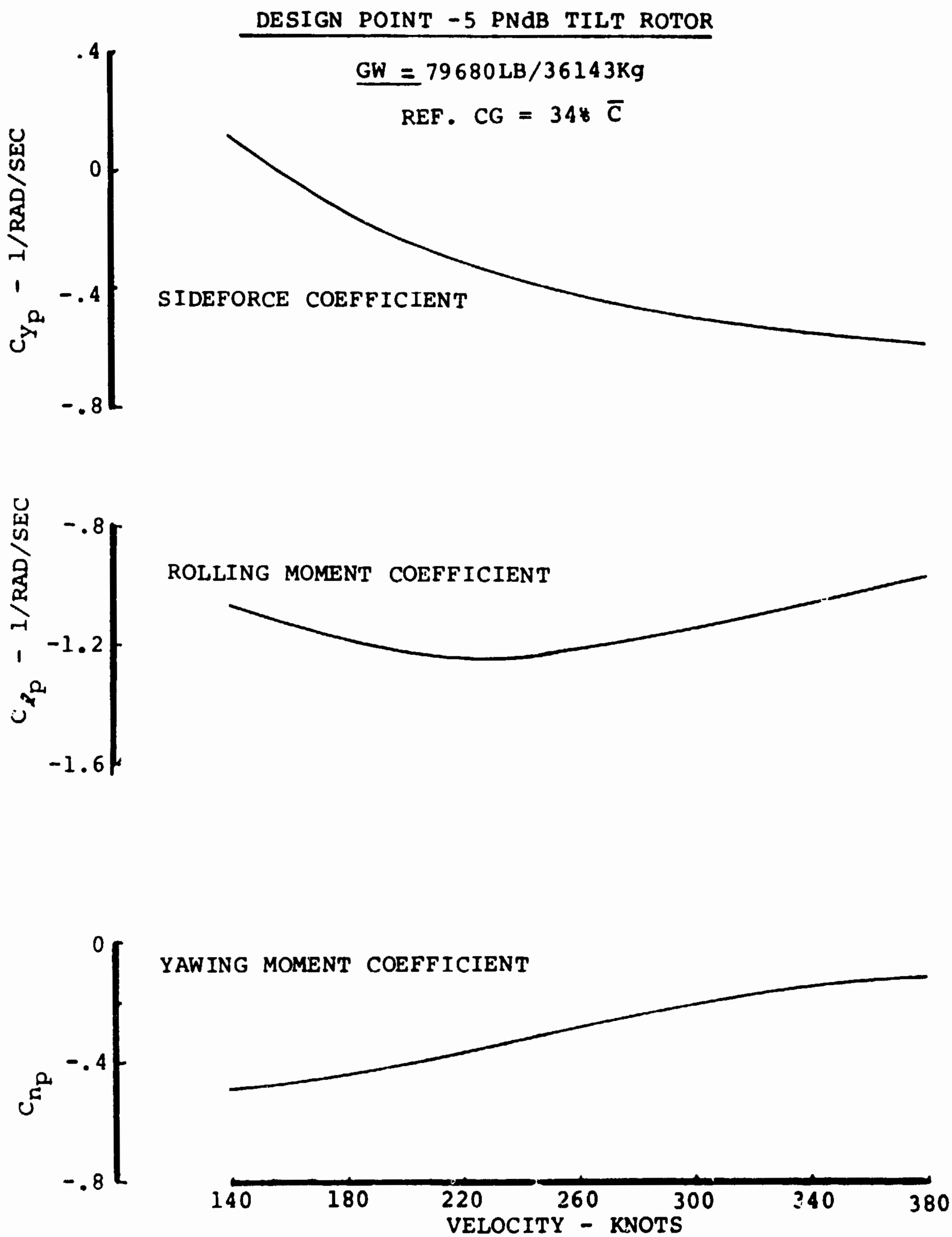


FIGURE 3.89. -5 PNdB TILT ROTOR STABILITY COEFFICIENTS DUE TO ROLL RATE.

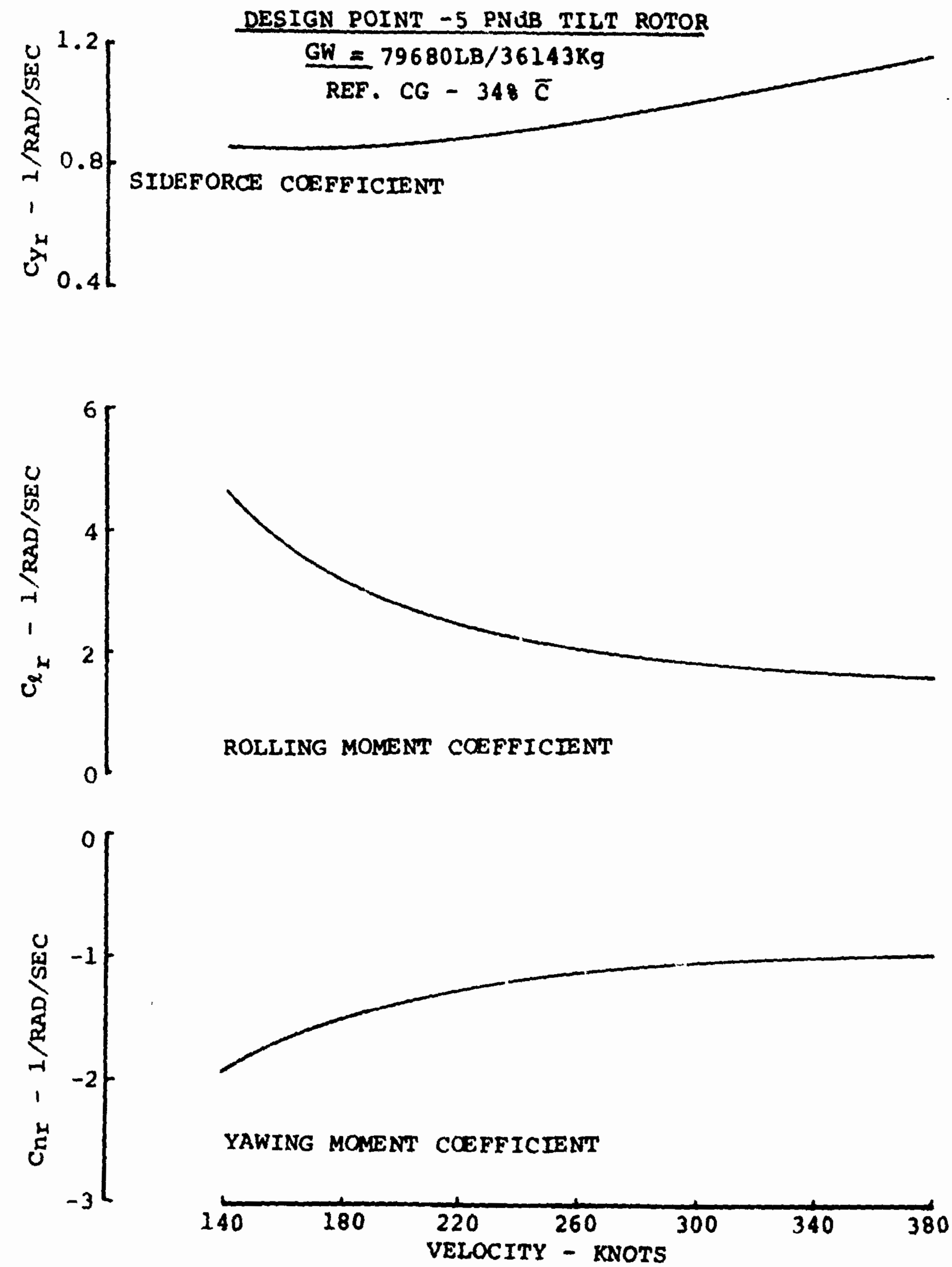


FIGURE 3.90. -5 PNdB TILT ROTOR STABILITY COEFFICIENTS DUE TO YAW RATE.

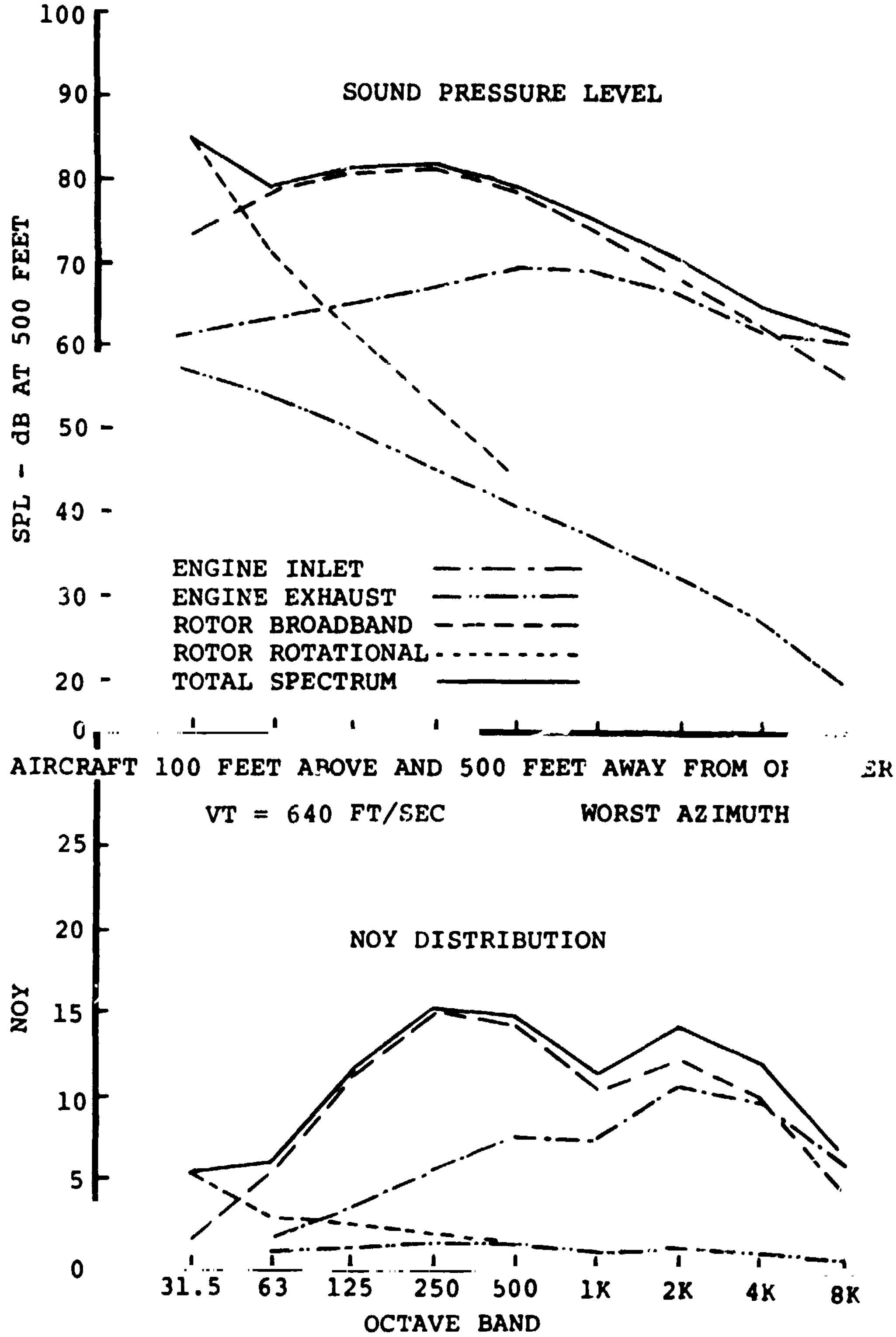


FIGURE 3.91. TILT ROTOR -5 PNdB HOVER NOISE SPECTRUM AND NOY DISTRIBUTION - PNdB = 93.4.

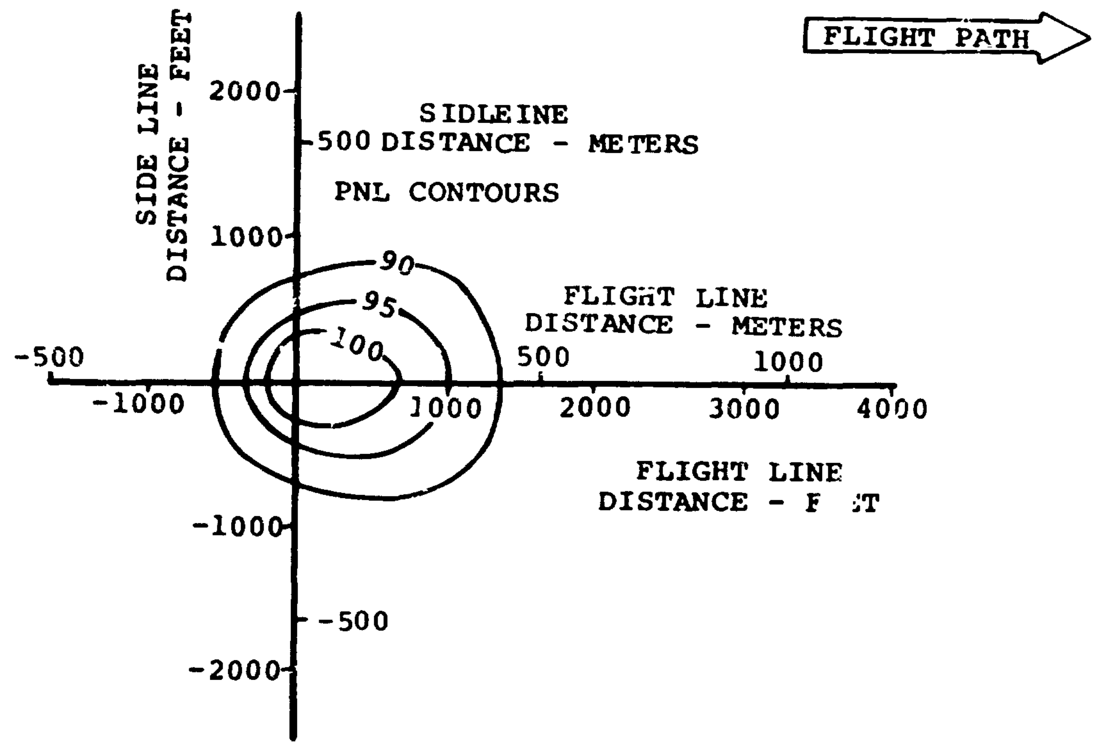


FIGURE 3.92. -5 PNdB TILT ROTOR STANDARD TAKEOFF - PNL CONTOURS.

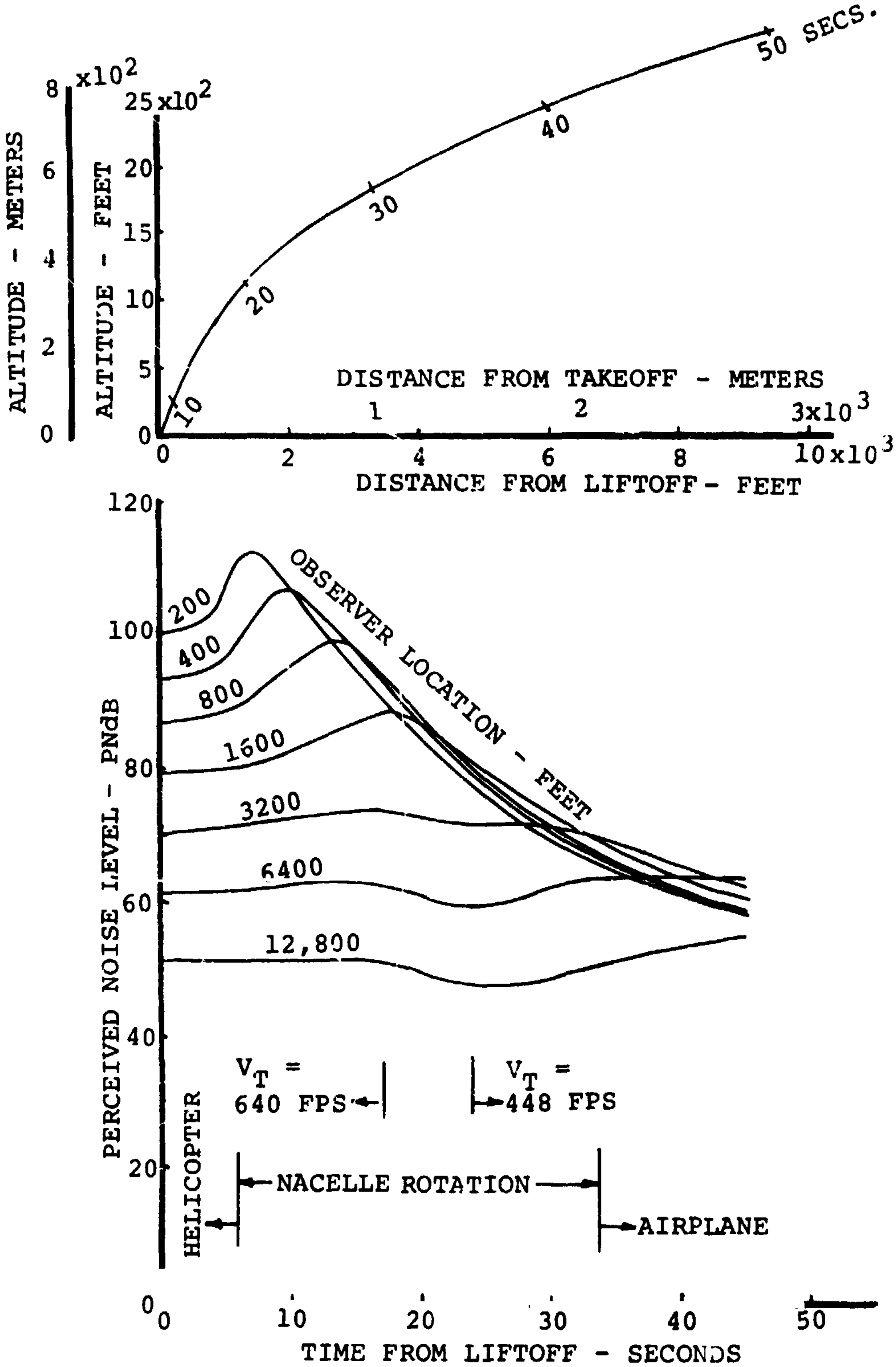


FIGURE 3.93. -5 PNdB TILT ROTOR STANDARD TAKEOFF - PERCEIVED NOISE.

