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CONCEPTUAL DESIGN STUDY OF A 1985 COMMERCIAL STOL TILT ROTOR TRANSPORT

C. A. WIDDISON

J. P. MAGEE

H. R. ALEXANDER

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LIST OF SYMBOLS

A	Disc area of prop/rotor
AEO	All engines operating
AR	Aspect ratio
b	Number of blades per prop/rotor
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_{TPROP}	Propeller thrust coefficient, $T/\rho n^2 D^4$
C_{PPROP}	Propeller power coefficient, $P/\rho n^3 D^5$
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_{n_p}	Yawing moment coefficient due to roll rate
C_{n_r}	Yawing moment coefficient due to yaw rate
C_{ℓ_p}	Rolling moment coefficient due to roll rate
C_{Y_p}	Side force coefficient due to roll 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
C	Chord (of wing, tail or prop/rotor blade)
$\bar{C}$	Mean chord

LIST OF SYMBOLS (CONTINUED)

D	Diameter of prop/rotor
DGW (or GW)	Design gross weight
DCP	Differential collective pitch
DOC	Direct operating cost
F/W	Force to weight ratio
Fe(fe)	Equivalent drag area
HP	Horsepower
I_{xx}	Aircraft inertia (roll)
I_{yy}	Aircraft inertia (pitch)
I_{zz}	Aircraft inertia (yaw)
IGE	In ground effect
J	Propeller advance ratio V/nD
KTAS	Knots, true airspeed
MAC	Mean aerodynamic chord
M_{mo}	Maximum operating Mach Number
M_{α}	Pitching 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 (or prop/rotor rotation speed)
nm	Nautical miles
OEI	One engine inoperative

LIST OF SYMBOLS (CONTINUED)

OWE	Operating weight empty
P	Power
PED	Pedal
PNL	Perceived noise level
PNdB	Perceived noise level in decibels
PSF	Pounds per square foot
R	Radius of prop/rotor
SAS	Stability augmentation system
SPL	Sound pressure level
T	Thrust of prop/rotor
$T_{1/2}$	Time to half amplitude
T_2	Time to double amplitude
T/W	Static thrust to weight ratio
t	Time (seconds)
t/c	Thickness/Chord Ratio
TAS	True airspeed
V	Forward flight speed
V_T (V_t)	Tipspeed
V_e	Equivalent airspeed
V_{M_O}	Maximum operating airspeed
$\bar{V}_H$	Horizontal tail volume ratio
W/S	Wing Loading
XMSN	Transmission

LIST OF SYMBOLS (CONTINUED)

$\theta, \dot{\theta}, \ddot{\theta}$	Pitch angle, rate, acceleration
$\psi, \dot{\psi}, \ddot{\psi}$	Yaw angle, rate, acceleration
$\phi, \dot{\phi}, \ddot{\phi}$	Roll angle, rate, acceleration
δe	Elevator deflection
θ_{75}	Collective pitch
$\Delta\gamma$	Differential cyclic
α_{trim}	Angle of attack to trim
β/δ_R	Sideslip angle per degree of rudder
γ	Climb angle
δ_R	Rudder deflection
ϕ/δ_R	Roll angle per degree of rudder
μ	Friction coefficient
$\delta_{S_\alpha}/\delta_R$	Lateral stick per degree of rudder
ρ	Air density
σ	Solidity of prop/rotor, $bc/\pi R$
ω_{n_N}	Natural frequency
ζ	Damping ratio

ABSTRACT

This document contains the results of conceptual engineering design studies of a STOL tilt rotor commercial aircraft for the 1985 time frame. The aircraft is sized to carry 100 passengers over a 200 nautical mile range. The field length was limited to 2,000 feet. The details of aircraft size, performance, flying qualities, noise and cost are included in this report.

The primary objective of the study was to determine the savings in terms of fuel economy resulting from STOL operation compared with VTOL vehicles.

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. Mr. D. Giulianetti and Mr. K. Edenborough were technical monitors for this work.

The Boeing project manager was J. P. Magee and project engineer was C. Widdison.

SUMMARY

The study reported in this document provides preliminary design data for a STOL tilt rotor aircraft intended for use in the short haul market in the mid 1980's.

It has been demonstrated in Reference 1 that, for the short haul mission, a VTOL tilt rotor aircraft is a good potential solution to the problems of rising fuel costs and increasing congestion at major airports. The ability of the VTOL tilt rotor to operate out of small airfields and its high cruise efficiency are properties shared by the STOL tilt rotor.

This study was initiated to investigate the fuel savings and other benefits and penalties of designing for STOL operation rather than for VTOL.

The STOL tilt rotor described in this document has been designed to perform the same mission as the VTOL tilt rotor aircraft of Reference 1. The same ground rules were used in the STOL design as were used for the VTOL with the exception that the vertical takeoff and landing capability was replaced by the requirement to operate from fields of 2,000 feet length or less. Thus, a meaningful comparison between VTOL and STOL designs is possible and the relative merits of each can be assessed on a consistent basis.

The STOL tilt rotor described in the report was designed to carry a payload of 100 passengers with baggage, over a block distance of 200 nautical miles. The design was evolved from the VTOL tilt rotor of Reference 1 by means of varying the

airframe and propulsion parameters in such a way that direct operating cost was minimized (subject to the constraints of field length and practical design considerations).

The fuel consumption of the STOL tilt rotor is 62.5 passenger miles per gallon compared with a value of 47.3 for the VTOL aircraft. The greater fuel economy of the STOL tilt rotor is reflected in the fact that for the design mission it consumes 25% less fuel than the VTOL.

The STOL tilt rotor has a design gross weight of 68,493 pounds and a maximum cruise speed of 311 knots at 14,000 feet altitude. The VTOL aircraft, designed for the same mission is heavier at 74,749 pounds, but has a higher cruise speed, 349 knots.

Because of its lower gross weight the STOL tilt rotor has a lower initial cost than the VTOL, but because of its lower speed the direct operating cost is only slightly lower. Based on an airframe cost of $90.00 per pound the STOL aircraft has an initial cost of $4.62M and a direct operating cost of 2.09 cents per seat mile at the design range. The corresponding costs of the VTOL tilt rotor are $5.15M and 2.19 cents per seat mile.

The STOL tilt rotor aircraft has a 500 foot sideline perceived noise level of 101.3 PNdB whereas that of the VTOL aircraft is 98.2 PNdB. The area subjected to noise levels greater than 95 PNdB at takeoff is slightly higher than that due to the VTOL aircraft, but on landing the STOL aircraft affects a smaller area at that sound level than does the VTOL.

The report contains a description of the aircraft design and a detailed description of its performance, flying qualities, weights and cost.

A detailed comparison is made with the VTOL tilt rotor aircraft defined in Reference 1. The table in this section summarizes the important design parameters of the STOL and VTOL tilt rotor aircraft discussed above.

An examination of the technical risks involved in the development of a 100 passenger STOL tilt rotor was carried out with respect to size, dynamic systems, aeroelastic phenomena and economics.

No insurmountable problems are envisioned that will make for a prohibitive technical risk in constructing such an aircraft for the mid 1980's provided a comprehensive component development program of flight hardware can be initiated by 1979.

The conclusions to be drawn from the study are that the benefits to be expected from designing for STOL include greatly improved fuel economy and initial costs, and a slight improvement in operating costs and productivity. The price of these benefits is the loss of vertical takeoff and landing capability, slightly more noise and longer block times.

		STOL TILT ROTOR	VTOL TILT ROTOR
GROSS WEIGHT	Kg(Lbs)	31,068(68,493)	33,905(74,749)
EMPTY WEIGHT	Kg(Lbs)	20,422(45,023)	22,710(50,068)
CRUISE SPEED	KTAS	310	349
CRUISE ALTITUDE	m(ft)	4,267(14,000)	4,267(14,000)
BLOCK TIME	Hours	0.82	0.747
DOC AT 200 N.M. RANGE*	cents/seat mile	2.09	2.19
500 FOOT SIDELINE PERCEIVED NOISE, PNdB		101.3	98.2
AREA OF 96 PNdB CONTOUR AT TAKEOFF	SqKm(SqMiles)	.30(.115)	.23(.09)
AREA OF 95 PNdB CONTOUR ON LANDING	SqKm(SqMiles)	.36(.14)	.39(.15)
BLOCK FUEL	Kg(Lbs)	1,086(2,391)	1,434(3,157)
ROTOR DIAMETER	m(ft)	13.53(44.4)	17.16(56.3)
DISC LOADING	Kg/m^2 (Lbs/ft^2)	108(22.12	73.26(15)
WING LOADING	Kg/m^2 (Lbs/ft^2)	489(100)	489(100)
TAKEOFF TIPSPEED	m/s(ft/sec)	244(800)	236(775)
CRUISE TIPSPEED	m/s(ft/sec)	171(560)	166(543)
INSTALLED POWER (SEA LEVEL, STANDARD)	Watts(HP)	8.31×10^6(11,142)	12.36×10^6(16,580)

*BASED ON AIRFRAME COST OF $90.00/POUND AND ANNUAL UTILIZATION OF 3,500 HOURS PER YEAR.

SUMMARY TABLE FOR STOL AND VTOL TILT ROTORS

1.0 INTRODUCTION

The tilt rotor aircraft is usually conceived as a vertical takeoff and landing aircraft with the ability to tilt the rotor from the hover or helicopter mode to the cruise or propeller flight configuration. This concept combines the low speed flight qualities of the helicopter with the cruise performance of a turboprop aircraft. The design studies reported in this document describe a STOL tilt rotor design and examine the effects of relaxing the VTOL constraints on the aircraft design, fuel economy, direct operating cost and vehicle performance. The aircraft is similar to a vertical takeoff tilt rotor vehicle except that it has a smaller rotor system and less installed power.

The objective in performing the design studies was to provide engineering, fuel consumption and cost data on an optimized vehicle of this type to complement the vertical takeoff designs performed under the same contract and reported in Reference 1. These design data are intended to be used as input data on a larger short haul air systems evaluation study to be done by NASA.

The studies provide definition of a vehicle with low fuel consumption (62.5 passenger miles/gallon) for its design range, short takeoff and landing and a relatively small area within which noise levels exceed 95 PNdB.

The improvement in fuel economy is largely due to the reduction in installed power. The diminishing availability and rising cost of fossil fuels make low fuel consumption an attractive asset of this concept.

The ability to operate on less than 2,000 feet field length opens up the possibility of scheduled transportation from many more small airfields and would assist in reducing the current congestion at large airports.

The reduction in installed power and thrust required for take-off for this vehicle tend to reduce the perceived noise level. However, the higher blade loading that results from the increased tip speed and reduced rotor solidity counteract this tendency and a higher perceived noise level results at the static thrust condition than was the case for the VTOL tilt rotor.

The noise level at static thrust is not the obvious parameter upon which to judge community acceptance. The STOL vehicle noise, for example, affects a smaller area with high noise than the VTOL tilt rotor or helicopter designs reported in Reference 1.

The details of the vehicle design, mission performance, noise levels and cost data are given in Section 2 of this report. Section 3 provides vehicle performance, weights and flying qualities information. The design data and performance are compared with a vertical takeoff tilt rotor and a tandem rotor helicopter from Reference 1 (as well as more conventional transports) in Section 4.

The development of a large aircraft of a new concept requires careful consideration of the technical risks involved and this subject is addressed in Section 5.

Appendix A summarizes the process by which the design point STOL tilt rotor aircraft was selected. The cost methodology has been detailed in Appendix B and the weight predictions are substantiated in Appendix C.

2.0 DESIGN POINT STOL TILT ROTOR AIRCRAFT

The tilt rotor concept is unique in that it combines the hover efficiency, low speed agility and low installed hover power of the helicopter with the cruise advantages of a conventional turboprop transport.

The prop rotors are mounted on the wing tip and tilt from the vertical in hover down to a conventional propeller configuration in cruise. In cruise the prop/rotor propulsive efficiency remains high which coupled with a high lift/drag ratio provides an efficient cruising aircraft.

Although the tilt rotor aircraft concept is a VTOL vehicle, the lifting capability of the low disc loading rotors provide attractive STOL performance. Removing the design constraints of vertical flight operation allow the installed horsepower to be further reduced. The vehicle described in this section is designed specifically for STOL operation.

2.1 AIRCRAFT CHARACTERISTICS

A general view of the design point STOL tilt rotor aircraft is shown in Figure 2.1 and a general arrangement (three view) drawing is given in Figures 2.2 and 2.3. Table 2.1 provides a list of the major aircraft dimensions and characteristics.

The aircraft has a takeoff gross weight of 68,493 pounds (31,068 kilograms) and an empty weight of 45,023 pounds (20,422 kilograms). The two rotors have three blades each of hingeless fiberglass composite construction. The rotor

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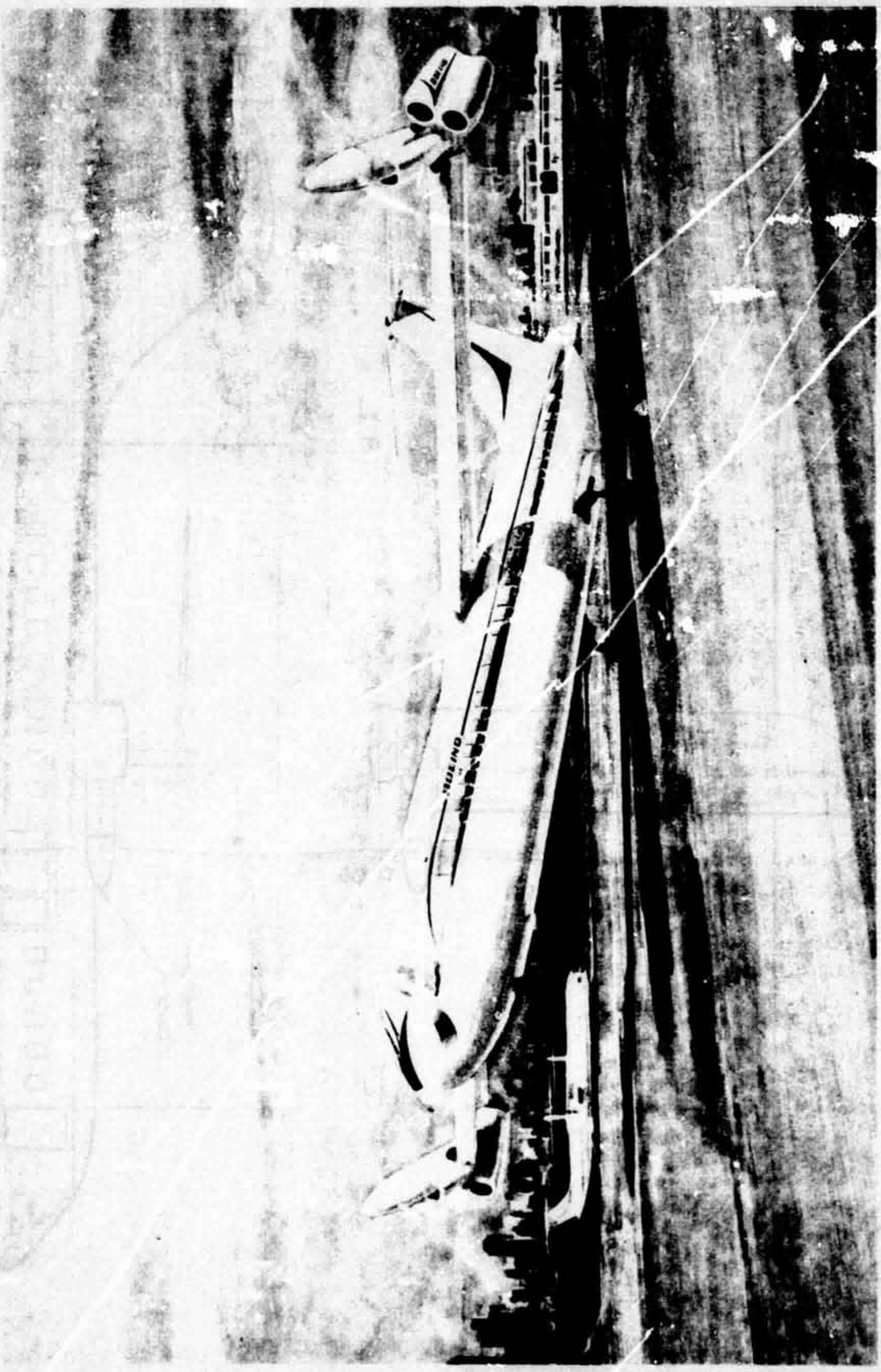

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FIGURE 2.1. **BOEING VERTOL 1985 100 PASSENGER STOL TILT ROTOR COMMERCIAL TRANSPORT**

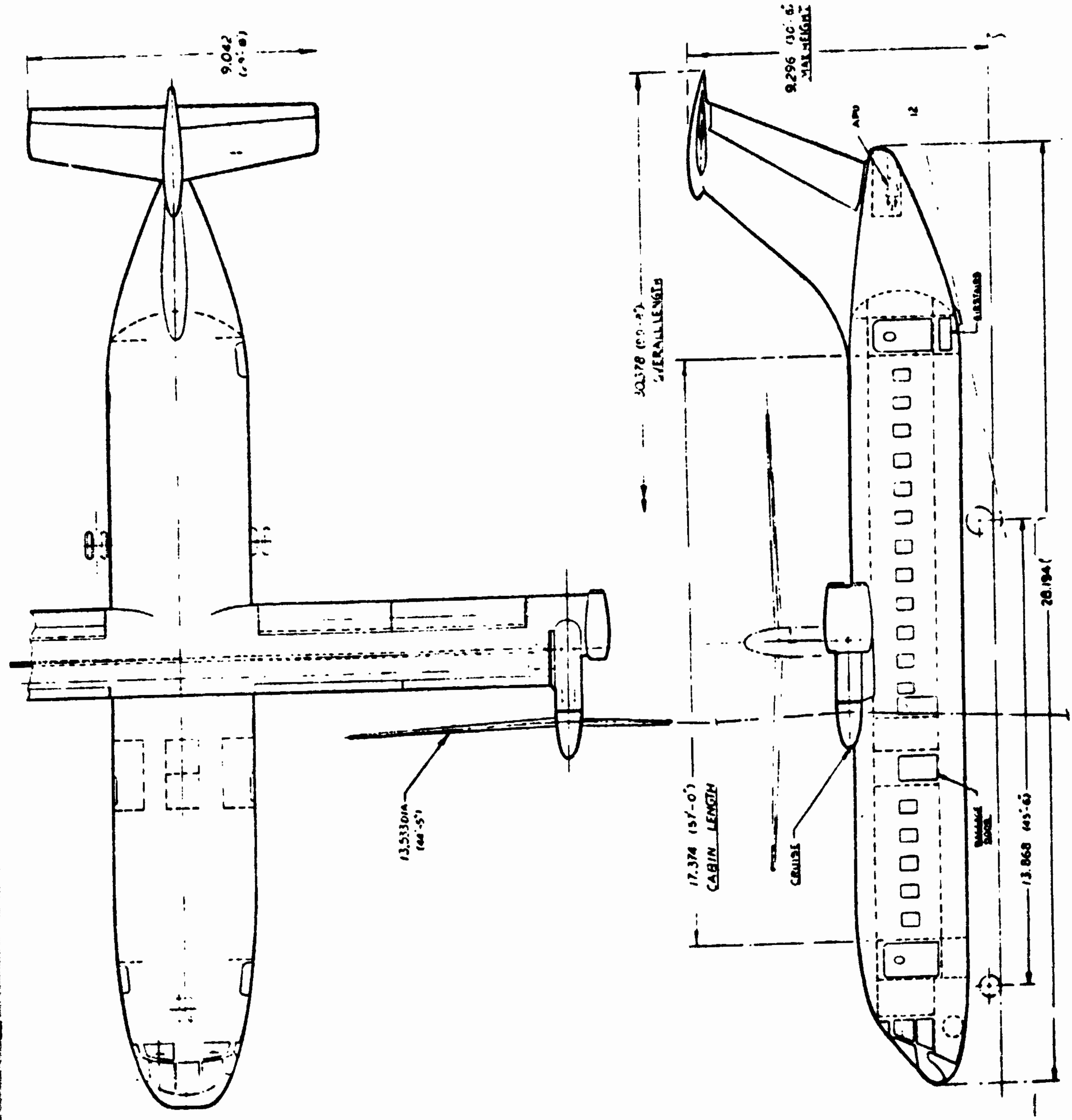


FIGURE 2.2. SIDE AND PLAN VIEWS.

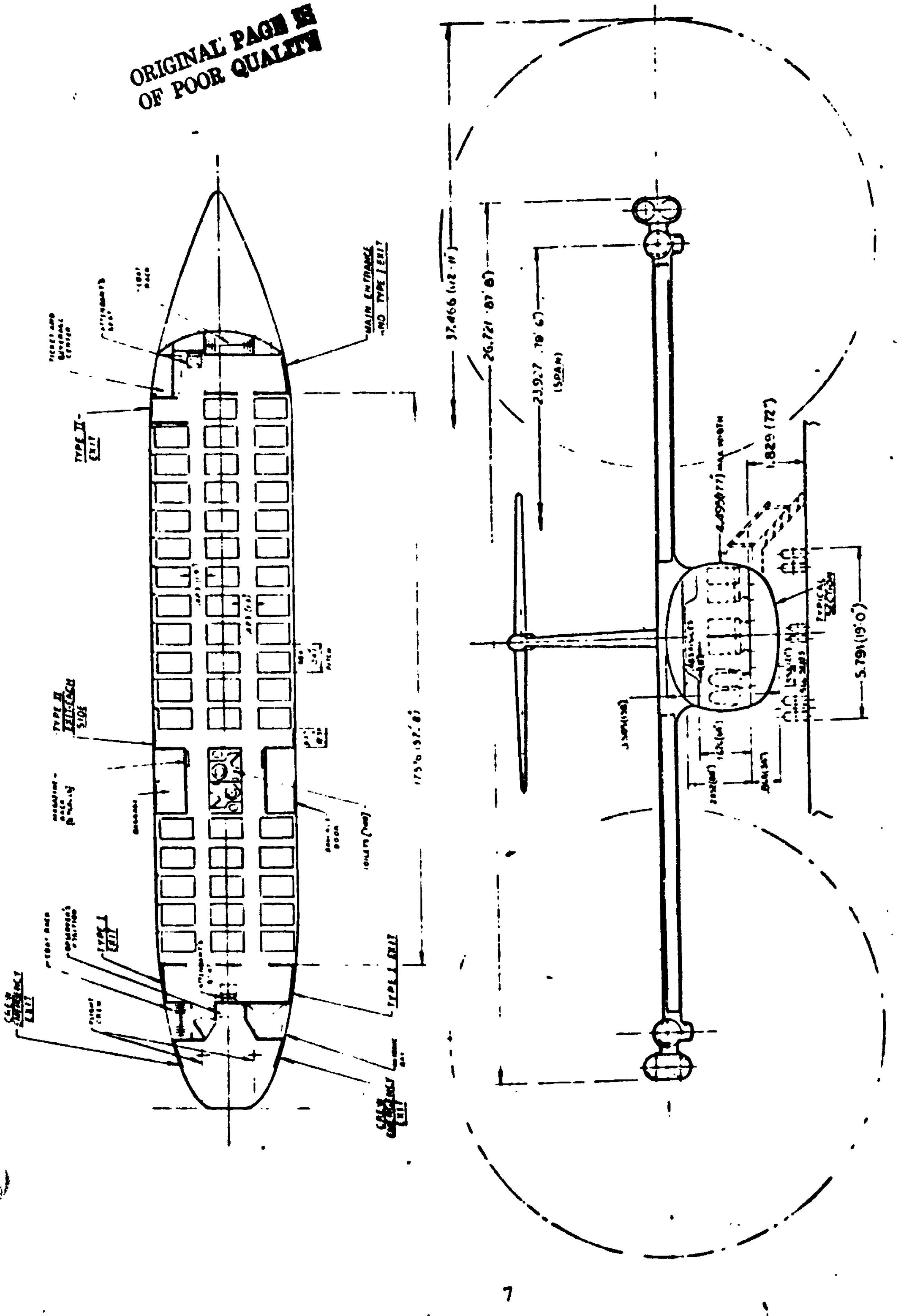


FIGURE 2.3. FRONT VIEW AND CABIN LAYOUT.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

	S.I. UNITS		U.S. UNITS	
WEIGHTS				
DESIGN GROSS WEIGHT	31,068	Kg	68,493	Lbs
WEIGHT EMPTY	20,422	Kg	45,023	Lbs
FUEL WEIGHT	1,554		3,425	Lbs
NUMBER OF PASSENGERS	100		100	
ROTORS				
DISC LOADING	108.0	Kg/m^2	22.12	PSF
DIAMETER	13.53	m	44.4	Feet
SOLIDITY	0.082		0.082	
BLADE NUMBER	3		3	
TWIST	36	Degs	36	Degs
TIPSPEED TAKEOFF/CRUISE	244/171	m/s	800/560	Ft/Sec
POWER				
NUMBER OF ENGINES	4		4	
RATED POWER/ENGINE	2.0775×10^6	Watts	2,786	HP
FUSELAGE				
LENGTH	28.19	m	92.5	Feet
WIDTH (MAX)	4.50	m	14.75	Feet
CABIN LENGTH	17.60	m	15.75	Feet
WING				
AREA	63.63	m^2	684.93	$Feet^2$
SPAN	23.93	m	78.5	Feet
TAPER RATIO	1.0		1.0	
CHORD	2.65	m	8.7	Feet
ASPECT RATIO	9.0		9.0	
AIRFOIL t/c	0.21		0.21	
HORIZONTAL TAIL				
AREA	15.89	m^2	171	$Feet^2$
SPAN	9.05	m	29.7	Feet
TAIL VOLUME RATIO	1.46		1.46	
ASPECT RATIO	5.16		5.16	
VERTICAL TAIL				
AREA	17.19	m^2	185	$Feet^2$
SPAN	4.75	m	15.6	Feet
TAIL VOLUME RATIO	0.145		0.145	
ASPECT RATIO	1.34		1.34	
PERFORMANCE				
NRP CRUISE SPEED	159.5	m/s	310	KTAS
CRUISE ALTITUDE	4,267	m	14,000	Feet
BLOCK TIME	0.82	Hours	0.82	Hours
NOISE				
SIDELINE NOISE - 500 FEET/ TAKEOFF	101.3	PNdB	101.3	PNdB

TABLE 2.1. STOL TILT ROTOR TABLE OF CHARACTERISTICS.

diameter is 44.4 feet (13.53 meters) and the solidity ratio is 0.082. In low speed flight, cyclic pitch control is applied to the rotor to provide control power and trim. The rotor blades are highly twisted (34 degrees) compared with those of a helicopter to provide efficient operation at high advance ratio as well as good static and takeoff performance.

For takeoff and landing, the rotor nacelles, containing the forward rotor transmission, tilt upwards toward the vertical. The engines, however, being mounted outboard of the tilting package remain fixed relative to the wing. This arrangement has the advantages of not requiring qualification of the engines for vertical (or severely inclined) operation and reduces the mass and inertia of the tilt package.

The aircraft has four engines, two at each wing tip. The left and right rotor transmissions are interconnected by a cross shaft which provides torque transfer across the aircraft in the event of an 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 78.5 feet (23.93 meters) measured between rotor axes. The wing is straight and untapered having an aspect ratio of 9. The wing section is a NACA 63_4221 airfoil set at an incidence of two degrees relative to the fuselage reference line.

Full span trailing edge flaperons of 30% chord are provided for use as both flaps and ailerons. The leading edge of the wing carries a full span 15% chord Kruger flap.

The empennage consists of a trimmable horizontal stabilizer whose tail volume ratio is 1.46 mounted atop of the vertical tail of volume ratio 0.145. The T tail configuration minimizes the effect of rotor downwash on the horizontal tail during transition.

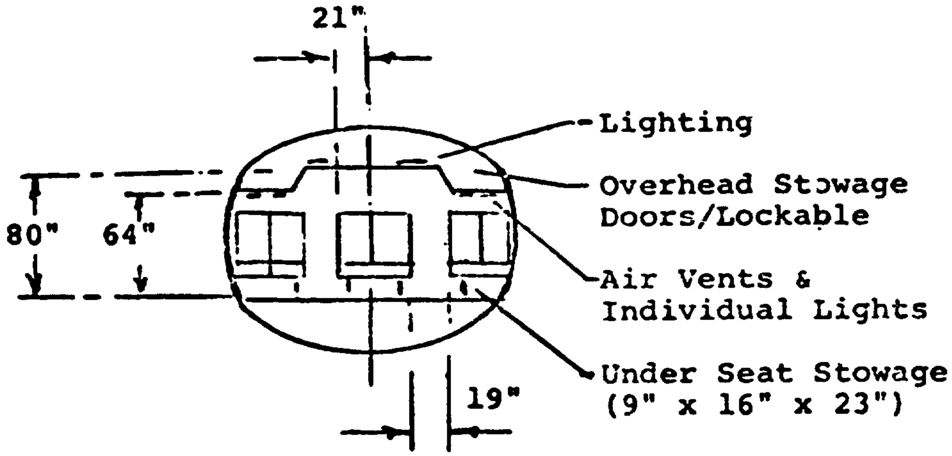
The landing gear is a tricycle configuration to provide good ground handling characteristics and is retractable into the lower fuselage.

Cabin layout and passenger accommodation details are shown in Figures 2.3 and 2.4 respectively. The passenger cabin has two main entrances on the port side of the aircraft. The rear entrance is equipped with an air stair in accordance with NASA guidelines; this 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, while a third 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.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

Cabin



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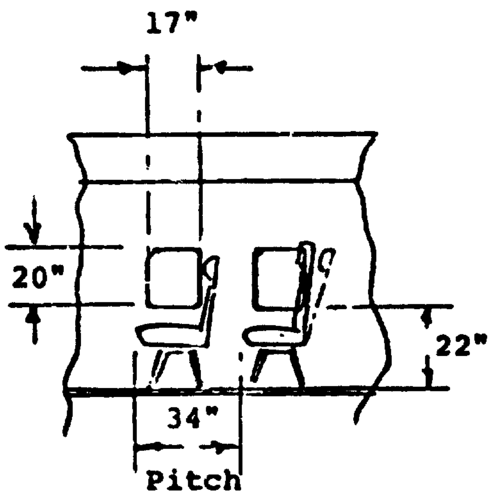


Systems

- 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 2.4. PASSENGER ACCOMMODATIONS.

An air vent, individual light, and a folding table are provided for each passenger in compliance with normal commercial aircraft practice. The cabin has dual 19 inch aisles and the main cabin lights are located over the aisles. A coat rack is provided at each end of the passenger cabin with accommodation for a total of 80 passengers. Two lavatories are provided in the center of the cabin in line with the baggage stowage area. The baggage and toilet facilities are located in the vicinity of the tip path plane of the rotors so that no passenger seats are subjected to excessive noise and vibration. External baggage loading doors are provided for ground crew access.

The beverage storage and service facilities are located aft. This unit is 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.

Seats for the cabin attendants are located against the forward passenger cabin bulkhead close to the exit and against the rear bulkhead close to the rear exit.

The 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 situated 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 access to the avionics/nav-aids bay. The cockpit has a crew emergency exit on each side.

2.2 MISSION PERFORMANCE

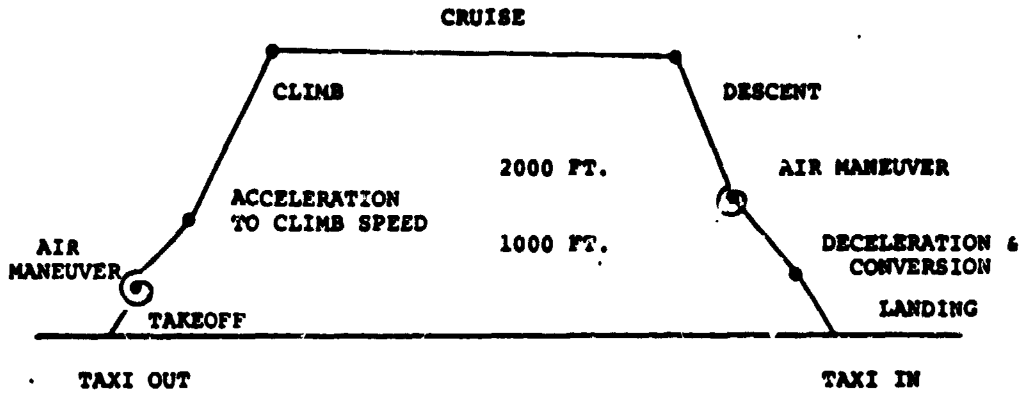
The STOL tilt rotor aircraft has been sized to the mission defined in Table 2.2 and illustrated in Figure 2.5. This aircraft carries 100 passengers over a short haul range of 371 kilometers (200 nautical miles).

A detailed account of the mission time history is provided in Tables 2.3 and 2.4.

The initial phases of the mission including taxi, takeoff, initial air maneuver and conversion to cruise flight require 150 pounds of fuel. The aircraft then climbs to 14,000 feet at an initial rate of climb of 3,105 feet per minute and a final rate of climb of 1,551 feet. At the end of the climb segment the aircraft has burned 538.7 pounds of fuel and has travelled 16.34 nautical miles down range.

The aircraft cruises at 14,000 feet at an initial weight of 67,954 pounds and a true airspeed of 310.7 knots. At the end of the cruise segment the aircraft has used 2,143.2 pounds of fuel and has travelled 173.88 nautical miles. The aircraft speed at the end of cruise is 312.5 knots TAS. The average specific range in the cruise segment is 0.0982 nautical miles per pound of fuel.

The descent to 2,000 feet altitude is initially at 4,388 feet per minute rate of descent falling to 2,137 feet per minute at 2,000 feet altitude. The fuel used at the end of descent amounts to 2,229.5 pounds for a range of 200 nautical miles.



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 speed
Cruise	As Calculated		At constant integral 1000 ft. altitudes (no enroute altitude change)
Descent to 2000 Feet	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 feet. 600 fpm maximum rate of descent below 35 feet.
Taxi In	1 Min.	0	

TABLE 2.2. STOL MISSION PROFILE DEFINITION.