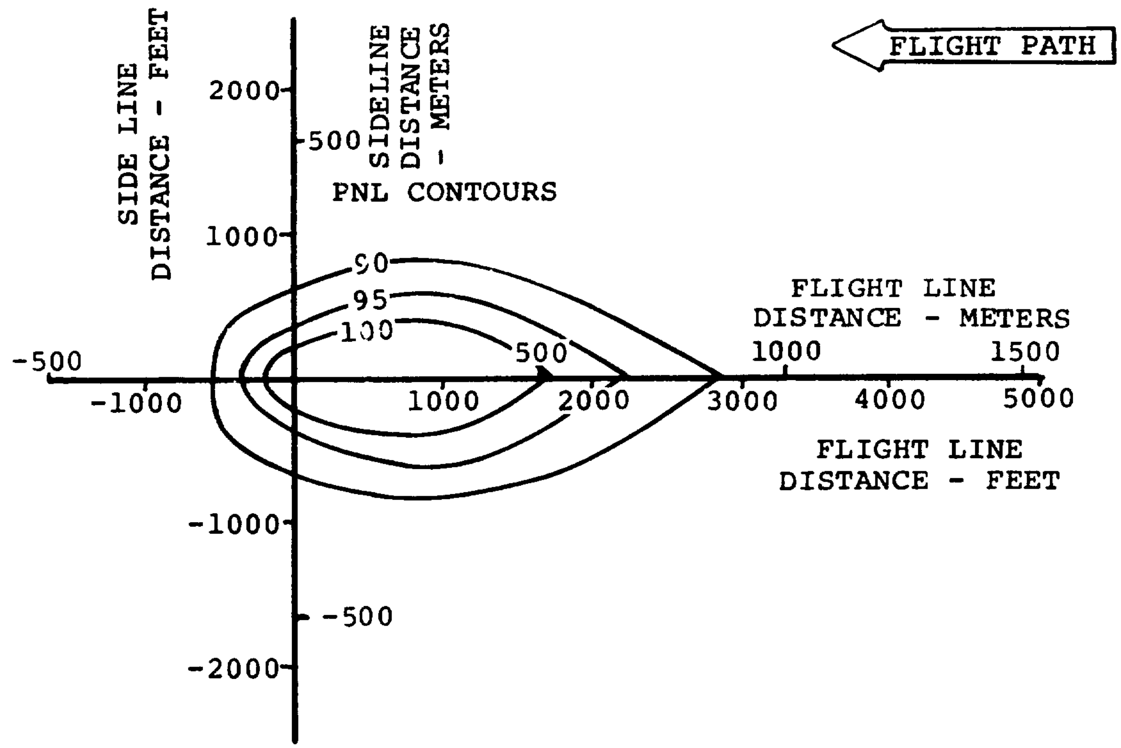


FIGURE 3.94. -5 PNdB TILT ROTOR STANDARD LANDING - PNL CONTOURS.

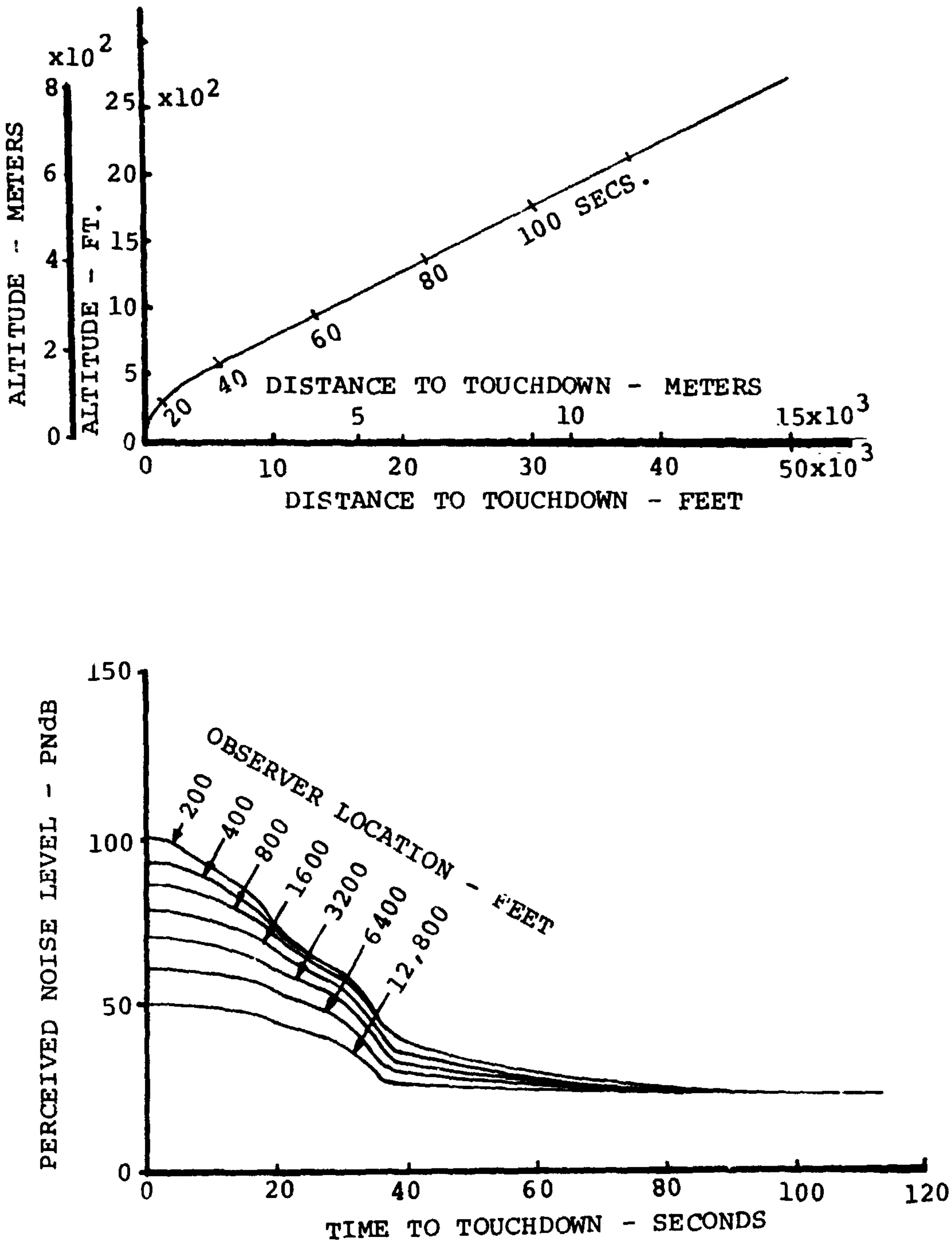


FIGURE 3.95. -5 PNdB TILT ROTOR STANDARD LANDING - PERCEIVED NOISE.

TR-100 (93.4) - Costs

Direct operating costs per seat mile and seat kilometer as a function of block distance are shown in Figure 3.96 for the specified combinations of aircraft utilization and airframe costs. Figure 3.96 also illustrates the impact of extending the design range of the TR-100 (93.4) to 460 statute miles. The increase in costs at the design point range (230 statute miles) is the result of the loss of 1 available seat due to the increased empty weight for the installation of larger fuel tanks. Although not shown in Figure 3.46 it should be noted that the larger fuel tanks will result in a small increase (less than 1%) in seat mile costs at ranges less than 230 statute miles due to increases in airframe maintenance and depreciation costs. In the extended range version of the TR-100 (93.4) seat mile costs show a continuing decline between 230 and 345 statute miles because the loss of available seats due to additional fuel requirements is offset by increasing block speed. Between 345 and 460 statute miles the delta block speeds become insufficient to offset seat losses and the seat mile costs begin to rise.

Table 3.29 shows the flyaway costs for the basic TR-100 (93.4) at $90.00 and $110.00 per pound of airframe. A breakout of the direct operating cost factors for the TR-100 (93.4) at 230 statute miles is shown in Table 3.29. Flyaway and direct operating cost breakouts for the extended range version of the TR-100 (93.4) are shown in Table 3.30.

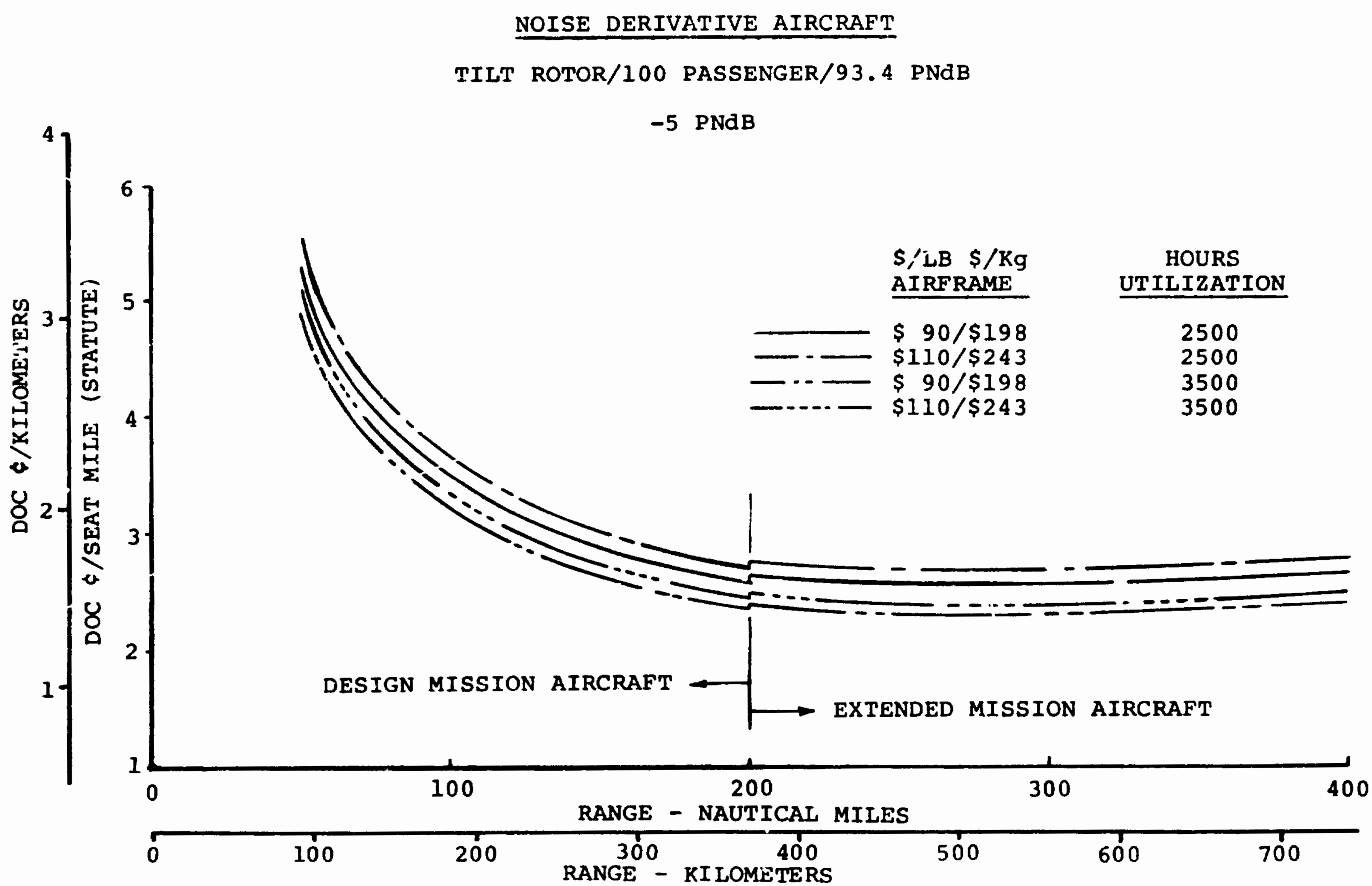


FIGURE 3.96. EFFECT OF OPERATING RANGE ON DIRECT OPERATING COST - -5 PNdB TILT ROTOR.

Flyaway Costs

Airframe Cost	$90.00/Lb	$110.00/Lb
Airframe	$3,279,780	$4,008,620
Dynamic System	1,196,320	1,196,320
Engines	878,736	878,736
Avionics	250,000	250,000
Total	$5,604,836	$6,333,676

Direct Operating Costs
Dollars/Seat Mile
Block Distance = 230 St. Miles

Utilization (Hrs/Yr)	2500		3500	
Airframe Cost ($/Lb)	90	110	90	110
Flying Operations				
Flight Crew	.0043	.0043	.0043	.0043
Fuel and Oil	.0037	.0037	.0037	.0037
Hull Insurance	.0014	.0016	.0010	.0011
Total Flying Operations	.0094	.0096	.0090	.0091
Direct Maintenance				
Airframe - Labor	.0014	.0014	.0014	.0014
- Material	.0012	.0014	.0012	.0014
Engines - Labor	.0006	.0006	.0006	.0006
- Material	.0009	.0009	.0009	.0009
Dynamic System - Labor	.0007	.0007	.0007	.0007
- Material	.0010	.0010	.0010	.0010
Total Direct Maintenance	.0058	.0060	.0058	.0060
Maintenance Burden	.0040	.0040	.0040	.0040
Total Maintenance	.0098	.0100	.0098	.0100
Depreciation	.0068	.0076	.0048	.0054
Total Direct Costs	.0260	.0272	.0236	.0245

TABLE 3.29. INITIAL AND DIRECT OPERATING COSTS - -5 PNdB TILT ROTOR AIRCRAFT.

TR-100(93.4)
EXTENDED RANGE VERSION

Flyaway Costs

Airframe Cost	$90.00/Lb	$110.00/Lb
Airframe	$3,295,980	$4,028,420
Dynamic System	1,196,320	1,196,320
Engines	878,736	878,736
Avionics	250,000	250,000
Total	$5,621,036	$6,353,476

Direct Operating Costs
Dollars/Seat Mile
Block Distance = 230 St. Miles

Utilization (Hrs/Yr)	2500		3500	
Airframe Cost ($/Lb)	90	110	90	110
Flying Operations				
Flight Crew	.0044	.0044	.0044	.0044
Fuel and Oil	.0037	.0037	.0037	.0037
Hull Insurance	.0014	.0016	.0010	.0011
Total Flying Operations	.0095	.0097	.0091	.0092
Direct Maintenance				
Airframe - Labor	.0014	.0014	.0014	.0014
- Material	.0012	.0014	.0012	.0014
Engines - Labor	.0007	.0007	.0007	.0007
- Material	.0009	.0009	.0009	.0009
Dynamic System - Labor	.0007	.0007	.0007	.0007
- Material	.0010	.0010	.0010	.0010
Total Direct Maintenance	.0059	.0061	.0059	.0061
Maintenance Burden	.0040	.0040	.0040	.0040
Total Maintenance	.0099	.0101	.0099	.0101
Depreciation	.0068	.0077	.0049	.0055
Total Direct Costs	.0262	.0275	.0239	.0248

TABLE 3.30. INITIAL AND DIRECT OPERATING COSTS - -5 PNdB TILT ROTOR (EXTENDED RANGE VERSION).

4.0 DESIGN DATA COMPARISONS

The results of the design studies conducted for the tandem helicopter and tilt rotor configurations applied to the specified mission allow two major comparisons to be made. The first is the effect of external noise design criteria on the design parameters of each configuration and the second is a comparative evaluation of the two configurations themselves.

The basic weight data for all six designs are shown in Table 4.1.

The effect of external noise design criteria on both configurations is to increase aircraft design gross weight as the sideline noise level decreases. The helicopter designs are between 5,000 and 8,000 pounds lighter than the tilt rotors. The helicopter is 6 PNdB less noisy at 500 foot sideline distance in hover, however, its noise is felt over a much wider area and longer time than is the case in the tilt rotor. Fuel consumption of the tilt rotors is approximately 65% of the helicopter usage.

The data shown are inconsistent in one aspect. The design maneuver load factor from FAR-25 specifies 3.5 for the tandem helicopter and FAR-29 specifies 2.5 for the tilt rotor. This effect favors the tilt rotor designs in terms of weight comparisons. Figure 4.1 shows the effect of maneuver load factor on the design gross weight of the helicopter to provide a more accurate representation of the weight comparison. At

TABLE 4.1. COMPARISON OF AIRCRAFT WEIGHTS - (U.S. UNITS).

CONFIGURATION	-5 PNdB	BASELINE	+5 PNdB
TANDEM HELICOPTER			
DESIGN GROSS WEIGHT	74,227 LBS	67,175 LBS	65,843 LBS
WEIGHT EMPTY	46,533 LBS	40,181 LBS	38,152 LBS
FUEL WEIGHT	7,708 LBS	7,007 LBS	7,705 LBS
500 FOOT SIDELINE NOISE - PNdB	87.3	92.3	97.3
TILT ROTOR			
DESIGN GROSS WEIGHT	79,682 LBS	74,749 LBS	73,217 LBS
WEIGHT EMPTY	54,718 LBS	50,068 LBS	48,757 LBS
FUEL WEIGHT	4,939 LBS	4,656 LBS	4,436 LBS
500 FOOT SIDELINE NOISE - PNdB	93.2	98.2	103.2

TABLE 4.1 CONTINUED COMPARISON OF AIRCRAFT WEIGHTS - (S.I. UNITS).

CONFIGURATION	-5 PNdB	BASELINE	+5 PNdB
TANDEM HELICOPTER			
DESIGN GROSS WEIGHT	33,669 Kg	30,471 Kg	29,866 Kg
WEIGHT EMPTY	21,107 Kg	18,226 Kg	17,306 Kg
FUEL WEIGHT	3,496 Kg	3,178 Kg	3,495 Kg
500 FOOT SIDELINE NOISE - PNdB			
TILT ROTOR			
DESIGN GROSS WEIGHT	36,144 Kg	33,906 Kg	33,211 Kg
WEIGHT EMPTY	24,820 Kg	22,711 Kg	22,116 Kg
FUEL WEIGHT	2,240 Kg	2,112 Kg	2,012 Kg
500 FOOT SIDELINE NOIDE - PNdB	93.2	98.2	103.2

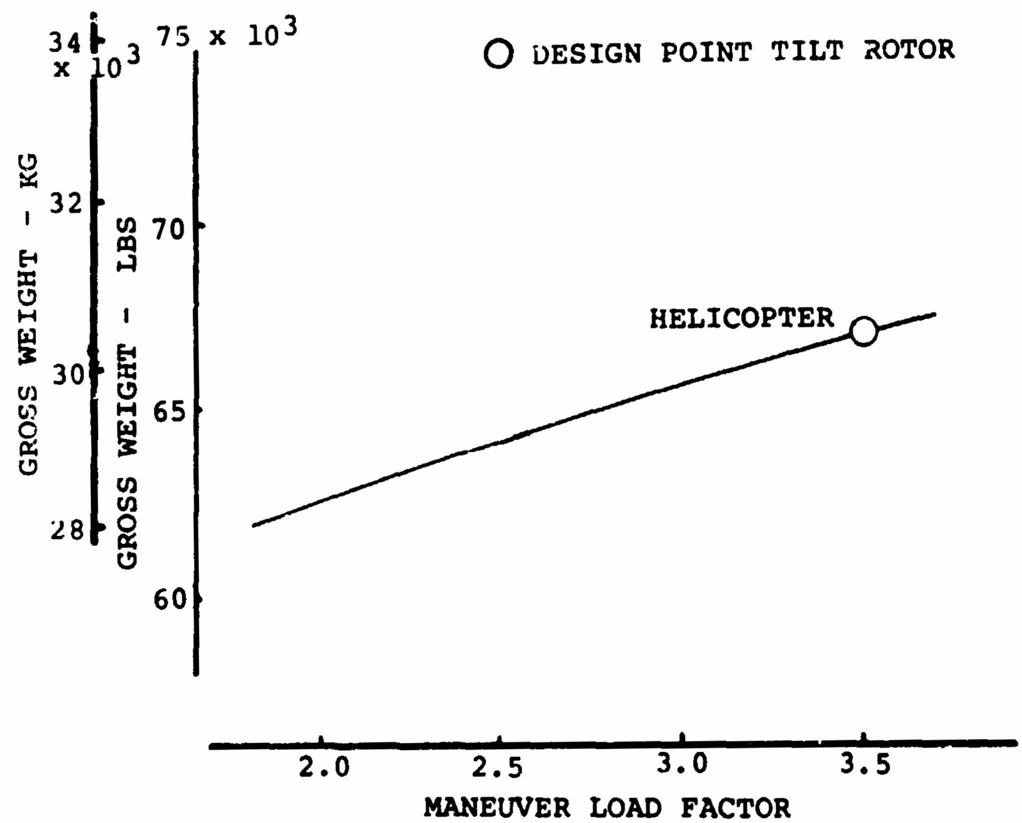


FIGURE 4.1 . EFFECT OF DESIGN MANEUVER LOAD FACTOR ON BASELINE HELICOPTER SIZE.

the same load factor weight of the baseline helicopter would be 64,200 pounds gross weight, 10,549 pounds lighter than the design point tilt rotor. The fuel weight is still larger than for the tilt rotor (6,715 pounds at maneuver load factor of 2.5).

Figure 4.2 shows the design gross weight of the aircraft as a function of PNL at 500 foot distance in hover. The minimum weight helicopter occurs at 98.4 PNdB. The tilt rotor design study indicates that minimum weight is achieved at higher noise levels - approximately 103 PNdB.

The impact of the requirement for an external noise reduction on aircraft gross weight is quite severe. For 5 PNdB reduction the baseline helicopter increases in weight by 7,052 pounds. The tilt rotor takes a 4,933 pound penalty for a 5 PNdB reduction. Comparisons of aircraft installed power and rotor diameter are shown in Figure 4.3.

These variations result from rotor noise criteria since the engine inlet noise has been attenuated. The basic design point disc loadings were retained at 9 pounds/square feet and 15 pounds/square feet for the helicopter and tilt rotor. Noise variations were obtained by changing tip speed and solidity. Comparison of tip speed, solidity and cruise speed at normal rated power associated with the different noise levels in the two configurations are given in Figure 4.4. The impact of noise criteria on parasite drag is given in Figure 4.5.

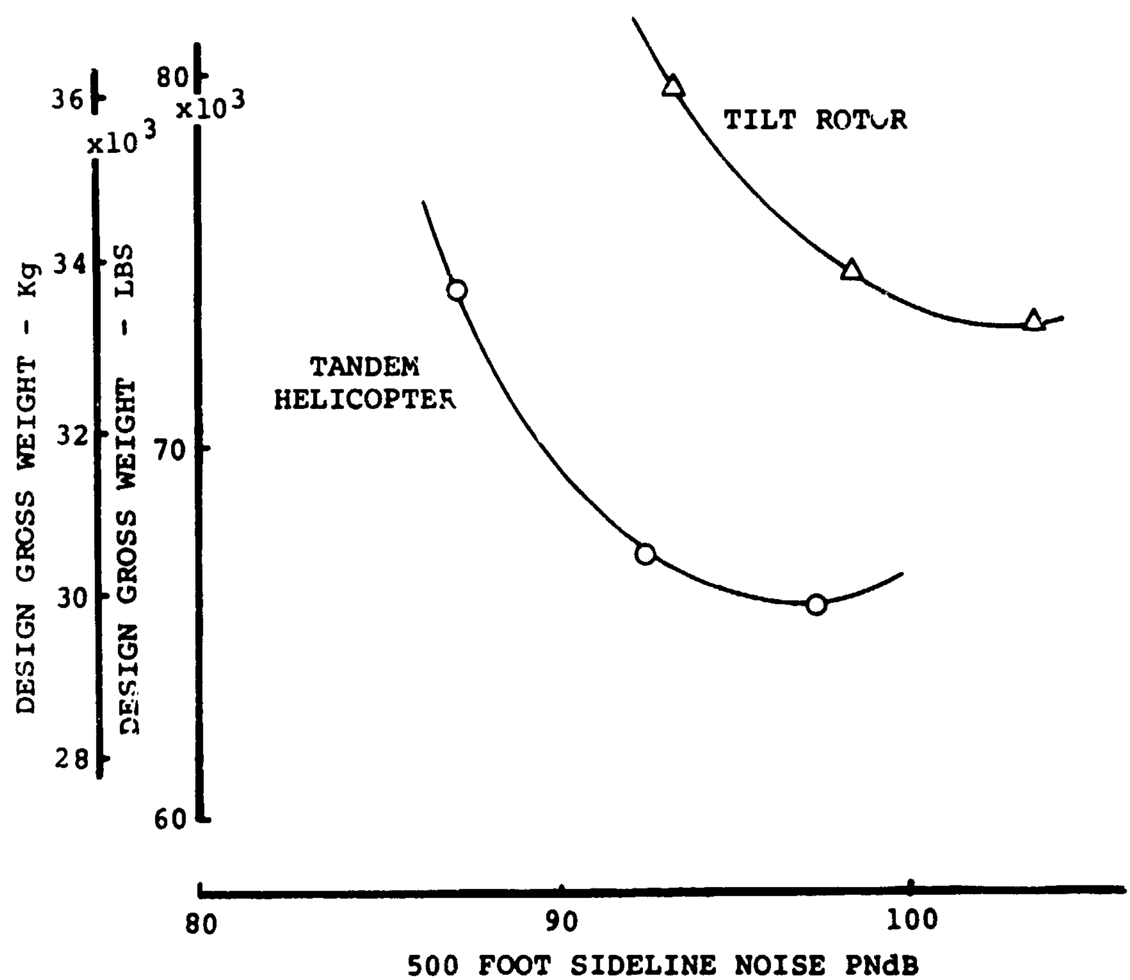


FIGURE 4.2 . THE EFFECT OF EXTERNAL NOISE DESIGN CRITERIA ON AIRCRAFT DESIGN GROSS WEIGHT.

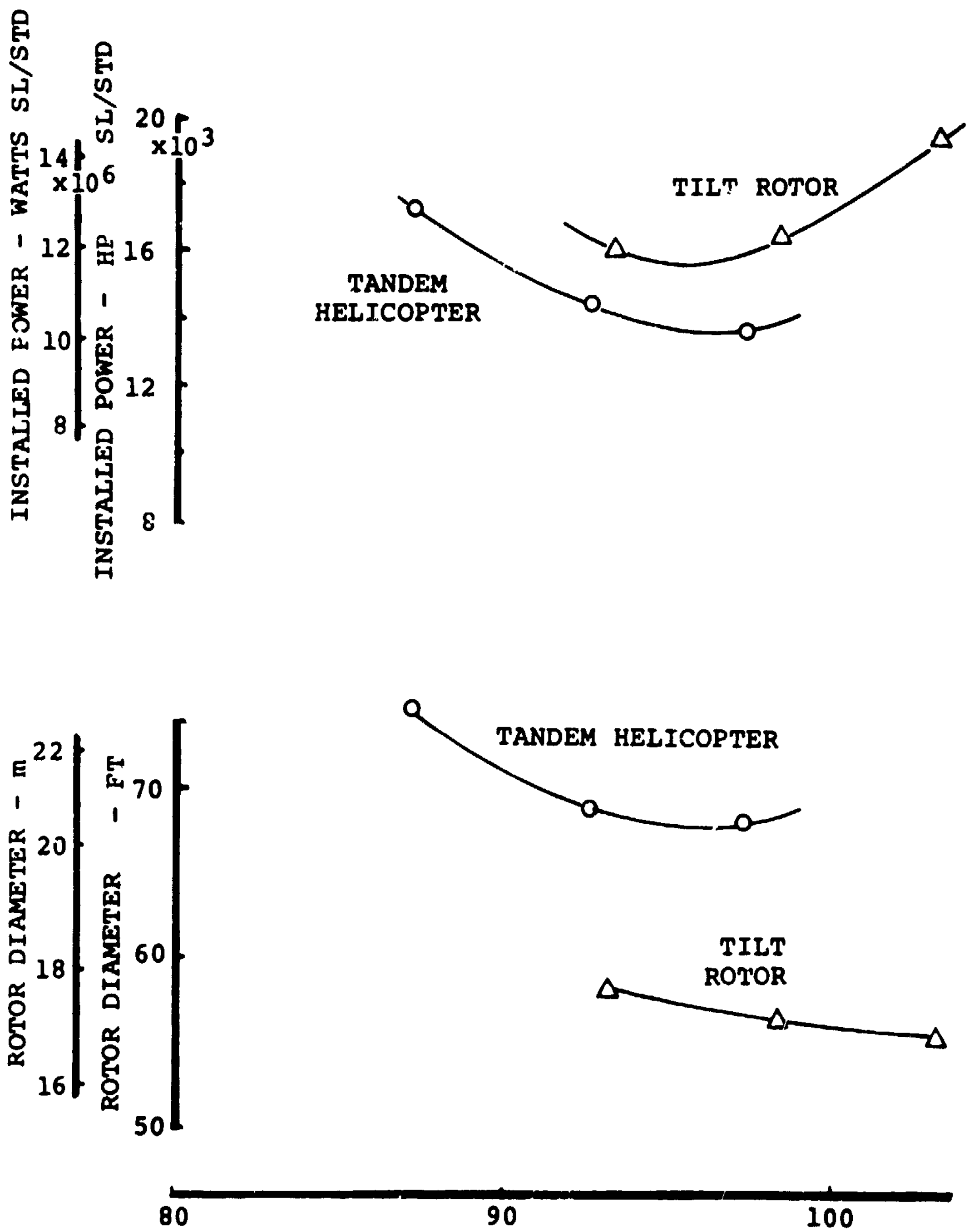


FIGURE 4.3 . THE EFFECT OF EXTERNAL NOISE DESIGN CRITERIA ON ROTOR DIAMETER AND TOTAL INSTALLED POWER.

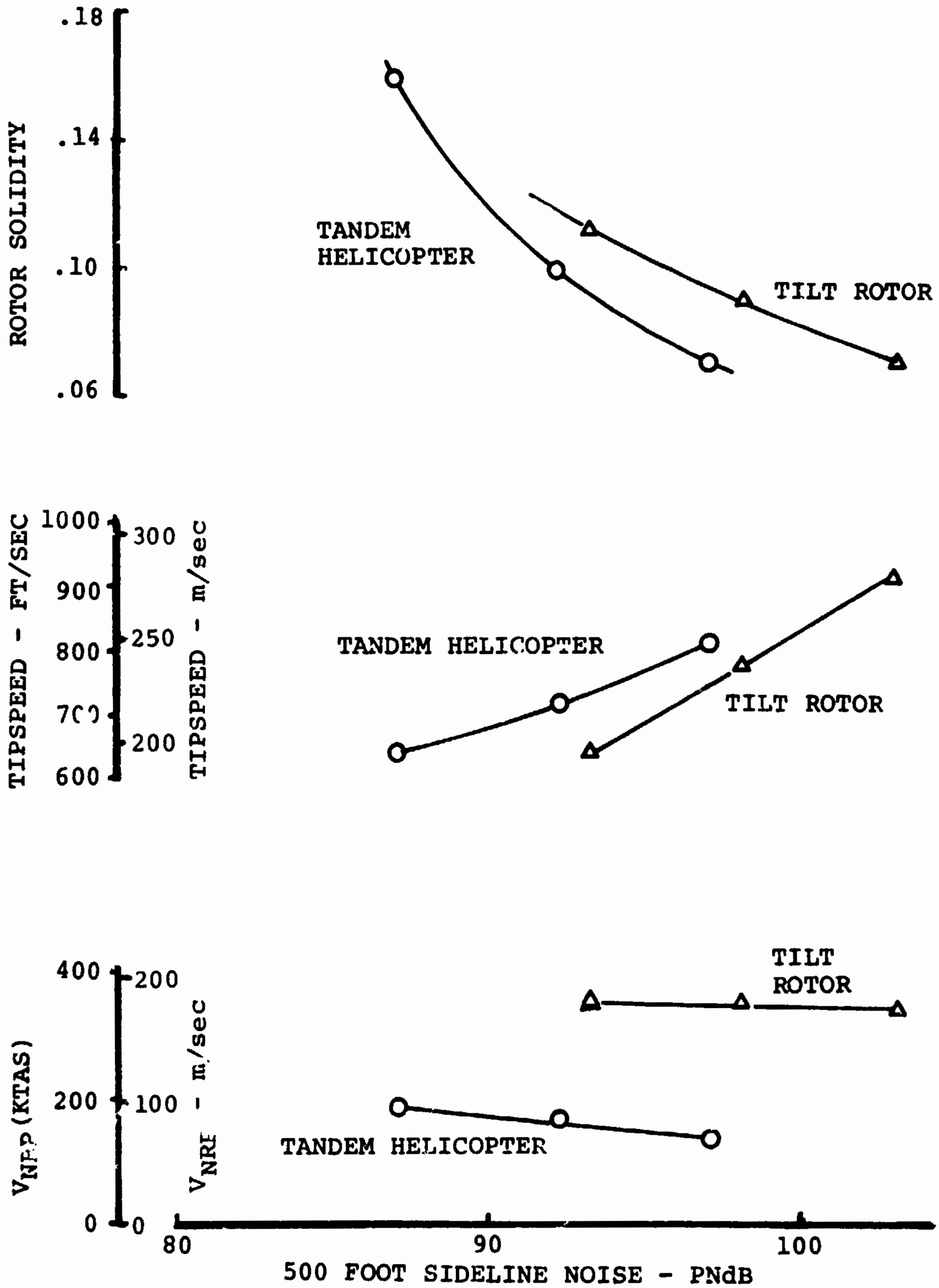


FIGURE 4.4 . EFFECT OF EXTERNAL NOISE CRITERIA ON DESIGN PARAMETERS.

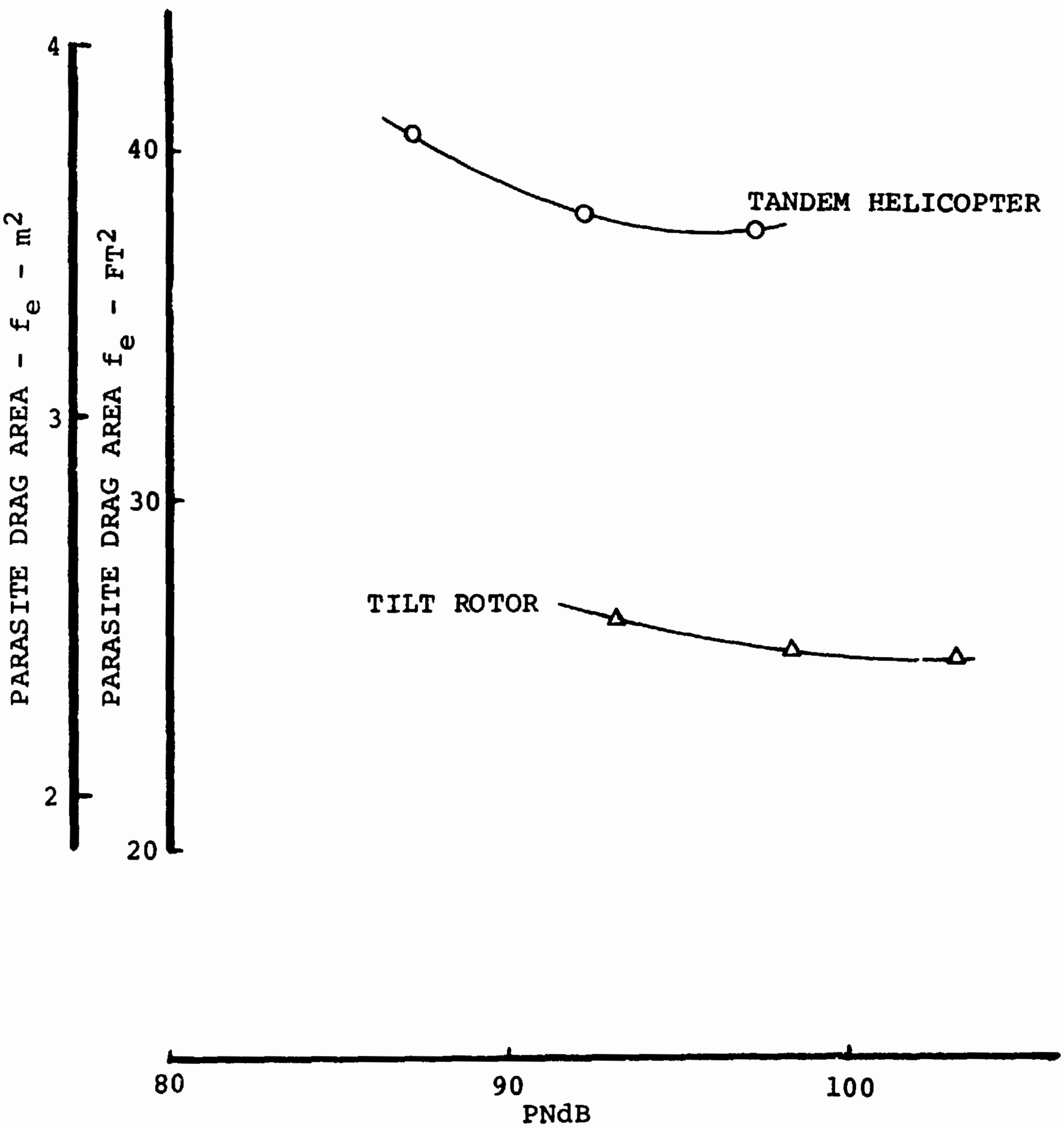


FIGURE 4.5 . EFFECT OF EXTERNAL NOISE CRITERIA ON PARASITE DRAG AREA f_e.

The variations in gross weight determine to a large extent the initial cost of the aircraft. The effect of external noise design criteria on the aircraft initial cost in shown in Figure 4.6 for both configurations with airframe costs estimated at both $90 per pound and $110 per pound.

The result of constraining the aircraft to a given perceived noise level is a more expensive aircraft in terms of initial cost. The tandem helicopter designs have a lower initial cost than the tilt rotor (by virtue of lower design gross weights).