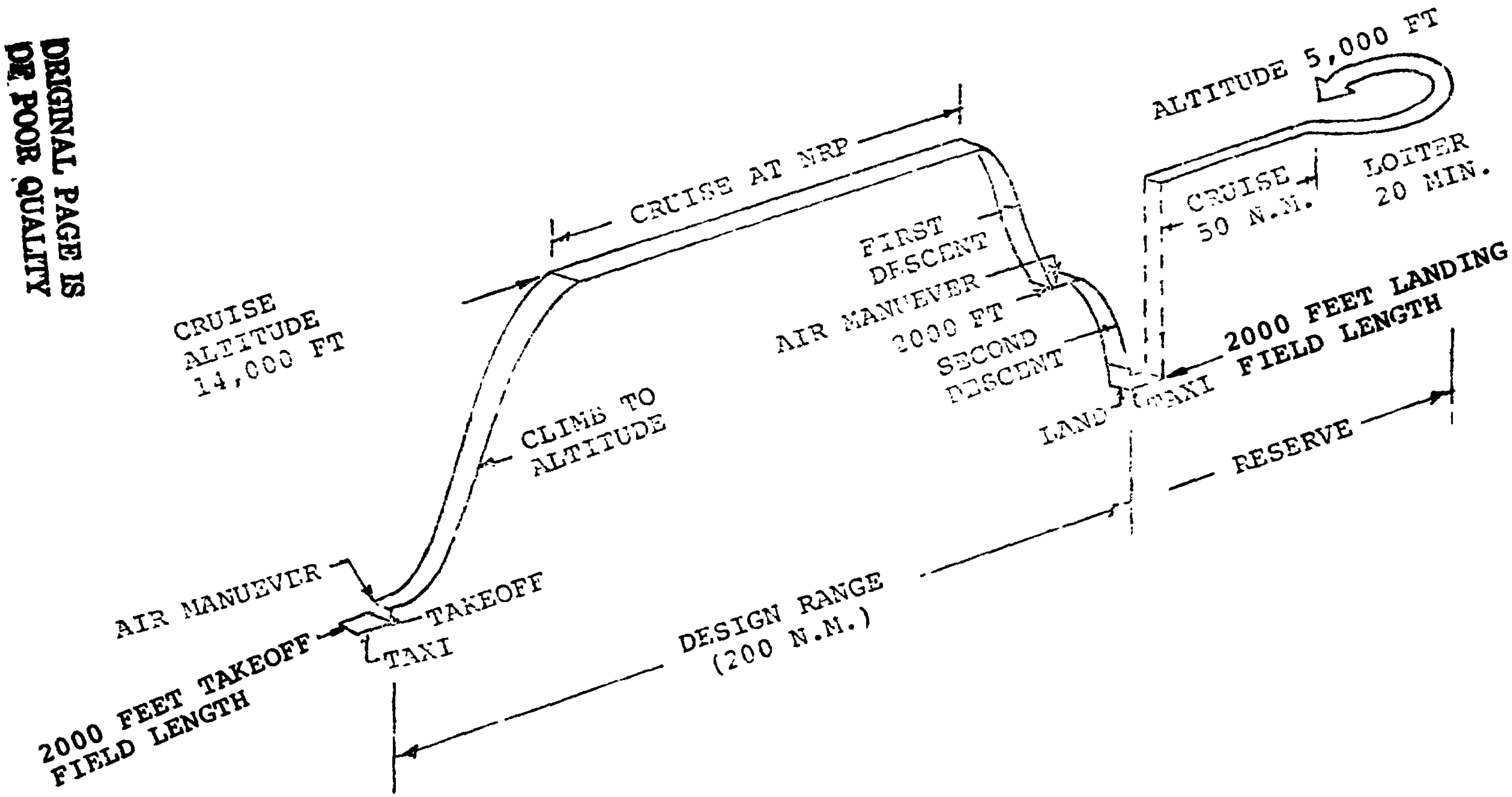


FIGURE 2.5. DESIGN SHORT HAUL MISSION.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

	TIME (HOURS)	DISTANCE (N.MI.)	WEIGHT (LBS)	FUEL (LBS)	V (KNOTS)	R/C (FT/MIN)
	0	0	68,493			
TAXI				9	-	-
	.017	0	68,484			
TAKEOFF				141	-	-
	.050	0	68,343			
CLIMB				389	153/181	3105/1551
	.152	16.34	67,954			
CRUISE				1605	311/313	0
	.657	173.88	66,349			
DESCENT				87	374/258	-4388/-2137
	.746	200	66,262			
AIR MANEUVER				43	143	0
	.771	200	66,219			
DESCENT				8	257/254	-2326/-2134
	.779	201.92	66,211			
LANDING				101	-	-
	.804	201.92	66,110			
TAXI				9	-	-
	.820	201.92	66,101			
RESERVE				1044	150/212	0
	1.373	250	65,057			

TABLE 2.3. STOL TILT ROTOR DESIGN MISSION PERFORMANCE (U.S. UNITS).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

	TIME (HOURS)	DISTANCE (KM)	WEIGHT (Kg)	FUEL (Kg)	V (KNOTS)	R/C m/s
	0	0	31,068			
TAXI				4	-	-
	.017	0	31,064			
TAKEOFF				64	-	-
	.050	0	31,000			
CLIMB				176	153/181	15.8/7.9
	.152	30.26	30,823			
CRUISE				728	311/313	0
	.657	322.03	30,095			
DESCENT				39	374/258	-22.3/-10.9
	.746	370.40	30,056			
AIR MANEUVER				20	143	0
	.771	370.40	30,036			
DESCENT				4	257/254	-11.8/-10.8
	.779	373.96	30,033			
LANDING				46	-	-
	.804	373.96	29,987			
TAXI				4	-	-
	.820	373.96	29,983			
RESERVE				474	150/212	0
	1.373	463.00	29,509			

TABLE 2.4. STOL TILT ROTOR DESIGN MISSION PERFORMANCE (S.I. UNITS).

The final air maneuver or loiter for 1.5 minutes increases the fuel used to 2,273 pounds. The descent to 1,000 feet altitude is done at an average rate of descent of 2,165 feet per minute followed by the descent from 1,000 feet conversion and landing. At touchdown the aircraft has used 2,381.2 pounds of fuel and after a final taxi segment completes the mission for 2,390.6 pounds of fuel.

The overall fuel consumption for the mission is 62.54 passenger miles per gallon of fuel.

Table 2.3 also shows the computation of reserve fuel which is 1,044.6 pounds for a total fuel load of 3,435.2 pounds. The mission block time is 0.820 hours.

2.3 COST ANALYSIS

The design point aircraft initial costs are tabulated in Table 2.6. The flyaway costs have been computed using $90 and $110 per pound of airframe weight. At $90 per pound the aircraft initial cost is $4.62 million and at $110 per pound it is $5.34 million. The basic airframe costs are $3.24 million and $3.97 million respectively with dynamic system, engines and avionics costs amounting to $1.37 million.

The direct operating costs of the aircraft are also shown in Table 2.5 for utilization of 2,500 hours per year and 3,500 hours per year and for both $90 and $110 per pound airframe costs.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

FLYAWAY COSTS

AIRFRAME COST	$90.00/LB	$110.00/LB
AIRFRAME	$3,244,230	$3,965,170
DYNAMIC SYSTEM	557,440	557,440
ENGINES	566,928	566,928
AVIONICS	250,000	250,000
TOTAL	$4,618,598	$5,339,538

DIRECT OPERATING COSTS
DOLLARS/SEAT MILE
BLOCK DISTANCE = 230 S. MILES

UTILIZATION (HRS/YR)	2500		3500	
AIRFRAME COST ($/LB)	90	110	90	110
FLYING OPERATIONS				
FLIGHT CREW	.0048	.0048	.0048	.0048
FUEL AND OIL	.0026	.0026	.0026	.0026
HULL INSURANCE	.0013	.0015	.0009	.0011
TOTAL FLYING OPERATIONS	.0087	.0089	.0083	.0085
DIRECT MAINTENANCE				
AIRFRAME - LABOR	.0014	.0014	.0014	.0014
- MATERIAL	.0012	.0015	.0012	.0015
ENGINES - LABOR	.0006	.0006	.0006	.0006
- MATERIAL	.0006	.0006	.0006	.0006
DYNAMIC SYSTEM - LABOR	.0003	.0003	.0003	.0003
- MATERIAL	.0005	.0005	.0005	.0005
TOTAL DIRECT MAINTENANCE	.0047	.0050	.0047	.0050
MAINTENANCE BURDEN	.0036	.0036	.0036	.0036
TOTAL MAINTENANCE	.0083	.0085	.0083	.0085
DEPRECIATION	.0061	.0070	.0044	.0050
TOTAL DIRECT COSTS	.0231	.0244	.0209	.0220

TABLE 2.5. INITIAL AND DIRECT OPERATING COSTS.

For 2,500 hours per year and $90 per pound the direct operating cost is $2.31 cents per seat mile. This cost breaks down into 0.87 cents per seat mile for flight operation, 0.83 cents per seat mile for maintenance and 0.61 cents per seat mile for depreciation.

At $110 per pound airframe cost, the direct operating cost rises to 2.44 cents per seat mile. The increase of 0.13 cents 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 2.09 cents per seat mile and at $110 per pound airframe cost the direct operating cost is 2.20 cents per seat mile. These reductions in direct operating cost are due to reduced insurance and depreciation costs per seat mile since these annual costs are spread over more passenger miles per year at the higher level of utilization.

Table 2.6 shows similar data for a modified aircraft with increased fuel tankage to provide a 400 nautical mile range capability.

The aircraft flyaway costs rise to $4.63 million at $90 per pound and $5.36 million at $110 per pound due to increased aircraft weight. This aircraft can carry 99 passengers over the 230 statute mile design mission.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

(EXTENDED RANGE VERSION)

	FLYAWAY COSTS	
AIRFRAME COST	$90.00/LB	$110.00/LB
AIRFRAME	$3,258,090	$3,982,110
DYNAMIC SYSTEM	557,440	557,440
ENGINES	566,928	566,928
AVIONICS	250,000	250,000
TOTAL	$4,632,458	$5,356,478

DIRECT OPERATING COSTS
DOLLARS/SEAT MILE
BLOCK DISTANCE = 230 S.MILES

UTILIZATION (HRS/YR)	2500		3500	
AIRFRAME COST ($/LB)	90	110	90	110
FLYING OPERATIONS				
FLIGHT CREW	.0049	.0049	.0049	.0049
FUEL AND OIL	.0026	.0026	.0026	.0026
HULL INSURANCE	.0013	.0015	.0009	.0011
TOTAL FLYING OPERATIONS	.0088	.0090	.0084	.0086
DIRECT MAINTENANCE				
AIRFRAME - LABOR	.0014	.0014	.0014	.0014
- MATERIAL	.0012	.0015	.0012	.0015
ENGINES - LABOR	.0006	.0006	.0006	.0006
- MATERIAL	.0006	.0006	.0006	.0006
DYNAMIC SYSTEM - LABOR	.0003	.0003	.0003	.0003
- MAT'L	.0005	.0005	.0005	.0005
TOTAL DIRECT MAINTENANCE	.0048	.0050	.0048	.0050
MAINTENANCE BURDEN	.0036	.0036	.0036	.0036
TOTAL MAINTENANCE	.0084	.0086	.0084	.0086
DEPRECIATION	.0062	.0071	.0044	.0051
TOTAL DIRECT COSTS	.0233	.0247	.0212	.0223

TABLE 2.6. INITIAL AND DIRECT OPERATING COSTS (EXTENDED RANGE).

Direct operating costs per seat mile and seat kilometer as a function of block distance are shown in Figure 2.6 for the specified combinations of aircraft utilization and airframe costs. Figure 2.6 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 2.6, 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, seat mile costs show a continuing decrease beyond 230 statute miles because the increase in block speed at the longer ranges more than offsets the effect of fewer available seats. The increased fuel requirements for 460 statute mile range reduce the available seats to 88.

2.4 NOISE

One of the most significant factors in community acceptance of a V/STOL vehicle is the annoyance level caused by external noise in the terminal area. This factor is difficult to assess since the level of annoyance appears to depend upon several parameters (e.g., overall SPL, frequency distribution and the exposure time involved).

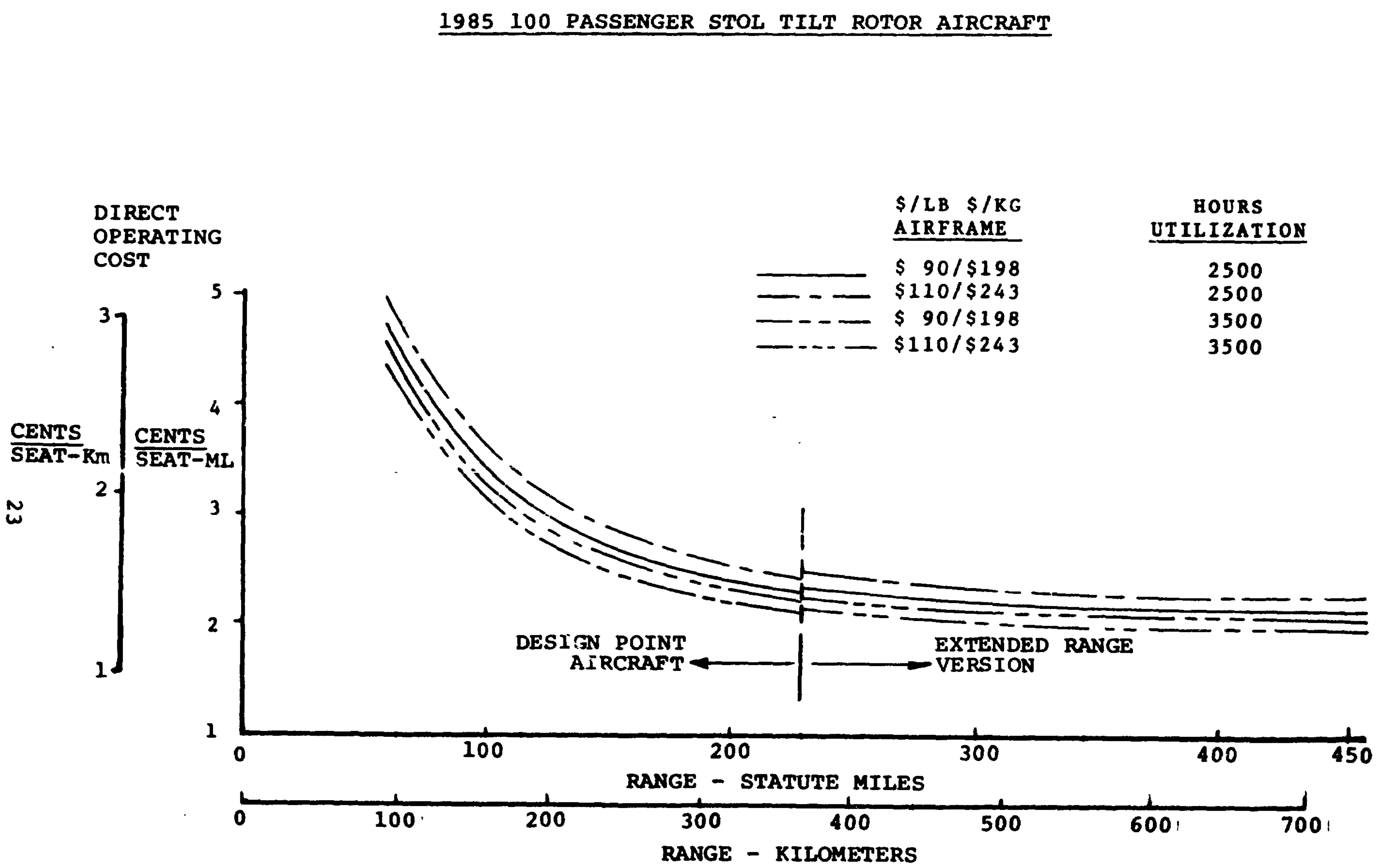


FIGURE 2.6. EFFECT OF OPERATING RANGE ON DIRECT OPERATING COST

The noise spectra produced by the STOL design in its static thrust condition at the start of the takeoff roll are shown in Figure 2.7. This figure also shows the contributions of each noise component to the overall sound pressure level at various octave band frequencies. At frequencies above 100 Hertz, the broadband noise predominates and at low frequencies the major contribution results from rotational noise. These noise spectra include the use of engine inlet noise suppression to prevent the inlet noise from dominating the high frequency band. The attenuation used is shown in Figure 2.8 and results from acoustical suppression linings in the engine inlet.

The sound pressure levels shown in Figure 2.7 result in a perceived noise level of 101.3 PNdB at 500 feet sideline distance in the static thrust condition.

A second and perhaps more effective way of adjudicating noise annoyance is to consider the noise footprint due to takeoff and landing operation. These data provide an indication of the ground area which will be subjected to a given perceived noise. These curves are shown in Figure 2.9 for both takeoff and landing conditions. The ground perceived noise level footprints show that the worst noise levels are observed on the flight ground track. The 95 PNdB contour encloses an area of 0.30 square kilometers (.115 square miles) on takeoff and 0.36 square kilometers (.14 square miles) on landing.

The time histories of perceived noise levels for various observer locations along the ground track are shown in Figures 2.10 and 2.11 for typical takeoff and landing profiles. The

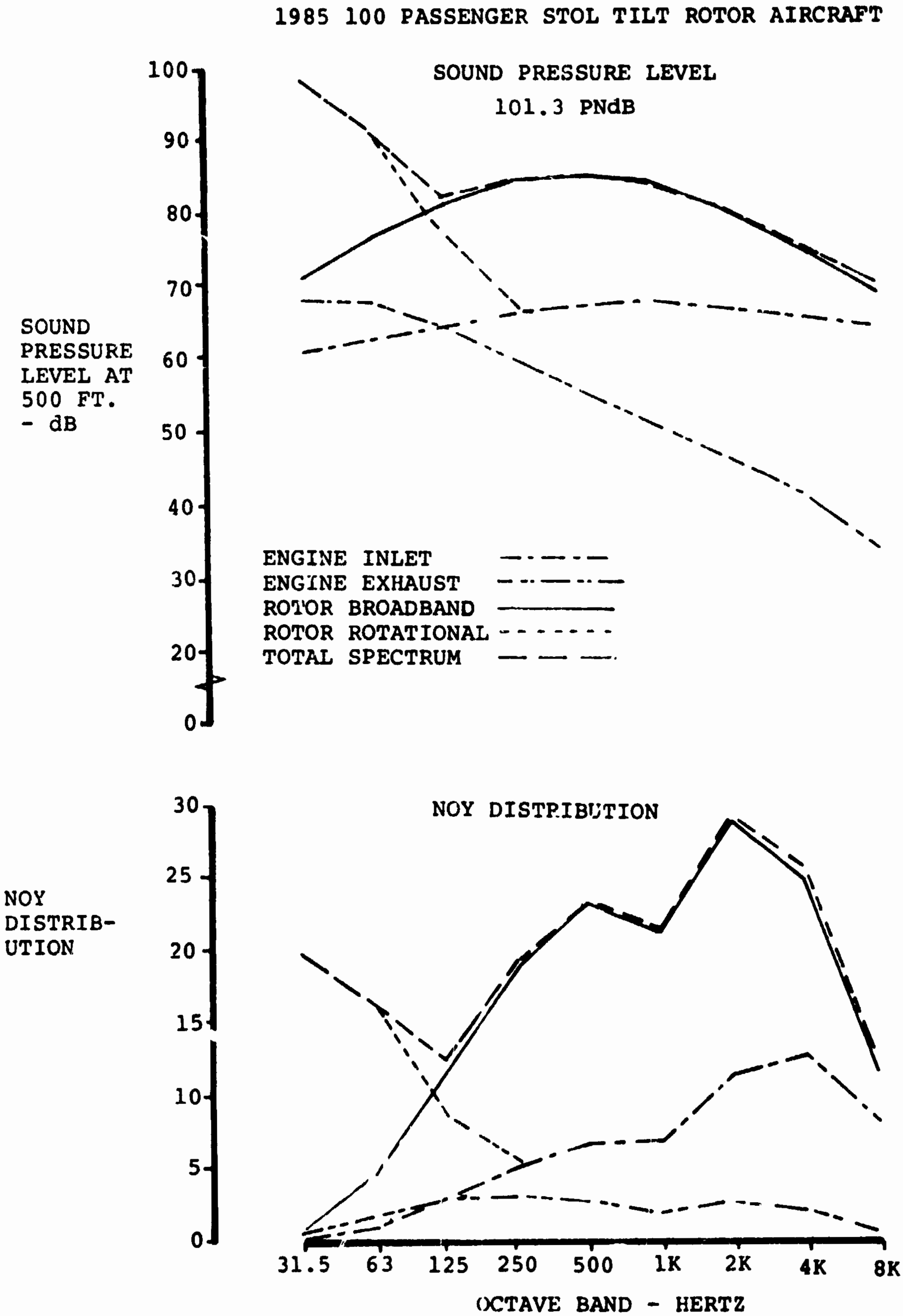


FIGURE 2.7. TAKEOFF NOISE SPECTRUM AND NOY DISTRIBUTION.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

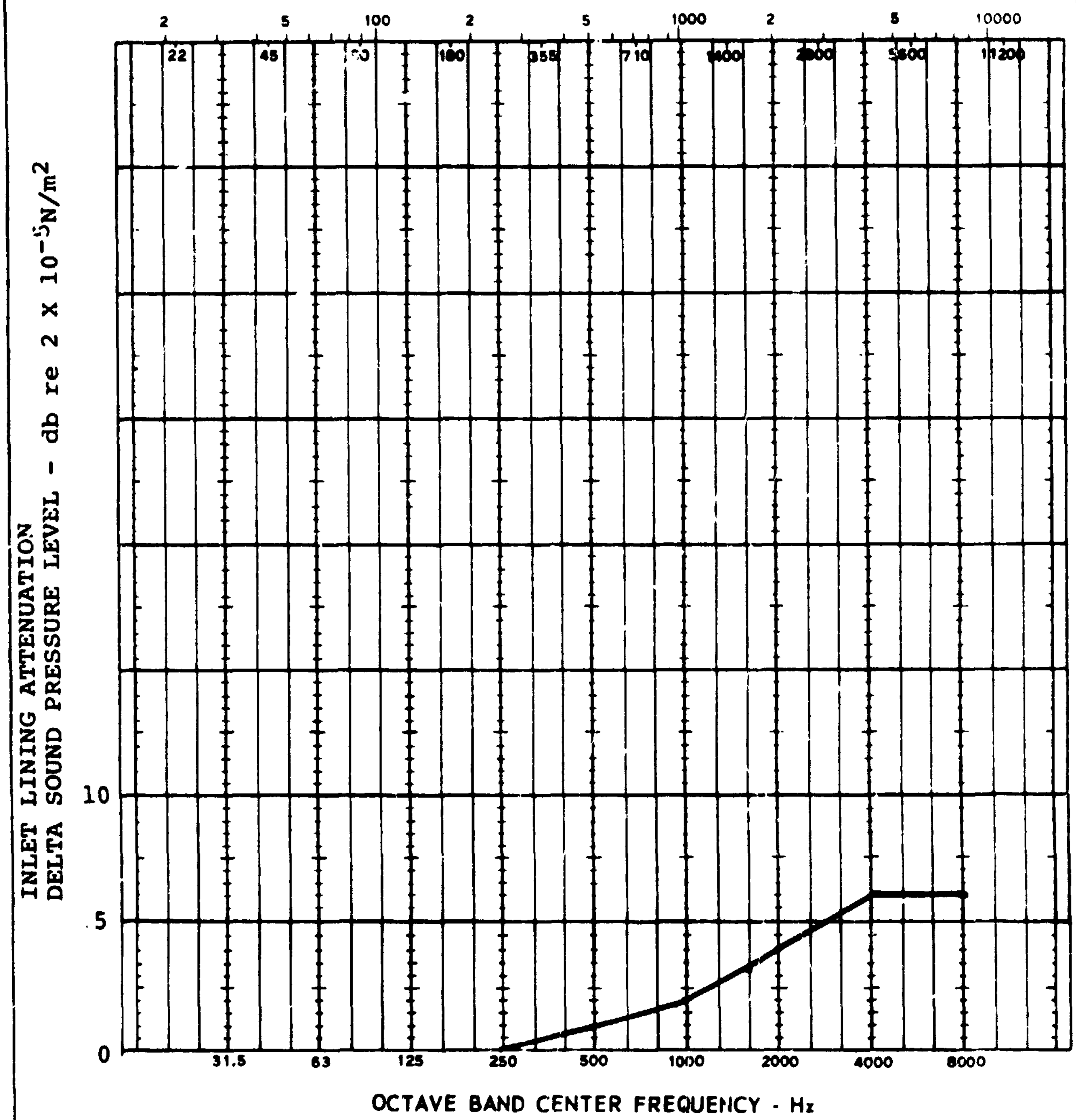
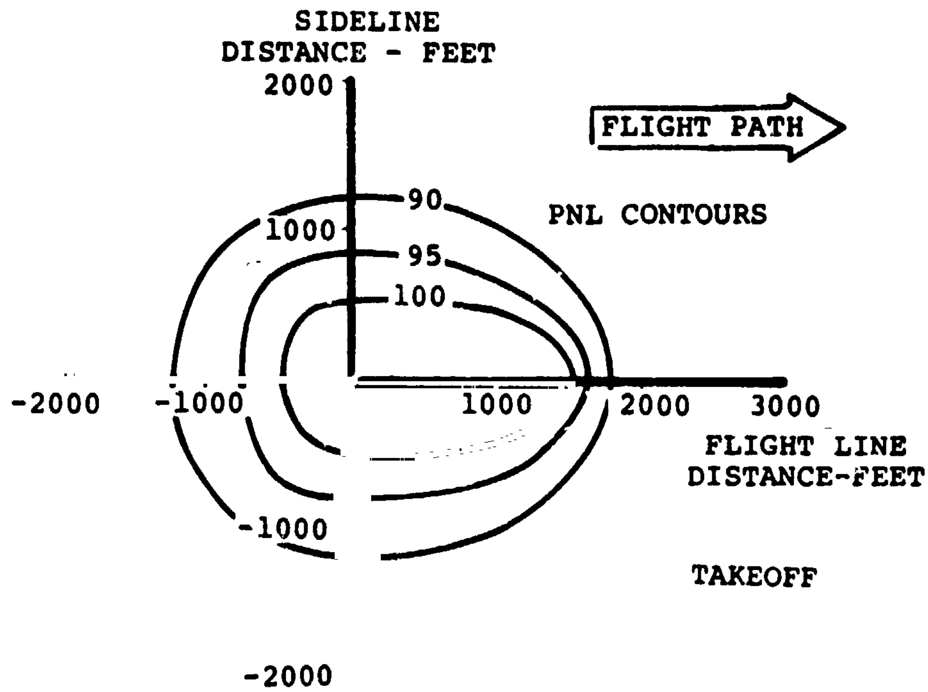


FIGURE 2.8. ENGINE INLET NOISE SUPPRESSION

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT



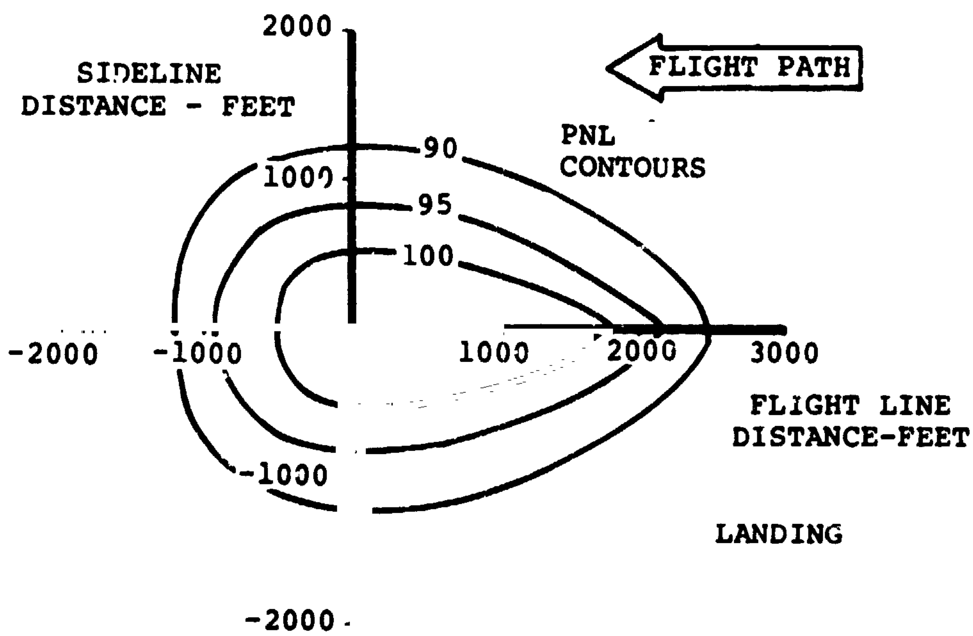


FIGURE 2.9. TAKEOFF AND LANDING PNL CONTOURS.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

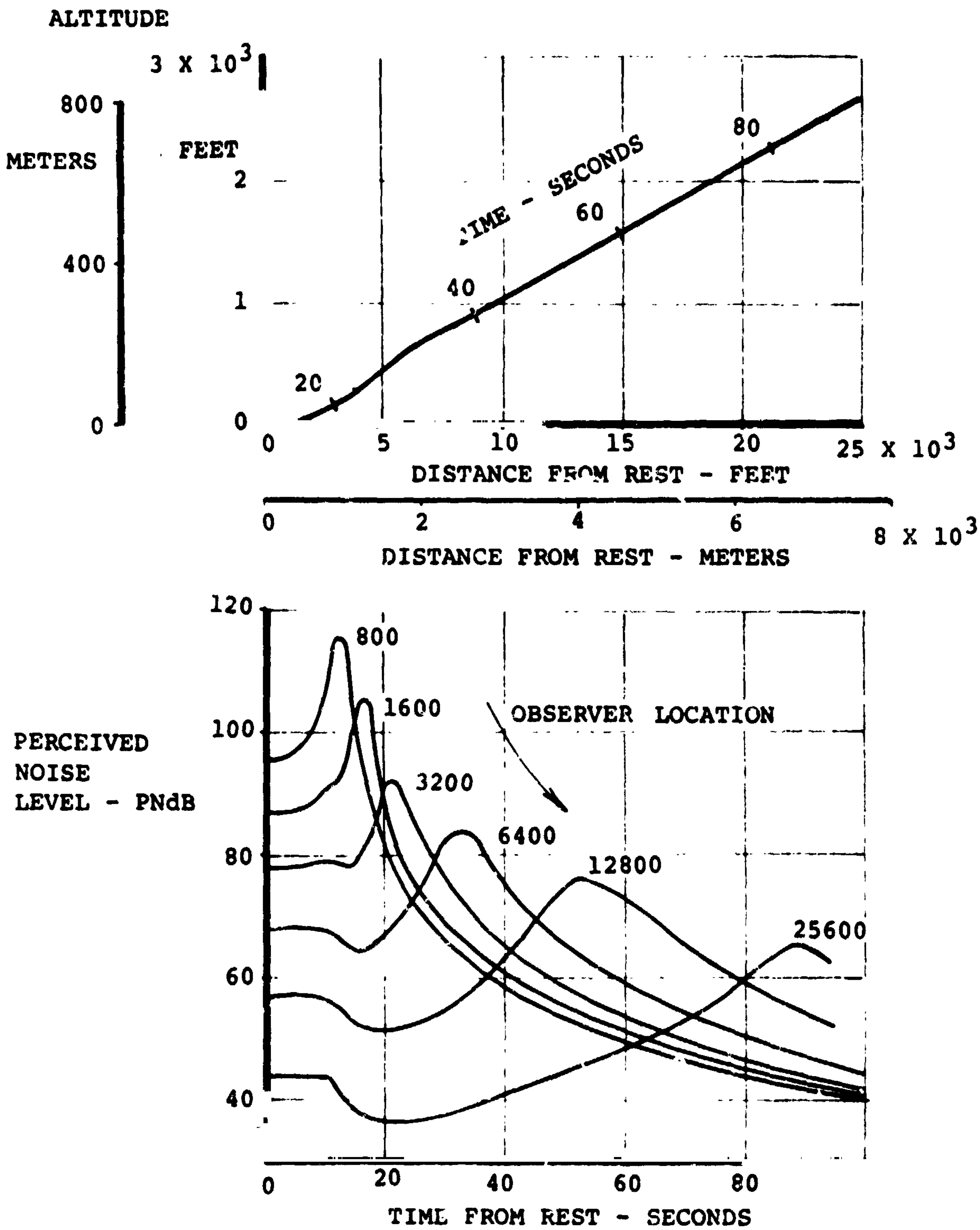


FIGURE 2.10. TAKEOFF PERCEIVED NOISE LEVEL HISTORY.