This advantage for the helicopter is lost, however, in the Direct Operating Cost (DOC) because of the speed and fuel consumption advantages of the tilt rotor. The DOC of both configurations are shown in Figure 4.7 for both 2500 and 3500 hours per year utilization.

The minimum DOC for tilt rotors is 2.18 cents per seat mile.

The tandem helicopter has a minimum DOC of 3.20 cents per seat mile. This minimum DOC requirement defines a design point tilt rotor which is about 6 PNdB more noisy than the helicopter. However, the tilt rotor noise can be reduced to the same level as the design point helicopter with only a slight degradation in DOC. This is illustrated by Figure 4.7 which indicates that a tilt rotor designed to 92 PNdB still shows a very significant advantage. A further advantage of the tilt rotor is that its noise area is significantly smaller than the helicopter. This is discussed in more detail in a

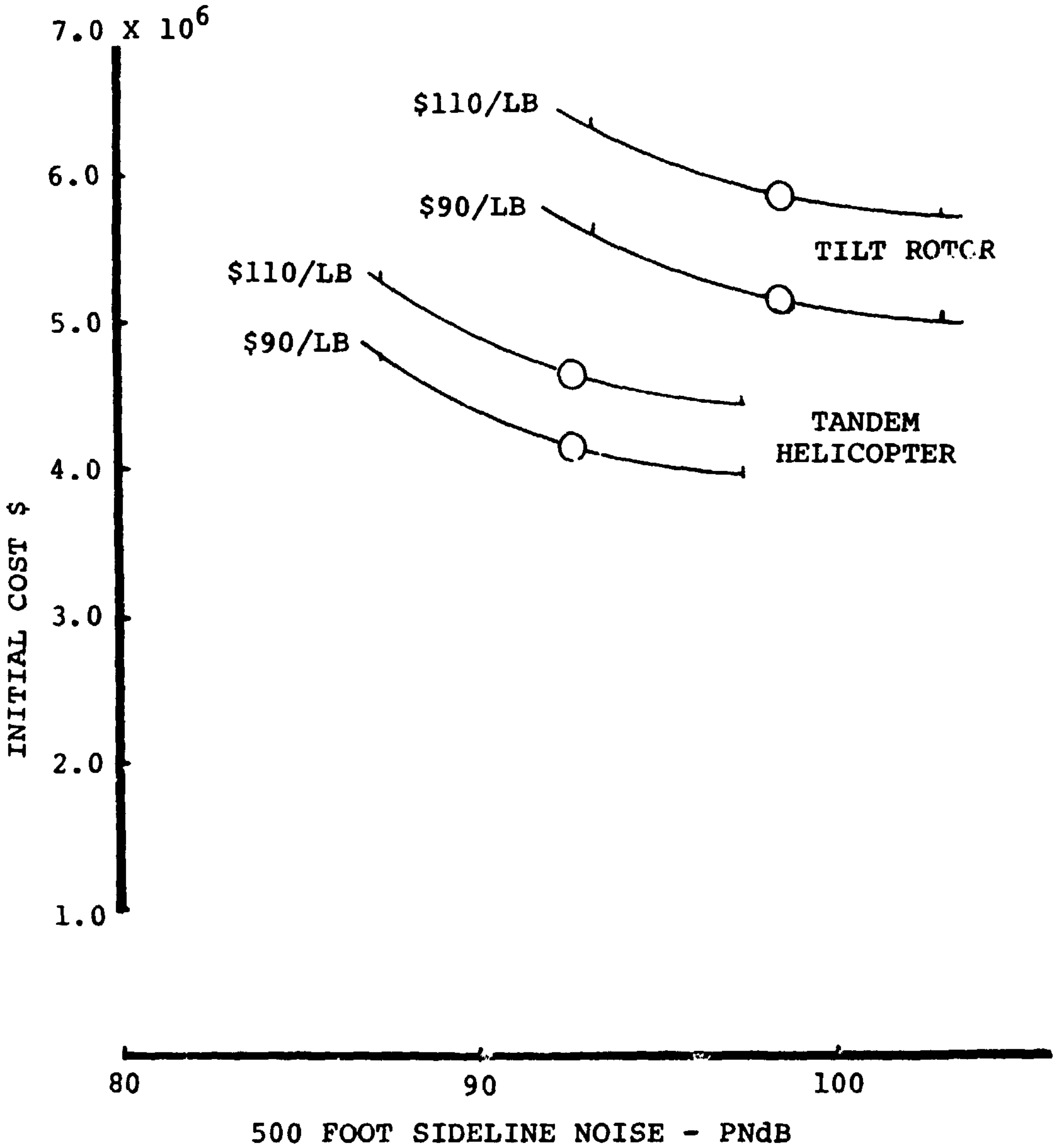


FIGURE 4.6 . EFFECT OF EXTERNAL NOISE DESIGN CRITERIA ON INITIAL COST.

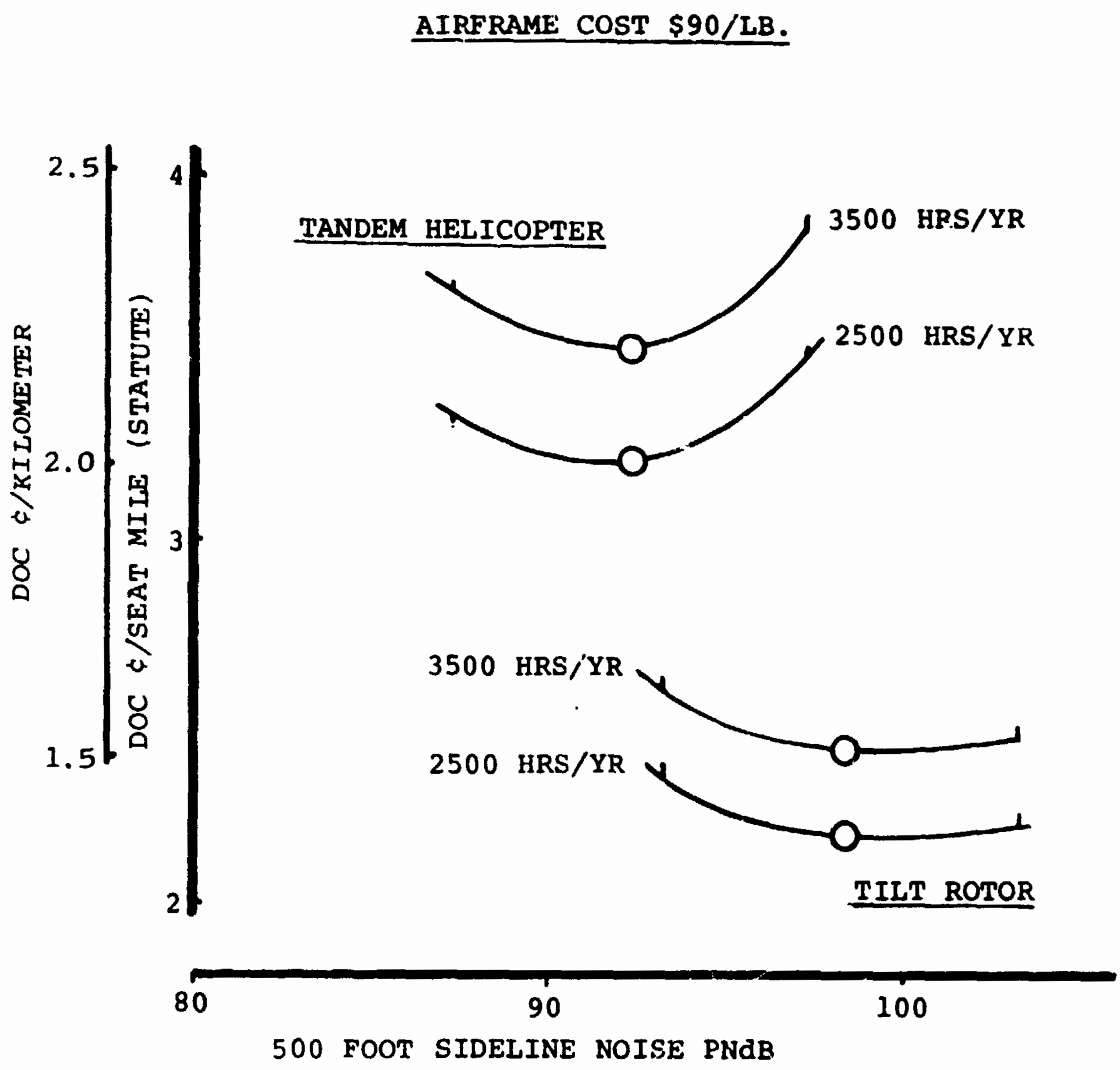


FIGURE 4.7 . EFFECT OF EXTERNAL NOISE CRITERIA ON DOC AT 230 STATUTE MILES RANGE.

later paragraph.

A comparison of direct operating cost as a function of range is given for the baseline aircraft in Figure 4.8.

The major cause of reduced DOC for the tilt rotors is the advantage of a higher block speed. The baseline tilt rotor and helicopter block speeds are shown as a function of range in Figure 4.9. At the design range of 230 statute miles the tilt rotor has a 317 miles per hour block speed compared with 169 miles per hour for the helicopter.

The productivity ratio of the two baseline configurations is shown in Figure 4.10 as a function of range. At the design range the tilt rotor performs 53% better than the tandem helicopter. At short ranges there is still an advantage, for example, at 50 miles the productivity ratio is still 18.5% better.

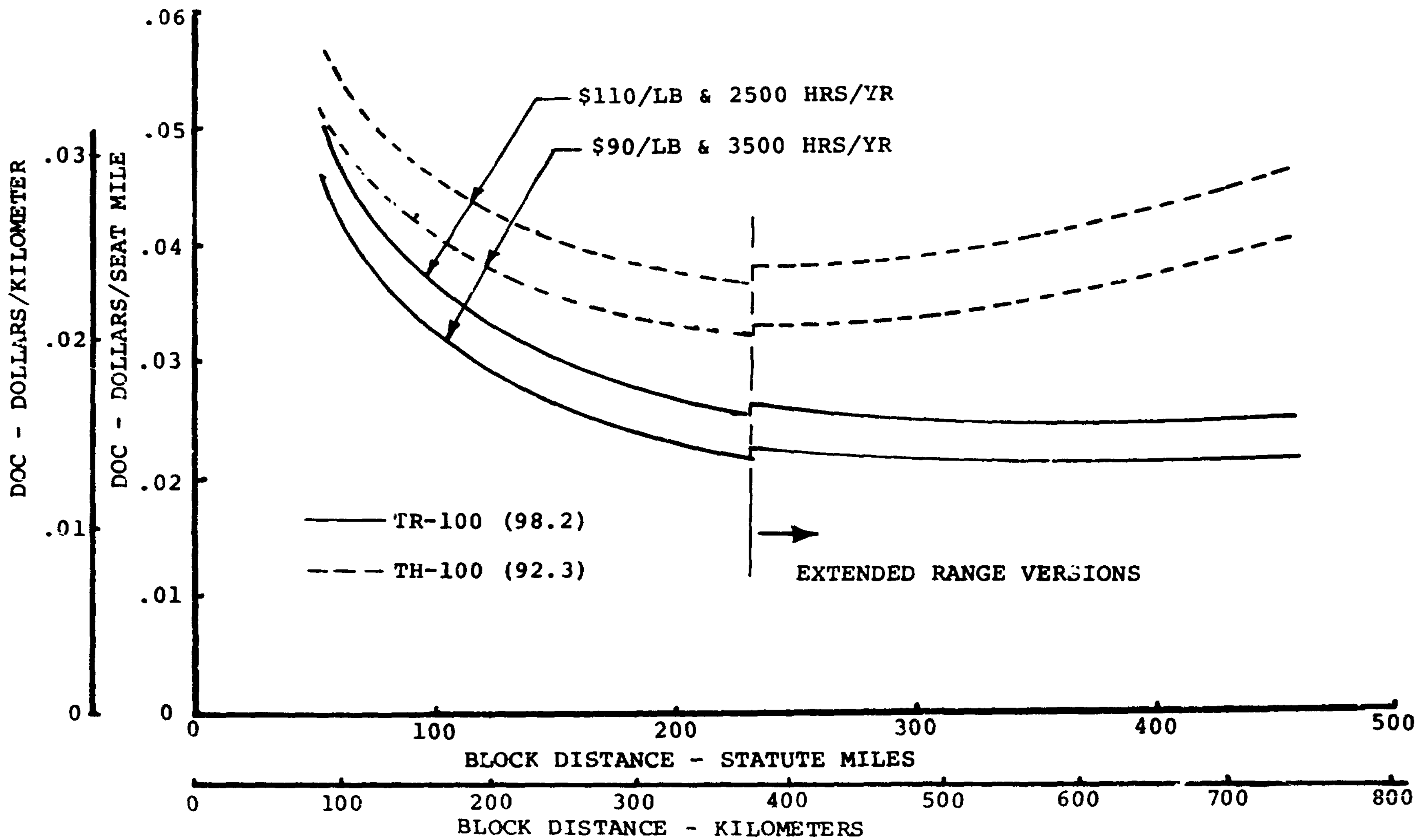
Further Comments on Noise Comparisons

The 500 foot sideline noise characteristics which have been used to classify the helicopter and tilt rotor configurations in the preceeding discussions, are of limited usefulness when applied as an indication of community acceptance. Two additional factors need to be considered when attempting to evaluate the impact of noise. These are:

(1) The area exposed to noise levels in excess of continuous acceptable levels.

(2) The time or duration of exposure.

FIGURE 4.8. DIRECT OPERATING COST COMPARISON BASELINE TANDEM HELICOPTER AND TILT ROTOR



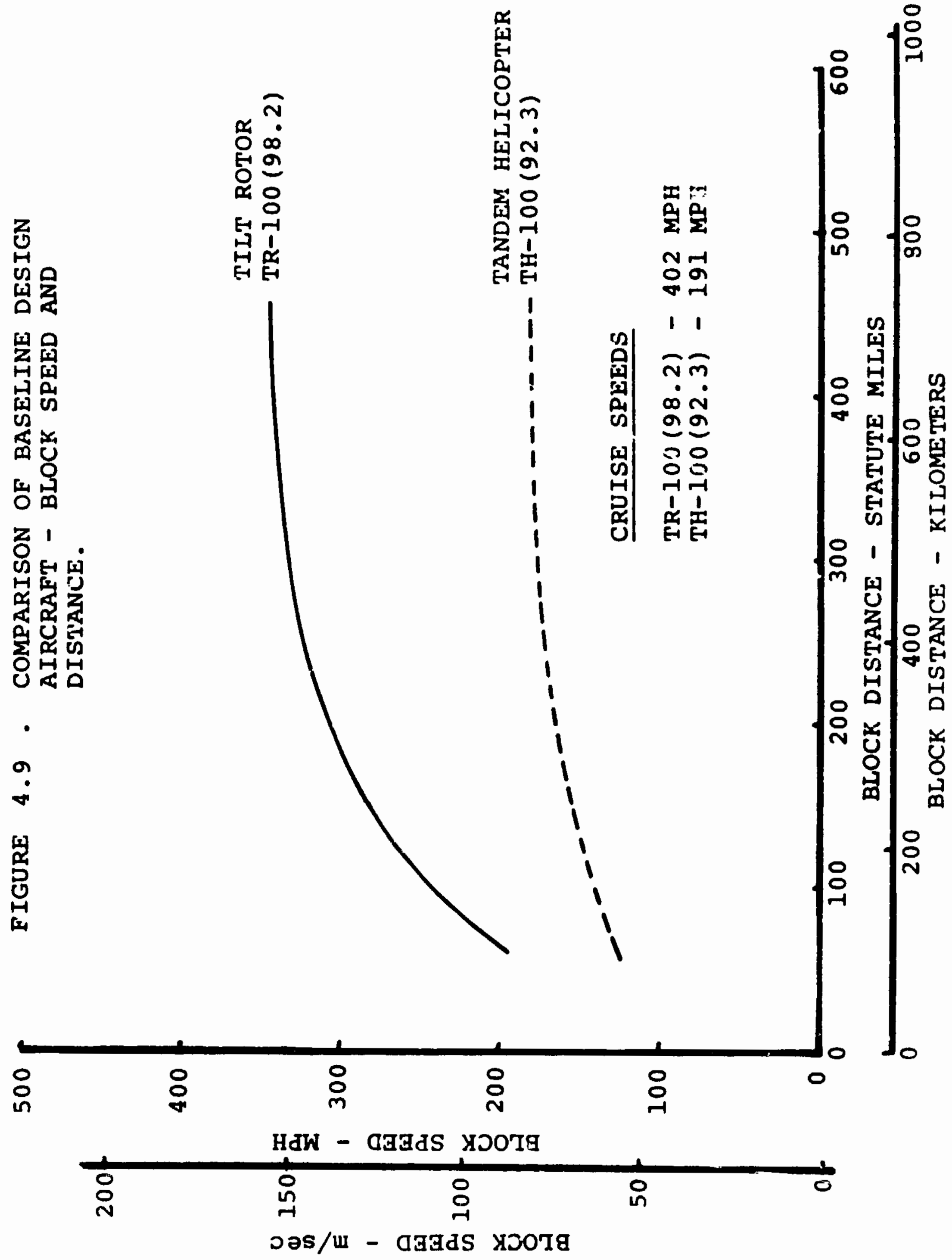


FIGURE 4.9 . COMPARISON OF BASELINE DESIGN AIRCRAFT - BLOCK SPEED AND DISTANCE.