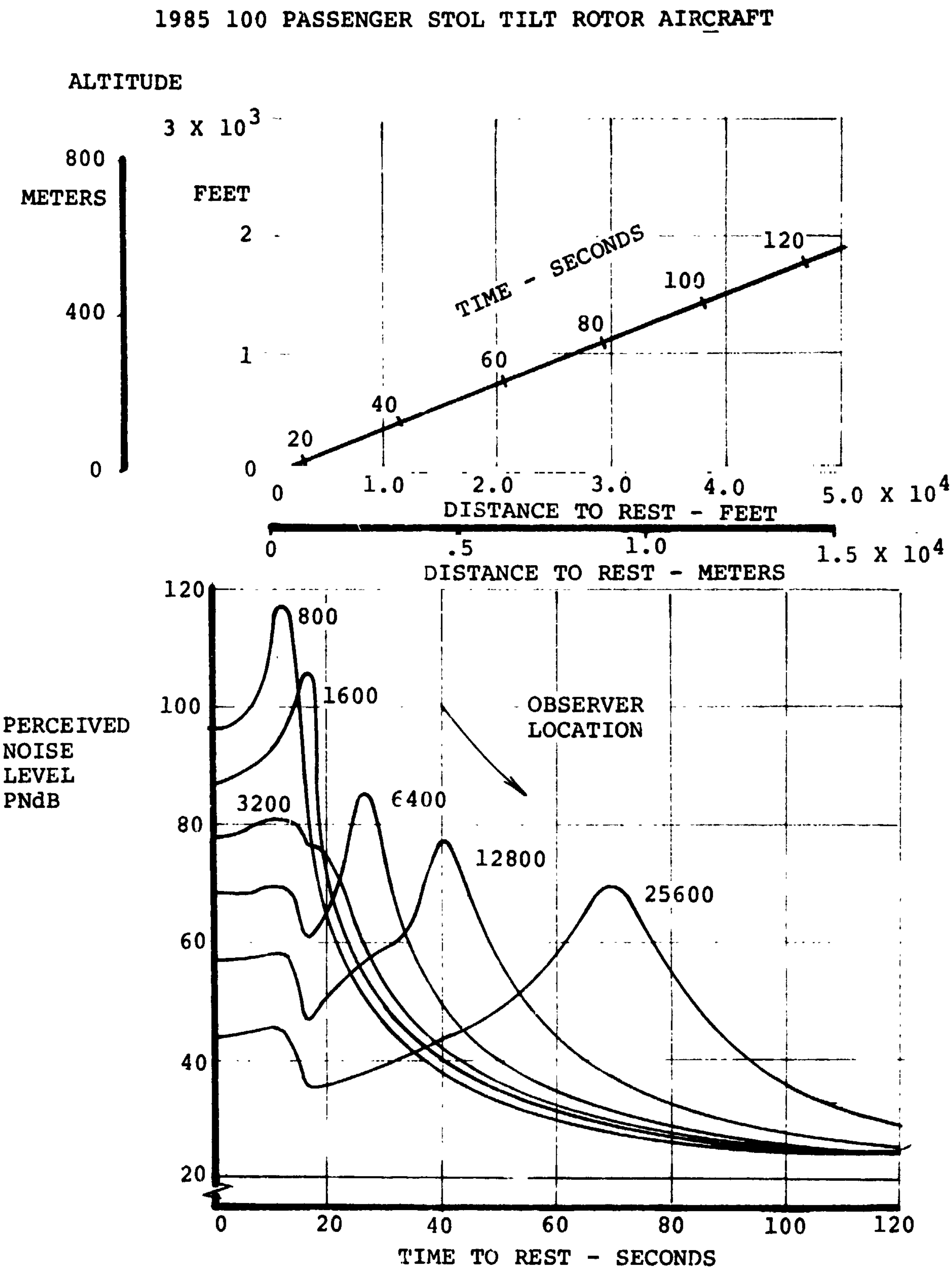


FIGURE 2.11. LANDING PERCEIVED NOISE LEVEL HISTORY.

flight profiles used in these computations are also shown. These data allow exposure time to various noise levels to be considered in evaluating community acceptance.

3.0 DESIGN DATA

This section of the report gives a detailed summary of the design point STOL tilt rotor performance capabilities, component weights, flying qualities in transition and cruise modes and some details of aircraft subsystems.

3.1 VEHICLE PERFORMANCE

The aircraft has been sized to carry 100 passengers over the 200 nautical mile (371 kilometers) mission. A summary of the performance of the aircraft while flying the mission has been given in Section 2.2 of this report. In this section of the report, a detailed assessment is given of the performance in takeoff, climb, cruise and landing.

3.1.1 Takeoff Performance

The ground rules for the calculation of the takeoff performance of the STOL tilt rotor aircraft are summarized in Table 3.1.

The takeoff technique assumed, in order to calculate the takeoff performance, consists of (i) holding the aircraft at the start of the runway, brakes on, while running up the engines to takeoff power, (ii) releasing the brakes and accelerating on the ground to a predetermined rotation speed while maintaining constant power, (iii) applying full up elevator and longitudinal cyclic to rotate the aircraft at a maximum fuselage rotation rate of 8 degrees per second until the fuselage is at 10 degrees angle of attack, and (iv) holding the same angle of attack until the threshold height of 35 feet is attained.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

TAKEOFF (@ S.L 90°F)

ACCELERATION: ROLLING FRICTION COEFFICIENT, $\mu = .03$
ALL ENGINES OPERATING

LIFT OFF SPEED:

$V_{LOF} \geq 1.05$ (V_{MCA} AND V_{MCG})

ROTATION:

8 DEGREES/SEC MAXIMUM

CLIMBOUT CONDITIONS TO OBSTACLE: (OBSTACLE HEIGHT = 35 FT)

AEO: (GEAR DOWN) CLIMB GRADIENT $\geq$ 6.7%(15:1)

OEI: CLIMB GRADIENT $\geq$ 6.7%(15:1) (GEAR UP)

$\alpha \leq (\alpha_{STALL} - 10°)$ (GEAR DOWN)

SPEED AT OBSTACLE:

$V_2 \geq V_{L.O} \geq 1.15\ V_{MCA} \geq V_{MCA} + 10$ KTS

FACTORS FOR FIELD LENGTH:

1.15 FOR AEO
1.00 FOR ENGINE CUT AT LIFT OFF
1.00 FOR ACCELERATE-STOP

LANDING (@ S.L 90°F)

APPROACH SPEED: (SPEED AT 35 FT OBSTACLE)

$V_{AP} \geq 1.15\ V_{MCA} \geq V_{MCA} + 10$ KTS
$\alpha \leq (\alpha_{STALL} - 10°)$

LANDING CLIMBOUT GRADIENT:

AEO: (GEAR DOWN) CLIMB GRADIENT = 3.33% (30:1)

OEI: (GEAR UP) CLIMB GRADIENT = 3.33% (30:1)

FLIGHT PATH FROM 35 FEET:

MAXIMUM R/D AT 35 FT = 800 FPM
MAXIMUM R/D AT TOUCHDOWN = 300 FPM

ROTATION: 8 DEGREES/SEC MAXIMUM

DECELERATION: 1 SECOND TIME DELAY
BRAKING FRICTION COEFFICIENT, $\mu = .25$
0.4 g MAXIMUM DECELERATION ON GROUND

FACTOR FOR FIELD LENGTH:

LANDING DISTANCE FROM 35 FT DIVIDED BY 0.7

TABLE 3.1. TAKEOFF AND LANDING GROUNDRULES.

The field length calculation required the evaluation of takeoff distance in the event of cutting one engine at the point of lift-off and the evaluation of the accelerate-stop distance.

The method of calculation of the engine out case assumes that a 9% (nine percent) power increase per remaining engine is available immediately.

To calculate the accelerate-stop distance, the distance to achieve lift-off speed, distance travelled during a one second delay at lift-off speed and the distance to stop at a constant deceleration of 0.35 g are summed. The lift-off speed is a variable dependent upon the nacelle incidence.

All takeoff calculations were made with a flap deflection of 40 degrees and with Kruger flaps deployed.

The rotation speed of 120 feet per second is determined by the requirement of sufficient dynamic pressure to enable rotation of the aircraft without having to apply too large an amount of cyclic control on the rotors which would cause excessive blade loads. All other critical speeds are considerably lower than the speed dictated by the rotation requirement.

The nacelle incidence required for takeoff is a compromise between the need for rapid forward acceleration and the requirement of rotating the aircraft once rotation speed is attained.

Figure 3.1 shows the effect of rotation speed on the takeoff performance of the aircraft. In addition to the variation of normal takeoff distance (to 35 feet height) the graph also shows the variation of required field length. The factors that determine the takeoff field length for the configuration considered in Figure 3.1 are (i) takeoff distance when one engine is cut at lift-off, and (ii) the accelerate-stop distance.

The former is predominant at speeds below 121 feet per second while above this speed the accelerate-stop distance is greater. In all cases, the field length is greater than 115% of the takeoff distance with all engines operating.

The variation of takeoff distance with atmospheric conditions is shown in Figure 3.2 for the case of all engines operating and the case when one engine is cut at lift-off. The degradation of engine performance and the effect of reduced air density on rotor and airframe aerodynamics are both reflected in the deterioration of takeoff performance as either altitude or ambient temperature is increased.

On a standard day the aircraft, at design gross weight, can operate from a 2,000 foot field at altitudes up to about 3,000 feet and on a hot day (standard plus 31 degrees F) at altitudes up to 1,000 feet above sea level.

Figure 3.3 shows that the takeoff distance is sensitive to the gross weight of the aircraft and becomes increasingly more sensitive as the design gross weight is approached and exceeded.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

TAKEOFF AT DESIGN GROSS WEIGHT
TAKEOFF AT SL/90°F (32° C)
FLAP SETTING = 40°
KRUGER FLAP DEPLOYED
NACELLE INCIDENCE = 66°

DGW = 68,493 LBS/31,068 Kg

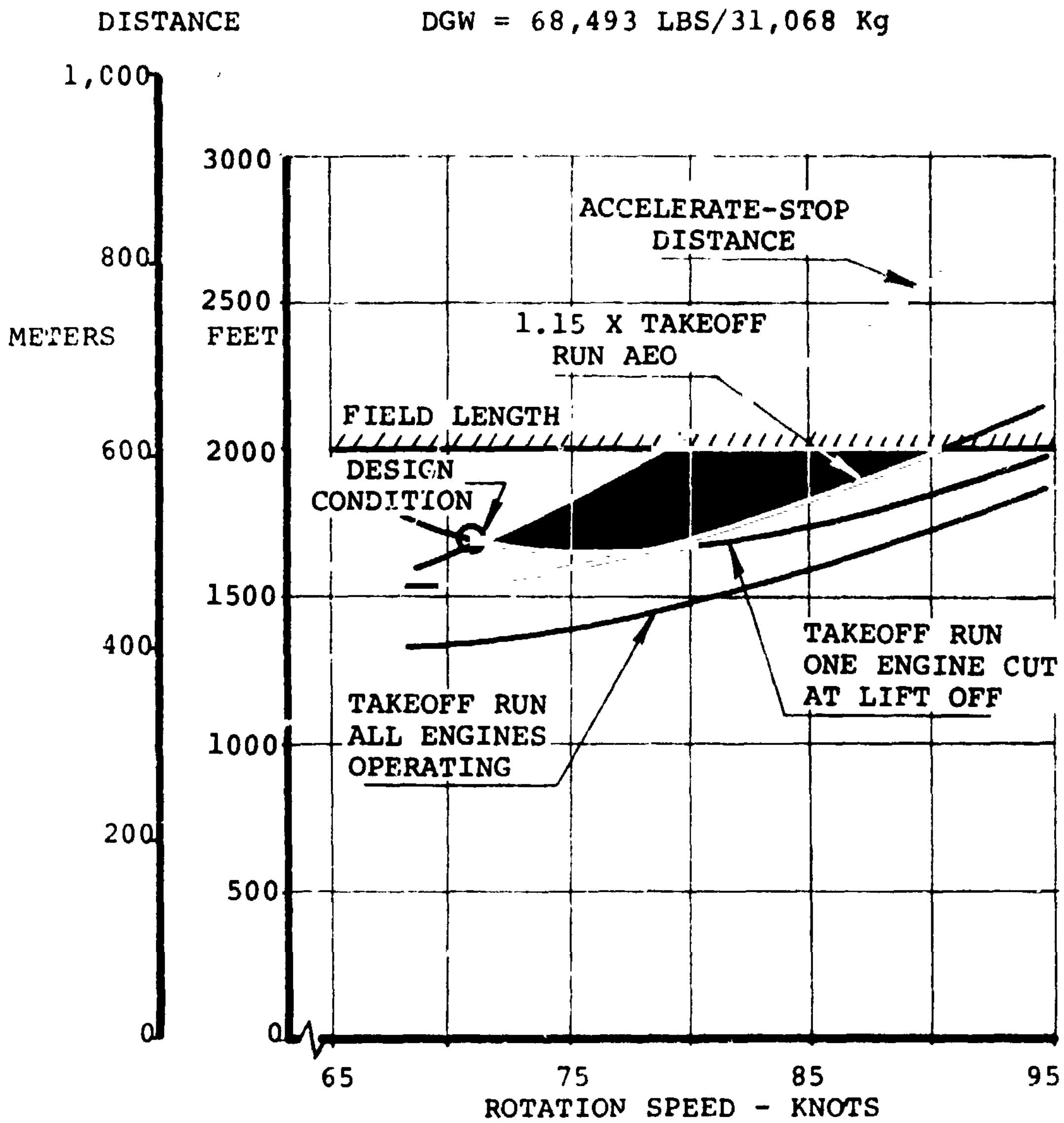


FIGURE 3.1. EFFECT OF ROTATION SPEED ON TAKEOFF PERFORMANCE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

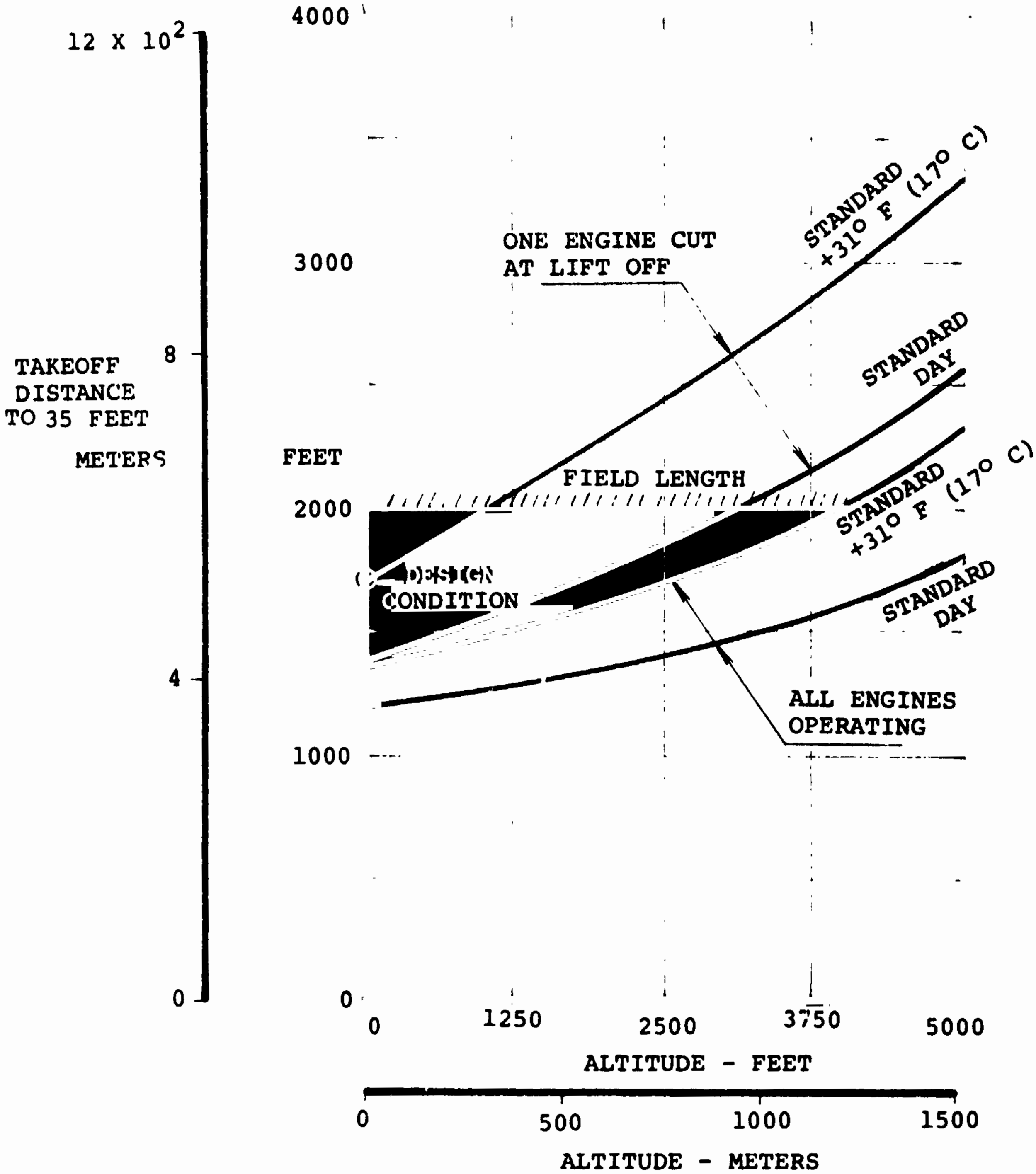


FIGURE 3.2. EFFECT OF ALTITUDE AND AMBIENT TEMPERATURE ON TAKEOFF PERFORMANCE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

TAKEOFF AT SL/90° F (32° C)
FLAP SETTING = 40°
KRUGER FLAP DEPLOYED
NACELLE INCIDENCE = 66°

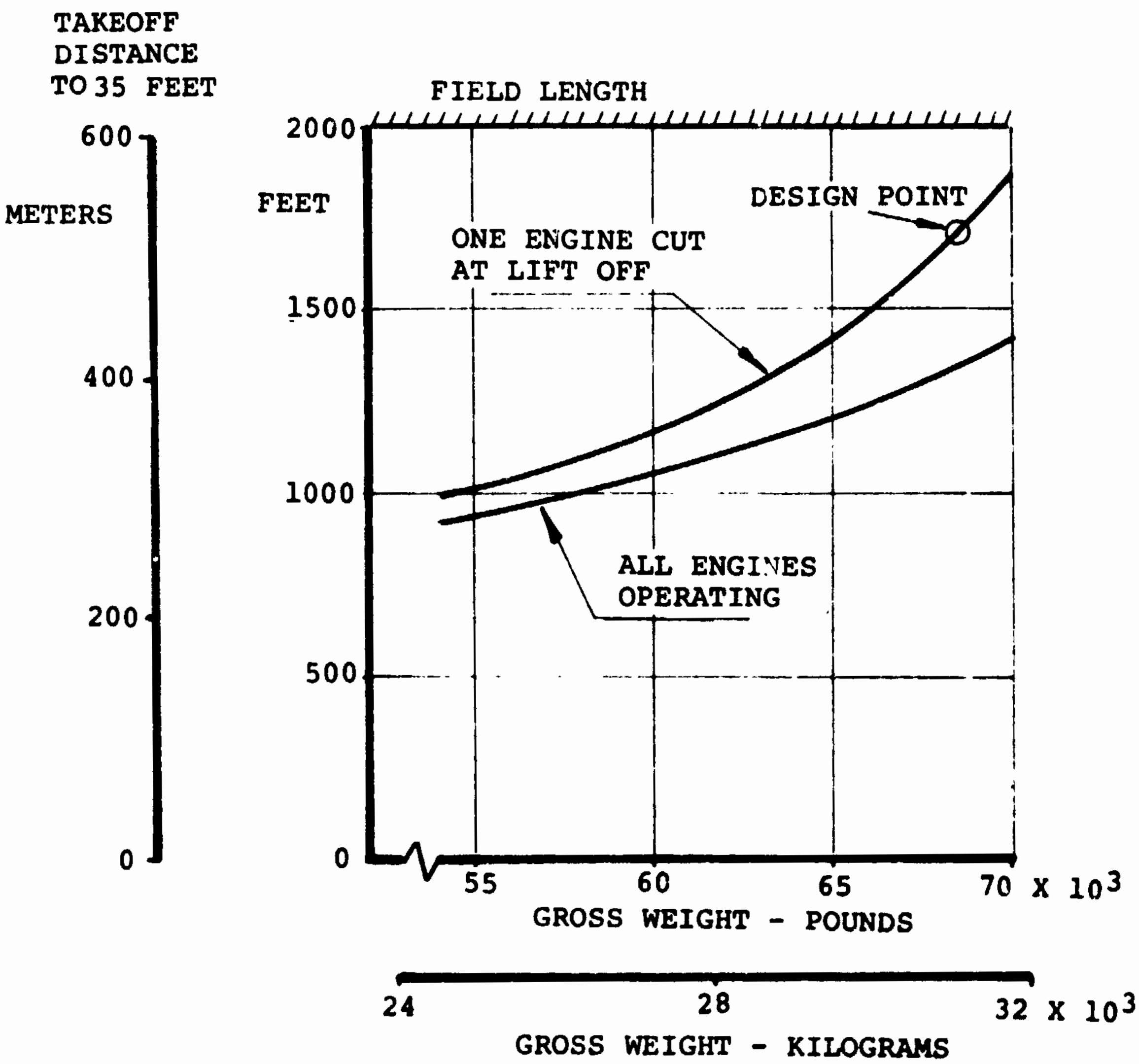


FIGURE 3.3. EFFECT OF GROSS WEIGHT ON TAKEOFF PERFORMANCE.

The effect of varying nacelle incidence on takeoff performance is illustrated in Figure 3.4. For the range of incidences shown, the normal takeoff distance decreases monotonically as the incidence is increased. This trend, however, would not continue indefinitely and, in fact, above an incidence of 70 degrees the takeoff distance must increase rapidly with incidence. This is because at high values of incidence (near 90 degrees say) the rotor thrust is nearly vertical and the longitudinal force component tends to become insufficient to accelerate the aircraft to the lift-off speed. The upper curve in Figure 3.4 shows the same type of trend and shows a minimum distance when the nacelle incidence is about 66 degrees.

Figure 3.5 shows the time history of three important takeoff parameters, speed, distance and height above the runway. The dashed portion of the graphs indicates the time history when one engine is cut at the lift-off point.

3.1.2 Transition Performance

Performance in transition is strongly dependent on the variation of rotor angle of attack with speed. The schedule of nacelle incidence with speed, in turn, depends upon the details of the control system.

Since a detailed design of the transition control system is beyond the scope of this conceptual study, the variation of power required with speed for a typical transition schedule (see Figures 3.23 and 3.24 in Section 3.3 of this report) has

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

TAKEOFF AT DESIGN GROSS WEIGHT
TAKEOFF AT SL/90° F
40° FLAP SETTING
120 FT/SEC ROTATION SPEED
KRUGER FLAP DEPLOYED
DGW = 68,493 LBS/31,068 Kg

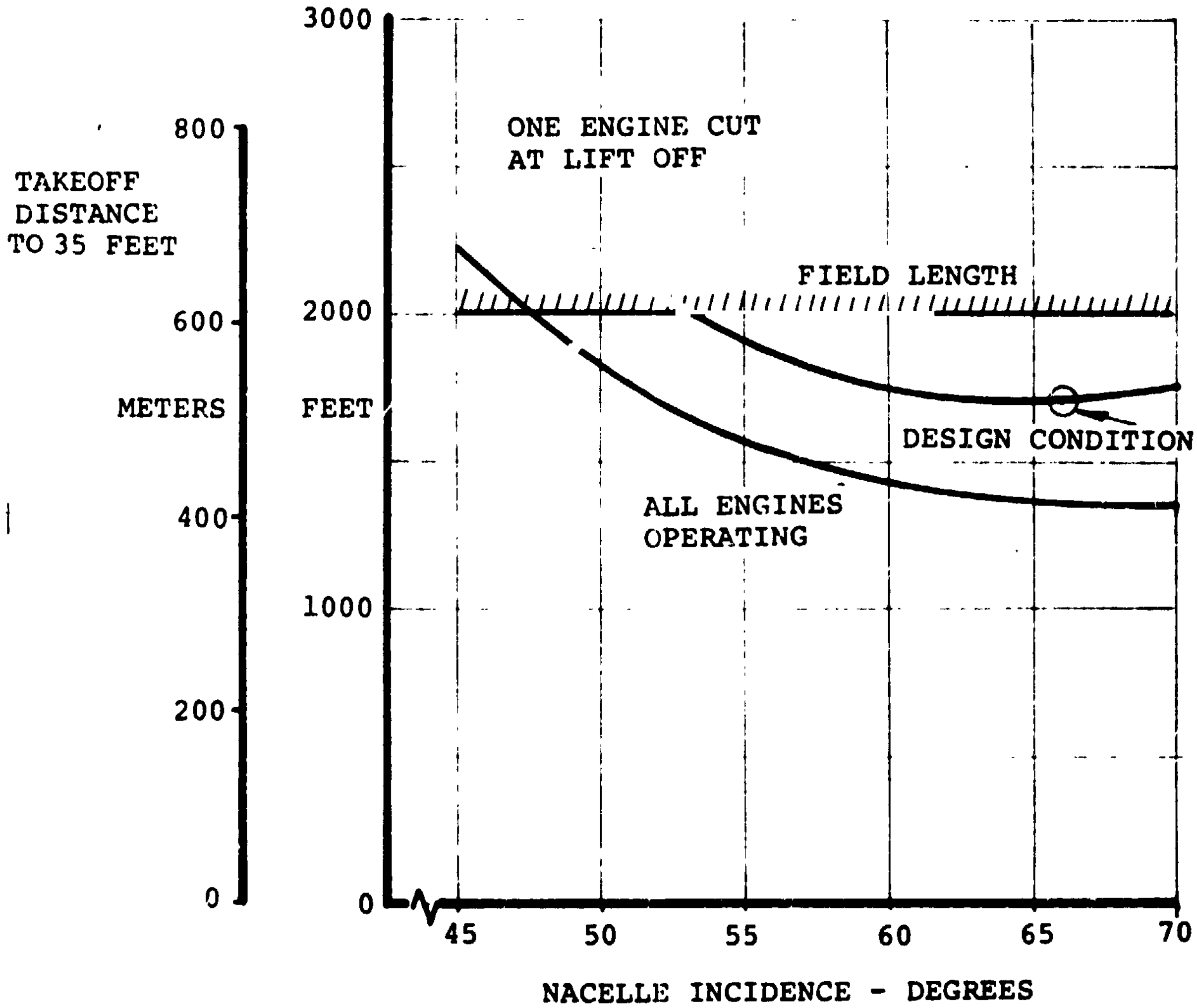


FIGURE 3.4. EFFECT OF NACELLE INCIDENCE SETTING ON TAKEOFF DISTANCE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

TAKEOFF AT SL/90° F (32° C)
FLAP SETTING = 40°
KRUGER FLAP DEPLOYED
TAKEOFF AT DGW = 68,49. LBS/31,068 Kg
NACELLE INCIDENCE = 66°

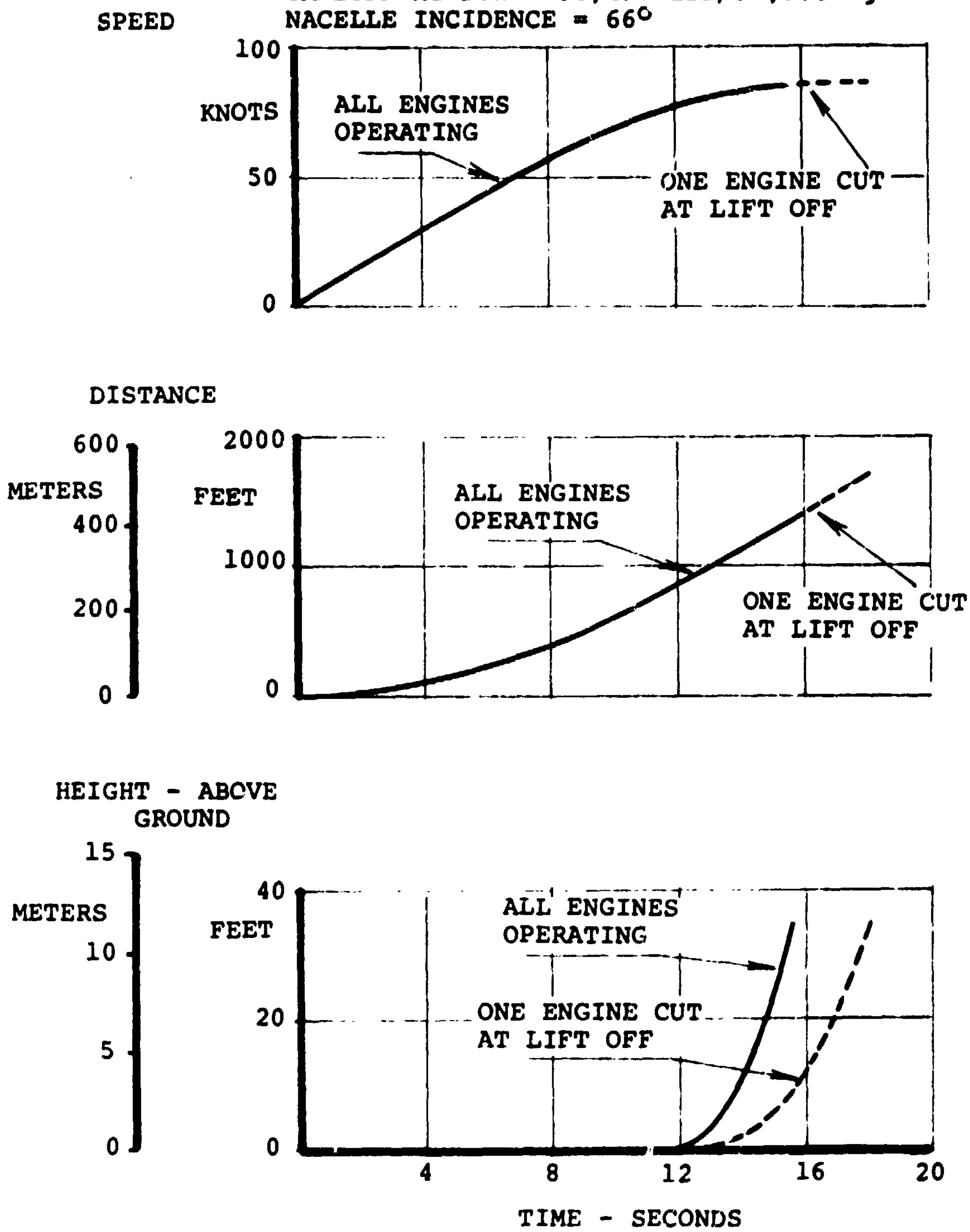


FIGURE 3.5. TAKEOFF TIME HISTORY.

been plotted in Figure 3.6. Superimposed on this curve is the transmission torque limit line. The variation of power along this line is due to the variation of rotor speed from 341 RPM, before, and 244 RPM after transition.

3.1.3 Climb Performance

The variation of rate of climb with altitude is shown in Figure 3.7. Climb rates with all engines operating (AEO) and with one engine inoperative (OEI) are shown for both the design gross weight (DGW) and the operating weight empty (OWE).

At the design gross weight, the rate of climb AEO is 2,958 feet per minute at sea level decreasing to a value of 1,625 feet per minute at the 14,000 feet cruise altitude. Extrapolation of the appropriate curve indicates a service ceiling of about 27,000 feet. Below about 400 feet altitude, the climb performance is limited by the torque capability of the transmission but at higher altitudes the limiting factor is the engine power available at the climb power setting.