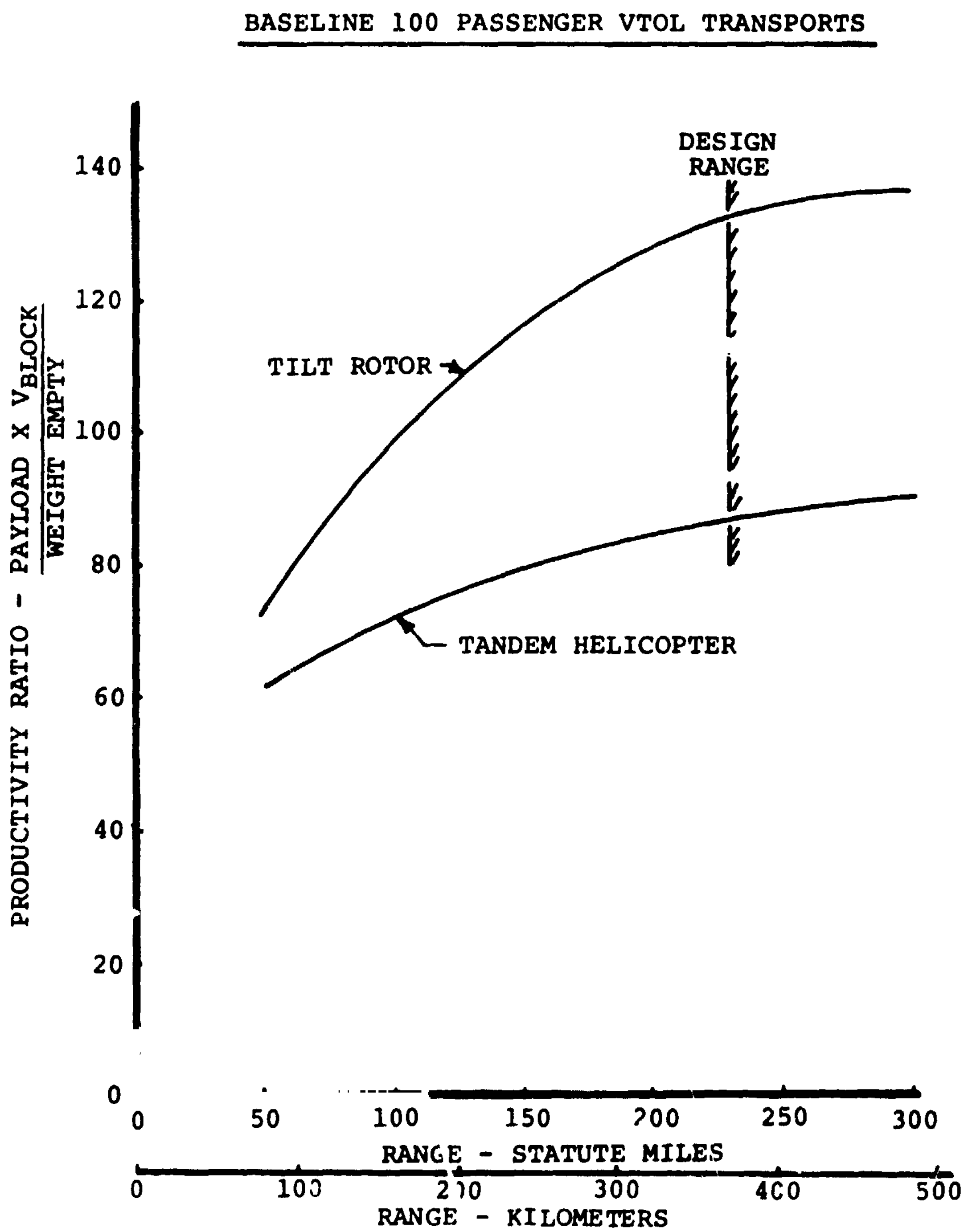


FIGURE 4.10 . PRODUCTIVITY RATIO COMPARISON: TANDEM ROTOR HELICOPTER AND TILT ROTOR.

When these are considered the design point tilt rotor is seen to have marked advantage in spite of the fact, mentioned above, that the minimum DOC criterion automatically selects a tilt rotor which has a 500 foot sideline noise 6 PNdB higher than the design point helicopter. This is illustrated by the comparison of the noise contours shown in Figures 4.11 and 4.12 for the tandem helicopter and tilt rotor at takeoff and landing. The areas exposed to 95PNdB and above for the two configurations are presented in Figure 4.13 as a function of specified levels of 500 foot sideline noise. This indicates that the takeoff 95 PNdB noise areas for a basic design point helicopter and tilt rotor are similar, but on landing the helicopter noise inputs an area approximately 3 times greater than the tilt rotor.

In Table 4.2 the total areas exposed on takeoff and landing to 95 PNdB or greater are tabulated for the design point and quiet and noisy aircraft.

TABLE 4.2. AREAS WITH NOISE LEVELS ≥95 PNdB

	-5 PNdB	DESIGN POINT	+5 PNdB
Helicopter	.29 Sq. Mi.	0.6 Sq. Mi.	1.05 Sq. Mi.
Tilt Rotor	.10 Sq. Mi.	.24 Sq. Mi.	.47 Sq. Mi.

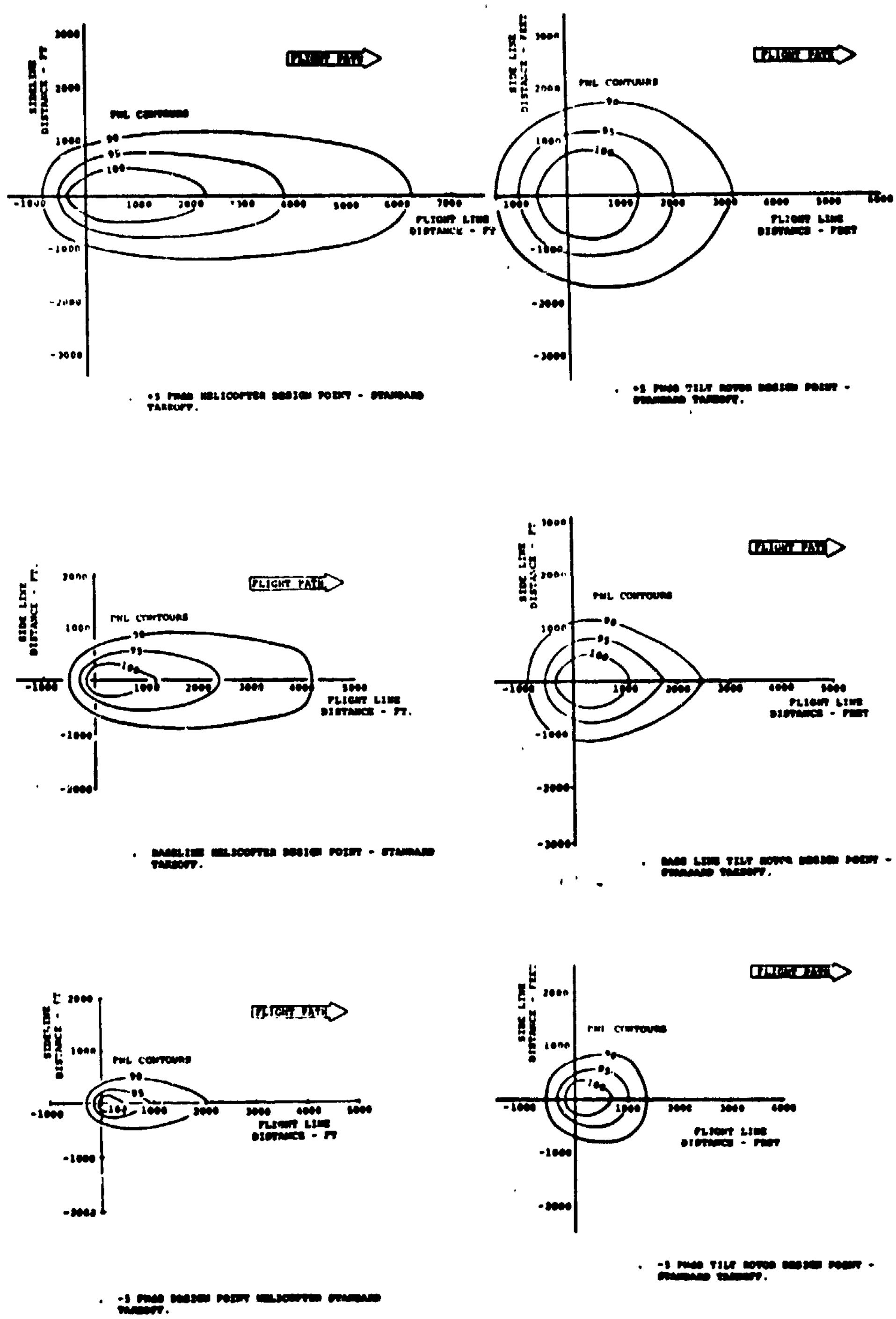


FIGURE 4.11. COMPARISON OF TILT ROTOR AND HELICOPTER TAKEOFF PNL CONTOURS

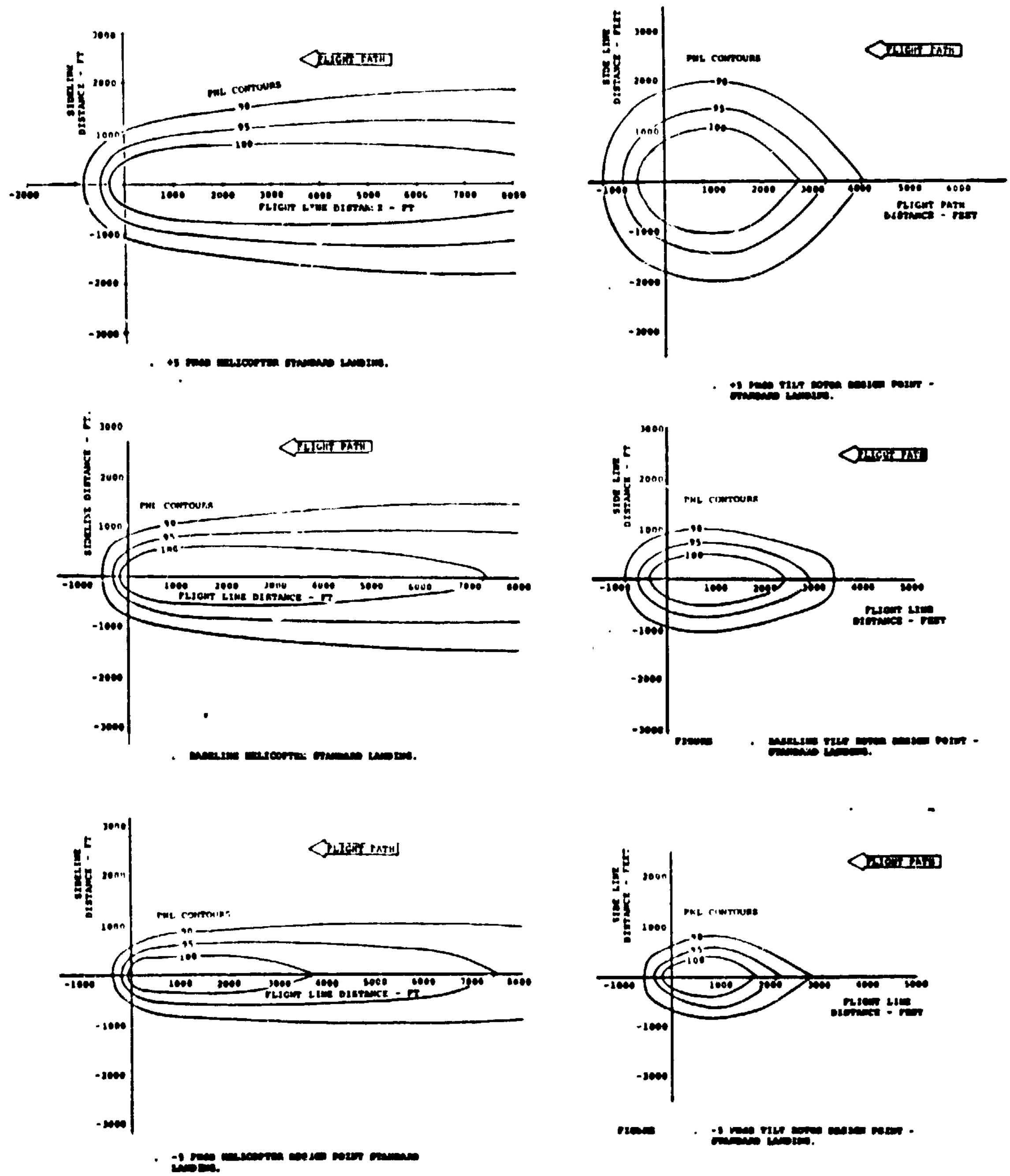


FIGURE 4.12. COMPARISONS OF TILT ROTOR AND HELICOPTER LANDING PNL CONTOURS

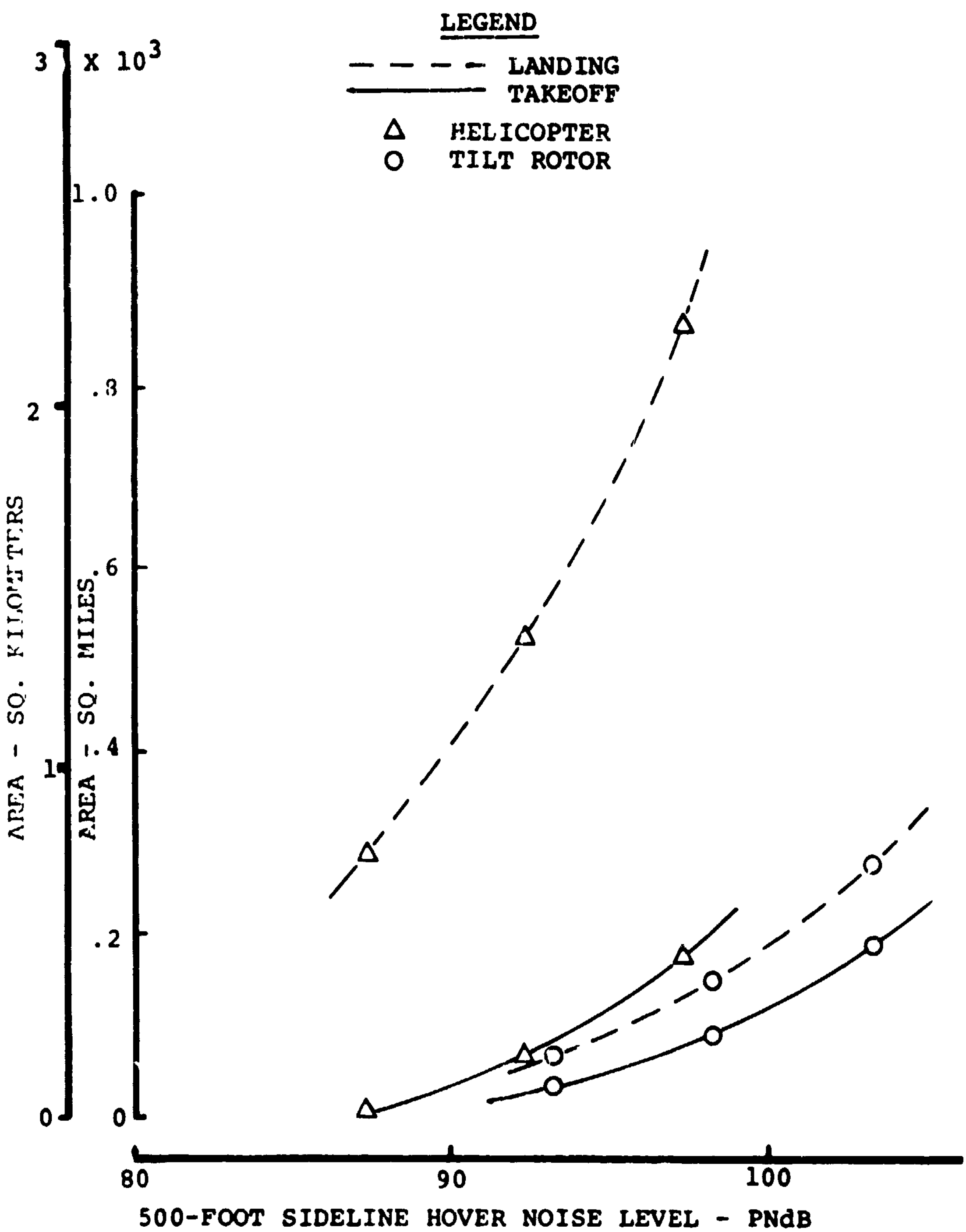


FIGURE 4.13 . COMPARISON OF AREA IMPACTED BY 95 PNdB NOISE LEVEL OR GREATER DURING TAKEOFF AND LANDING.

This indicates a substantial advantage for the tilt rotor on the basis of area alone. Further, more rational noise comparisons come readily to mind. For example, a better index of community acceptance would integrate time exposure at the different noise levels and apply weighting factors to the PNdB levels. This is beyond the scope of the present contract but sufficient basic data is included in Volume I and II to permit a computation of more refined acceptance indices.

Trip Time

An important facet of short haul operation is convenience which in part is related to the terminal operation and the block time of the vehicle. Figure 4.14 compares the trip time for various forms of transportation over varying range trips. An initial time log is shown for conventional jet transports and trains taken from Reference 2. It has been assumed that the VTOL flexibility of the baseline aircraft would allow operation from dispersed VTOL terminals with the same access time as for high speed intercity trains. The tilt rotor aircraft provides shorter trip times for ranges in excess of 73 miles than any of the other vehicles considered. The helicopter overtakes the private car at 83 miles. For the design 230 statute mile mission the total trip time for the tilt rotor is 1.45 hours compared with 2.05 hours for the helicopter, 2.18 hours for conventional air transport, 3.52 hours by train and 4.7 hours by car at an average speed of 50 miles per hour.

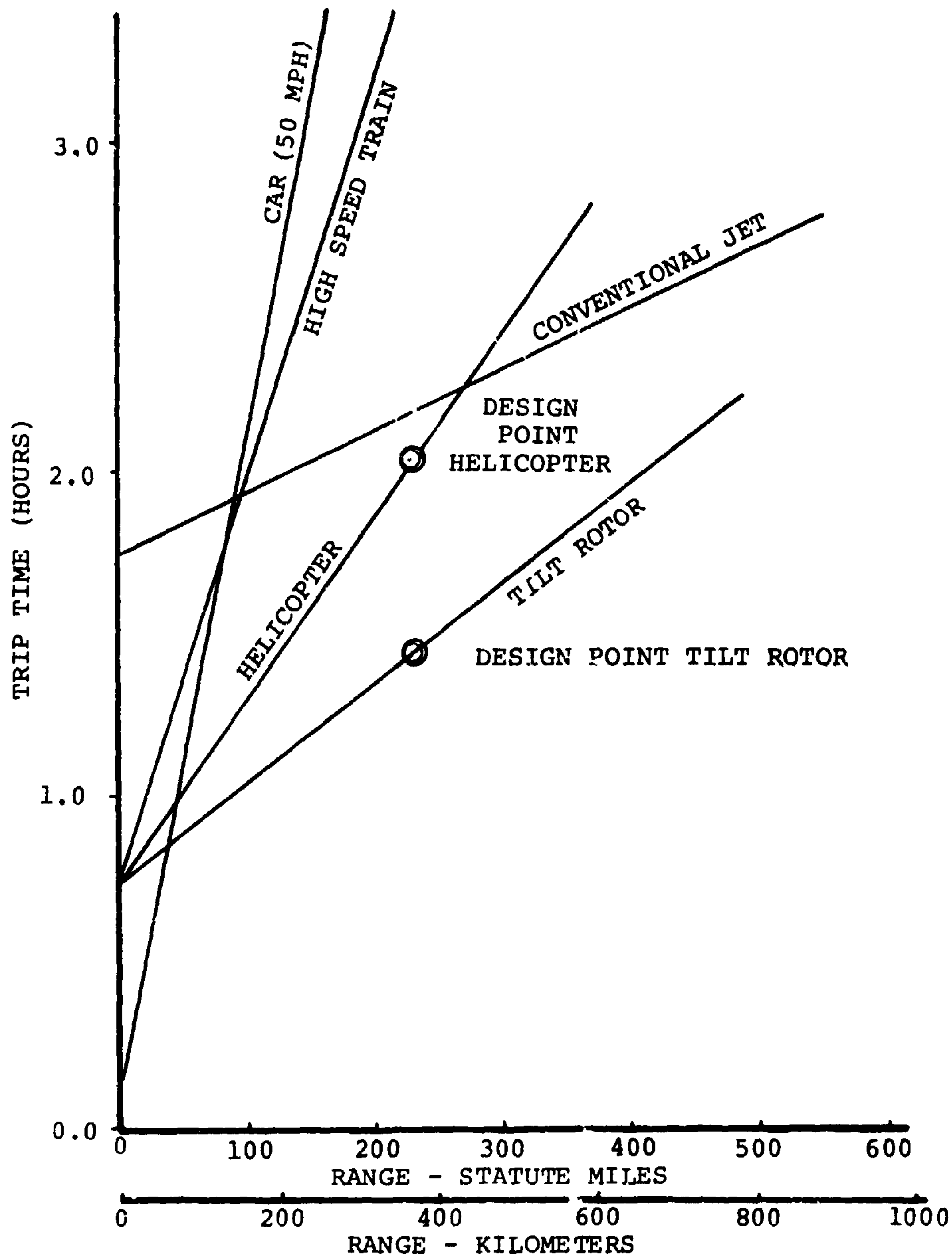


FIGURE 4.14. COMPARISON OF TRIP TIMES WITH CONVENTIONAL TRANSPORTATION.

Apart from the actual savings in trip time a further factor appears to govern the public's selection of a given form of transportation. The ratio of vehicle travel time to access time also plays an important role. This factor is much better for both of the design configurations than for conventional air transportation.

Figure 4.15 shows the design point aircraft fuel consumption in terms of passenger miles per gallon compared with other conventional air transports and helicopters. The data indicate a general trend of increased passenger miles per gallon as the aircraft design range increases.

The baseline tandem helicopter has a lower fuel consumption than existing helicopters and thus operates at higher passenger miles per gallon. In this respect the design point tandem helicopter is in the lower end of the conventional transport band but higher than the broad band trend of recent helicopters.

The tilt rotor has extremely good fuel performance and plots above the general trend formed by the conventional transports giving 50 passenger miles per gallon at 100% load factor. These better than trend performances are important in the current and continuing climate of energy conservation. Both the design point helicopter and tilt rotor performances reflect 1985 engine technology which partly explains their better performance than the trend of comparable transports.

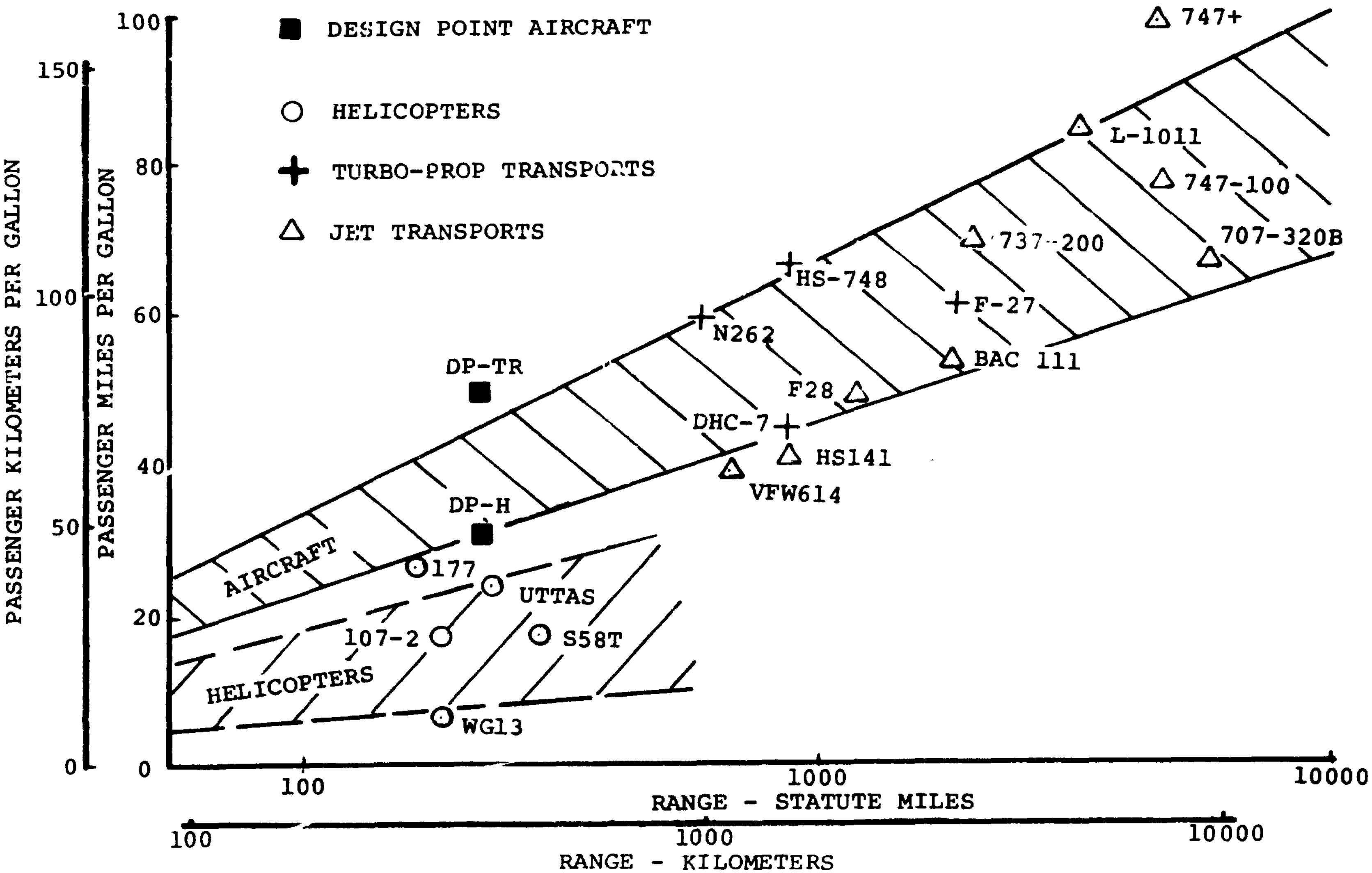


FIGURE 4.15 . FUEL CONSUMPTION COMPARISION.

5.0 MAXIMUM SIZE AND TECHNICAL RISK

Introduction

One of the items in the study Statement of Work which has provoked much thought and discussion is that pertaining to the number of passengers for which the aircraft were to be designed. This stated that the maximum payload should not exceed 100 passengers and that restrictions to a lower number should be governed by technological constraint only. Economic factors such as minimum operating cost per available seat mile were not to be considered in setting a size limit for the aircraft. The study has been fully responsive to this ground-rule, which might, under some circumstances, have forced the selection of uneconomic designs. However, careful examination of technology issues has not resulted in the identification of any serious impediments to the maximum size aircraft. In fact, only the 100 passenger constraint has been found to be more restrictive than either technological or economic considerations in both the helicopter and tilt rotor configurations. In both configurations the optimum operating costs occur around the 100 passenger mark and there is no specific evidence of technological phenomena, or difficulties with fabrication techniques or component manufacture which would limit the helicopter or tilt rotor to some intermediate number of passengers. The 100 passenger size vehicles were accordingly selected for detailed study.

Having arrived at this aircraft study size it may be worthwhile to review some of the other issues which might be involved in the selection of an aircraft to build. A large sized aircraft requires more development funds and more time to bring into service than a smaller sized aircraft. This might provide a persuasive argument for the development of a smaller design which fell within some set of budgetary and schedule constraints. Another factor to be considered is the credibility of the size selected and support among the technical community. It will be more difficult to generate and sustain support for a larger rather than a smaller sized development. Other issues which intrude into the area of economics are such questions as passenger density and frequency of schdule, and the availability of the initial capital costs to the commercial carrier. For example, the advantages of low direct operating cost could be overcome if the acquisition cost of the aircraft is more than the commercial carrier has at its disposal.

On the other hand an aircraft that is too small will be uneconomical to operate and will require a premium fare structure which may preclude use by the desired market. Some of these issues are not readily quantified and are in many cases outside the defined scope of the study.

Nevertheless, economics are of such importance that this discussion of risk has been expanded to include the effects of direct operating cost as well as an evaluation of the technical risks.

As stated above no identified technological problems restrict either the tandem rotor helicopter or the tilt rotor configurations to sizes less than 100 passengers, and the figures for direct operating costs strongly suggest 100 passengers or above.

5.1 MAXIMUM SIZE AND TECHNICAL RISK - TANDEM HELICOPTER

No limitation of tandem helicopter size based on technical risk exists within 100 passenger range. This conclusion is based upon examination of the elements of the tandem helicopter and comparison with current industrial experience.

The components and systems of a tandem helicopter to which a size dependent technical risk might be ascribed are the rotor system and the drive train.

Rotor System

The rotor system used in the design point tandem helicopter is a four-bladed 68.9 foot diameter rotor with a solidity of 0.099. The rotor is fully articulated and of conventional design. Table 5.1 shows the rotor characteristics compared with existing rotor designs. The design point aircraft is 8.9 feet larger in diameter than the CH-47 aircraft and considerably smaller than the other examples shown.

The rotor solidity is 0.099 which is almost identical to the XCH-62 (0.092). Rotor blades for the XCH-62 have already been

TANDEM HELICOPTER - ROTOR SYSTEM

AIRCRAFT	DIAMETER (FT)	σ	CHORD (INS.)	θ^{o}	TIPSPEED FT/SEC	V_{NRP} KTAS
DESIGN POINT	68.0	0.099	31.72	12	725	165
+5 PNdB	68.2	0.07	22.5	12	810	131
-5 PNdB	72.5	0.159	54.32	12	640	181
XCH-62	92	.09226	40	12	750	146
CH-47C	60	.062	25.25	-9	770	165
MODEL 347	60	.0827	25.25	-9	691	169
CH-53A	72	.115	26	-8	698	170
YCH-53E	79	.136	29	10.6	700	191

TABLE 5.1 TANDEM HELICOPTER - ROTOR SYSTEM.

fabricated using composite structures, which demonstrate that the rotor size is a minimal risk from a fabrication viewpoint. The only risk element involved in the rotor system is whether or not an adequate weight allowance has been made in the aircraft design. The rotor system weight is shown on a statistical weights trend comparison in Volume II and demonstrates that the weight allowance used is consistent with actual weights of existing large rotors in this size class.

This trend is based on prototype composites, built and tested at Boeing Vertol. While some weight reduction can be anticipated as construction techniques improve, it is not anticipated that the gain will be large by the 1985 time frame.

Drive Train

The drive train used in the tandem helicopter design is modelled on the XCH-62 helicopter system designed by Boeing and currently undergoing development testing. The design point aircraft installed power is lower than the XCH-62 and the torque levels required in the combiner box are modest by comparison with the existing design as shown in Figure 5.1. The rotor transmission is required to transmit a maximum of 207,847 foot-pounds of torque which is comparable to the CH-53A (210,000 foot-pounds) and much less than the XCH-53E (342,000 foot-pounds) and the XCH-62 (358,000 foot-pounds).

The critical components of the lift/propulsion package are therefore within the range of experience of the Boeing Vertol Company.

One method of reducing the risks in the development of large aircraft is by a component development program approach. The ongoing program on the HLH is developing the critical com-

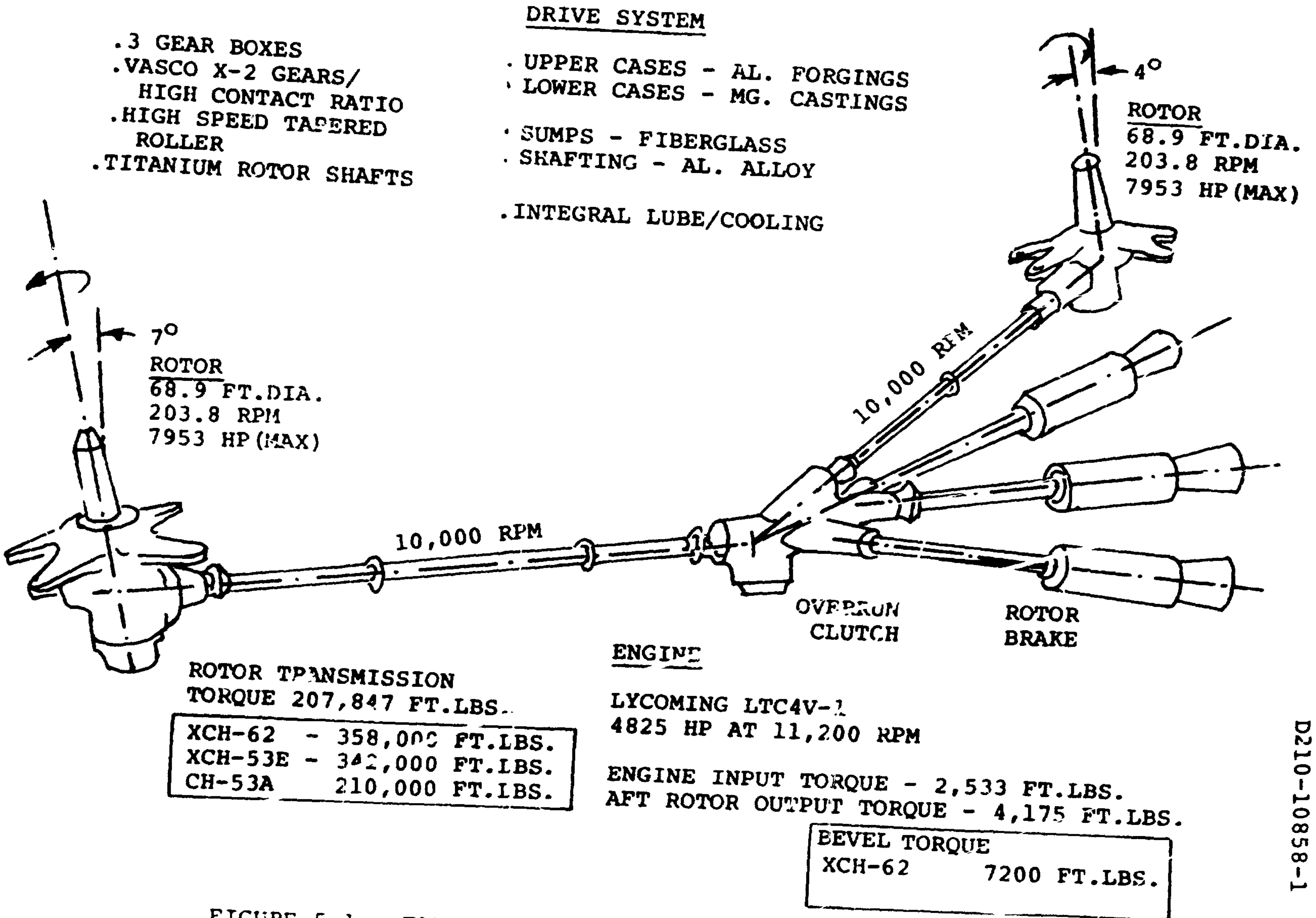


FIGURE 5.1. TANDEM ROTOR HELICOPTER DRIVE SYSTEM

ponents of the vehicle and will produce a prototype aircraft of much larger size than the tandem helicopter design selected for the short haul mission. In view of this experience, the risks for a vehicle whose fabrication is to start in 1980 must be considered small provided that the experience gained at Boeing Vertol in large tandem helicopter designs is utilized in the design and fabrication of the commercial aircraft defined in this study.

The only element of risk associated with the designs is the structural weight reduction of 25% to allow for advanced composite materials design. This reduction is thought to be optimistic and a maximum weight reduction of 16% is considered to be more appropriate based on Boeing experience. The 25% reduction was used however after discussions with NASA to preserve common groundrules between these designs and those produced by other contractors.

5.2 MAXIMUM SIZE AND RISK - TILT ROTOR

Introduction

The evaluation of risk and the selection of maximum capacity for the tilt rotor transport requires careful reasoning and is approached under a number of groundrules which rely on certain assumptions, including the successful completion of the NASA-Army XV-15 program.

The directive of the study guidelines was to select the largest aircraft (up to 100 passengers) limited only by technical risk.

An examination of the risk elements associated with a 100 passenger size vehicle is summarized below in this context to meet the guideline directive. Although the letter of the study guidelines required only technical risk to be considered, an implicit requirement of the study was to identify the aircraft which should be built to meet short haul transportation needs. Such a decision cannot be realistically made without reference to the economic environment within which the aircraft must operate. For this reason additional information is provided to show the impact of economic considerations on aircraft size.

Technical Risk -- Tilt Rotor

Background and Assumptions

The fundamental assumption in the evaluation of risk for the tilt rotor aircraft has been that the XV-15 program will be successful. That is to say that performance, handling qualities and structural integrity are demonstrated to be within an acceptable and predictable range. Specifically, it is assumed that the behavior of currently identified phenomena which define design conditions peculiar to the configuration (such as whirl flutter and rotor dynamic interactions with the flight mode dynamics) will be found to be as predicted by analysis and model and component testing. In summary, it is assumed that configuration problems will be resolved by the XV-15 program and therefore the discussion of risk for the 1985 tilt rotor transport may be limited to those issues which are functions only of size.

Technical Evaluation of Risk

It is not considered to be a useful exercise to speculate on the possible emergence of new phenomena and design difficulties as size is increased since if such difficulties are not predicted, quantification and evaluation is impossible. The potential for such development problems is recognized, but it is proposed that the development plan for the commercial transport vehicle should be structured to obtain an orderly resolution of design problems to minimize their impact. Before discussing such a development program which insures against the intangible risks, it is necessary to examine the known problem areas such as dynamic system design and predictable phenomena to determine whether any predictable limits exist.

The potential for risk in the fuselage, empennage, and aircraft systems must be considered minimal since structure and systems of this type are not significantly different from existing aircraft practice. The wealth of information in these areas in size ranges of the same magnitude and for much larger aircraft than the 100 passenger aircraft provides a solid basis for design and development.

Developmental difficulties in previous experience where large steps in size have been made in rotary wing design have been related to the aircraft dynamic systems. For this reason it is useful to briefly examine these areas in tilt-rotor design.

The components and systems which have the highest potential for developmental risk are:

1. Drive System

   Can large transmission with large torques and low rational frequencies be successfully designed?

2. Rotor System

   Does the rotor blade strength keep pace with rotor loads as size is increased?

3. Rotor - Nacelle - Wing Aeroelastic Considerations

   As size is increased, do the design constraints of wing strength and frequency become more or less restrictive?