At the operating empty weight, the rate of climb AEO varies from 4,887 feet per minute at sea level to 3,354 feet per minute at cruise altitude. Again at the lower altitudes (below about 2,000 feet), the climb rate is limited by the transmission torque capability. At this light weight, the power to weight ratio is relatively large and a very high climb angle would be possible. To avoid excessively large

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

SEA LEVEL/STANDARD DAY

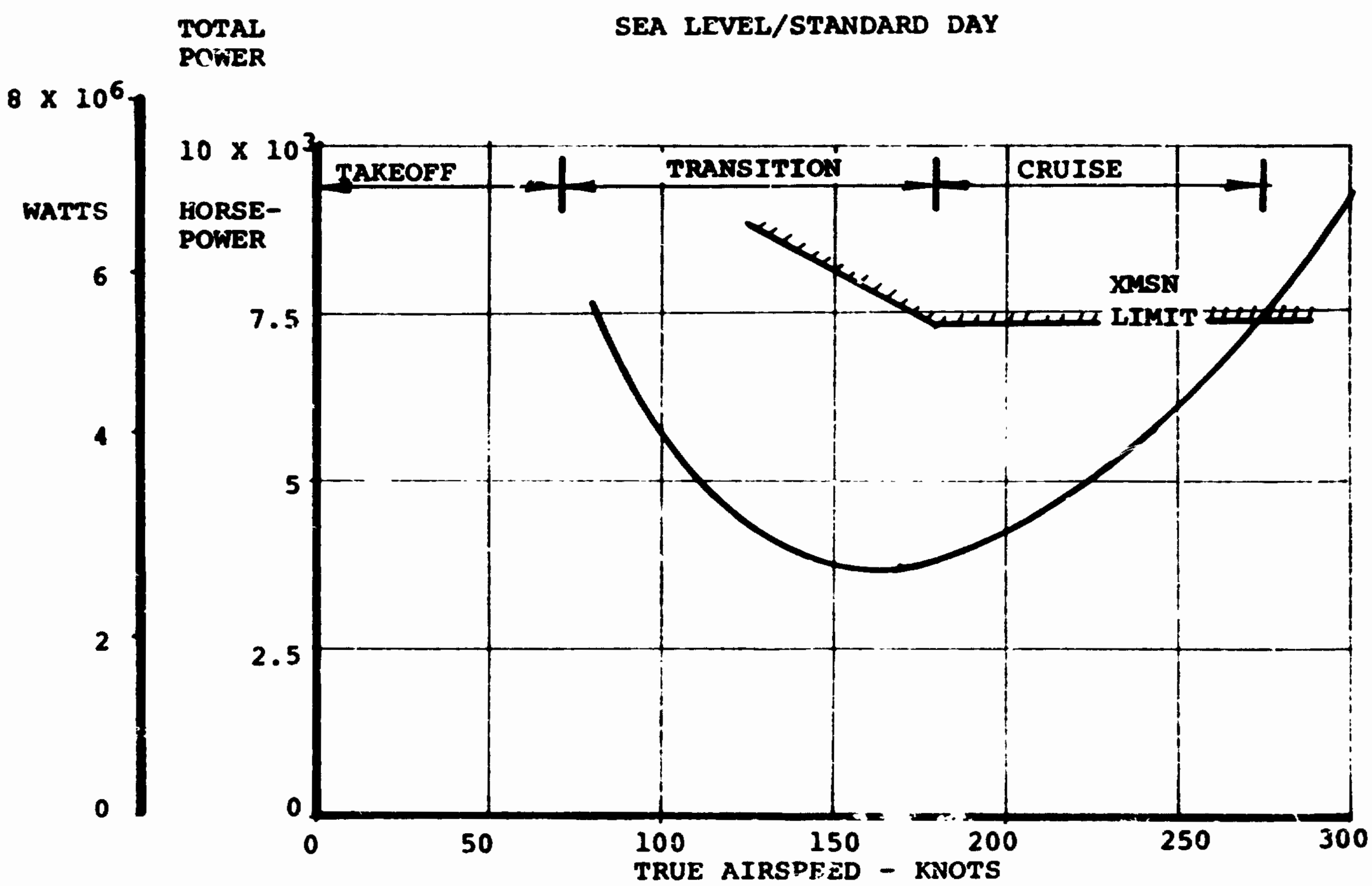


FIGURE 3.6. POWER REQUIRED FOR EQUILIBRIUM FLIGHT IN TRANSITION AND CRUISE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

STANDARD DAY - TAKEOFF RPM - TAKEOFF POWER

DGW = 68,493 LBS/31,068 Kg
OWE = 47,068 LBS/21,350 Kg

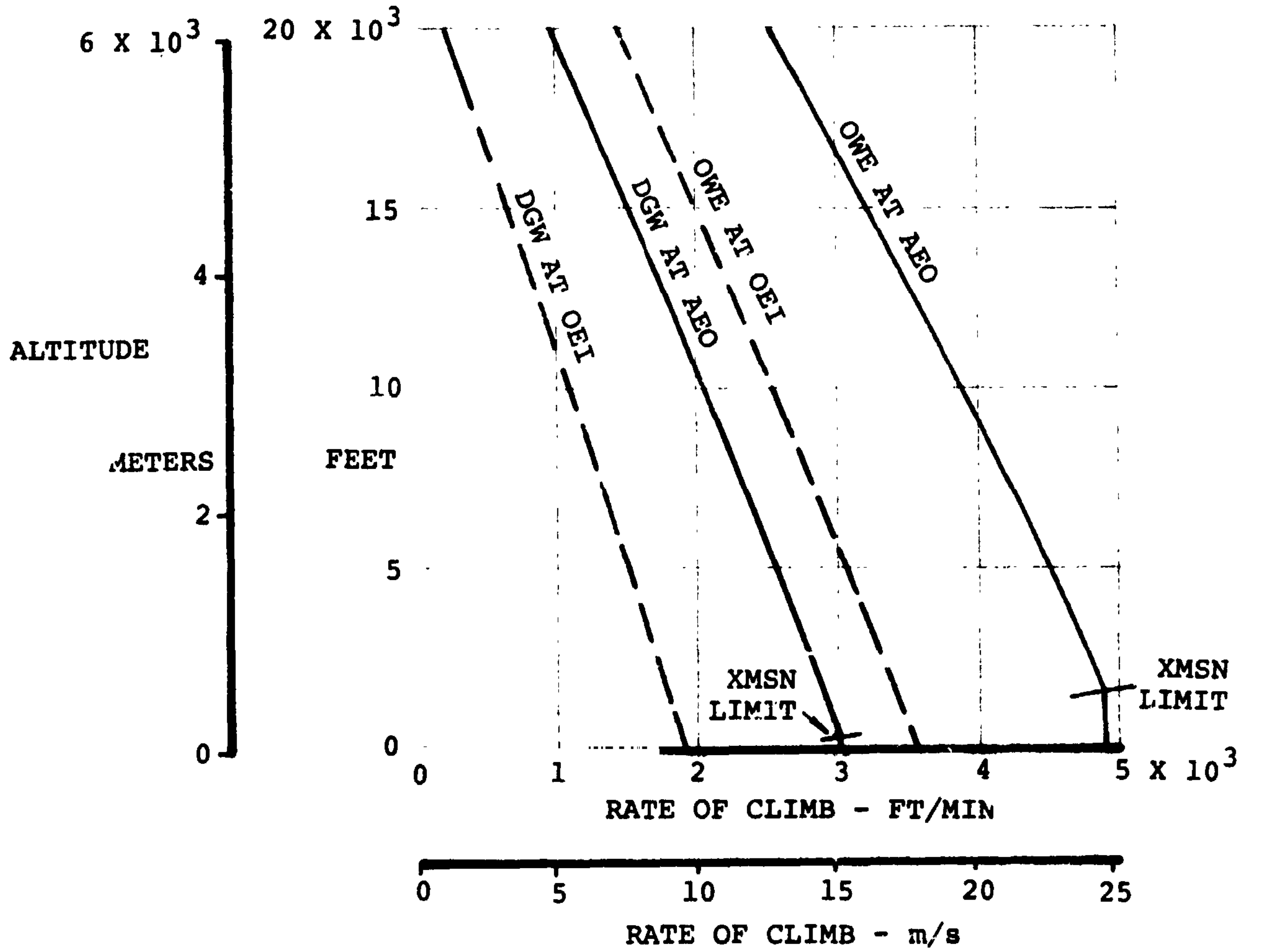


FIGURE 3.7. CLIMB PERFORMANCE - ALL ENGINES OPERATING AND ONE ENGINE INOPERATIVE.

fuselage inclinations, a limit of 20 degrees was imposed on the fuselage floor angle. This limit is in force up to an altitude of 10,000 feet.

With one engine inoperative, the climb performance is degraded by about 1,000 feet per minute in all cases. At the design gross weight, the rate of climb is 1,910 feet at sea level and falls to 730 feet per minute at cruise altitude. Extraploation of this line indicates a service ceiling of about 21,000 feet altitude. At the operating empty weight, the sea level climb rate is 3,566 feet per minute and the value at cruise altitude is 2,094 feet per minute. Despite the loss of power from one engine, the climb rate at the operating weight empty is limited by the fuselage angle restraint up to an altitude of 10,000 feet.

3.1.4 <u>Cruise Performance</u>

In the cruise attitude the rotor nacelles are set at zero incidence and the rotors operate as propellers. The rotor speed is reduced to 70% of the takeoff value.

The variation with speed of power required and available is shown in Figures 3.8 and 3.9 for altitudes of 5,000 feet and 14,000 feet respectively. Each graph shows the variation of power required for three gross weights; the design gross weight, a mid weight of 58,000 pounds and the operating empty weight. Also shown on each graph is a pair of lines showing the variation of power available with all engines operating (AEO) and with one engine inoperative (OEI).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

STANDARD DAY - CRUISE RPM

DGW = 68,493 LBS/31,068 Kg
MID WT = 58,000 LBS/26,309 Kg
OWE = 47,068 LBS/21,350 Kg

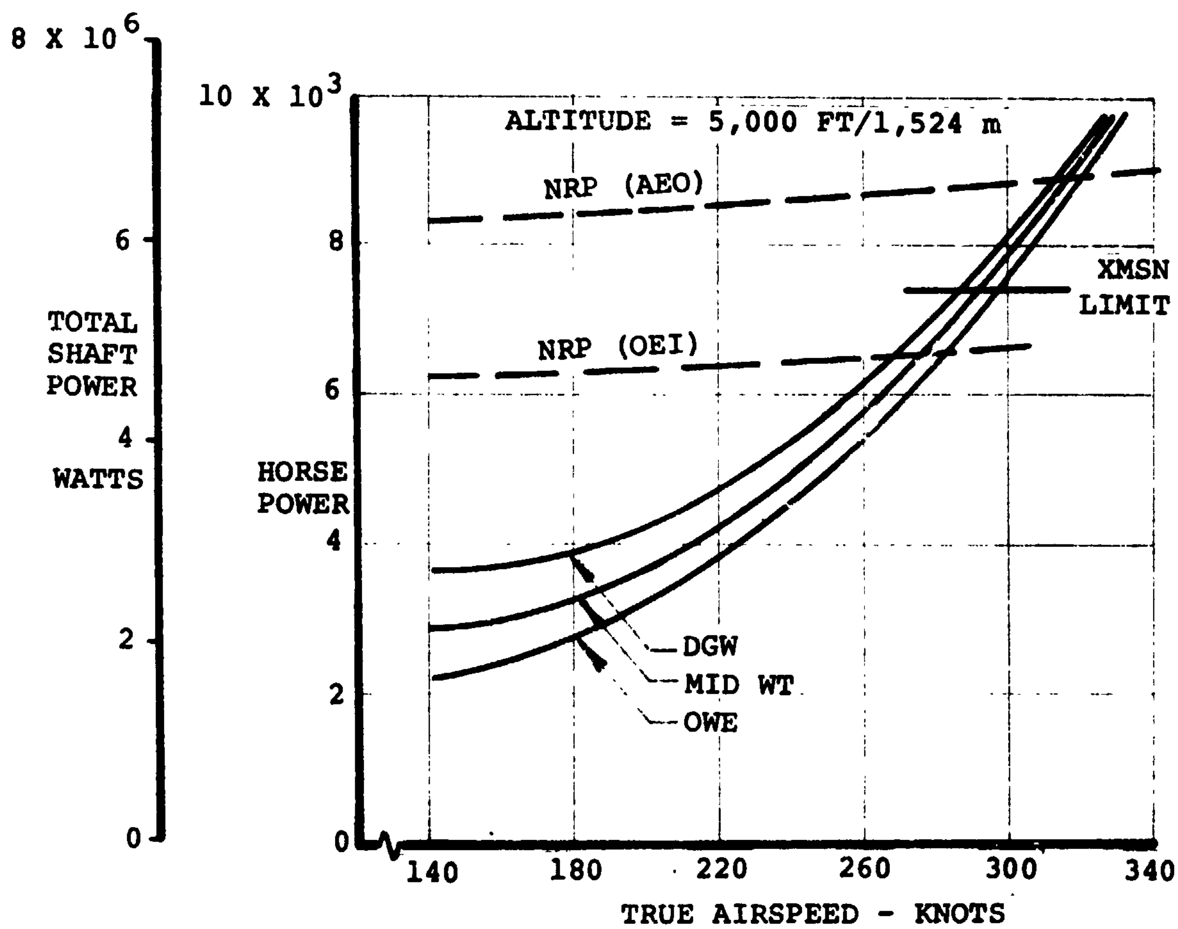


FIGURE 3.8. CRUISE PERFORMANCE - POWER REQUIRED/ AVAILABLE. (5000 FT.)

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

STANDARD DAY - CRUISE RPM

DGW = 68,493 LBS/31,068 Kg
MID WT = 58,000 LBS/26,309 Kg
OWE = 47,068 LBS/21,350 Kg

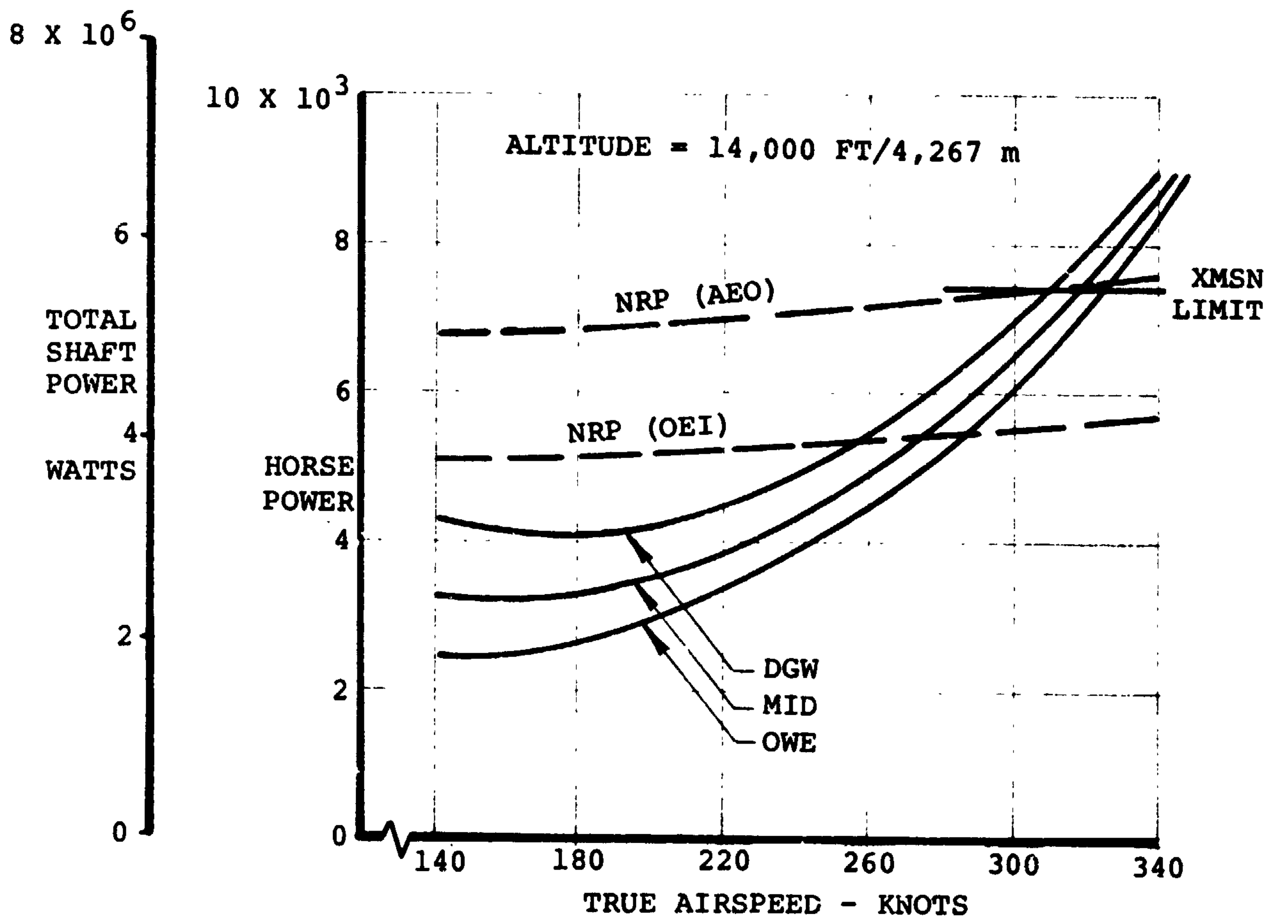


FIGURE 3.9. CRUISE PERFORMANCE - POWER REQUIRED/ AVAILABLE. (14,000FT.)

At 5,000 feet altitude the maximum speed AEO is transmission limited to 287 knots at design gross weight and 298 knots of the operating empty weight. The corresponding speeds OEI, which are not transmission limited are 269 knots and 283 knots respectively.

At design gross weight AEO the maximum cruise speed at 14,000 feet altitude is 310 knots limited by both the power available at normal rated power and the transmission torque capability. At the operating weight empty the maximum cruise speed, AEO, is increased to 324 knots and is transmission limited.

The corresponding OEI cruise speeds are 256 knots and 288 knots respectively.

Figure 3.10 summarizes the maximum cruise speed capability of the STOL tilt rotor as it varies with altitude. At design gross weight the maximum speed AEO is 310 knots at 14,000 feet altitude. Below this altitude the speed is limited by the transmission torque capability, and at higher altitudes by the normal rated power available. At the operating empty weight the transmission limit extends to an altitude of 14,500 feet and the maximum speed is 326 knots with all engines operating.

With one engine inoperative the cruise performance is not transmission limited at any altitude. At design gross weight the maximum speed (OEI) of 270 knots occurs at sea level. The

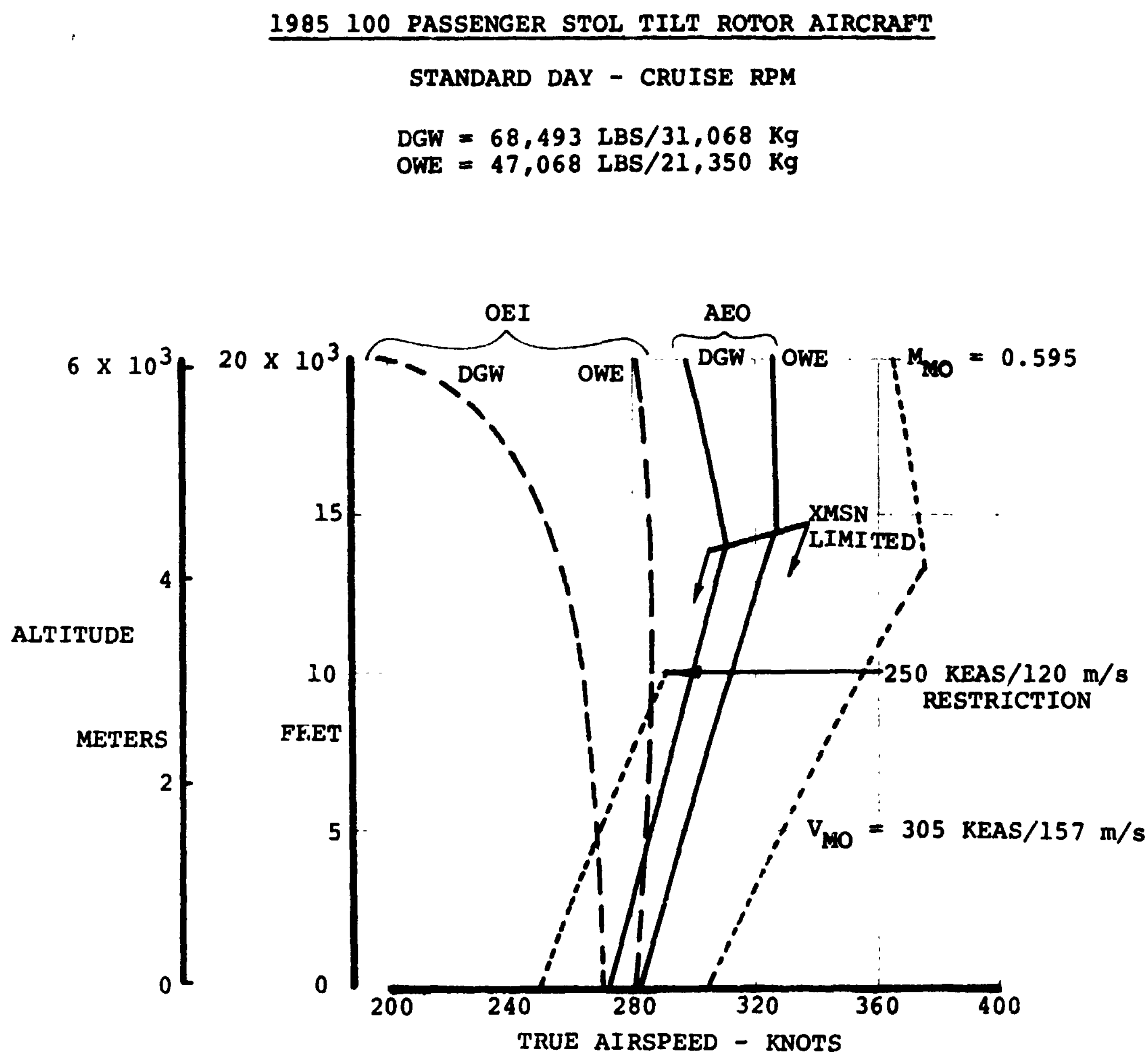


FIGURE 3.10. LEVEL FLIGHT CRUISE SPEED ENVELOPES.

reduction in speed capability at higher altitudes reflects the degradation of engine performance (power available) with altitude.

In no case is the cruise speed capability closer than 20 knots below the structural maximum operating speed and Mach number, shown as a boundary on the extreme right of Figure 3.10.

The aircraft has the capability to exceed the operational constraint of 250 knots equivalent airspeed imposed at altitudes of less than 10,000 feet over a wide range of power and weight conditions.

Figures 3.11 and 3.12 show the specific range performance achieved by the aircraft in the cruise mode as a function of true airspeed. Figure 3.11 shows the variation for the AEO case whereas Figure 3.12 is for OEI. Each of the two figures includes data for altitudes of 5,000 feet and 14,000 feet and for three gross weights; design gross weight, a mid weight and the operating empty weight.

For both the AEO and OEI cases the strong effect of gross weight in reducing the specific range is evident from Figures 3.11 and 3.12. For example, the best specific range at 14,000 feet altitude, AEO, falls from 0.1462 to 0.1122 nautical miles per pound of fuel as weight is increased from the operating weight empty to the design gross weight. At the same time the best range speed increases from 205 knots to 237 knots. The corresponding values of speed for 99% best range are 223 knots and 255 knots.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

STANDARD DAY - CRUISE RPM

DGW = 68,493 LBS/31,068 Kg

MID WT = 58,000 LBS/26,309 Kg

OWE = 47,068 LBS/21,350 Kg

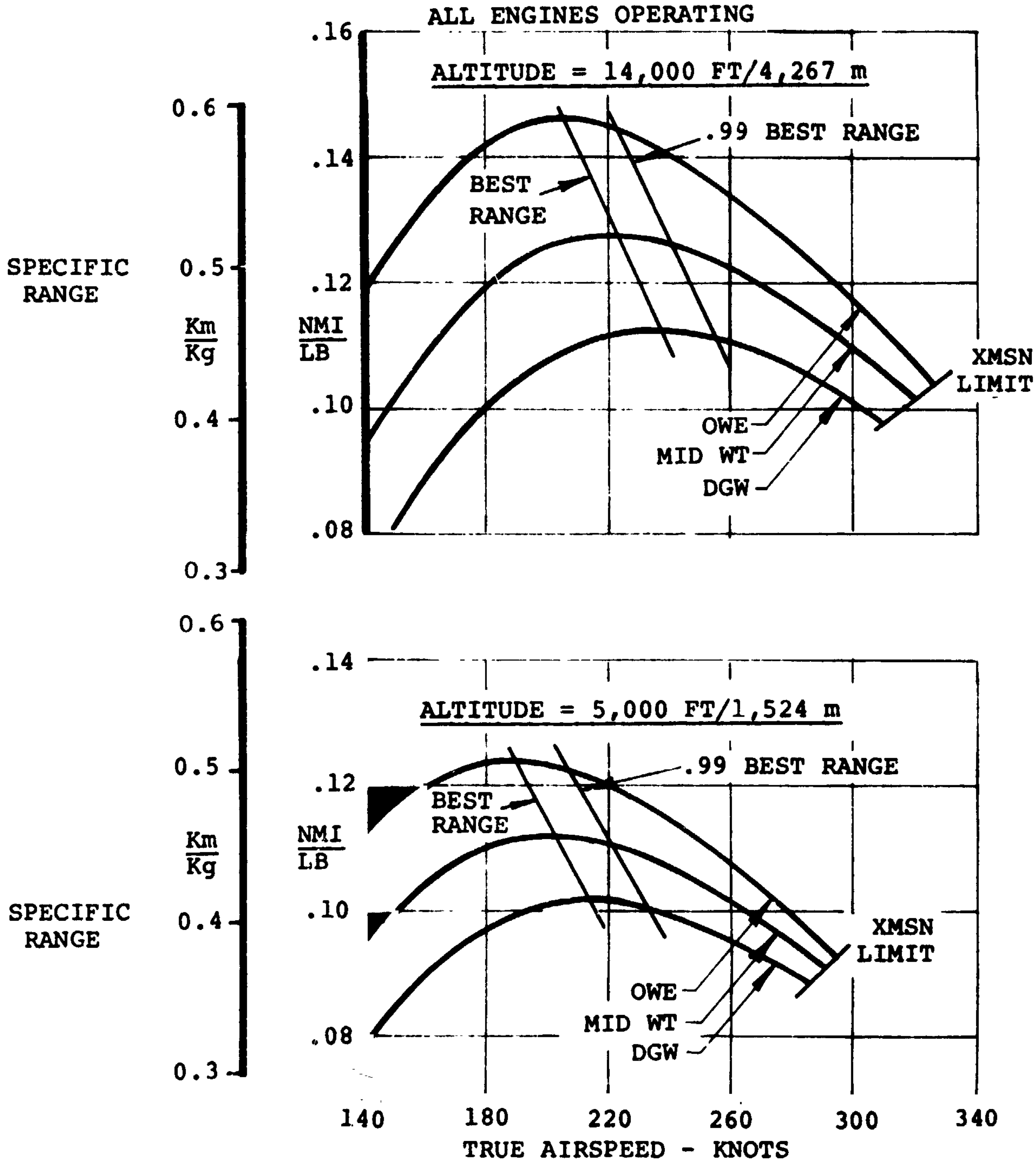


FIGURE 3.11. CRUISE PERFORMANCE - SPECIFIC RANGE. (AEO)

1985 100 PASSENGER STOL TILT RCTOR AIRCRAFT

STANDARD DAY - CRUISE RPM
DGW = 68,493 LBS/31,068 Kg
MID WT = 58,000 LBS/26,309 Kg
OWE = 47,068 LBS/21,350 Kg

ONE ENGINE INOPERATIVE

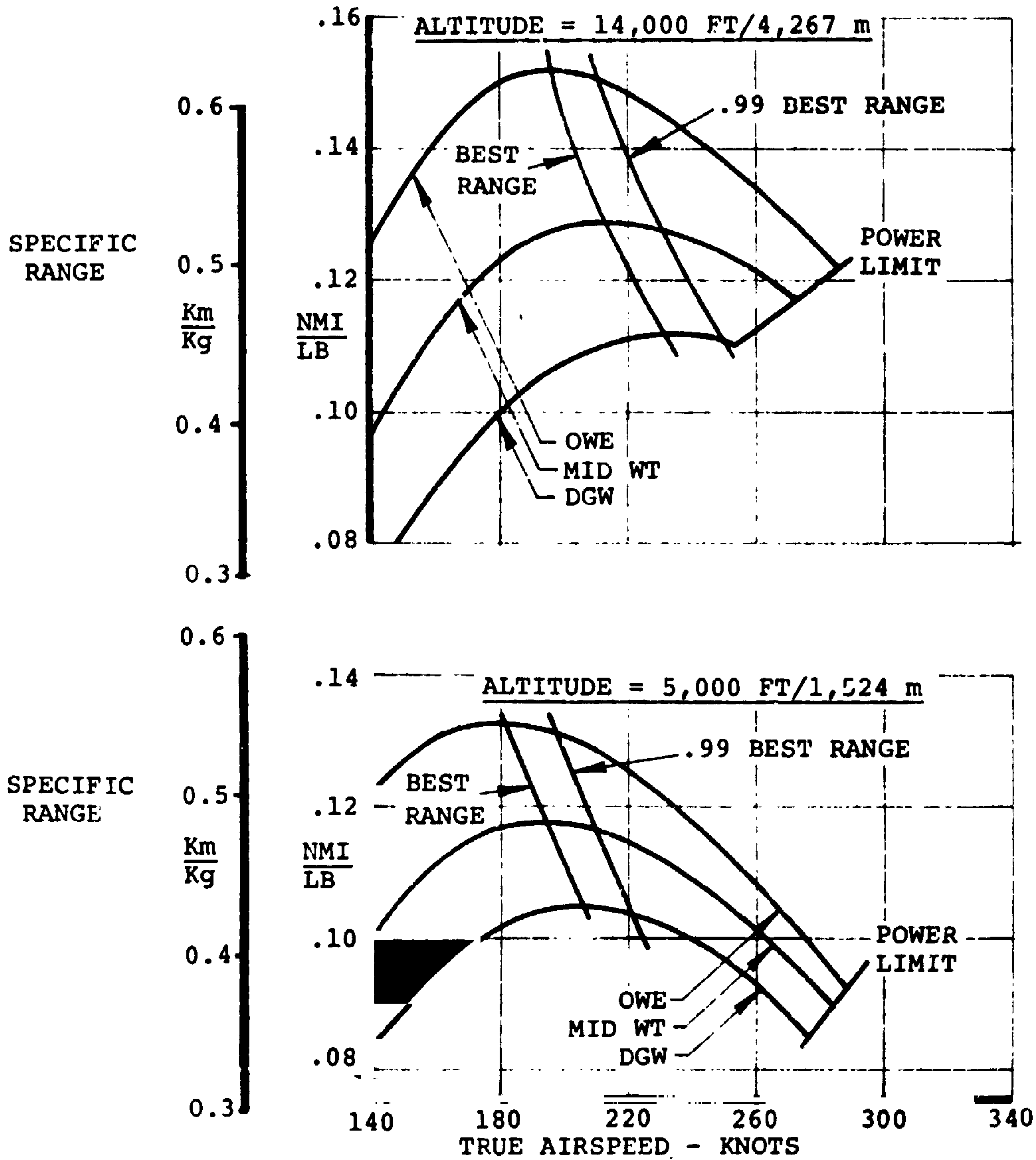


FIGURE 3.12. CRUISE PERFORMANCE - SPECIFIC RANGE. (OEI)

The effect of altitude on specific range can be assessed by comparing the upper and lower graphs in Figures 3.11 and 3.12. Changing altitude from 5,000 feet to 14,000 feet the best specific range is increased from 0.1240 nautical miles per pound of fuel to 0.1462 while the best range speed rises from 190 knots to 205 knots.

It should be noted that increasing the cruise altitude above 14,000 feet would not continue the apparent trend to increasing specific range. This altitude is close to the optimum and further increases of altitude would lead to reduced levels of specific range.

The effect of operating the aircraft on three engines instead of four can be assessed by comparing the curves of Figure 3.11 with those of Figure 3.12. At all but the highest speeds the aircraft has a higher specific range when flying with one engine inoperative. This is due to the improved specific fuel consumption resulting from operating the three engines at an increased power level.

For the design mission range of 200 nautical miles the fuel consumption when flying AEO is 62.54 passenger miles per gallon compared with 68.26 when flying with OEI during the cruise and loiter segments. The overall mission fuel consumption for the aircraft flying AEO is shown as a function of cruise speed in Figure 4.12.

The payload range capability of the STOL tilt rotor is shown in Figure 3.13. The takeoff weight used in the calculation

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

CRUISE AT NRP - CRUISE RPM
ALL ENGINES OPERATING - STANDARD DAY
DGW = 68,493 LBS/31,068 Kg

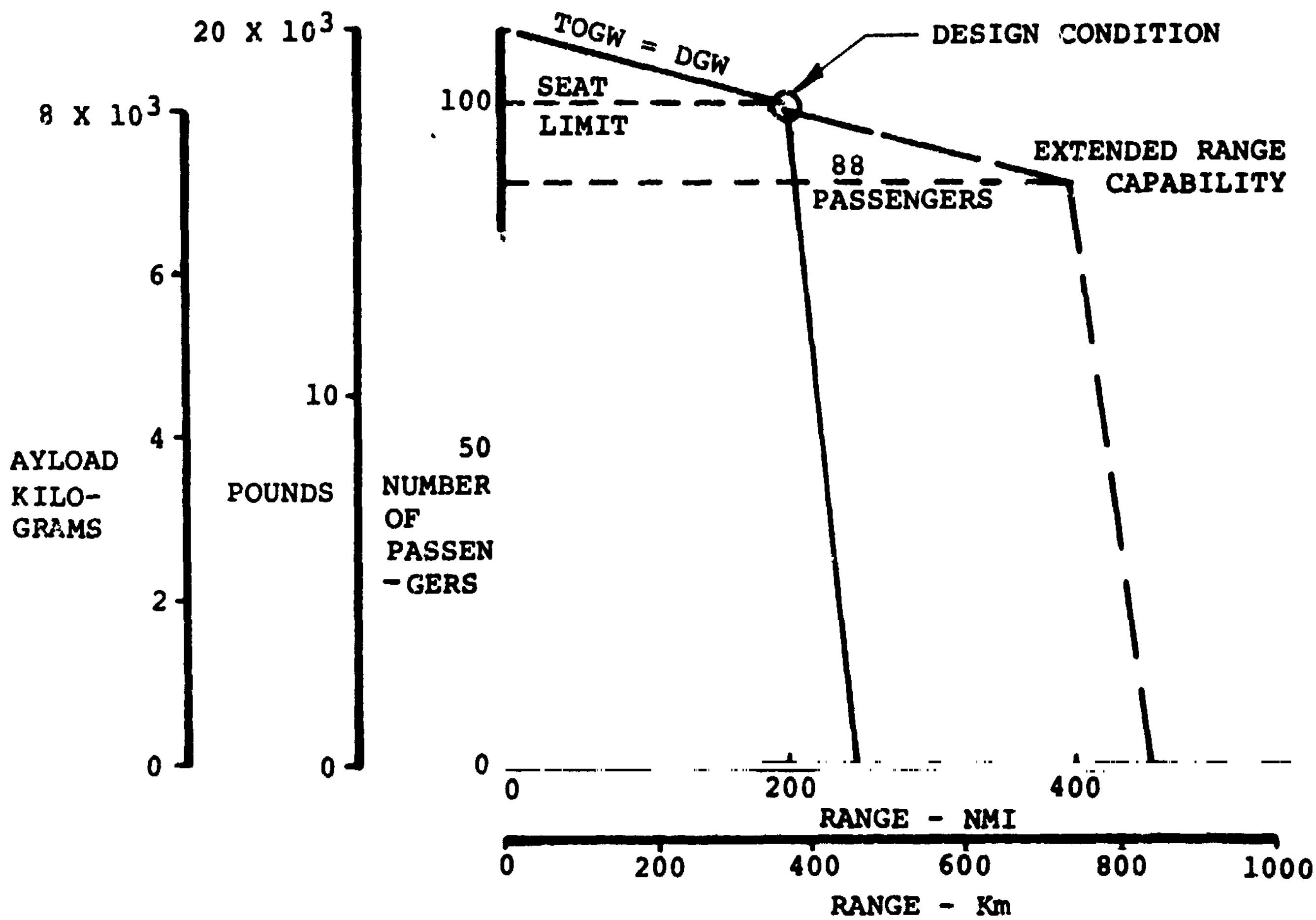


FIGURE 3.13. PAYLOAD RANGE CAPABILITY (ALL ENGINES OPERATING).

of these data was the design gross weight, 68,493 pounds and the mission was flown with all engines operating. The data for the "off-design" range points was evaluated by changing the length of the cruise portion of the basic mission.

The graph indicates that the design condition is a payload of 18,000 pounds (100 passengers) at a range of 200 nautical miles (230 statute miles). The maximum range of the aircraft, as designed, is 250 nautical miles with zero payload. For flights of lesser range than 200 nautical miles the payload capability is greater than the design value of 100 passengers. But since no cargo carrying provision has been made in the design and the seating is limited to 100 passengers the additional capability is unusable.

Further calculations have been made for an "extended range" version of the selected aircraft. For ranges of greater than 200 nautical miles,additional fuel tankage has been added to enable the range to be increased to 400 nautical miles. This entailed the addition of 154 pounds of fuel tanks with a capacity of 306 gallons of fuel.

Thus, at the 200 nautical mile range the payload capability is reduced from 100 to 99 passengers and at 400 nautical miles the passenger capacity is 88. The additional fuel tankage allows the zero payload range to be increased to 450 nautical miles.

In Figure 3.14 the payload range capability is shown for missions in which one engine is shut down during the cruise and loiter segments. Since the cruise is flown with the engines set at normal rated power, the fuel consumption will be considerably less with one engine shut down. This is reflected in the increased range performance shown in Figure 3.14. The maximum range with 100 passengers has increased from 200 to 230 nautical miles and the zero payload range of the basic aircraft has increased by 50 to 300 nautical miles. The corresponding ranges for the "extended range" version are 458 and 530 nautical miles.

3.1.5 Landing Performance

Figures 3.15 through 3.18 summarize the landing performance of the STOL tilt rotor aircraft. The ground rules dictating the landing performance are summarized in Table 3.1. Figure 3.15 shows the effect of atmospheric conditions on the landing distance and field length at the design gross weight. The effect of both altitude and temperature are small in terms of landing distance or field length when compared with the effect on takeoff. The effect is due entirely to the change of aerodynamic forces that result from the density changes associated with altitude and temperature variation. The effect of altitude and temperature on the engine performance is of small consequence because the landing is effected at a low power level (in the region of 50% of takeoff power.)

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

CRUISE AT NRP WITH ONE ENGINE INOPERATIVE

STANDARD DAY CRUISE RPM

DGW - 68,493 LBS/31,068 Kg

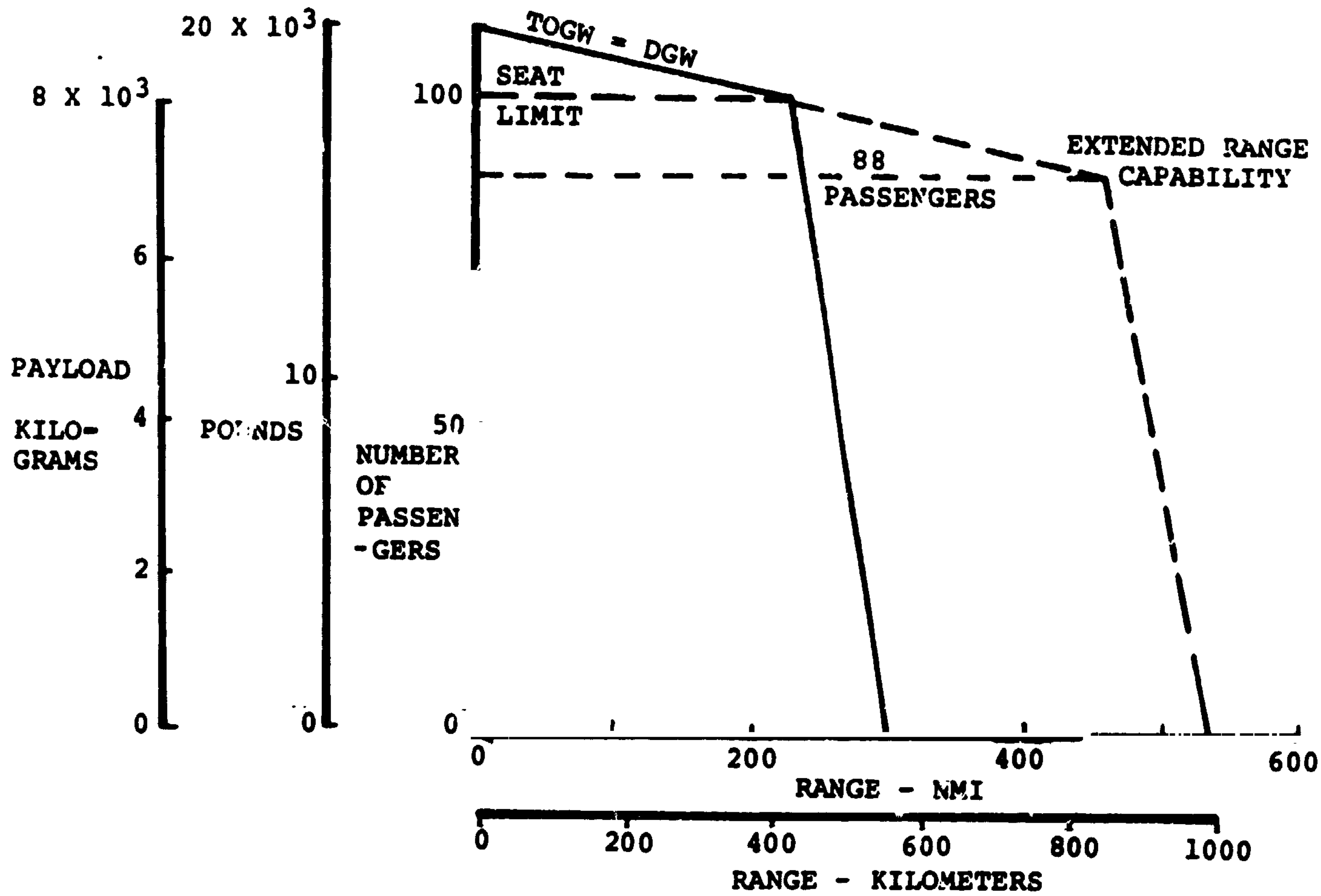


FIGURE 3.14. PAYLOAD - RANGE CAPABILITY - OEI.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

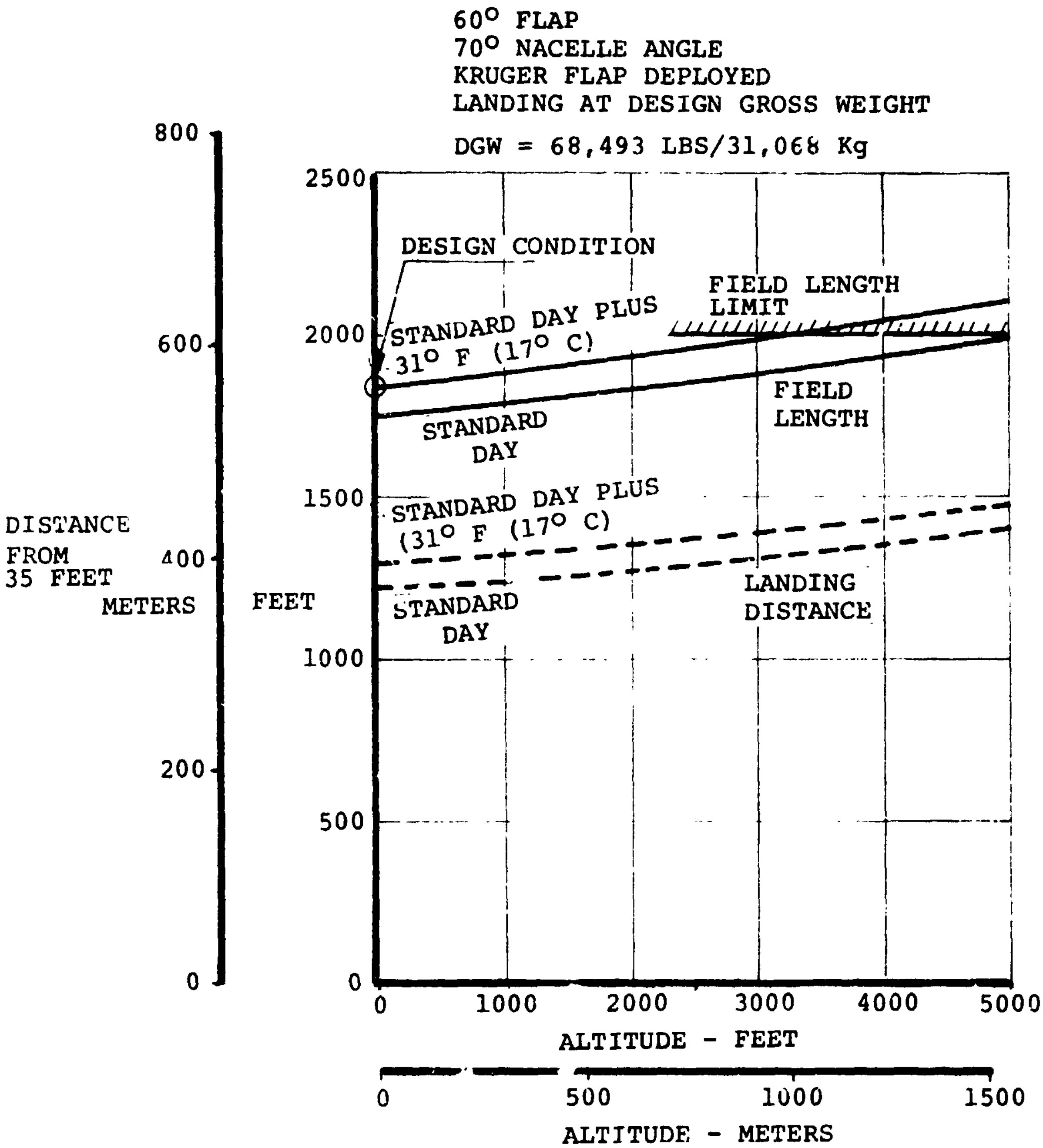


FIGURE 3.15. EFFECT OF ALTITUDE AND AMBIENT TEMPERATURE ON LANDING PERFORMANCE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

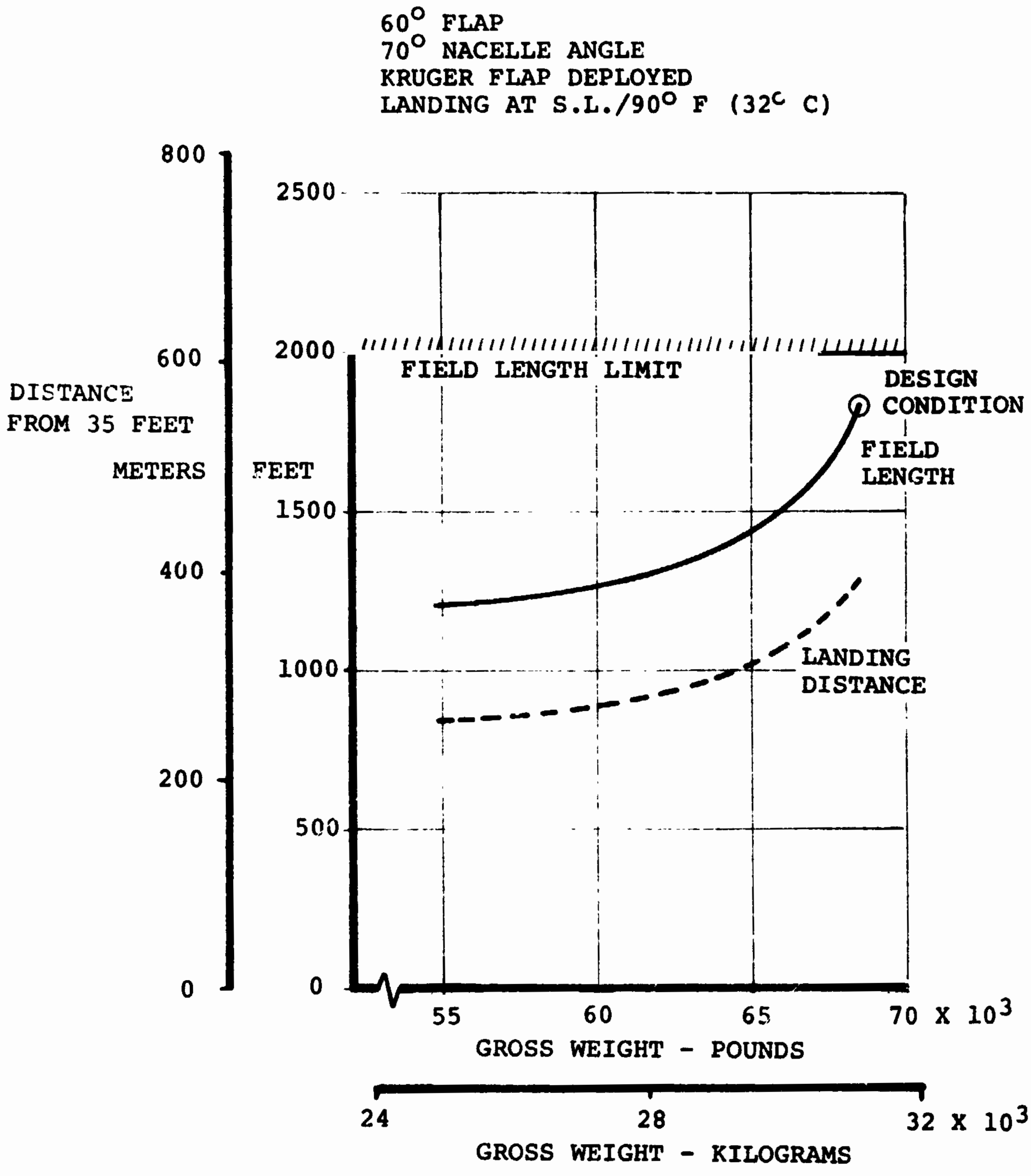


FIGURE 3.16. EFFECT OF GROSS WEIGHT ON LANDING PERFORMANCE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

60° FLAP
KRUGER FLAP DEPLOYED
LANDING AT DESIGN GROSS WEIGHT

DGW = 68,493 LBS/31,068 Kg

S.L./90° F (32° C)

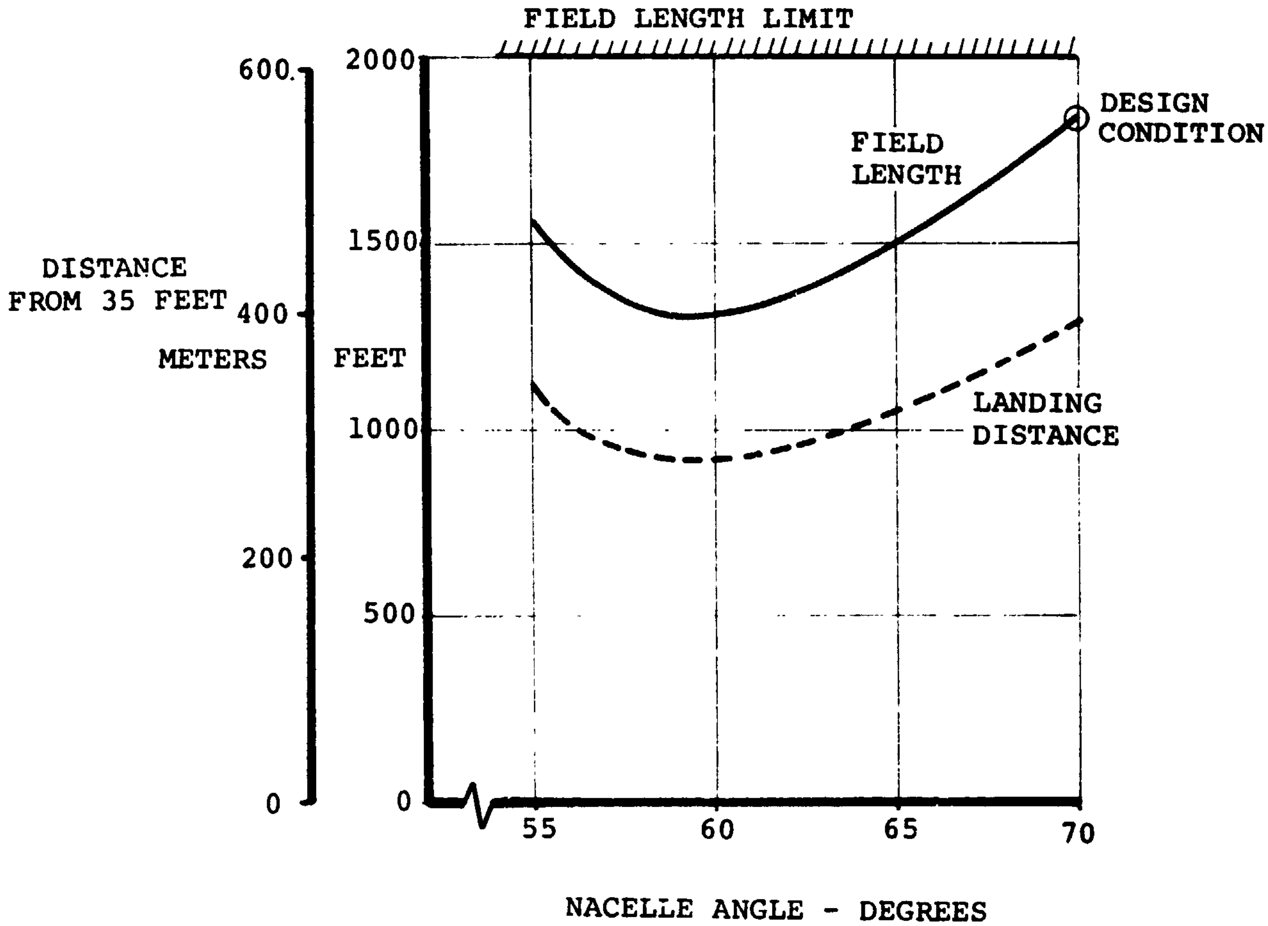


FIGURE 3.17. EFFECT OF NACELLE ANGLE ON LANDING PERFORMANCE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
60° FLAP KRUGER FLAP DEPLOYED
70° NACELLE ANGLE S.L./90° F (32° C)
LANDING AT DESIGN GROSS WEIGHT
DGW = 68,493 LBS/31,068 Kg

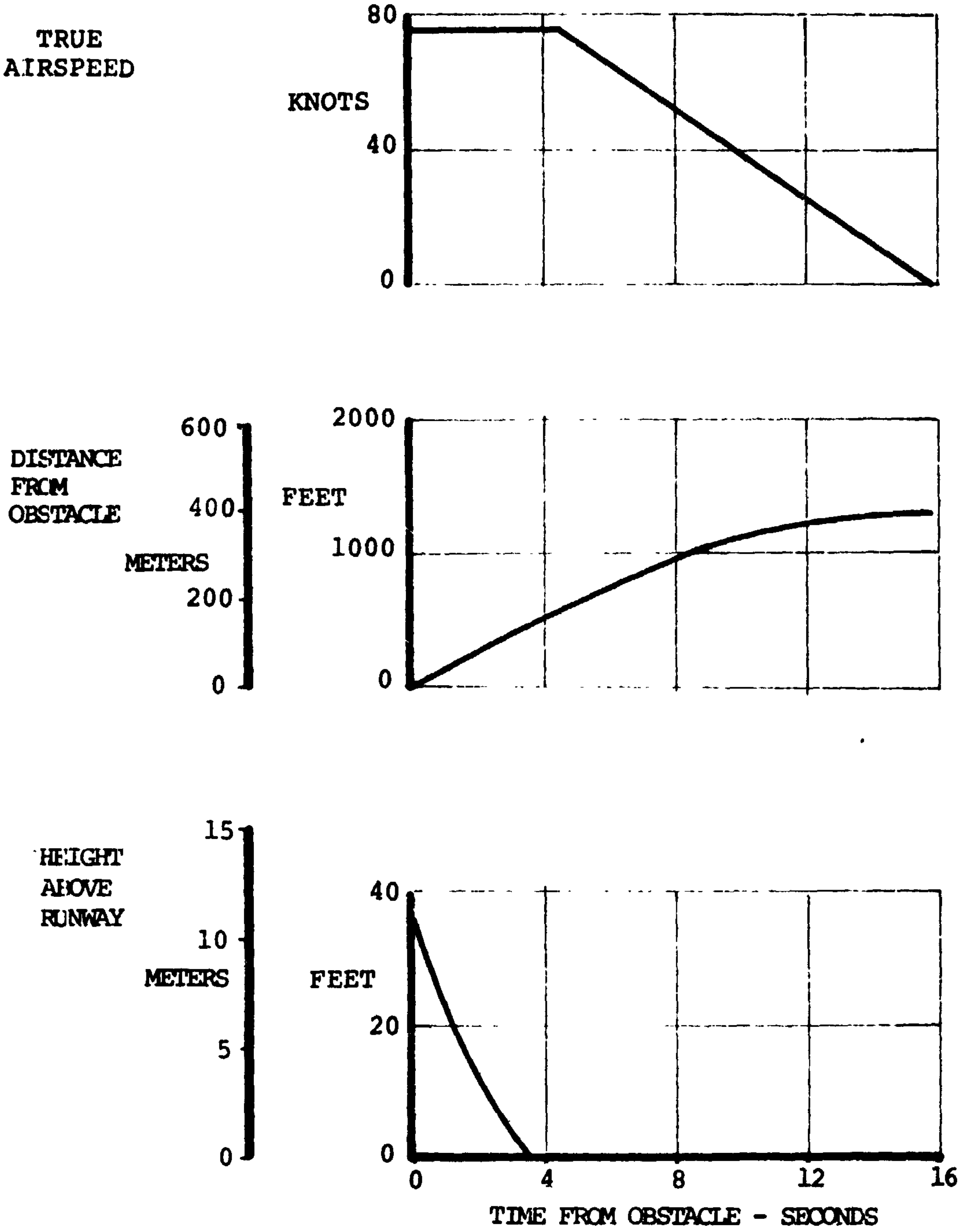


FIGURE 3.18. LANDING TIME HISTORY.

The effect of aircraft weight on the landing performance is shown in Figure 3.16. The rapid increase of landing distance with gross weight is due to the increase of approach speed required to achieve the required descent angle. This higher approach speed dictates a larger braking distance on the ground.

Figure 3.17, the effect of nacelle angle on landing performance, shows that the design point landing performance is by no means the smallest distance, or field length apparently obtainable. However, to obtain the shorter distances corresponding to nacelle angles lower than 70 degrees is impractical as the fuselage attitude required to keep the rotor at the optimum angle of attack becomes excessive. Consequently, all other landing data have been calculated for a nacelle incidence of 70 degrees in order to keep fuselage angles down to about 10 degrees in the final approach and flare.

The time history of a landing at design gross weight at sea level, 90 degrees F is shown in Figure 3.18. Speed, distance from the obstacle and height above the ground are all plotted as a function of time. It can be seen from the top graph that a constant airspeed is maintained until touchdown and for one second thereafter. The one second delay is to allow time for the reduction of rotor collective pitch and the application of wheel brakes. Thereafter, the aircraft decelerates on the ground at a constant deceleration of 0.35 g.

During the airborne part of the landing, a slight flare from a height of 25 feet is required in order to limit the touchdown sink speed to 300 feet per minute.

3.1.6 Airframe Drag

The contribution of the major airframe components to the total drag is shown in Table 3.2 in terms of equivalent "flatplate drag area".

The STOL tilt rotor aircraft has an equivalent drag area of 24.37 square feet (2.264 square meters) and a gross weight to drag area ratio of 2,811 pounds per square foot (13,722 kilograms per square meter).

3.1.7 Prop/Rotor Performance

The static and cruise performance of the prop/rotor is shown in Figures 3.19 and 3.20 respectively. The static performance, shown in Figure 3.19, shows that the figure of merit achieved at the start of takeoff is 0.76 which is very close to the maximum attainable value.

The performance in axial flight (cruise, loiter, climb, etc.) is presented in Figure 3.20 in the form of curves of power coefficient versus thrust coefficient for given values of propeller advance ratio.

The rotor blade geometry is illustrated in Figure 3.45 in Section 3.4 of this report.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

COMPONENT	DRAG AREA FT^2	DRAG AREA m^2
FUSELAGE	10.39	0.965
WING	7.20	0.669
VERTICAL TAIL	1.35	0.125
HORIZONTAL TAIL	1.41	0.131
ROTOR NACELLES	1.04	0.097
ENGINE NACELLES	2.14	0.199
OIL COOLER MOMENTUM LOSS	0.28	0.026
AIR CONDITIONING	0.50	0.046
TRIM	0.06	0.006
TOTAL DRAG AREA	24.37	2.264

TABLE 3.2. DRAG SUMMARY.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

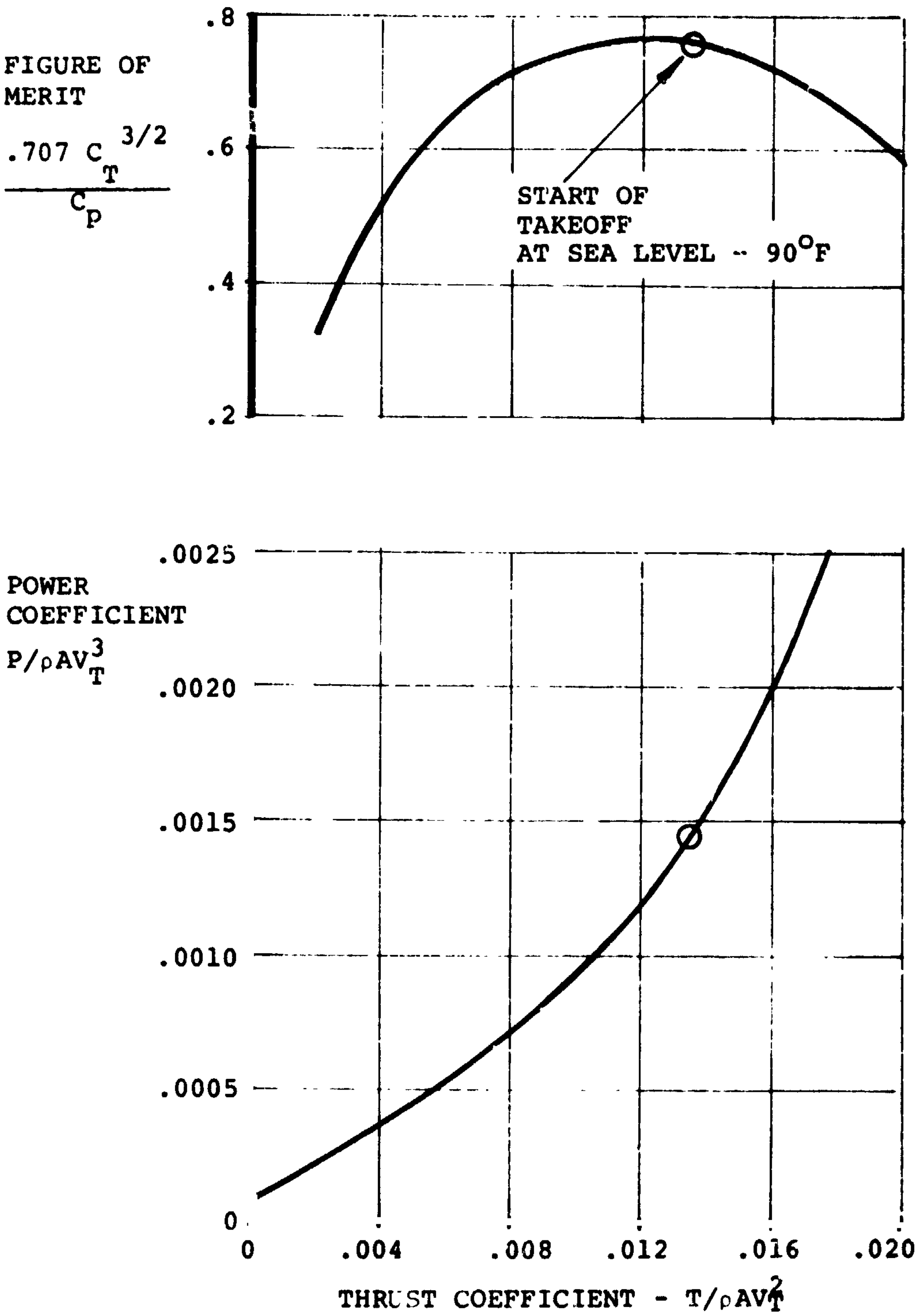


FIGURE 3.19. STATIC ROTOR PERFORMANCE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

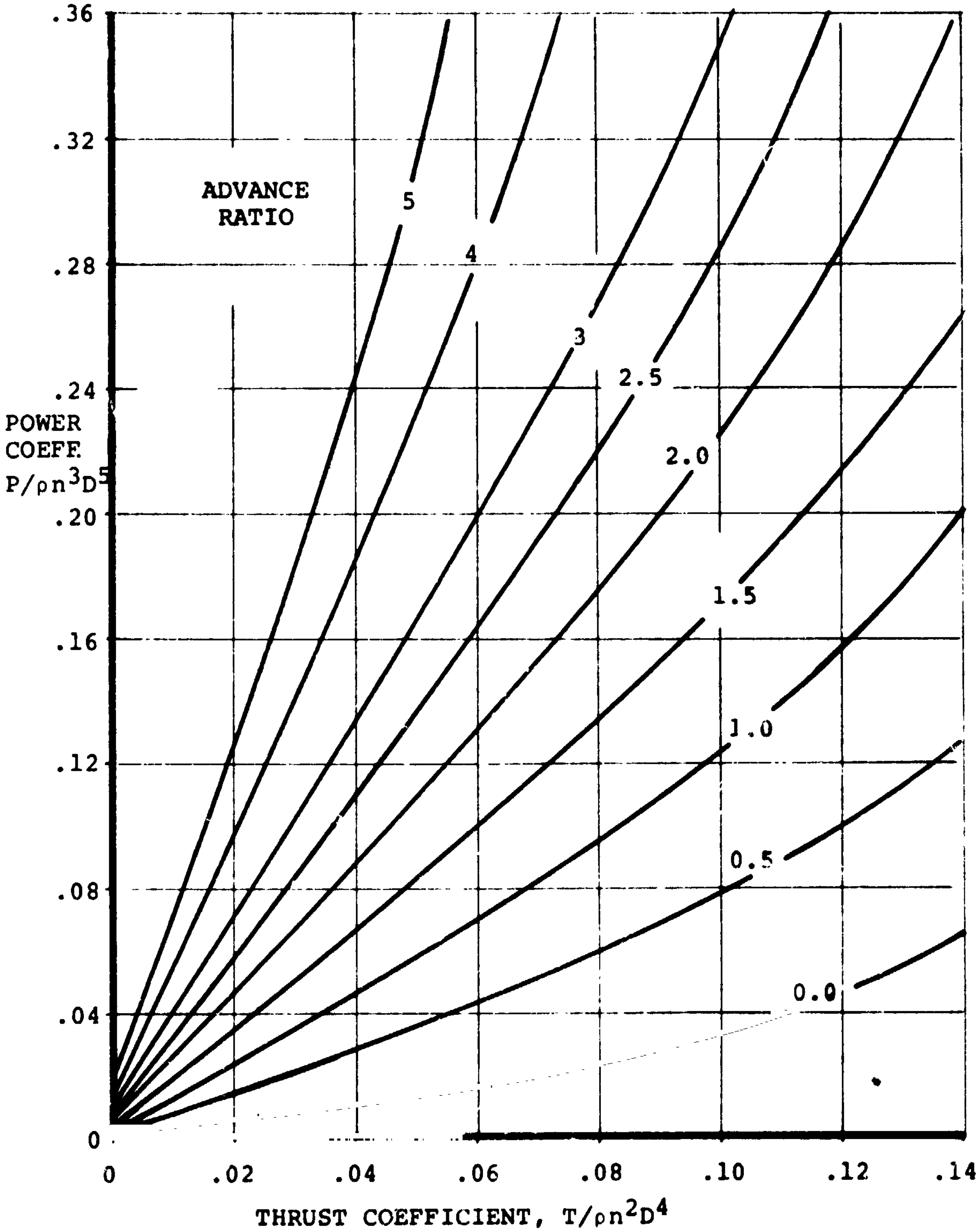


FIGURE 3.20. CRUISE PERFORMANCE OF ROTOR.

3.2 WEIGHT

The STOL tilt rotor aircraft design gross weight is 31,067 kilograms (68,493 pounds). The weight breakdown in terms of the structural and system categories is shown in Table 3.3.

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

The component and system weights are verified in Appendix C 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 Appendix C. 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.
4. Accommodation and equipment shall be provided for a flight crew of two and for one cabin attendant per 50 passengers. In addition,

	KILOGRAMS	POUNDS
WING	2397.7	5286
ROTOR	1877.4	4139
TAIL	520.3	1147
SURFACES	520.3	1147
ROTOR		
BODY	3889.5	8575
BASIC		
SECONDARY		
ALIGHTING GEAR GROUP	1242.8	2740
ENGINE SECTION	288.9	637
PROPULSION GROUP	3000.9	6616
ENGINE INST'L	796.1	1755
EXHAUST SYSTEM *		
COOLING		
CONTROLS *		
STARTING *		
PROPELLER INST'L	*246.8	*544
LUBRICATING *		
FUEL	762	168
DRIVE	1881.9	4149
FLIGHT CONTROLS	1567.2	3455
AUX. POWER PLANT	288.5	636
INSTRUMENTS	191.9	423
HYDR. & PNEUMATIC	308.4	680
ELECTRICAL GROUP	423.7	934
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	20431.0	45043
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	1553.5	3425
GROSS WEIGHT	31067.7	68493

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TABLE 3.3. STOL TILT ROTOR WEIGHT SUMMARY.

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4. continued

   some provision shall be made on the flight deck for an occasional flight observer. Each crew man 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.

Standard Weight Items

The weights of specified standard items shall be as provided in Table 3.4, Tilt Rotor Weights Guidelines.

Fly-By-Wire Control Systems

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

Gearboxes

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

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 3.4. TILT ROTOR WEIGHTS GUIDELINES.

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.

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 minu 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. 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 3.5 for both takeoff and cruise flight at the extremes of the weight envelope, i.e., weight empty and design gross weight.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

	WEIGHT EMPTY	GROSS WEIGHT	
	ROTOR VERTICAL	TAKEOFF ROTOR AT 66-DEGREES	CRUISE FLIGHT ROTOR HORIZONTAL
WEIGHT	20,431.0 Kg (45,043 LBS)	31,067.8 Kg (68,493 LBS)	31,067.8 Kg (68,493 LBS)
CENTER OF GRAVITY*			
FUSELAGE STATION	13.14 M(517.4 IN.)	13.13 M(517.1 IN.)	13.03 M(513.0 IN.)
% MAC		31.2	23.3
WATER LINE	3.96 M(156.0 IN.)	3.43 M(135.1 IN.)	3.26 M(128.2 IN.)
MOMENT OF INERTIA			
I_{xx} (ROLL)	1,129,550 Kg M^2 833,254 Slug Ft^2	1,217,556 Kg M^2 898,175 Slug Ft^2	1,242,404 Kg M^2 916,505 Slug Ft^2
I_{yy} (PITCH)	504,545 Kg M^2 372,196 Slug Ft^2	543,854 Kg M^2 401,194 Slug Ft^2	554,954 Kg M^2 409,382 Slug Ft^2
I_{zz} (YAW)	1,354,534 Kg M^2 99,222 Slug Ft^2	1,460,068 Kg M^2 1,077,073 Slug Ft^2	1,489,866 Kg M^2 1,099,054 Slug Ft^2

*FUSELAGE STATION O IS NOSE OF BODY - WATER LINE O IS GROUND LINE.