Each of these areas are addressed in the following discussion.

The structural weight reductions of 25% used in the study in accordance with the guidelines is thought to constitute a technical risk. A weight reduction of 16% maximum would be more in line with Boeing experience.

Drive Train

The drive train required by the design point tilt rotor aircraft is shown schematically in Figure 5.2. The technical risks may be evaluated as before by comparing each transmission box or gear train with existing hardware.

The engine transfer case critical mesh torque is 2,525 foot-pounds. A similar spur torque mesh exists in the AH-56 transmission designed to 9,895 foot-pounds.

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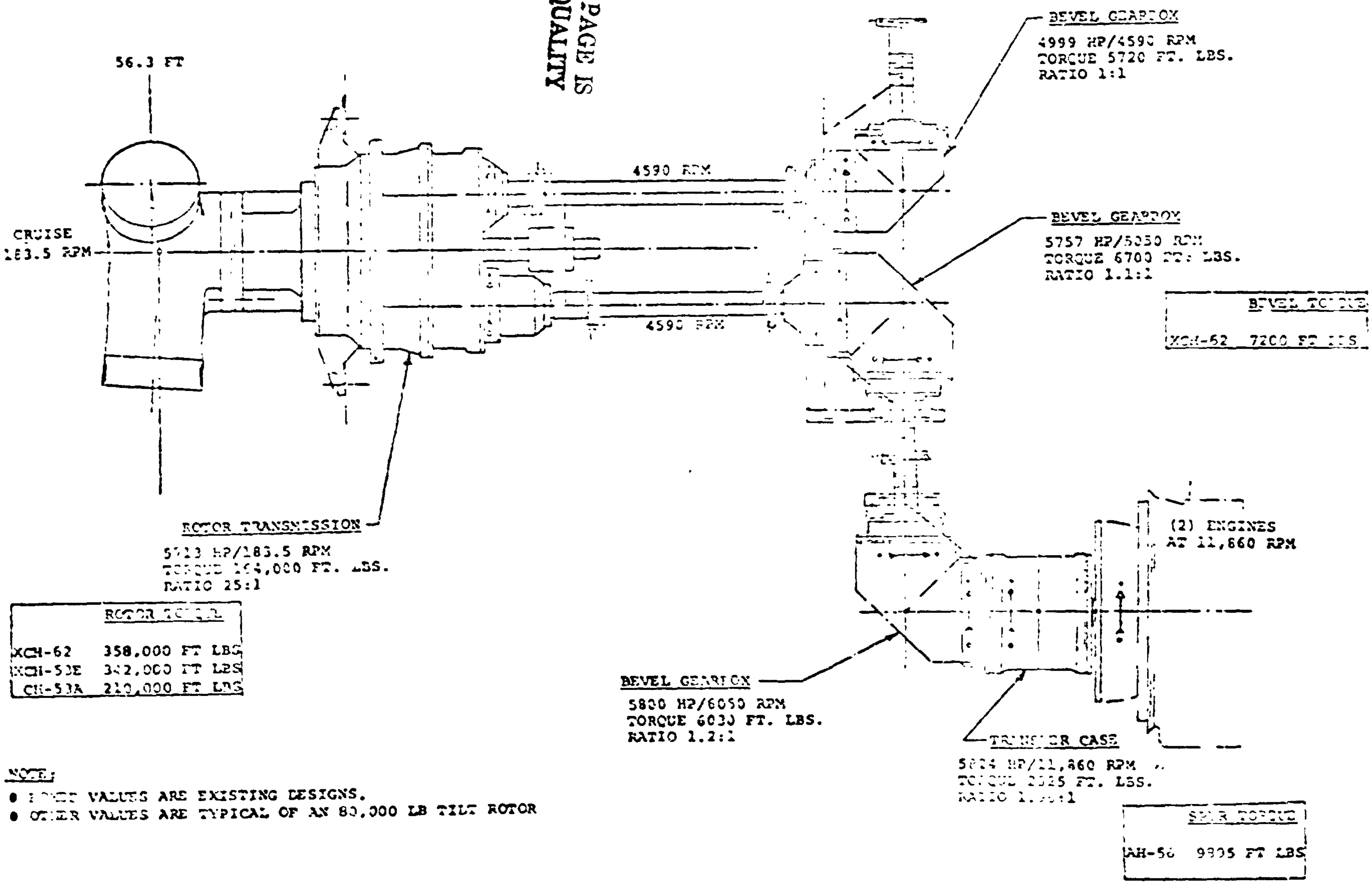


NOTE:
- [illegible] VALUES ARE EXISTING DESIGNS.
- OTHER VALUES ARE TYPICAL OF AN 80,000 LB TILT ROTOR

FIGURE 5.2 COMPARISON OF TRANSMISSION DESIGN TECHNOLOGY FOR TILT ROTOR.

The largest of the bevel boxes requires the transmission of 6,700 foot-pounds of torque which can be compared to a bevel set in the transmission of the XCH-62 which is designed to 7,200-foot-pounds.

The main rotor transmission requires a maximum torque of 165,000 foot-pounds which is much smaller than the CH-53A at 210,000 foot-pounds or the XCH-53E at 342,000 foot-pounds or the XCH-62 at 358,000 foot-pounds.

The spur torque which drives the cross shaft from the main transmission collector is sized at 7,200 foot-pounds which is again less than the AH-56 spur torque of 9,895 foot-pounds.

The rotor transmission requires a reduction ratio of 25:1. The XCH-53E main rotor transmission has a reduction ratio of 35.8:1 and the CH-53A 32.5:1. The XCH-62 reduction ratio is 51.2:1..

The maximum reduction ratio required for the bevel boxes is 1.2:1 which is quite low. Typically bevel boxes can be designed up to 3:1 and at low power even 5:1 reduction ratios are not uncommon.

The transfer case spur gearing has a 1.96:1 reduction ratio which again is modest by industry experience (up to 5:1 ratios).

These comparisons indicate that the elements of the drive system are well within industry experience in terms of size, torque transfer and reduction ratio.

The design of the individual gear boxes and shafting can not be considered a size limiting risk item although the operation of these components in the configuration specific to the tilt rotor would require development as is the case for any new transmission.

Rotor Blade Design

The design of a hingeless rotor for a tilt rotor aircraft requires the compromise of blade root strength and blade root stiffness in order to provide a finished design which has acceptable rotating blade frequencies as well as adequate blade fatigue bending strength. The detailed design of the rotor is beyond the scope of this conceptual design study, however, estimates of blade loads and strength have been made to show that such a design is feasible. Based on experience with the Boeing Model 222 design the 8.5% radial station on the blade is the probable fatigue critical section. Since the rotor will be of fiberglass construction the allowable alternating stress may be taken as 12,000 psi. The modulus of elasticity for unidirectional fiberglass is 6.2×10^6 pounds-square inch giving an allowable alternating strain of 1905 μ i/inch. These data reflect today's technology and are, therefore, reasonably conservative for the 1985 time frame. Assuming that the blade spar cross section at 8.5% R is circular, and 9.4 inches in diameter, then the spar stiffness is $EI = 1575 \times 10^6$ pounds-square inch and the allowable total alternating blade bending moment is 527,000 inch-pounds.

The blade root stiffness is compatible with blade rotating first mode frequencies in the design criteria range used in the Model 222 design.

Figure 5.3 shows the alternating total blade bending moments for the design point tilt rotor aircraft in cruise flight at both sea level and 14,000 feet altitude for 1g level flight at design gross weight.

The alternating blade loads are about 50% of the estimated fatigue allowable. The rotor loads have been computed from the measured 26-foot diameter loads using Mach scaling and accounting for the difference in rotor solidity. Cyclic pitch is assumed to be input as a function of longitudinal stick.

Figure 5.4 shows the estimated normal load factor at which endurance limit loads on the blade root occur. For speeds in excess of 216 knots the aircraft can pull its design maneuver limit with no fatigue damage and at the worst case can pull 1.8 g's before fatigue damage occurs.

The criterion used in the past for conventional propeller design is that the blade should be able to tolerate loads corresponding to 1200 Aq (i.e., angle of attack times dynamic pressure) with no damage. This line is also shown in Figure 5.4 to provide a comparison.

The maximum normal maneuver in hover requires 5.6 degrees cyclic. A normal maneuver is defined by passenger comfort levels quoted in the study guidelines (.1g lateral, .4g vertical). At this condition, the resulting blade stresses are

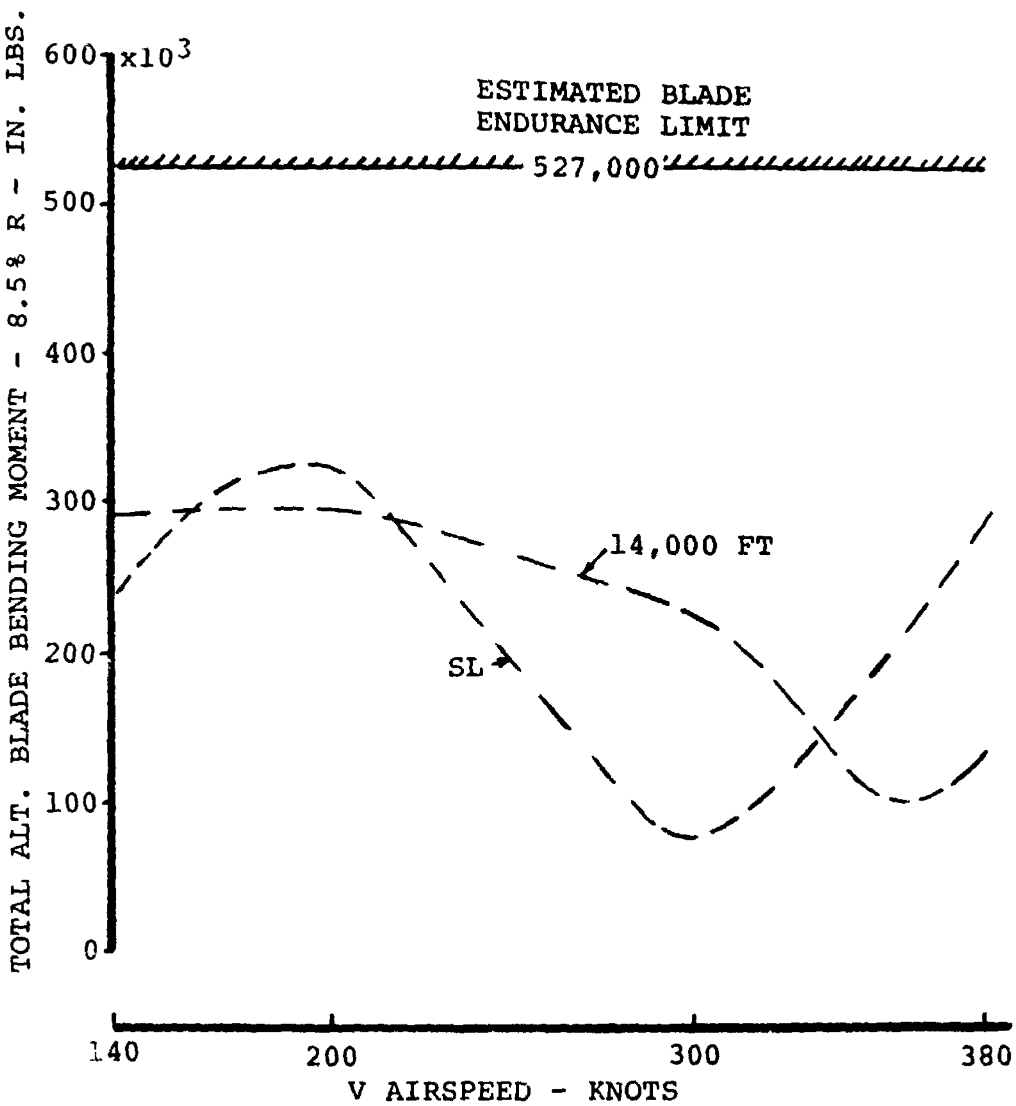


FIGURE 5.3 . BLADE ALTERNATING BENDING MOMENTS IN CRUISE 1g FLIGHT AT SEA LEVEL AND 14,000 FEET.

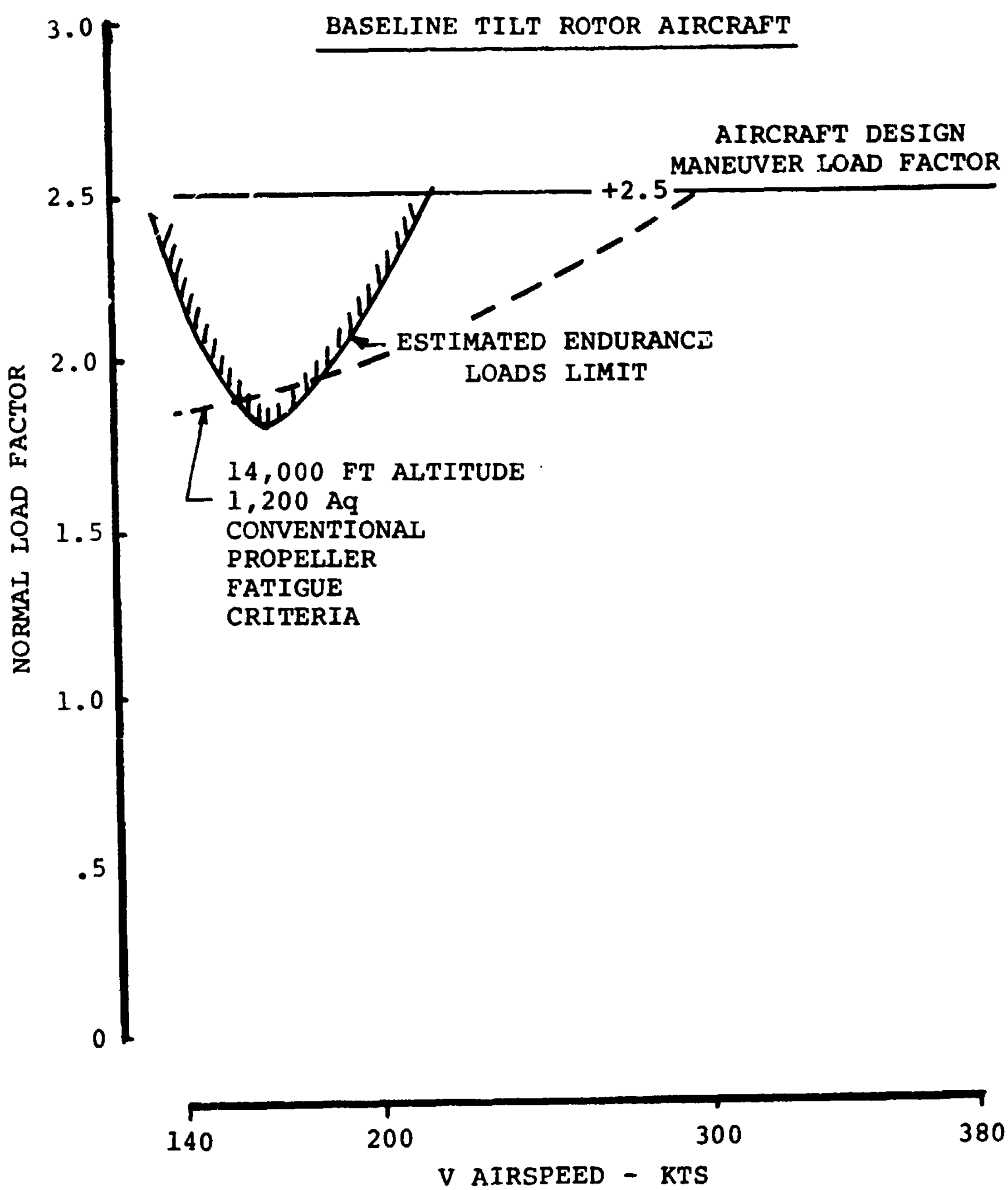


FIGURE 5.4 . MANEUVER LOAD FACTOR ENVELOPE TILT ROTOR - BLADE LOAD LIMITS.

approximately 84% of the fatigue allowable.

Normal hover loads with worst cyclic to trim produce 167,000 inch-pounds of blade bending moment of 31.6% of the endurance limit loads. The detailed design of the blade and the aircraft control system in transition would be required to compute the blade fatigue life. However, the magnitude of the loads estimated in relationship to the fatigue endurance limit provides a reasonable indication that this blade could be designed to give an adequate fatigue life in commercial service.

Scaling

In discussing possible problems which may be a function of size, the question will be asked whether XV3 and XV15 experience as well as the growing body of full scale component and scaled model test data can be extrapolated or scaled up to the size associated with the 100 passenger tilt rotor aircraft. It is the position of Boeing Vertol that experience gained in any well conducted tilt rotor test program is indeed relevant to others of larger scale and that the series of results of tests of scaled models and full scale rotors which have been conducted in support of the NASA-Army Research Vehicle competition and subsequently, may be applied in two ways: (a) by direct application using scaling laws and (b) by validating general methodology which may be applied in widely different situations.

The validity of scaling model data to full scale has been demonstrated at Boeing Vertol by experience with the 1/9th scale version of the 26 foot diameter rotor which was tested in the NASA-Ames 60 by 80 foot wind tunnel. This experience is summarized in Figure 5.5 and shows that the small scale test was an adequate indicator of the aeroelastic behavior of the full scale wing and rotor system.

A relatively smaller jump is involved in going from the 25-26 foot diameter level to a 56 foot diameter rotor system.