TABLE 3.5. WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA SUMMARY.

The excursions of center of gravity travel are shown for both the takeoff and cruise configurations in Figure 3.21. The center of gravity envelopes for this aircraft assume that window seats are filled first, followed by aisle seats. In the takeoff configuration the nacelle incidence is set at 66 degrees.

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 3.22. This plot provides the change in aircraft gross weight design for increasing or decreasing fixed weight items.

3.3 FLYING QUALITIES

Transition

Although the hover control requirements do not influence the design of the STOL tilt rotor configuration, the transition from takeoff nacelle incidence to cruise flight retains the same elements of the tilt rotor transition controls design problem.

The optimization and design of the control system in this flight regime are beyond the scope of a conceptual study. In order to provide visibility on available control powers in transition, a typical transition schedule has been considered. The variation of nacelle incidence and thrust

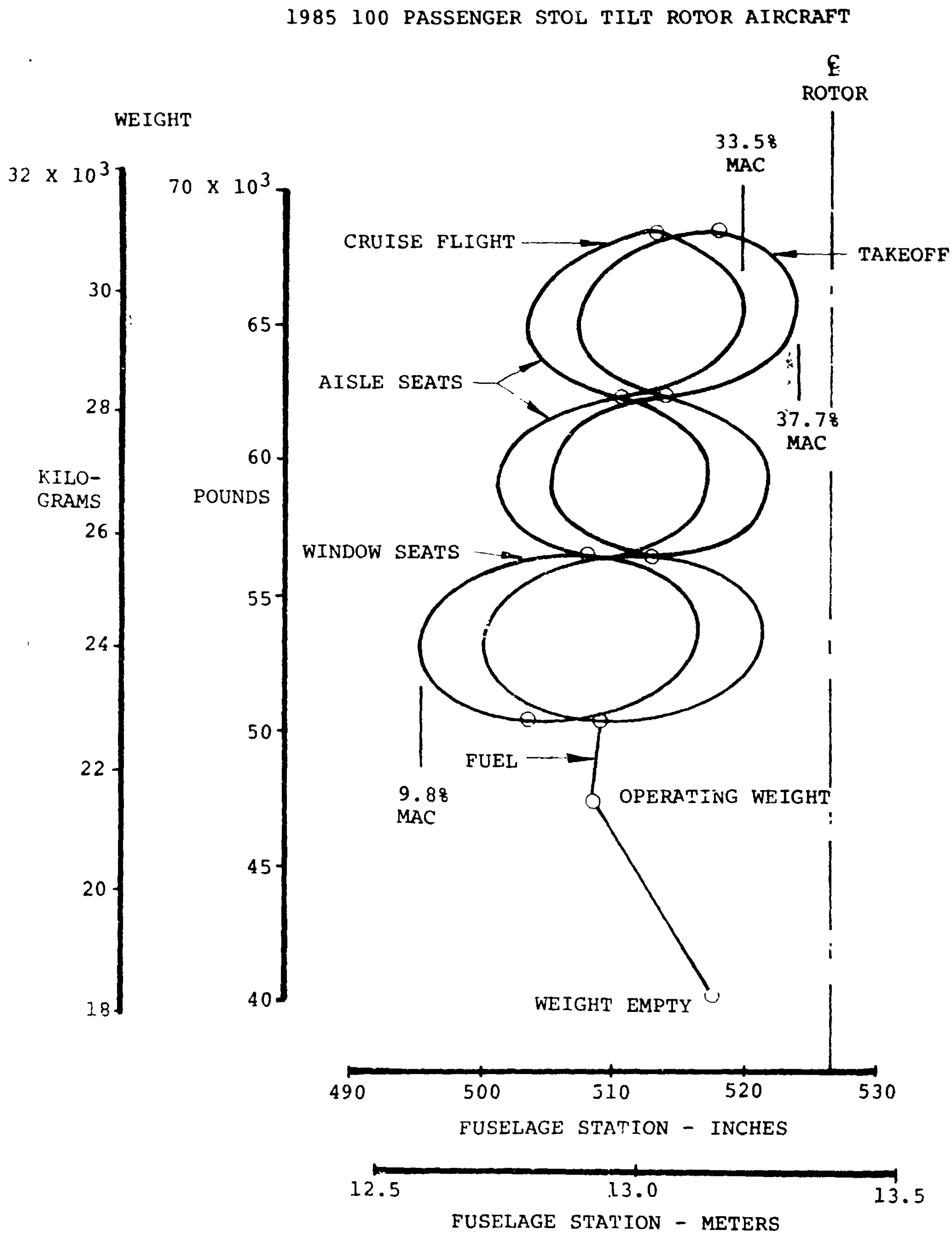

FIGURE 3.21. CENTER OF GRAVITY ENVELOPE.

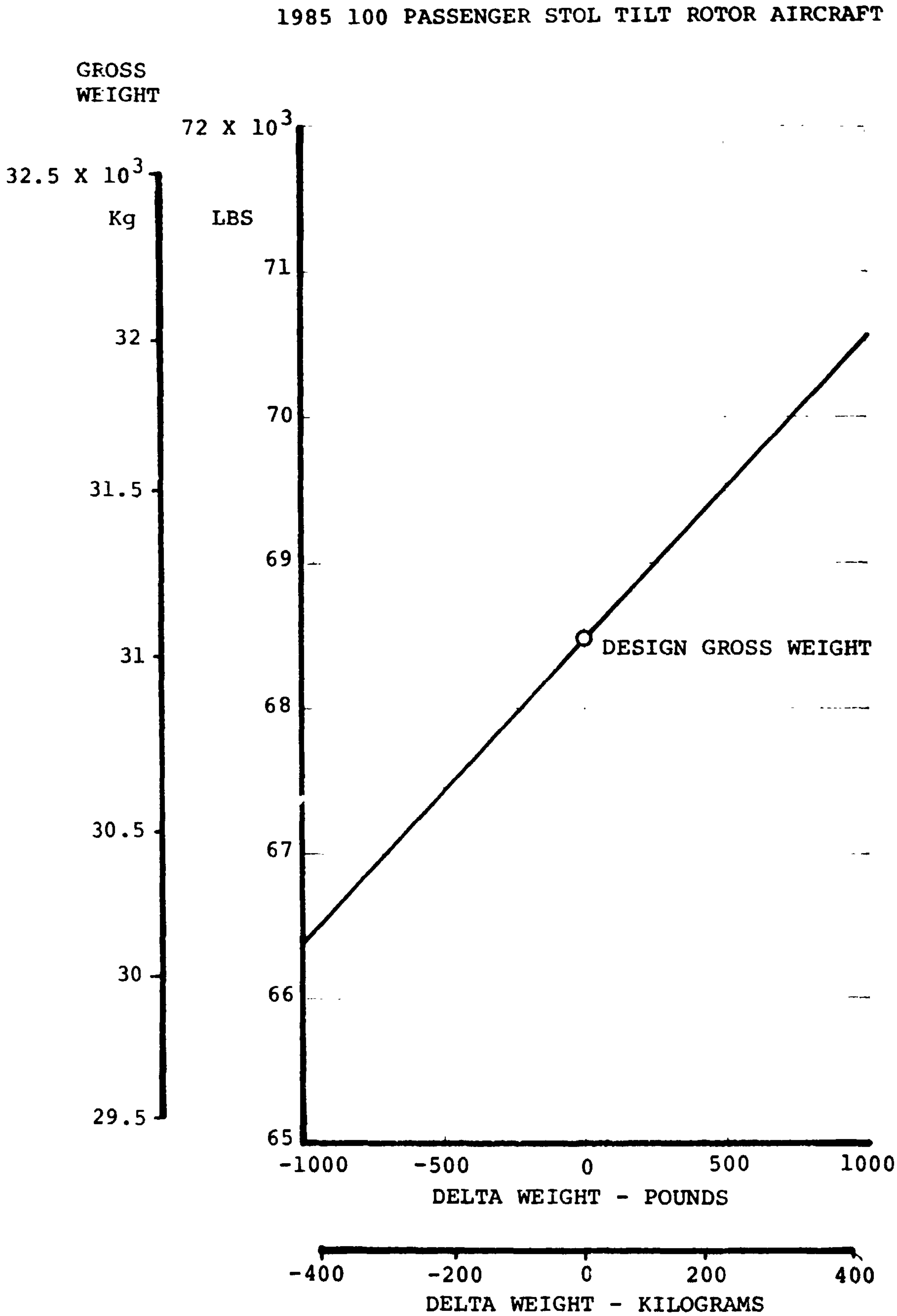

FIGURE 3.22. WEIGHT GROWTH AT CONSTANT PERFORMANCE AND STRENGTH.

with airspeed for 1g level trimmed flight is shown in Figure 3.23 and the control deflections and resulting fuselage attitude are given in Figure 3.24. In order to facilitate rotation at takeoff a trimmable horizontal tail has been assumed and this is trimmed back to zero incidence at 100 knots.

Pitch control power is obtained from elevator and longitudinal cyclic pitch controls. The control power available based upon the trim schedule is shown in Figure 3.25. The guideline requirement stipulated 0.3 radians per second per second above 40 knots and is superimposed in Figure 3.25. The available control power exceeds this requirement at all speeds above the aircraft rotation speed.

Roll control is achieved by differential thrust or collective and differential longitudinal cyclic pitch. As airspeed increases the aileron and spoiler controls become effective and the rotor controls are phased out as the nacelle incidence approaches cruise condition. The roll angular acceleration available is shown in Figure 3.26 and meets or exceeds the 0.4 radians per second per second requirement throughout the transition range.

Yaw control at low speed transition is also achieved by differential collective pitch and differential longitudinal cyclic as shown in Figure 3.27. At higher airspeeds the rotor controls are replaced by the rudder control as the cruise configuration is approached.

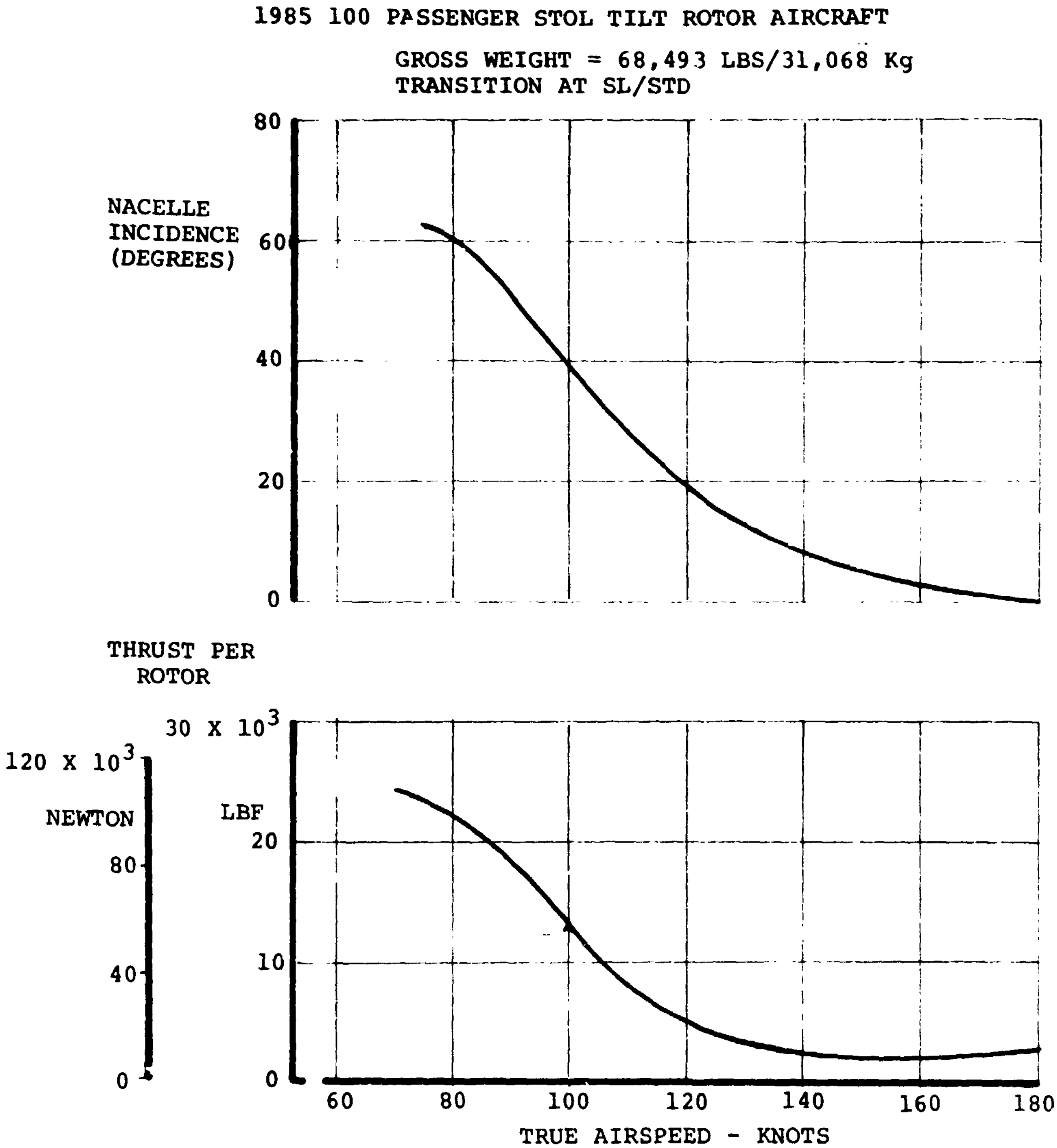

FIGURE 3.23. TYPICAL TRIM CHARACTERISTICS IN TRANSITION - THRUST AND NACELLE INCIDENCE.

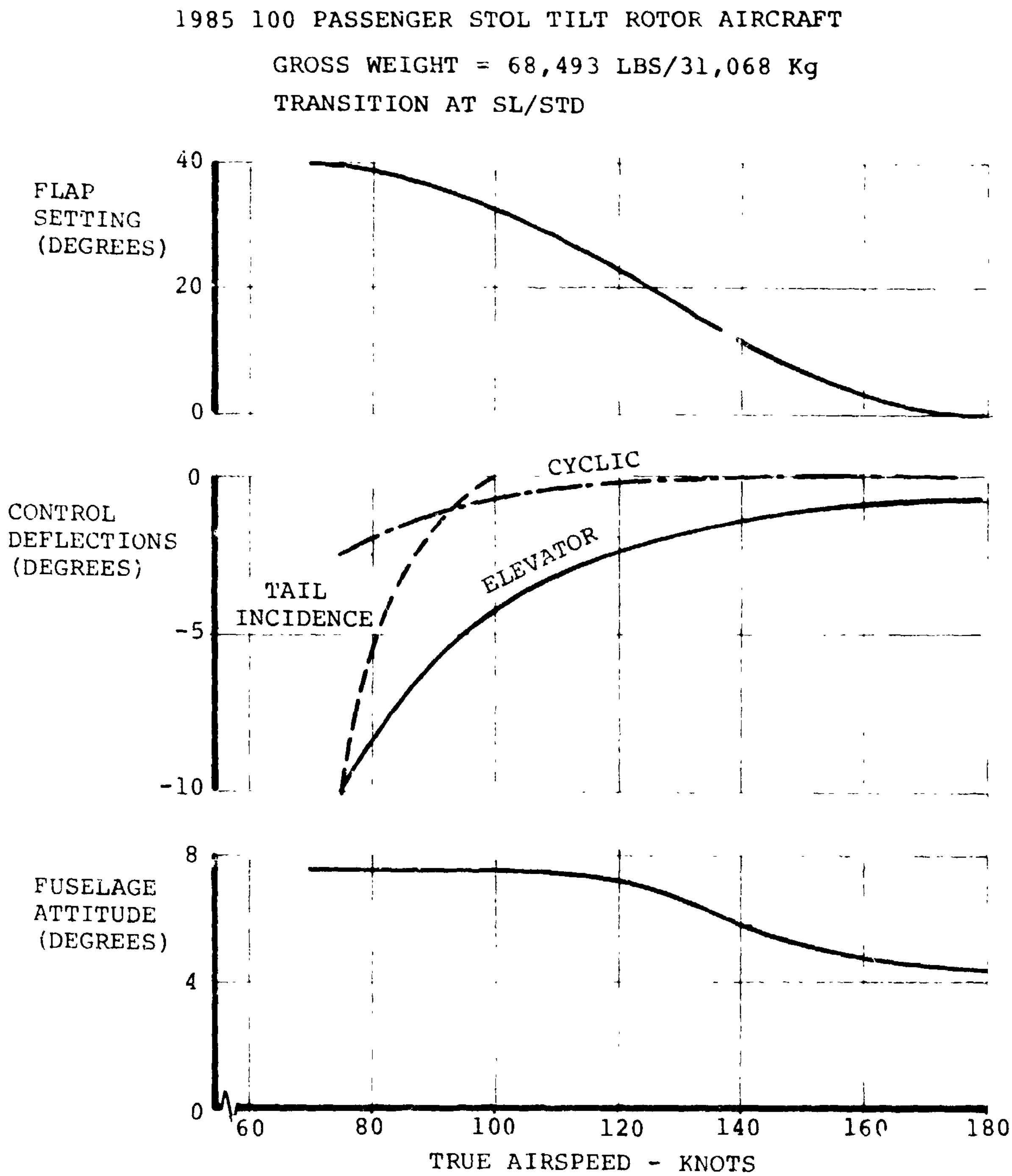

FIGURE 3.24. TYPICAL TRIM CHARACTERISTICS IN TRANSITION - CONTROL DEFLECTIONS AND FUSELAGE ATTITUDE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

GROSS WEIGHT = 68,493 LBS/31,068 Kg
TRANSITION AT SL/STD

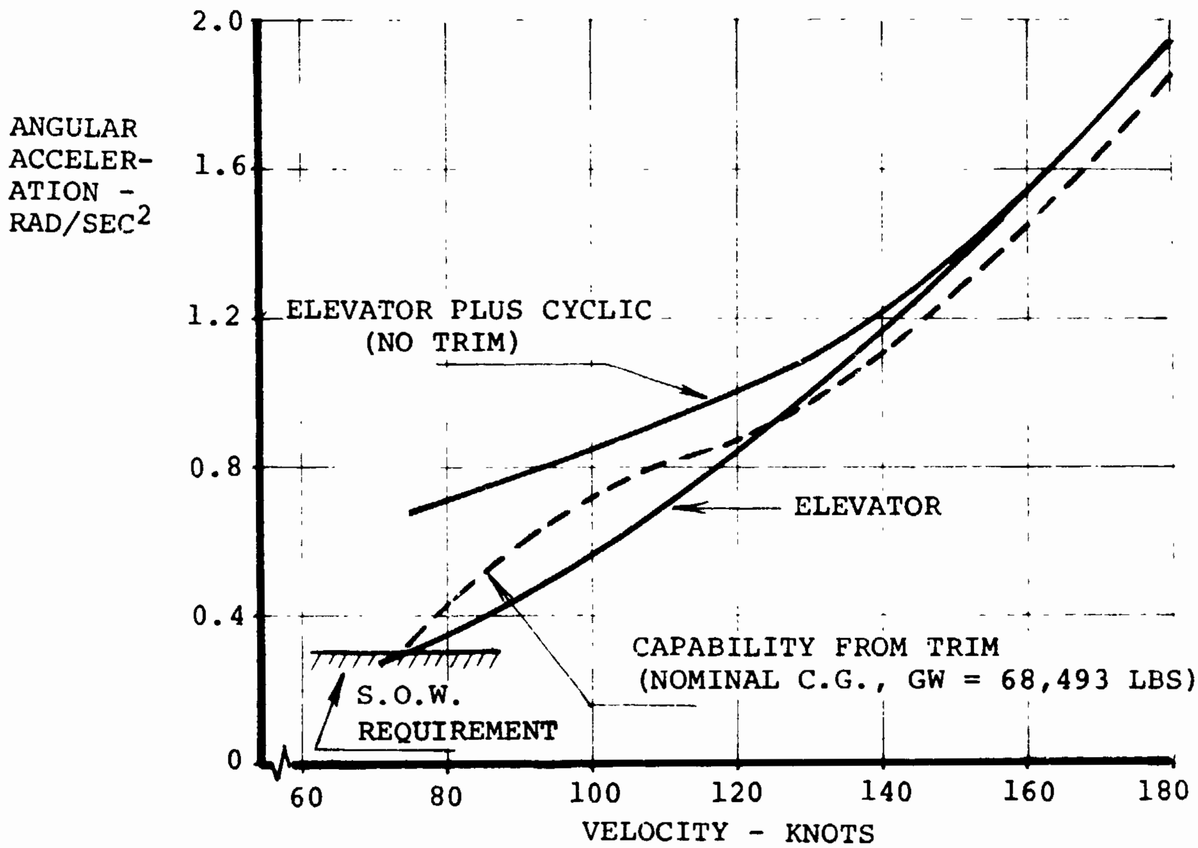

FIGURE 3.25. PITCH ANGULAR ACCELERATION CAPABILITY IN TRANSITION.

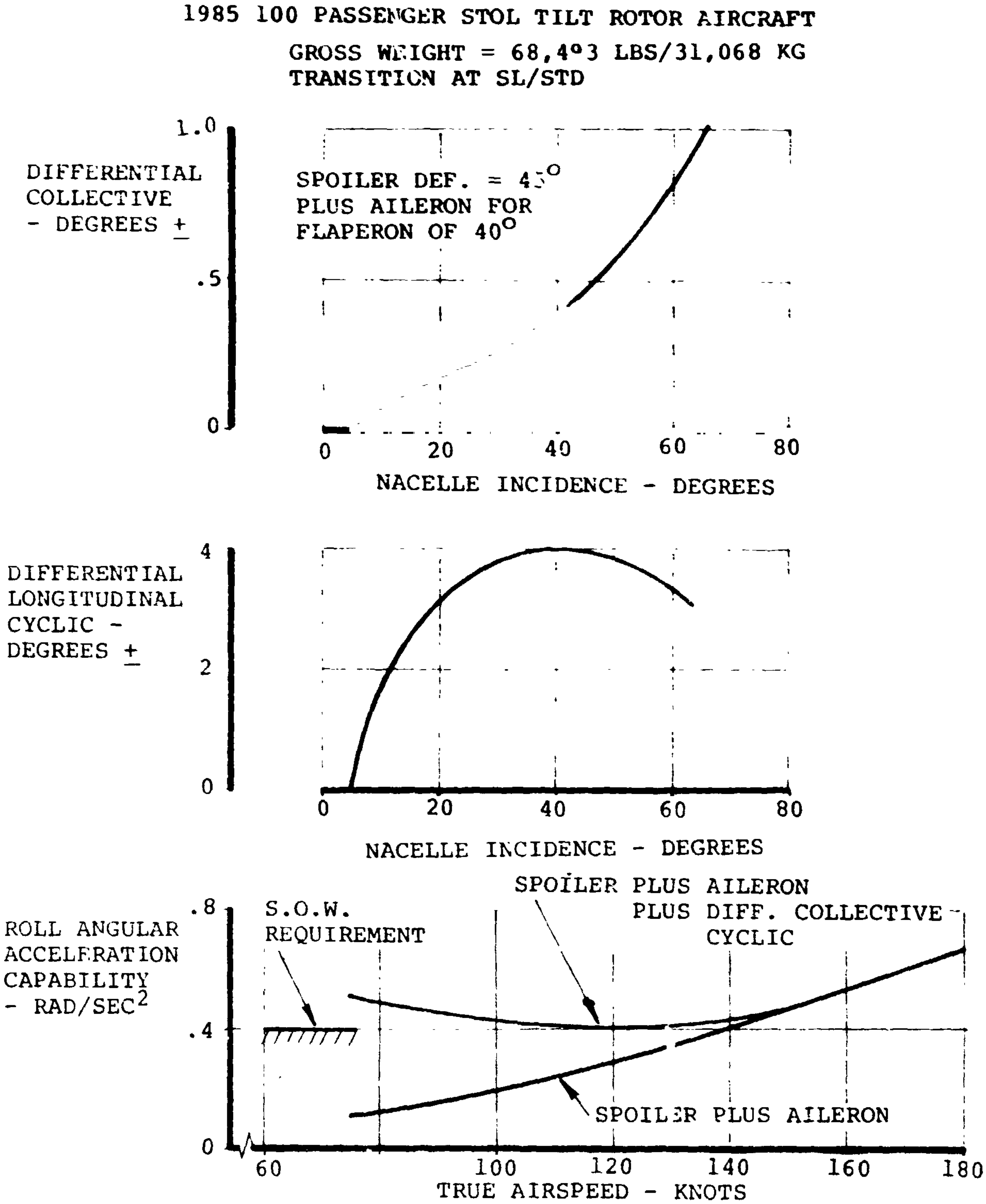

FIGURE 3.26. ROLL CONTROL IN TRANSITION.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

GROSS WEIGHT = 68,493 LBS/31,068 Kg
TRANSITION AT SL/STD
RUDDER DEFLECTION = 20°

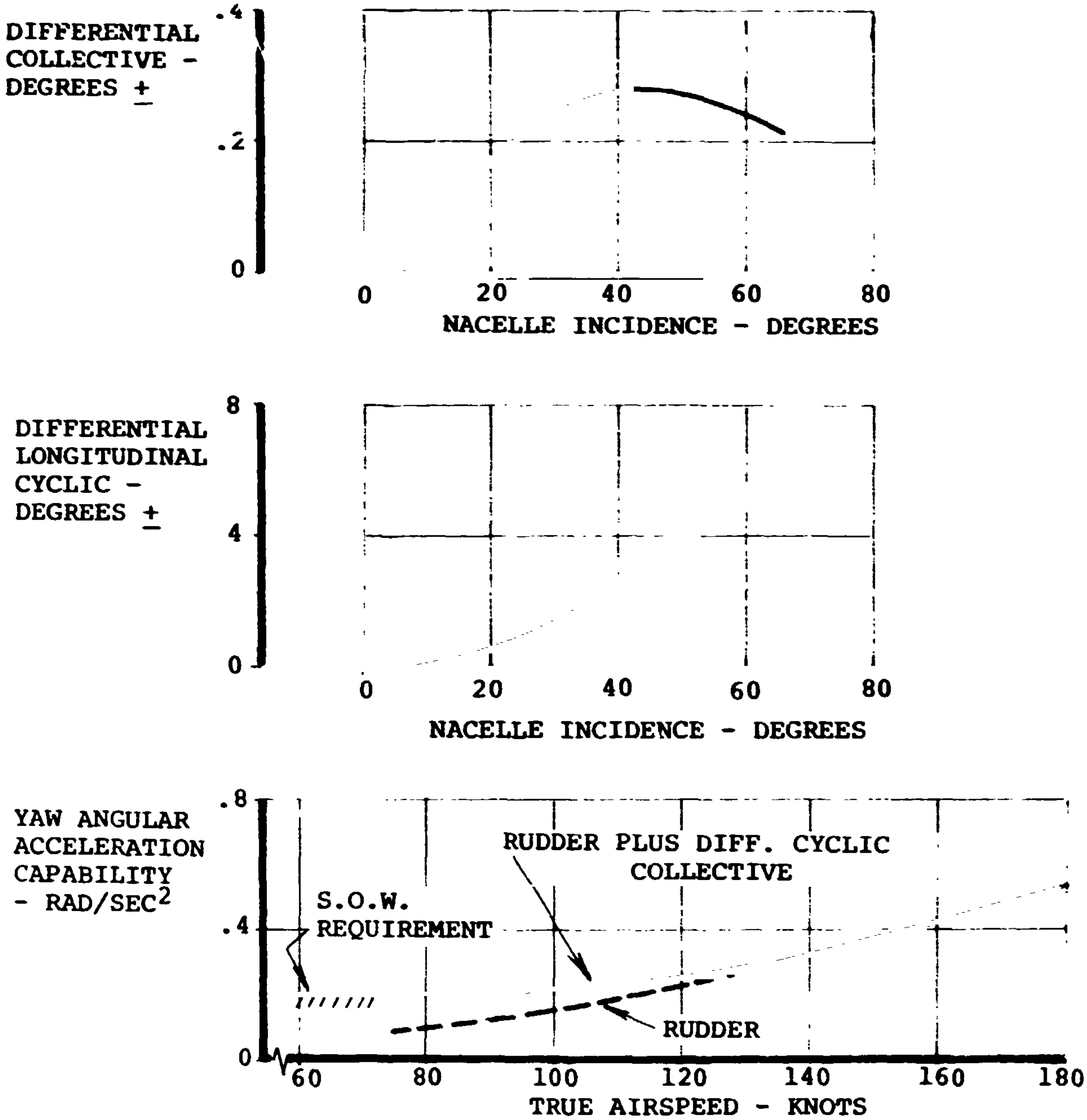

FIGURE 3.27. YAW CONTROL IN TRANSITION.

Cruise Flight Stability and Control

The longitudinal static stability of the design point STOL aircraft is shown in Figure 3.28. This figure shows the excursion in neutral point as a percentage of wing chord for various flight speeds and horizontal tail volume ratios. In cruise flight the most aft CG location is 33.5% MAC. The horizontal tail volume ratio of the STOL design is 1.46 and this provides a static margin in excess of 5% C at airspeeds as low as 140 knots. As airspeed increases the static margin increases.

Figures 3.29 and 3.30 present the cruise flight trim data for forward and aft CG locations. The trim aircraft angle of attack reduces as airspeed increases.

At the aft CG condition the trim angle of attack at normal rated power is 0.9 degrees at 14,000 feet altitude. At the forward CG location the trim angle of attack reduces to -0.2 degree.

Figure 3.31 and 3.32 show the aircraft pitch change and elevator required per g of maneuver load factor in a coordinated turn. Both pitch change and elevator required reduce as airspeed increases. At high speed the elevator travel per g is small and would result in a high stick force per g. This situation would require a longitudinal "feel" system to provide the pilot with a more displacement oriented control.

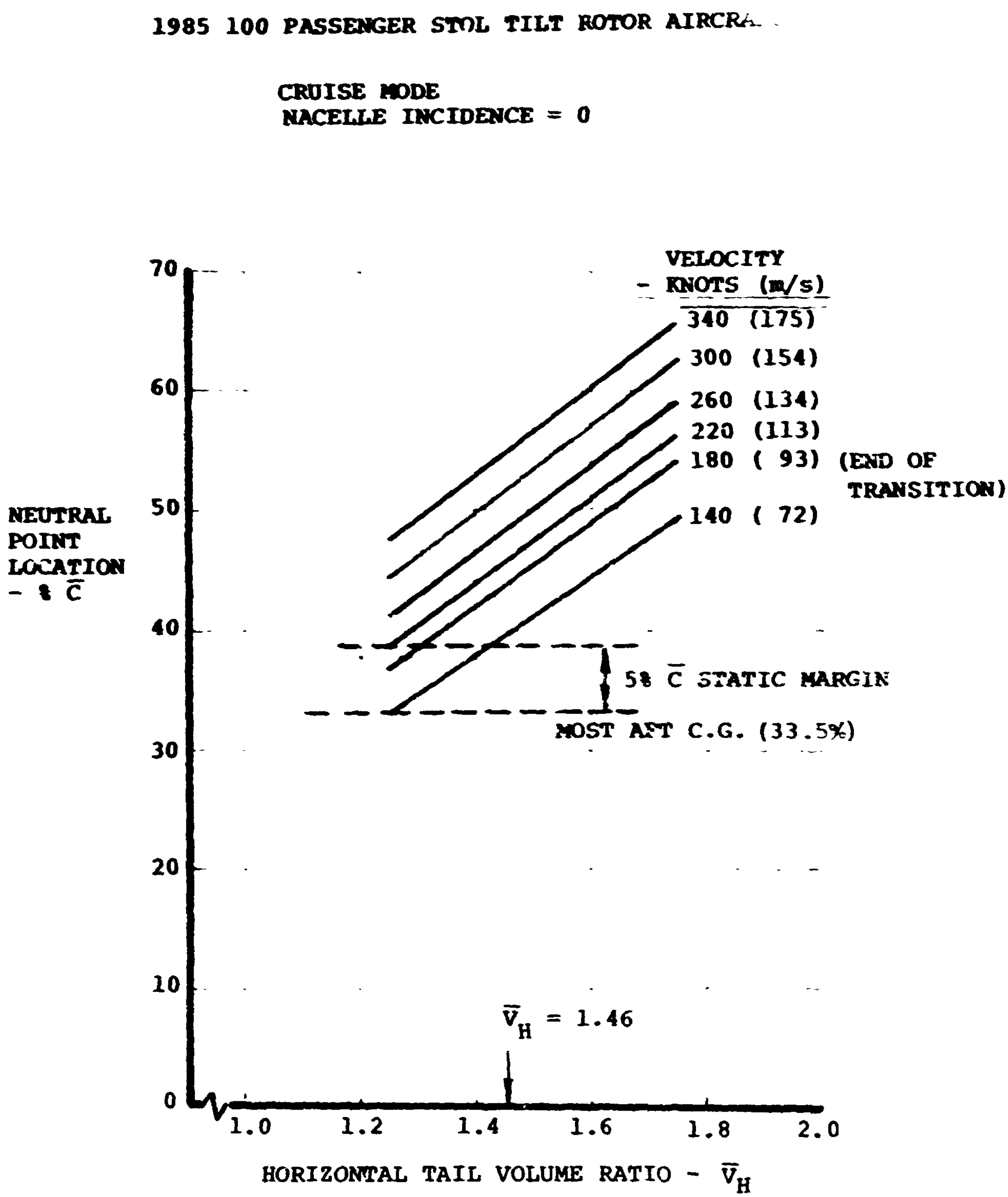

FIGURE 3.28. NEUTRAL POINT LOCATION AS A FUNCTION OF TAIL VOLUME RATIO AND VELOCITY.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
GROSS WEIGHT = 68,493 LBS/31,068 Kg

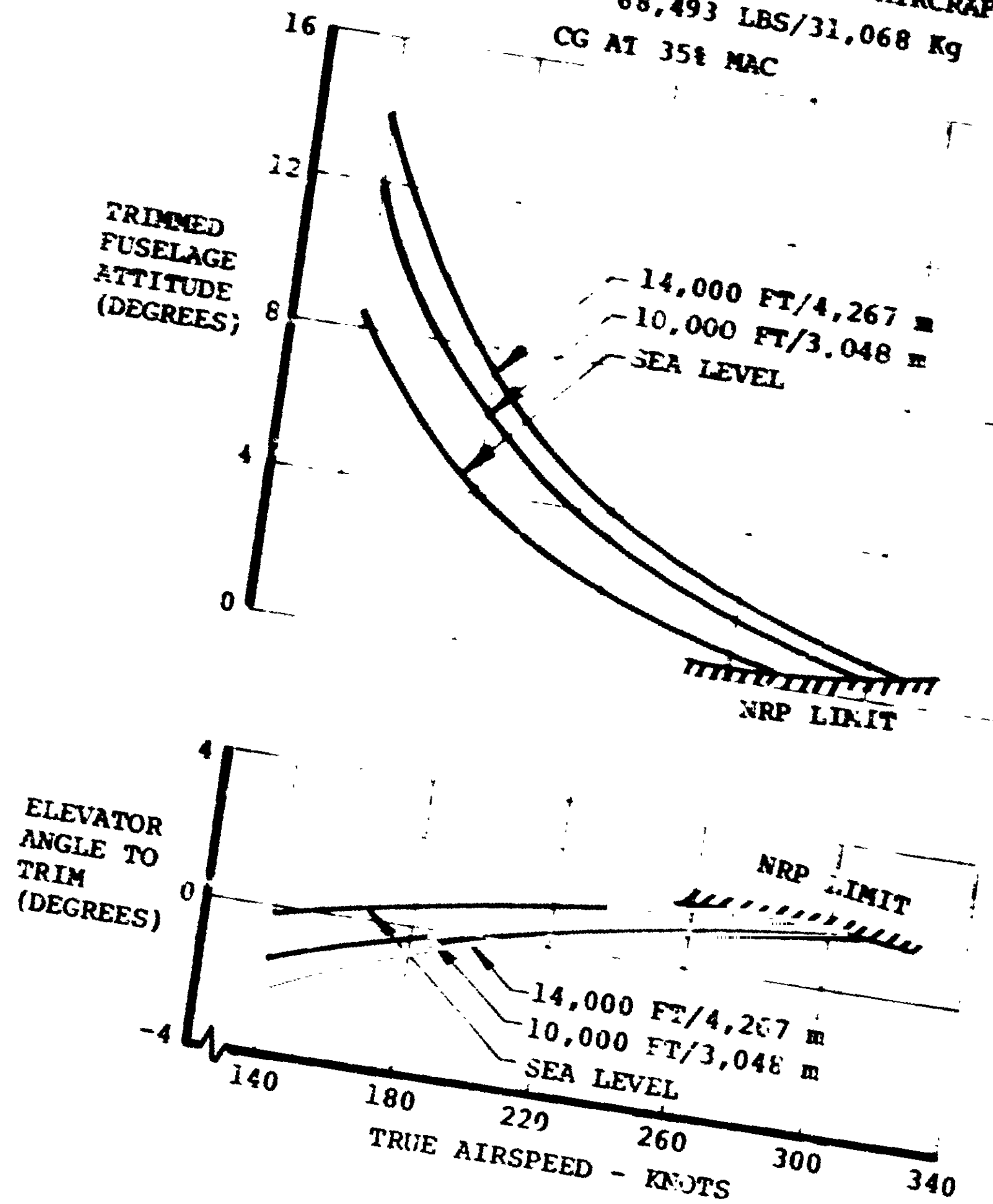

FIGURE 3.29. CRUISE TRIM CHARACTERISTICS - AFT CG.

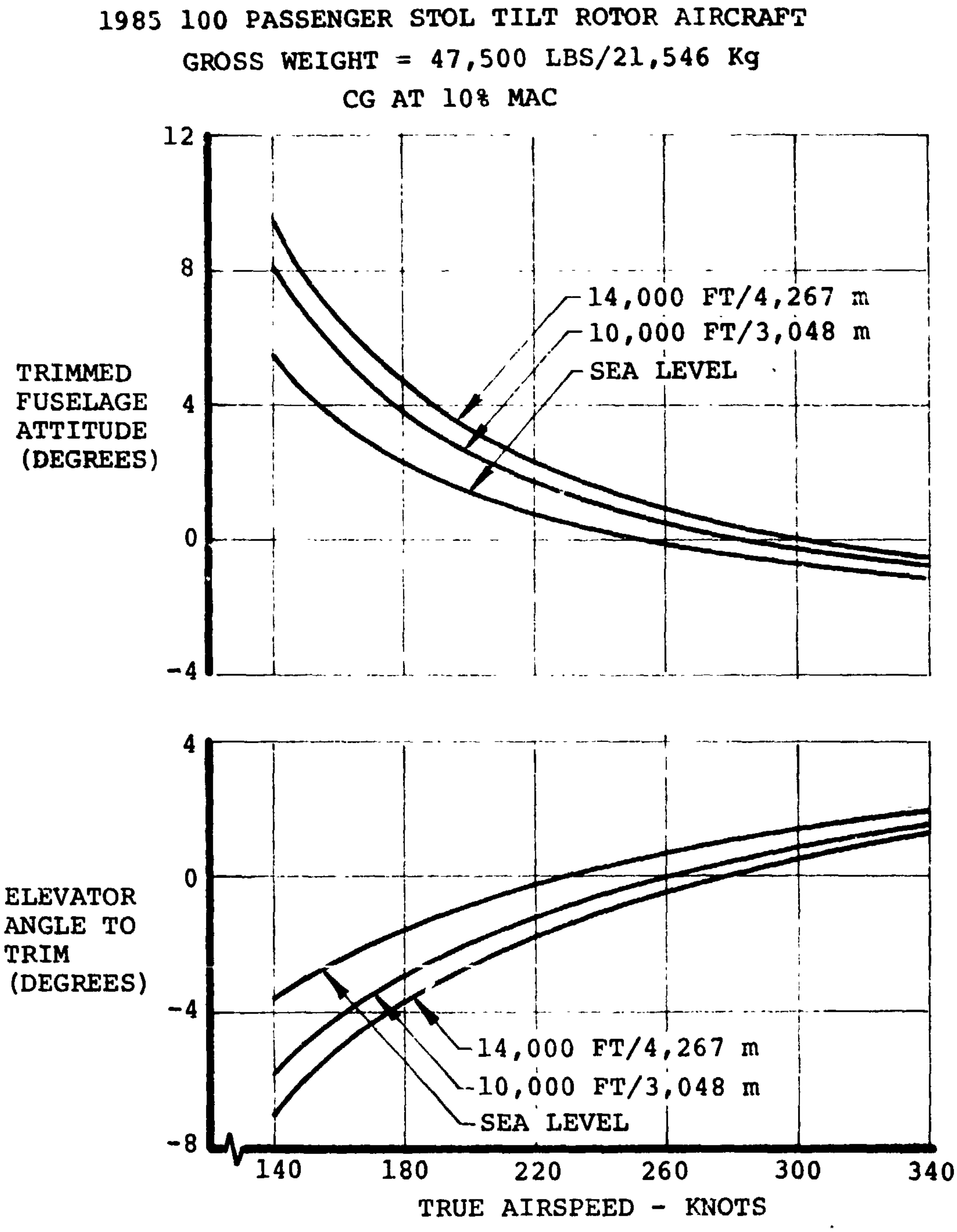

FIGURE 3.30. CRUISE TRIM CHARACTERISTICS - FORWARD CG.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

GROSS WEIGHT = 68,493 LBS/31,068 Kg

CG AT 35% MAC

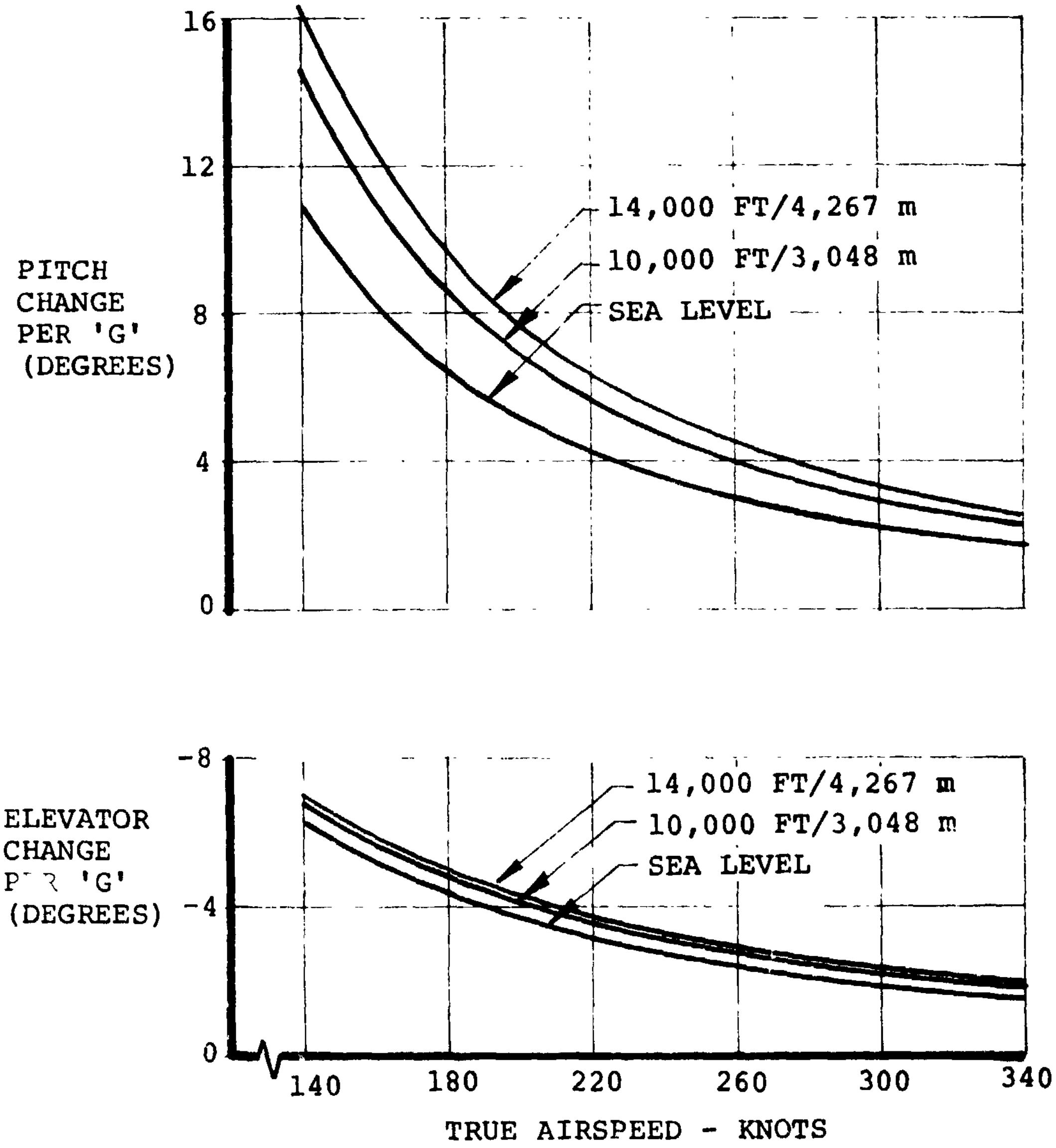

FIGURE 3.31. LONGITUDINAL CONTROL IN CRUISE - AFT CG.

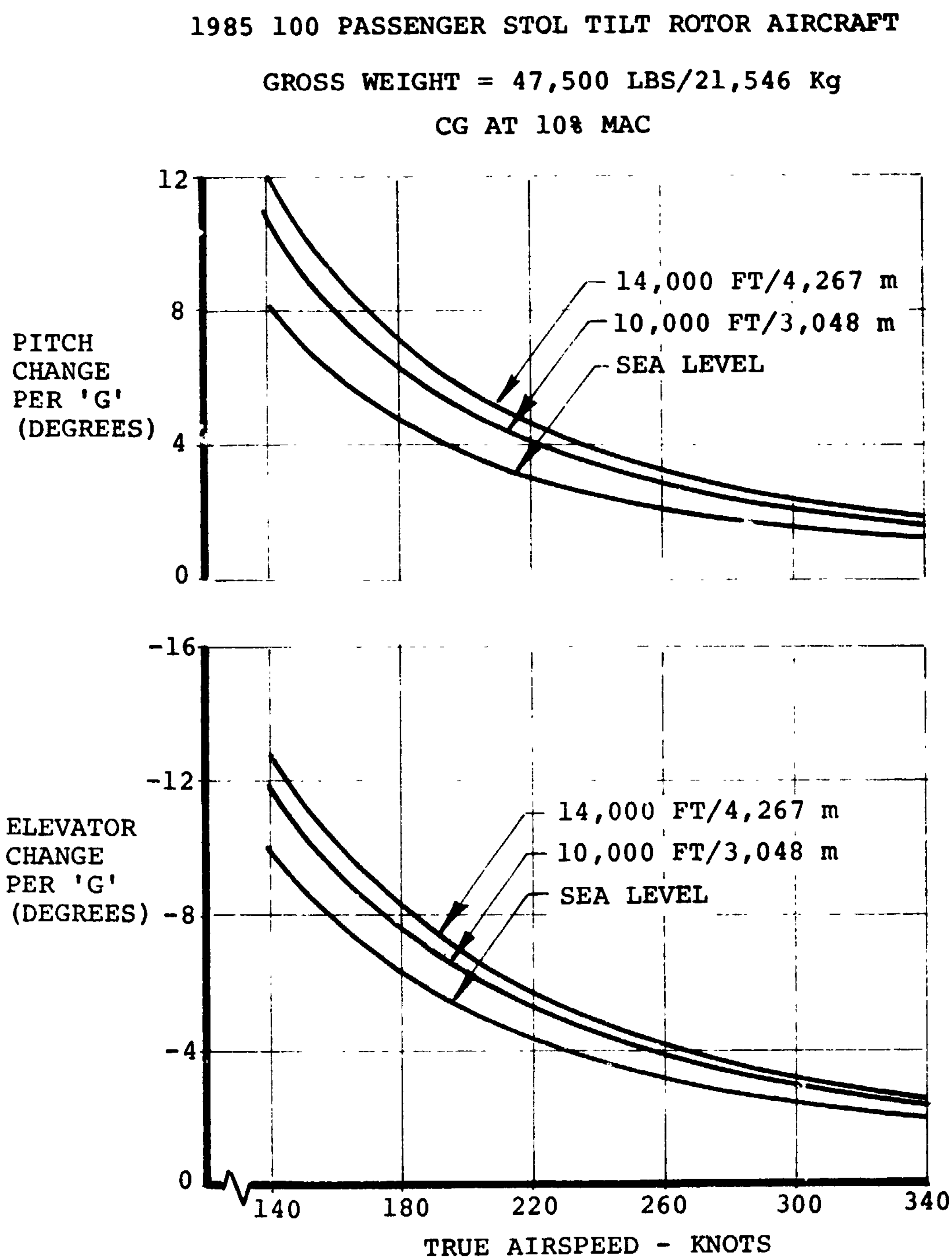

FIGURE 3.32. LONGITUDINAL CONTROL IN CRUISE - FORWARD CG.

The longitudinal dynamic response of the aircraft in the short period mode is shown in Figure 3.33. The roots are given for variations in aircraft gross weight and CG position. All cases meet the requirements of level 1 flying qualities (AGARD 577). The forward CG data show well damped periodic behavior.

For the aft CG case the roots become aperiodic at low speed. This effect manifests itself as an increasing pitch response time constant.

The pitch rate due to a unit elevator input is shown for the aft CG case in Figure 3.34 and shows acceptable behavior through the cruise flight range. At 180 knots the pitch time constant is 1.05 seconds and this reduces to 0.55 seconds at 300 knots.

The MIL-F-8785B(ASG) criteria for the short period mode is a response type of criteria and the STOL design point vehicle is shown in Figure 3.35 to meet level 1 criteria at all gross weights and CG positions.

The dynamic characteristics of the phugoid mode are shown in Figure 3.36. The phugoid roots are periodic and damped except for the low speed condition at design gross weight at 14,000 feet. This root splits and one root appears on the real axis in the right hand plane. This is of no practical significance since the time to double amplitude for this case is 42 seconds.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

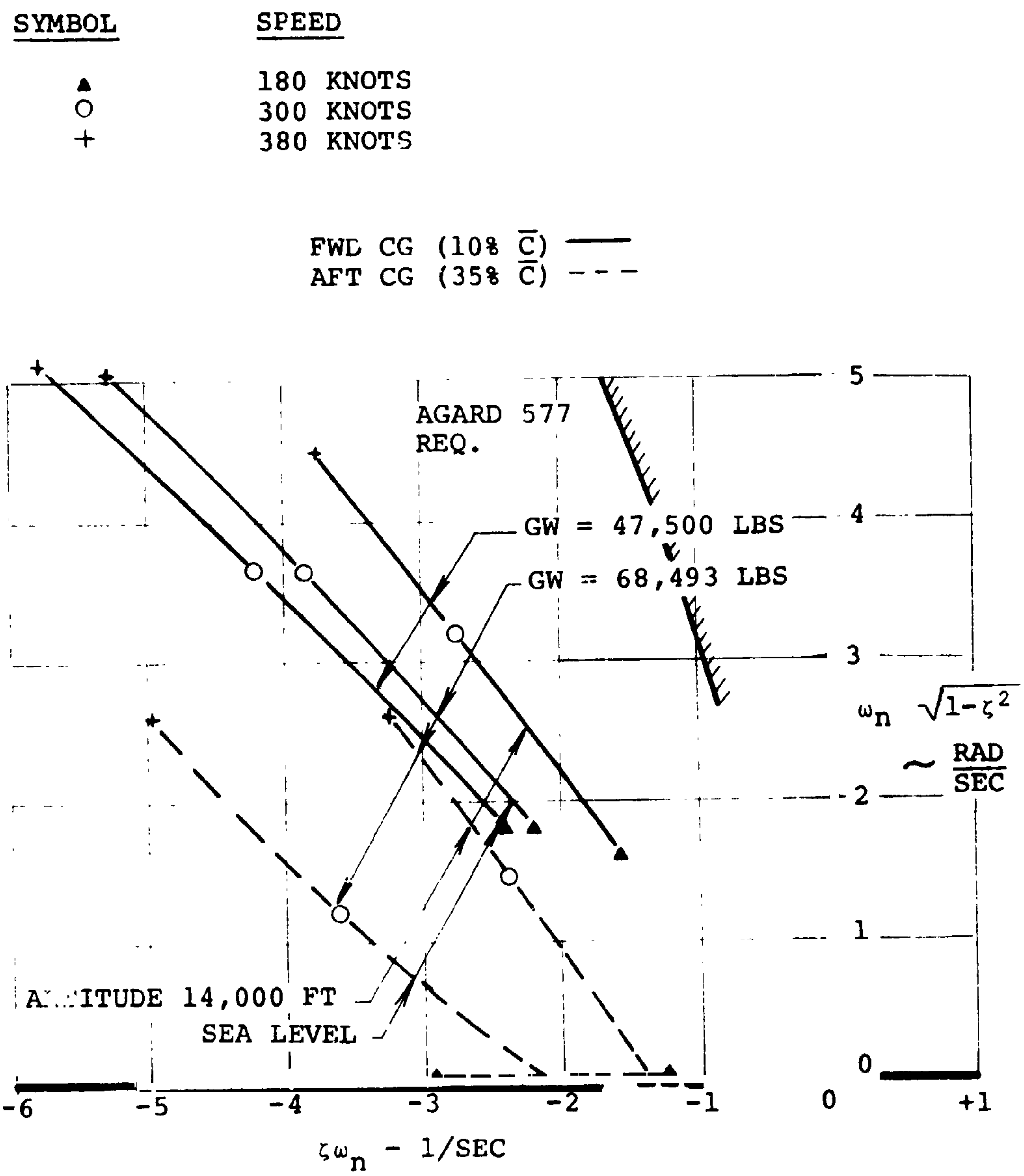

FIGURE 3.33. LONGITUDINAL DYNAMIC RESPONSE CHARACTERISTICS - SHORT PERIOD.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

GROSS WEIGHT = 68,493 LBS/31,068 Kg
C.G. AT 35% MAC
CRUISE AT SL/STD

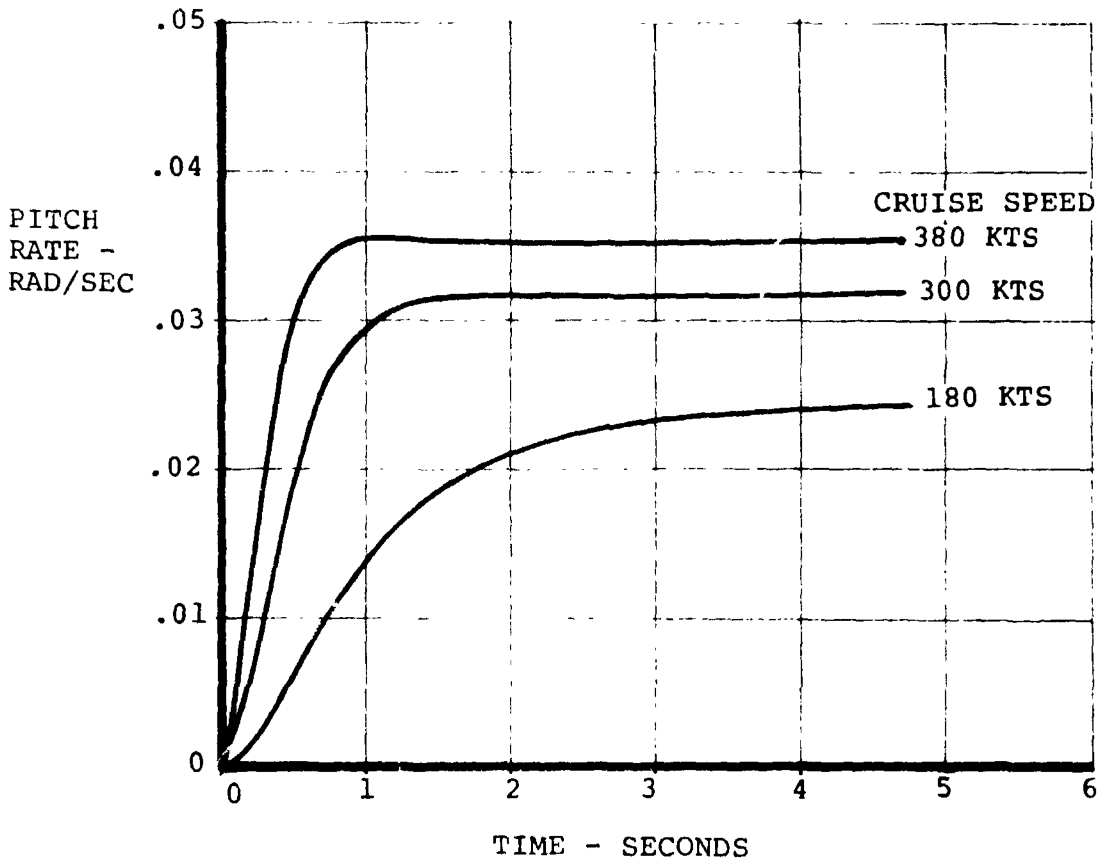

FIGURE 3.34. PITCH RATE RESPONSE TO UNIT STEP ELEVATOR INPUT.

MIL-F-8785B(ASG)

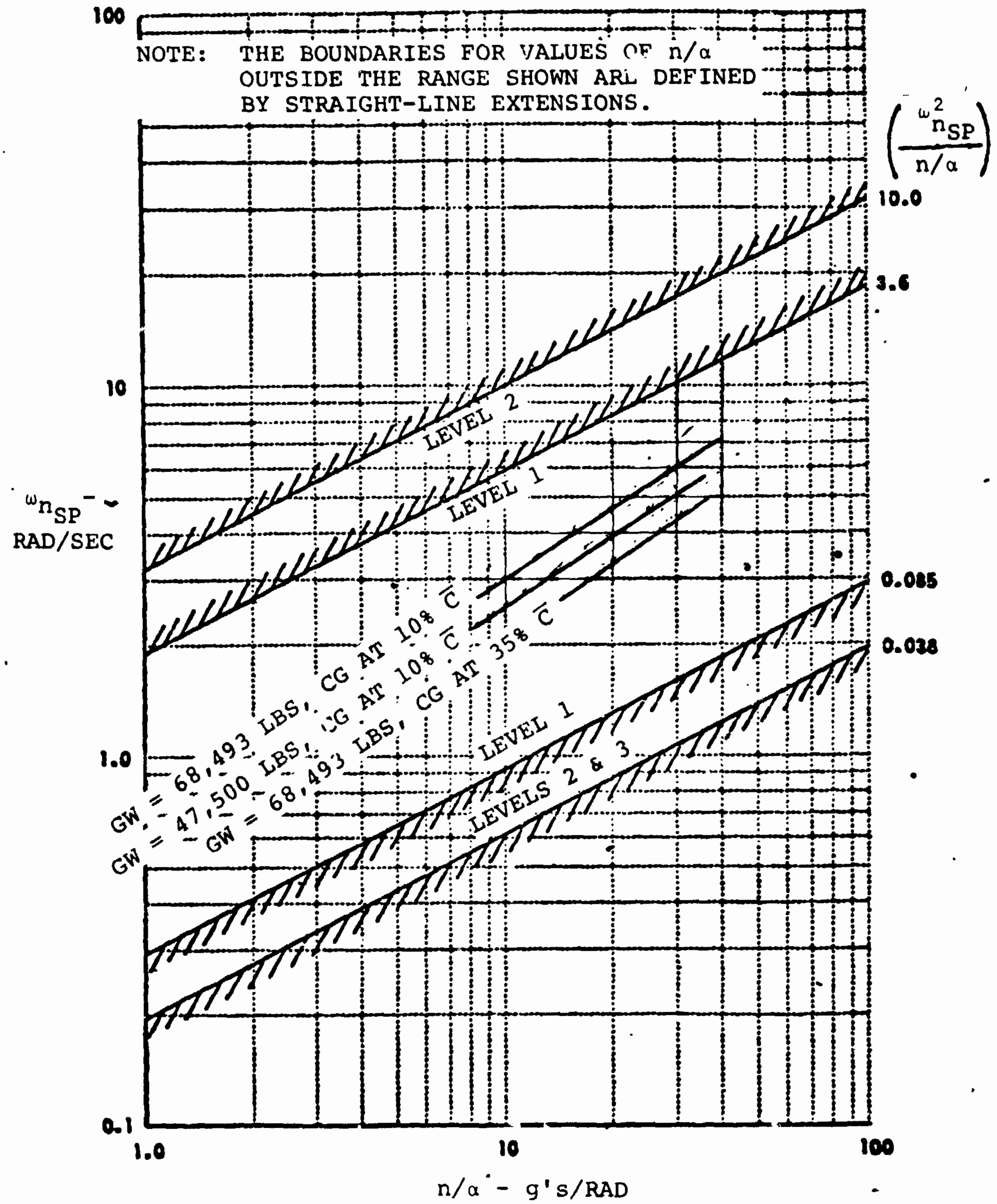

FIGURE 3.35. SHORT-PERIOD FREQUENCY REQUIREMENTS.

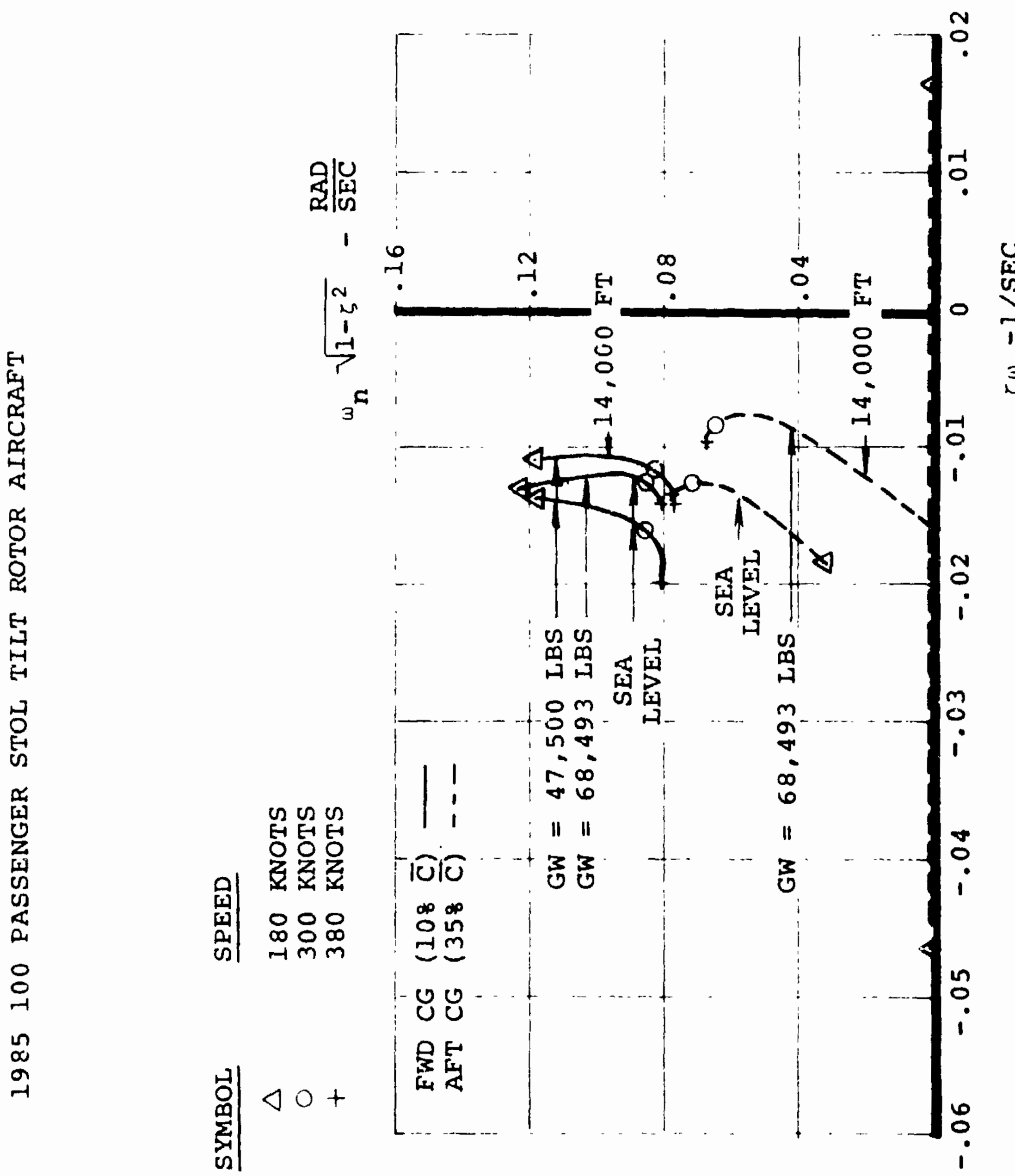

FIGURE 3.36. LONGITUDINAL D.NAMIC RESPONSE CHARACTERISTICS - PHUGOID.

The lateral static stability derivatives in cruise are shown in Figure 3.37. The aircraft is statically directionally stable (positive C_{n_β}) and the level increases with airspeed. The dihedral effect is positive (i.e. negative C_{ℓ_β}), but this effect is reduced as airspeed increases. This is the result of the rotor contribution. The decreasing C_{ℓ_β} and the increase in C_{n_β} as airspeed increases tend to improve the aircraft dutch roll stability and reduce the spiral mode stability.

The side force due to sideslip derivative C_{y_β} is large and results in a relatively large roll angle to compensate in flying a straight ground track in sideslip.

This effect is in evidence in the sideslip characteristics in cruise shown in Figure 3.38. At 300 knots the roll angle per degree of rudder is 3.3 degrees to maintain a straight ground track. The lateral stick per degree of rudder is always positive indicating normal control direction in sideslip.

The rudder effectiveness in sideslip is high and decreases as airspeed increases.

The roll rate derivatives are shown in Figure 3.39. The roll rate damping is high due to the rotor contribution. The yawing moment due to rate of roll decreases as airspeed increases.

The yaw rate derivatives are shown in Figure 3.40 and indicate a high C_{n_r} or yaw damping.