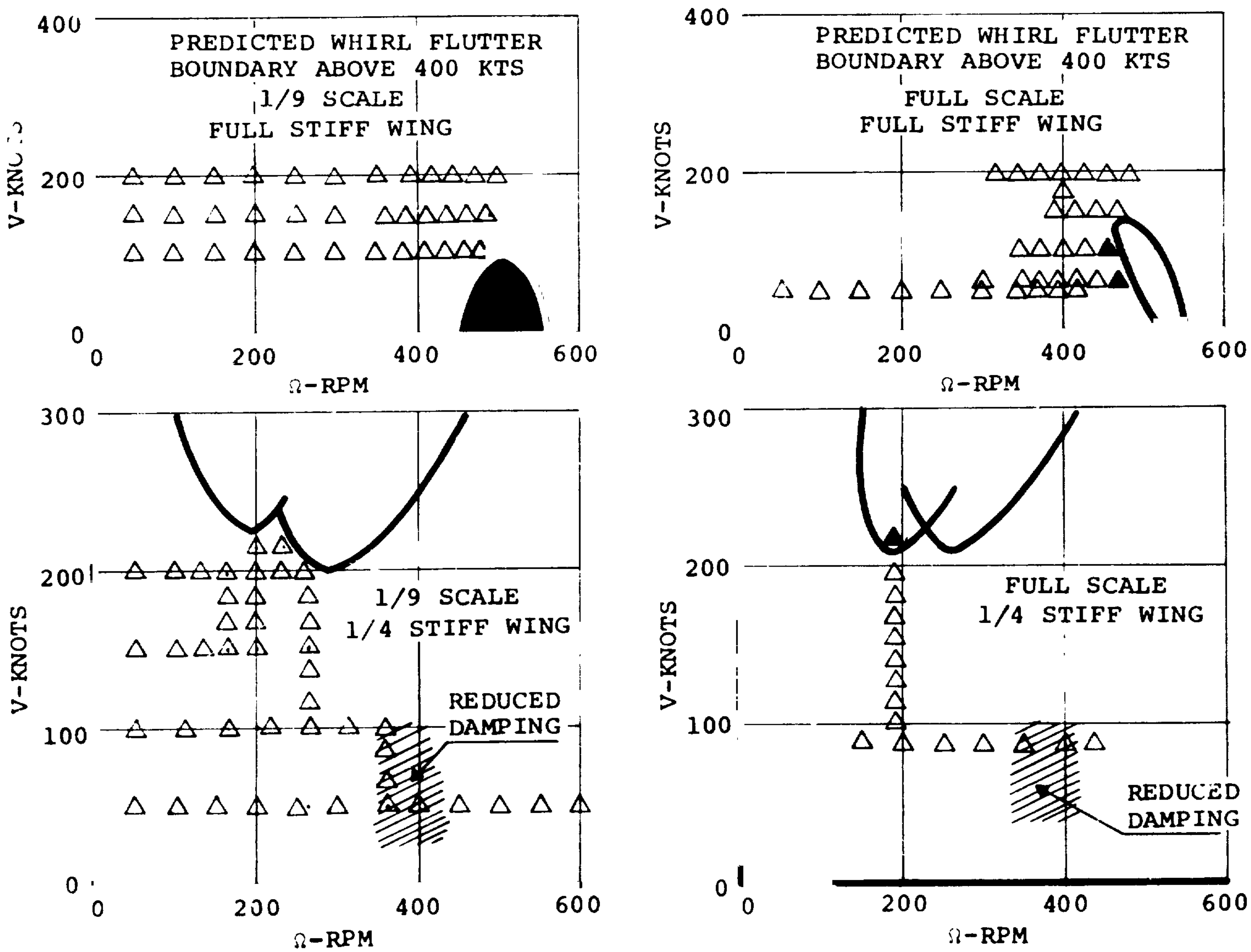
The more general question of validition of methodology has been addressed at length in other Boeing documents (e.g., Reference 3, Wind Tunnel Tests) and will not be repeated here, except to state that good predictive capability has been shown in all technology areas including blade loads, rotor derivatives and aeroelastic stability.

Aeroelastic Stability

At an early stage of the study, aeroelastic stability was reviewed as a potential area of risk as aircraft size grew from levels which had been studied in depth (e.g., Boeing Vertol M222 and Bell M301). The concern was that the parameters which determine aeroelastic behavior might grow in such a manner that aeroelastic requirements would become governing, and that the structural weights required would be substantially higher than that indicated by the usual sizing and weight trend procedures.

In growing a hingeless rotor from the 26 foot diameter size to 56 foot diameter, tip speed is held constant and blade per rev frequency maintained at the values selected for the M222

FIGURE 5.5. SCALING SUMMARY
FULL SCALE & 1/9 SCALE TESTING FOR HINGELESS ROTOR AEROELASTIC STABILITY



and other Boeing Vertol designs. Lock number remains effectively constant or is slightly reduced because of the lower solidity proposed for the 1985 vehicle. Wing aspect ratio is rather higher than M222. Rather than attempt a deduction of aeroelastic behavior on the basis of parameter changes it was considered desirable to conduct a detailed study using methodology which was validated by model and full scale tests. (Reference 4).

This detailed calculation of aeroelastic behavior was made when the final design point aircraft was selected.

In making this evaluation, constant wing bending stiffnesses were used in the beam and chordwise directions. These were derived from the strength requirements associated with jump takeoff and landing load considerations. A range of torsional stiffness was then explored and the minimum torsional stiffness required to meet its FAA requirement of stability up to 1.2 V_{Dive} at the cruise RPM was identified. The behavior at two altitudes, sea level and 14,000 feet was examined with substantially the same results. Using stiffness parameters providing adequate aeroelastic behavior are as follows;

EI_{Beam}	80×10^9 lb-in.
EI_{Chord}	475×10^9 lb-in.
$GJ_{Torsion}$	22×10^9 lb-in.

These stiffness characteristics can be provided for an acceptable structural weight.

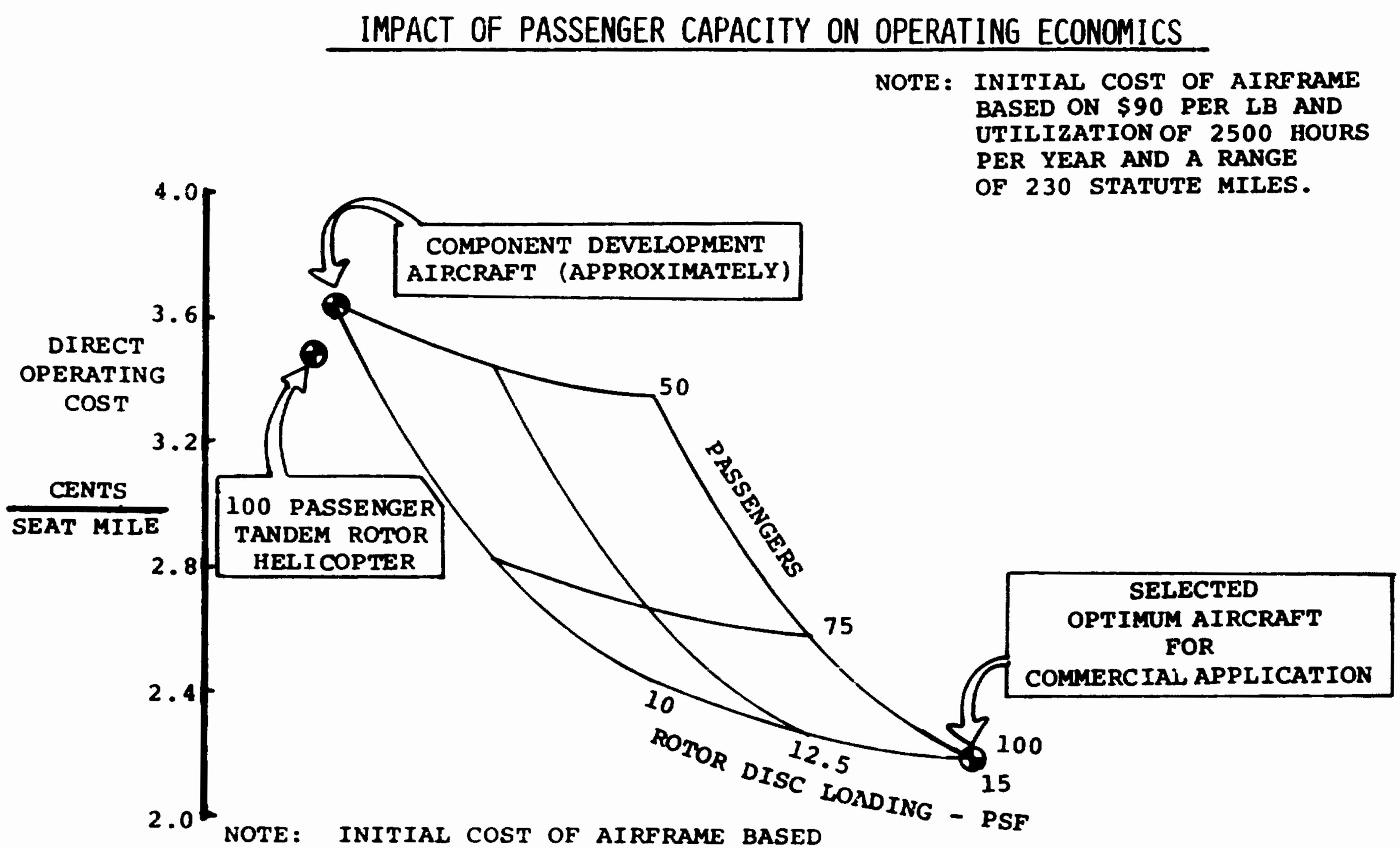
An increased margin could be provided for the cost of additional structural weight, however, the 1.2 V_{Dive} criterion already provides a 44% margin over the speeds at which the aircraft is designed to operate and this is considered adequate in an aircraft intended for civil commercial operation.

Economics

The single most important parameter in selecting a successful commercial vehicle is cost.

As the payload (i.e., number of passengers) and size of the aircraft increase, the direct operating costs decrease. This is illustrated by Figure 5.6. For example, the costs of operation per passenger mile of a 50 passenger aircraft would be 43% higher than its 100 passenger counterpart.

Since no major technology issues are identified limiting size in the study range, the optimization of vehicle cost clearly indicates that a 100 passenger vehicle (maximum allowed by the study guidelines) must be selected. A compromise decision to offer commercially an intermediate sized aircraft would set back the acceptance of the concept. For example, a 50 passenger vehicle would demonstrate economics which are slightly worse than the 100 passenger helicopter which can almost be considered as follows within the current state of the art. This comparison would therefore tend to eliminate the tilt rotor from contention.



NOTE: INITIAL COST OF AIRFRAME BASED ON $90 PER LB AND UTILIZATION OF 2500 HOURS PER YEAR AND A RANGE OF 230 STATUTE MILES.

FIGURE 5.6. TILT ROTOR DIRECT OPERATING COSTS AS A FUNCTION OF DISC LOADING AND NUMBER OF PASSENGERS

In addition, an intermediate sized vehicle would not compare favorably with conventional aircraft in terms of operating cost whereas a one hundred passenger ve icle is potentially superior as shown in Figure 5.7.

In the commercial situation, these economic facts require that unless compelling technical and engineering reasons are clearly identified which will limit the size of the aircraft, the selected vehicle must be of 100 passenger size if the aircraft concept is to realize its potential and successfully compete in the short haul marketplace. This position does not preclude the construction of an intermediate sized vehicle for component development and technology demonstration purposes and a program of this sort involving component development and testing is proposed below.

Outline of Component Development Program

The phenomena which are considered to be the largest known risk items have been examined and although much more detailed analyses need to be performed the initial studies indicate no identifiable limits which would prevent the development of a 100 passenger aircraft.

The very serious question remains as to how we can surmount the unexpected design and fabrication problems that might arise in the development of large rotors and drive systems. It is considered that the rational approach to the problem is by an orderly well planned component development program initiated to support the aircraft program, in a timely manner. Such a program is outlined below.

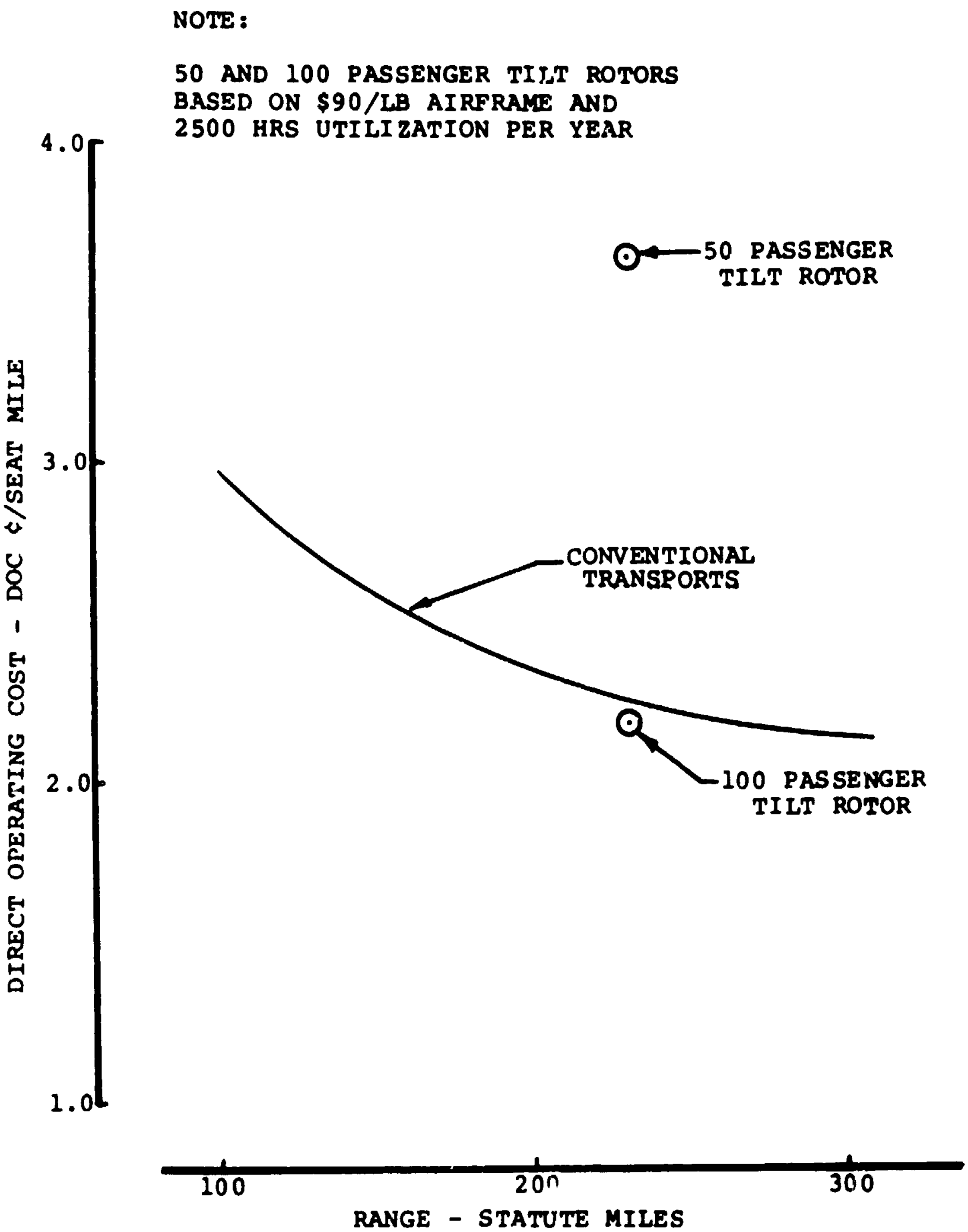


FIGURE 5.7. COMPARISON OF 50 AND 100 PASSENGER TILT ROTOR AIRCRAFT DOC WITH CONVENTIONAL TRANSPORTS.

Support of the development of large advanced aircraft by means of a component development program has a precedent set by the XCH-62 (HLH) program.

Phase I - Component Development and Ground Testing

The initial phases of the program would be the design, fabrication and testing of a full scale rotor system and drive system. These components will be subjected to the standard range of laboratory tests such as operational testing on a whirl tower to validate structural integrity and performance, and transmission qualification testing. Fatigue testing of strength critical component will also be accomplished du ng this phase. Phase I will also include a comprehensive program of scaled model tests to evaluate empirically aircraft performance, flight controls and flying qualities and aeroelastic stability.

Phase II

The second stage of development is a flight program using the full scale rotor and drive system on an intermediate size test bed aircraft of approximately 45,000 - 50,000 pounds gross weight. An existing fuselage would be used, either the CH-47 fuselage or one of similar size. This intermediate weight aircraft would have a low disc loading of about 9 to 10 pounds per square foot and will permit the initial flight experience with the rotor and drive system to take place under minimum risk conditions since the rotor power train and engines will be operating initially at levels well below their performance and structural limits.

This intermediate aircraft will also have excellent agility and control power in hover which may be presumed to enhance flight safety.

While this intermediate sized vehicle is not commercially interesting, it is definitely of military interest, and an additional persuasive argument for this approach is that the test bed vehicle itself is of a size which qualifies it as a prototype for application in several military roles.

For example, the LTTAS, SAR and long range, long endurance surveillance aircraft required by anti-submarine warfare all fall into this general weight range, and the low disc loading of the test bed aircraft is considered a necessity for these applications.

The Phase II program would include testing up to the weight of the 100 passenger aircraft by gradual weight increments using ballast. At the conclusion of this program, few risks would remain in the dynamic system and the fabrication of the 100 passenger aircraft could proceed.

Phase III - Fabrication and Test of the Commercial Tilt Rotor

The fabrication and experimental flight test evaluation of the 100 passenger vehicle would be a relatively straightforward development following Phases I and II and should not entail any more risk than that routinely accepted in the development of new versions of conventional fixed wing or rotary wing aircraft.

This phase would entail repetition of some Phase II items such as detailed redesign of airframe structure and a model testing, but not the major expenses associated with rotor and drive system development.

Program Schedule

To meet a 1985 deadline for the 100 passenger transport, the program outlined would require initiation in 1978, with laboratory work and whirl tests during 1979 and 1980. The fuselage for the intermediate sized aircraft will be selected from existing inventory since cruise performance will not be critical on the test bed vehicle. This phase would need to be started in 1979 to produce flight data by 1981. The orderly development of hardware in this way and the acquisition of flight experience will provide a necessary background to fly commercially successful passenger tilt-rotor aircraft by 1985.

6.0 CONCLUSIONS AND RECOMMENDATIONS

CONCLUSIONS

Both tandem rotor helicopter and tilt rotor aircraft have been designed to carry 100 passengers over 200 nautical miles. These aircraft meet the specifications of the guidelines and are viable competitors in the short haul market.

Comparison of the aircraft leads to the following generalized conclusions.

1. The helicopter is lighter, and less expensive in terms of initial cost than the tilt rotor.
2. The tilt rotor is faster and less expensive in terms of direct operating cost than the helicopter.
3. The helicopter has lower perceived noise levels at 500 feet sideline distance than the tilt rotor. However, comparison of equal intensity perceived noise level contours indicate that the tilt rotor noise affects a much smaller area.
4. The aircraft are competitive with other forms of transportation in terms of trip time, fuel consumption and direct operating cost.
5. The preponderance of experience on helicopters makes the tandem helicopter a lower developmental risk than the tilt rotor, however, no identifiable risks can be quantified for either configuration.

The tilt rotor would require a component development program approach to minimize the risk of developmental difficulties.

RECOMMENDATIONS

The utility of these vehicles in the short haul market rests to a large degree on their ability to takeoff and land vertically. The terminal area operation requires investigation to identify operational and environmental problems associated with this type of operation.

It is recommended that terminal area simulation be done to investigate this aspect of the vehicle operation.

The tilt rotor will require the development of gust alleviation systems to improve passenger comfort. Such a system would involve both rotor and wing controls. Design studies and tests to optimize such a control element should be performed.

The assessment of risk is such that the helicopter development could start as late as 1980 for opera[illegible] in the 1985 time frame. The tilt rotor would require more lead time and a component development program should be initiated as soon as the XV-15 flight program provides a successful demonstration of the concept.

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