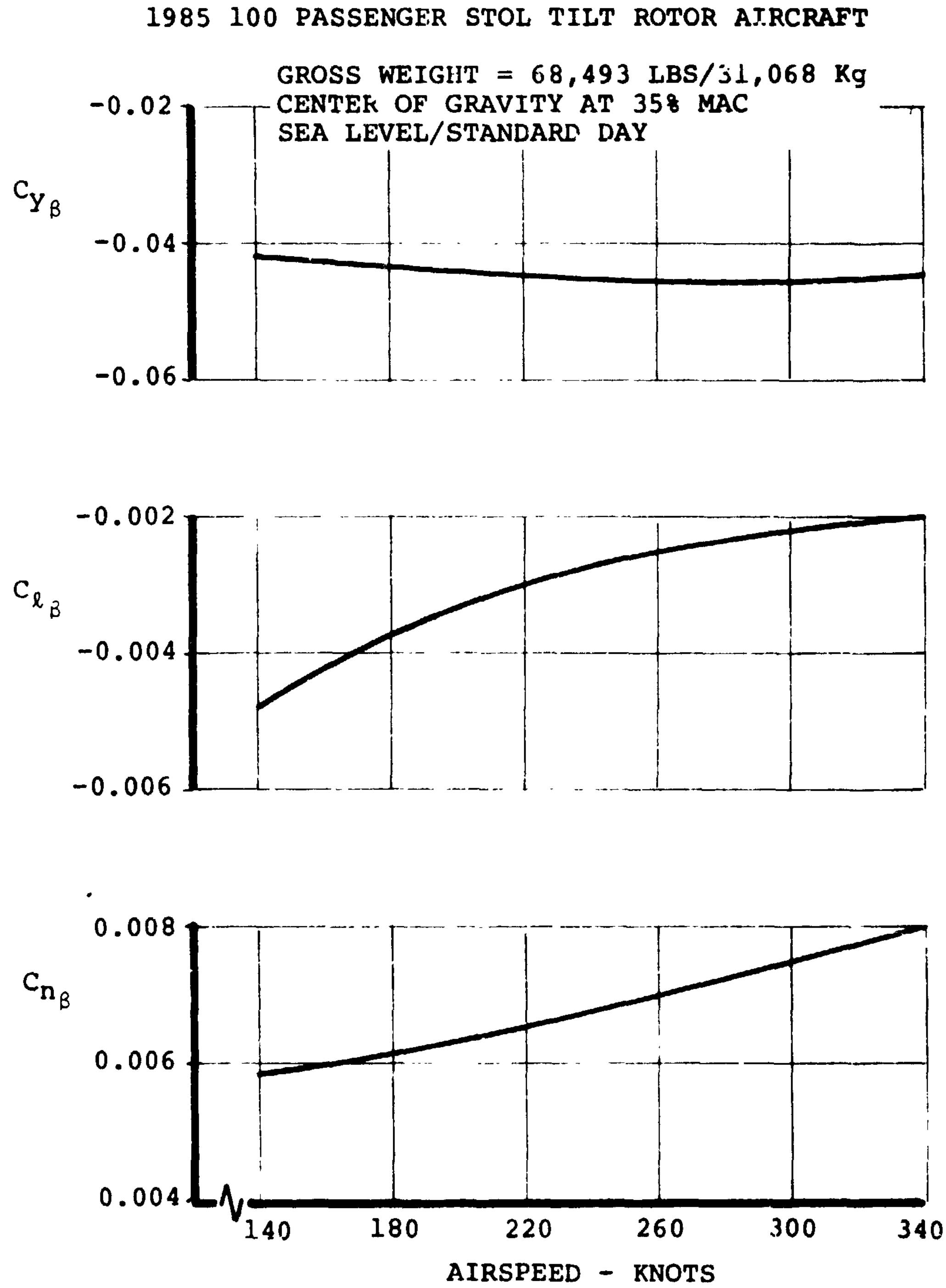

FIGURE 3.37. STATIC LATERAL-DIRECTIONAL DERIVATIVES IN CRUISE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

GROSS WEIGHT = 68,493 LBS/31,068 Kg
CENTER OF GRAVITY AT 35% MAC
SEA LEVEL/STANDARD DAY

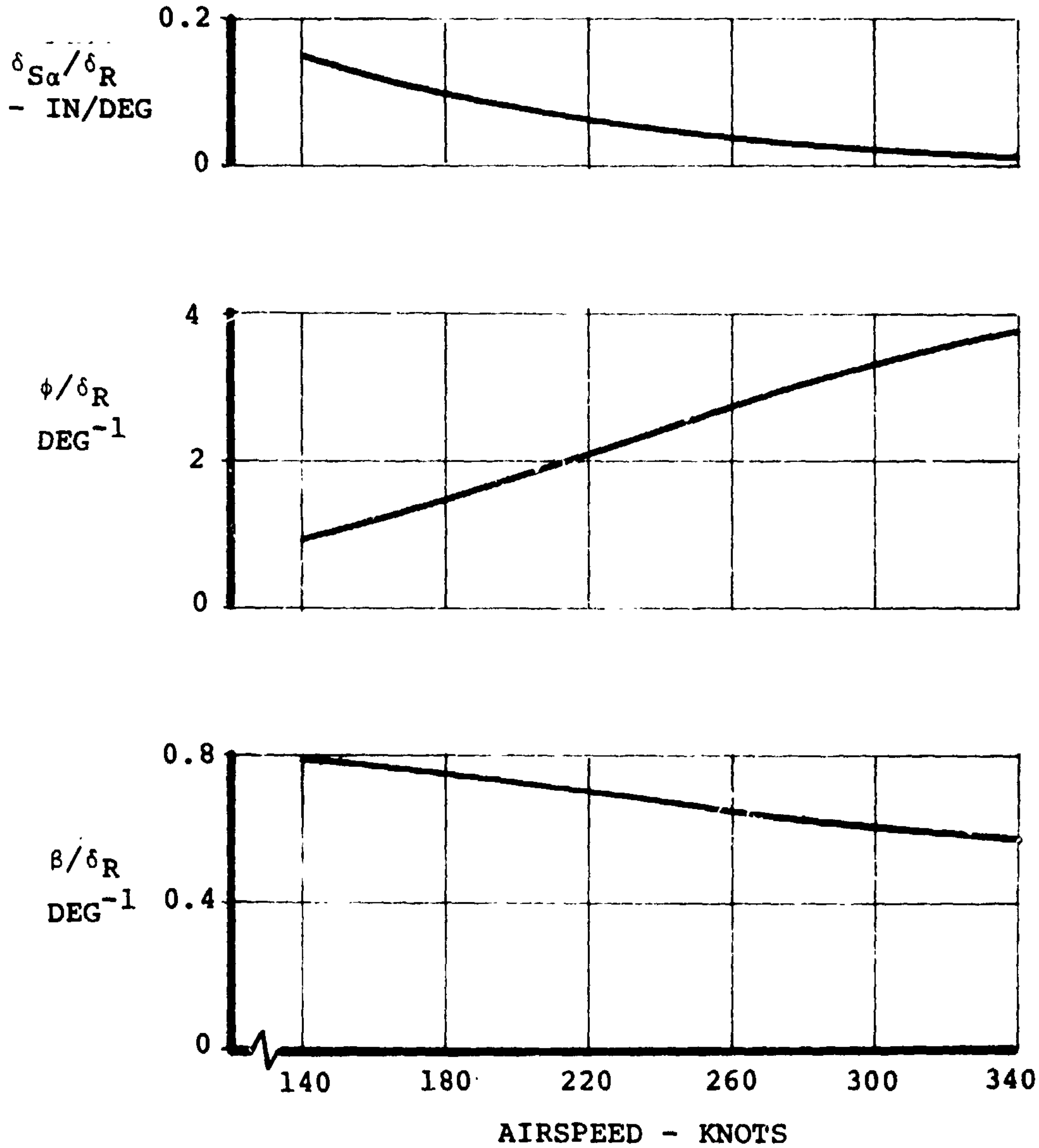

FIGURE 3.38. SIDE SLIP CHARACTERISTICS IN CRUISE.

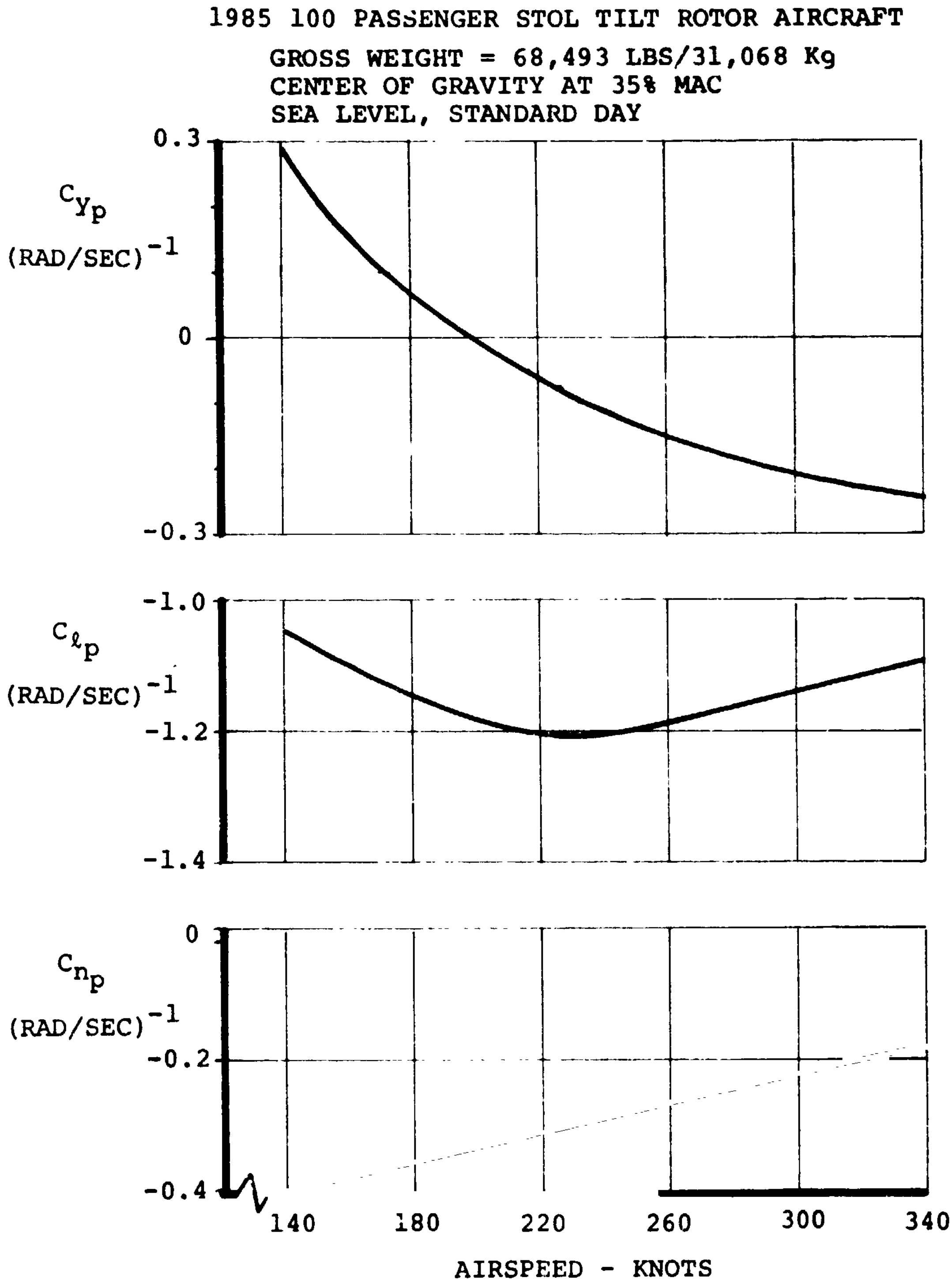

FIGURE 3.39. LATERAL-DIRECTIONAL ROLL RATE DERIVATIVES IN CRUISE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

GROSS WEIGHT = 68,493 LBS/31,068 Kg
CENTER OF GRAVITY AT 35% MAC
SEA LEVEL/STANDARD DAY

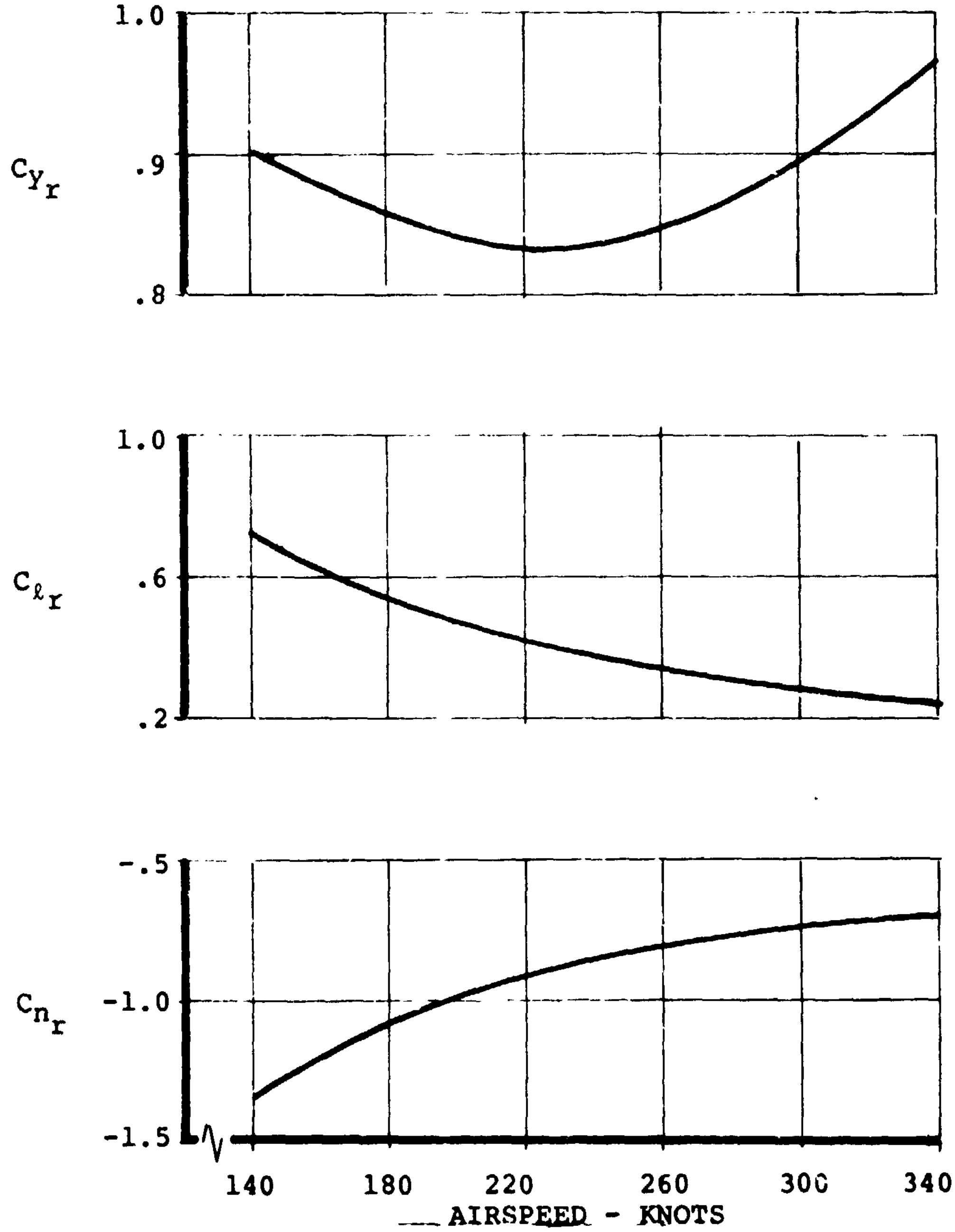

FIGURE 3.40. LATERAL-DIRECTIONAL YAW RATE DERIVATIVES IN CRUISE.

The data plotted in Figures 3.37, 3.39 and 3.40 all apply to flight at sea level, standard day, at the design gross weight with the center of gravity at the 35 percent mean aerodynamic chord location. Corresponding data for alternate conditions of weight, altitude and center of gravity location are tabulated in Tables 3.6, 3.7 and 3.8.

The response of the aircraft in the dutch roll mode is shown in root locus form in Figure 3.41. The roots indicate a periodic, well damped mode for all weight and altitude conditions calculated.

Figure 3.42 shows the roll mode time constant and spiral mode data. The roll mode time constant meets MIL-F-8785B for the cruise flight envelope and over most of the range is less than 1 second. The spiral mode is mildly unstable and Figure 3.42 shows the reciprocal of the time to double amplitude. In the worst case the time to double amplitude is greater than two minutes which easily meets the MIL-F-8785B specification for level 1 flying qualities.

Gust Sensitivity and Direct Lift Control

The 100 passenger STOL tilt rotor exhibits a similar degree of gust sensitivity as was apparent in the VTOL study (Reference 1). The situation without alleviation shown in Figure 3.43 for minimum and maximum operating gross weights at 10,000 feet and 14,000 feet, indicates that substantial amounts of lift must be dumped if the gust sensitivity criterion is to be met. It is envisioned that this will be

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

HEIGHT - FT	SL	SL	SL	14,000	14,000
WEIGHT - LB	68,493	68,493	47,500	68,493	47,500
C.G. % MAC	35%	10%	10%	35%	10%

C_{y_β} - Sideforce Due to Sideslip

Speed-Knots					
140	-.04217	-.04217	-.04217	-.04288	-.04288
180	-.04394	-.04394	-.04344	-.04479	-.04479
220	-.04494	-.04444	-.04494	-.04588	-.04588
260	-.04511	-.04511	-.04511	-.04610	-.04610
300	-.04512	-.04512	-.04512	-.04615	-.04615
340	-.04435	-.04435	-.04435	-.04538	-.04538

C_{ℓ_β} - Rolling Moment Due to Sideslip

Speed-Knots					
140	-.00482	-.00460	-.00476	-.00454	-.00449
180	-.00381	-.00342	-.00371	-.00357	-.00348
220	-.00310	-.00304	-.00298	-.00290	-.00278
260	-.00249	-.00245	-.00237	-.00227	-.00217
300	-.00221	-.00219	-.00210	-.00201	-.00191
340	-.00204	-.00204	-.00195	-.00186	-.00177

C_{n_β} - Yawing Moment Due to Sideslip

Speed-Knots					
140	.00595	.00686	.00686	.00588	.00681
180	.00622	.00718	.00718	.00618	.00715
220	.00661	.00759	.00759	.00661	.00762
260	.00708	.00808	.00808	.00711	.00813
300	.00751	.00850	.00850	.00759	.00862
340	.00803	.00900	.00900	.00818	.00918

TABLE 3.6. STATIC LATERAL-DIRECTIONAL DERIVATIVES IN CRUISE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

HEIGHT - FT	SL	SL	SL	14,000	14,000
WEIGHT - LB	68,493	68,493	47,500	68,493	47,500
C.G. % MAC	35%	10%	10%	35%	10%

C_{Y_p} - Sideforce Due to Roll Rate

Speed-Knots					
140	.2882	.2882	.2514	.3634	.3070
180	.0722	.0722	.0500	.1116	.0769
220	-.0605	-.0605	-.0753	-.0417	-.0644
260	-.1503	-.1503	-.1608	-.1438	-.1600
300	-.2095	-.2095	-.2173	-.2111	-.2231
340	-.2426	-.2426	-.2487	-.2490	-.2623

C_{ℓ_p} - Rolling Moment Due to Roll Rate

140	-1.0460	-1.0468	-1.0436	-1.0898	-1.0803
180	-1.1455	-1.1455	-1.1435	-1.1876	-1.1830
220	-1.2017	-1.2015	-1.1992	-1.2451	-1.2414
260	-1.1862	-1.1861	-1.1833	-1.2282	-1.2246
300	-1.1402	-1.1402	-1.1369	-1.1797	-1.1759
340	-1.0921	-1.0922	-1.0888	-1.1296	-1.1258

C_{n_p} - Yawing Moment Due to Roll Rate

140	- .4026	- .4072	- .3544	- .5100	- .4337
180	- .3611	- .3611	- .3293	- .4334	- .3845
220	- .3146	- .3116	- .2904	- .3673	- .3316
260	- .2633	- .2582	- .2432	- .3032	- .2748
300	- .2212	- .2047	- .2035	- .2524	- .2284
340	- .1893	- .1820	- .1734	- .2144	- .1936

TABLE 3.7. LATERAL-DIRECTIONAL ROLL RATE DERIVATIVES IN CRUISE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

HEIGHT - FT	SL	SL	SL	14,000	14,000
WEIGHT - LB	68,493	68,493	47,500	68,493	47,500
C.G. % MAC	35%	10%	10%	35%	10%

C_{y_r} - Sideforce Due to Yaw Rate

Speed-Knots					
140	.9022	.9566	.9566	.8877	.9455
180	.8576	.9114	.9114	.8415	.8987
220	.8313	.8834	.8834	.8151	.8705
260	.8461	.8946	.8946	.8318	.8834
300	.8945	.9395	.9395	.8858	.9336
340	.9681	1.0057	1.0057	.9668	1.0067

C_{ℓ_r} - Rolling Moment Due to Yaw Rate

140	.7287	.7338	.6061	.9714	.7795
180	.5400	.5388	.4621	.6917	.5717
220	.4323	.4279	.3769	.5373	.4536
260	.3387	.3320	.2957	.4644	.3515
300	.2800	.2723	.2452	.3385	.2879
340	.2430	.2343	.2132	.2898	.2476

C_{n_r} - Yawing Moment Due to Yaw Rate

140	-1.3492	-1.3429	-1.3429	-1.4655	-1.4031
180	-1.0738	-1.0703	-1.0617	-1.1355	-1.1113
220	- .9135	- .9125	- .9087	- .9537	- .9435
260	- .8068	- .8074	- .8054	- .8358	- .8317
300	.7424	- .7434	- .7423	- .7660	- .7644
340	.7000	- .7016	- .7009	- .7209	- .7211

TABLE 3.8. LATERAL-DIRECTIONAL YAW RATE DERIVATIVES IN CRUISE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

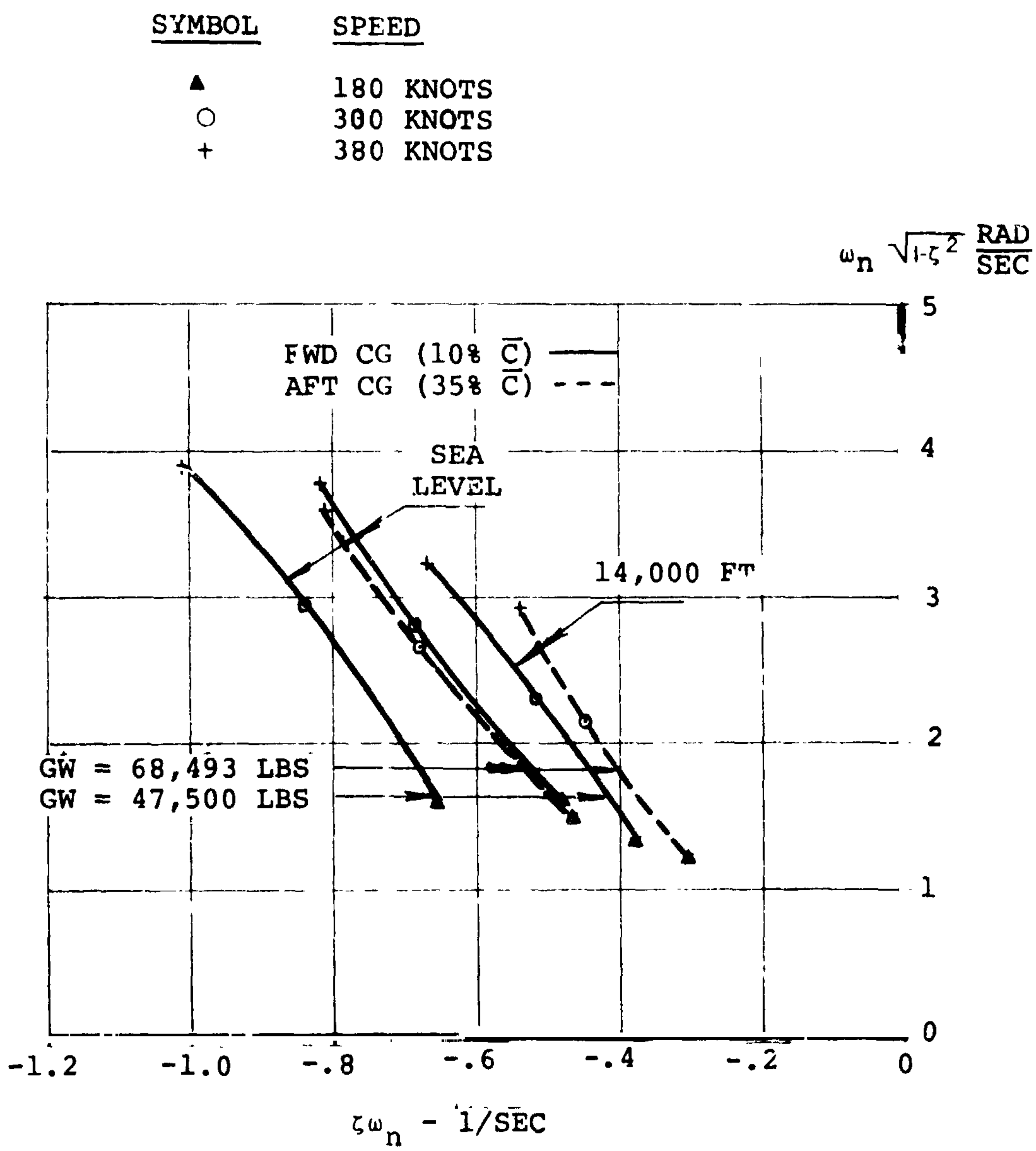

FIGURE 3.41. LATERAL-DIRECTIONAL DYNAMIC RESPONSE CHARACTERISTICS - DUTCH ROLL MODE.

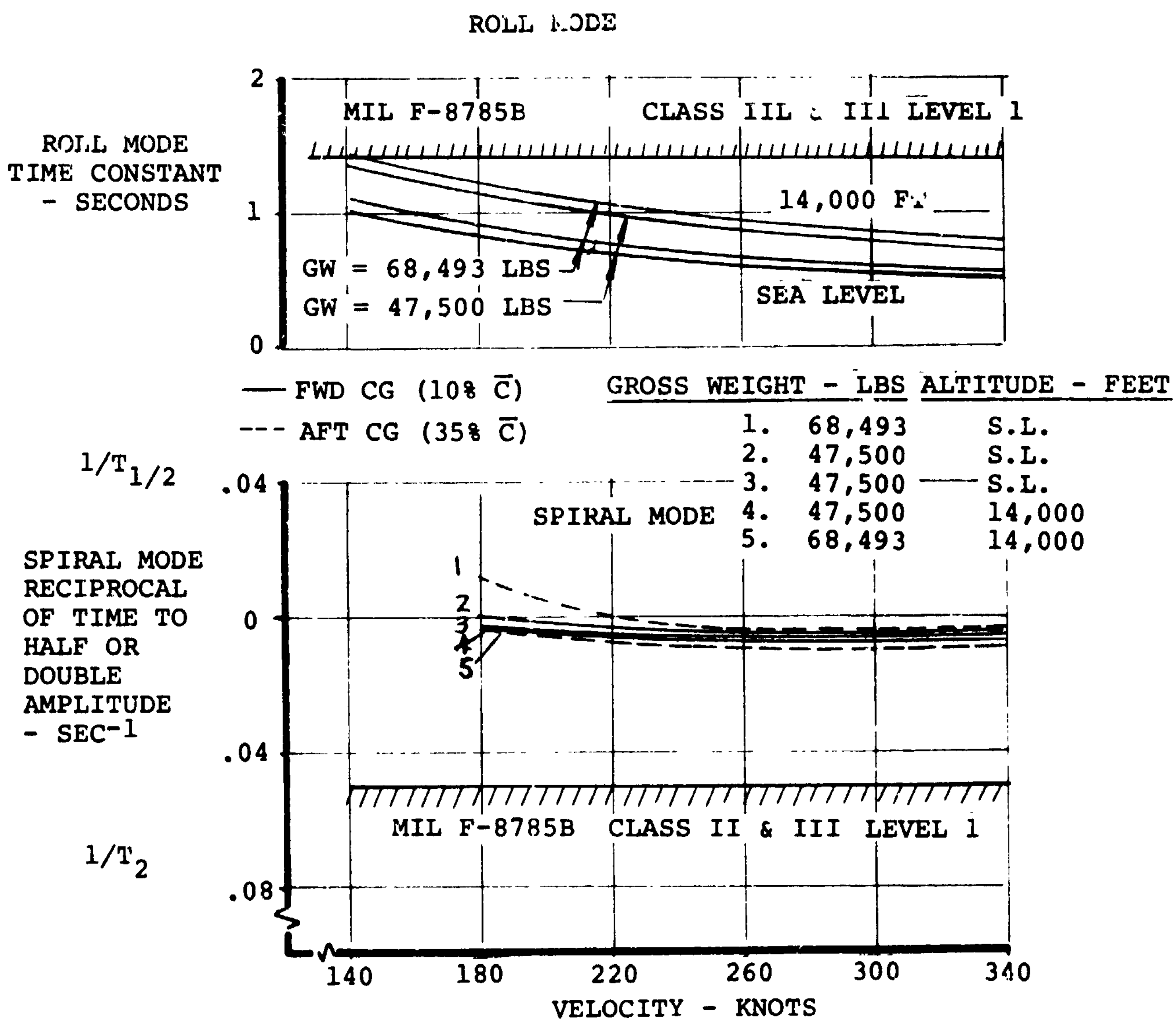

FIGURE 3.42. LATERAL-DIRECTIONAL DYNAMIC RESPONSE CHARACTERISTICS - ROLL AND SPIRAL MODES.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

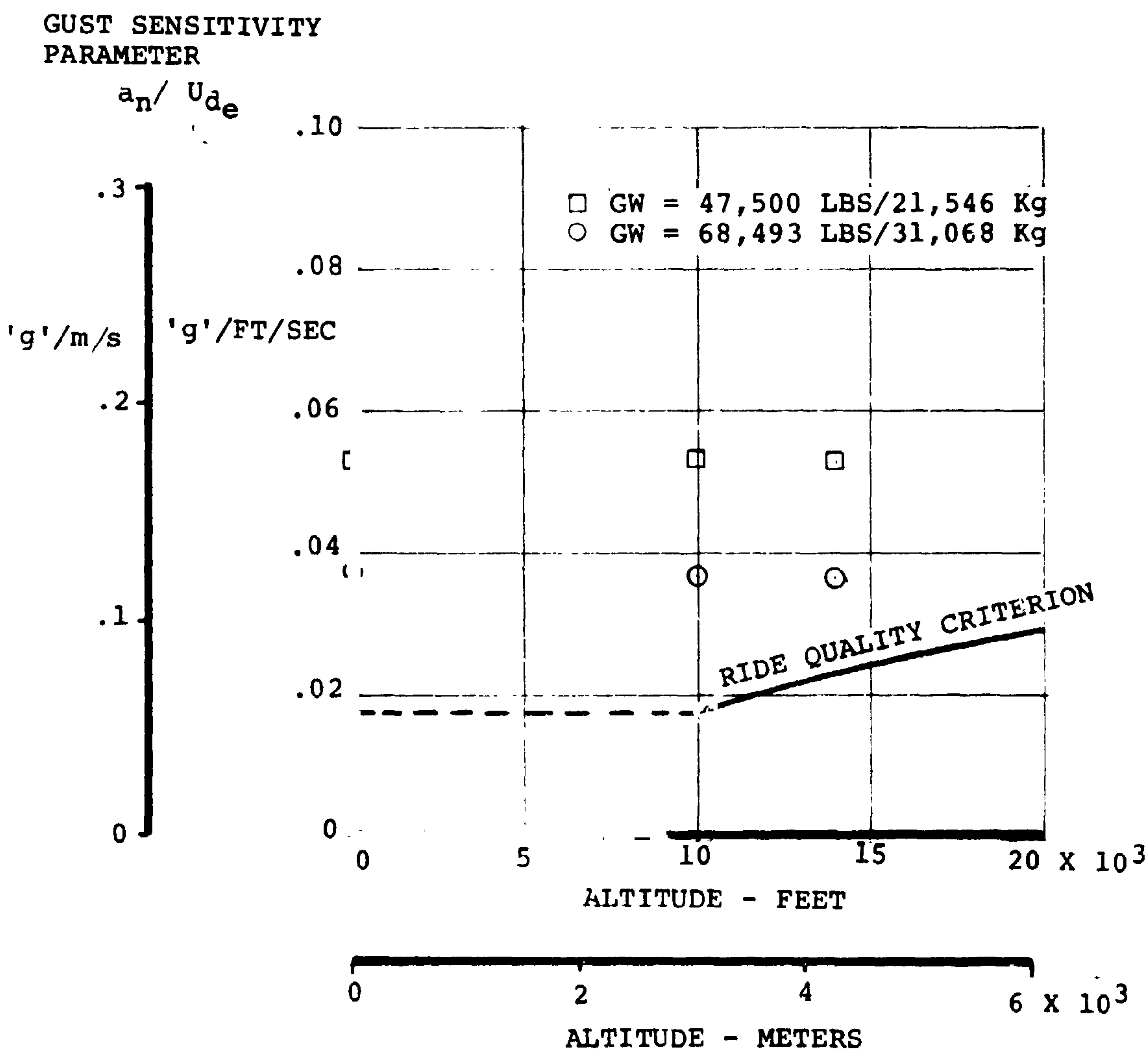

FIGURE 3.43. GUST SENSITIVITY AT MAXIMUM CRUISE SPEED.

primarily accomplished by the automatic application of spoilers and flaps in amounts proportional to the angle of attack change produced by the gust. The spoilers and flap function will be supplemented by similar operation of rotor cyclic pitch controls and the application of elevator controls to counteract pitching tendencies.

Rates of application and authority requirements were investigated in detail for the design point VTOL tilt rotor and the results of this investigation are presented in Reference 1. This study indicated that existing installed control powers were more than adequate for the gust alleviation function and this conclusion is assumed to be valid also for the STOL aircraft. 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 actuators. These are estimated to be approximately 35 pounds.

3.4 SUBSYSTEMS

Some of the subsystem requirements need definition in order that estimates of the component weights and performance can be made on a realistic basis. The main systems that impact weight, cost and performance are -

a) drive system

b) rotor system

c) control system, and

d) fuel system.

Drive System

The transmission of the STOL tilt rotor is an identical layout to the VTOL tilt rotor of Reference 1, except that the installed power is lower and the RPM reduction is less. The drive system components are listed in Table 3.9 and a schematic is shown in Figure 3.44.

Two engines are mounted in each nacelle and drive through overrunning clutches into a transfer case. The transfer case is sized at normal rated power at 14,000 feet altitude and cruise RPM. The transfer case reduction ratio is 2:1 and at normal rated power of 1,870 horsepower per engine and a cruise engine RPM of 14,430 gives a critical mesh torque of 1,362 foot-pounds.

The output of the transfer case drives into the engine bevel box which has a reduction ratio of 1.3:1 and a critical torque of 3,540 foot-pounds.

The next component is the rotor transmission bevel box which has a 1.1:1 reduction ratio and a critical torque of 3,910 foot-pounds.

The main rotor drive system input is 5,040 RPM and reduces to a rotor system RPM of 241 in cruise, a ratio of 20.9:1. At cruise normal rated power the horsepower available is 3,740 horsepower giving a transmission torque of 81,000 foot-pounds.

The cross shaft bevel is sized by the one engine inoperative requirement and the application of roll control in the adverse

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

COMPONENT	INPUT RPM	REDUCTION RATIO	CRITICAL TORQUE FOOT-POUNDS	SIZING CONDITION
ENGINE TRANSFER CASE	14430	2:1	1362	NRP, CRUISE, 14,000 FEET
ENGINE BEVEL BOX	7215	1.3:1	3540	NRP, CRUISE, 14,000 FEET
TRANSMISSION BEVEL	5550	1.1:1	3910	NRP, CRUISE, 14,000 FEET
ROTOR TRANSMISSION	5040	20.9:1	81,000	NRP, CRUISE, 14,000 FEET
CROSS SHAFT BEVEL	7200	1:1	3318	TAKEOFF, OEI, FULL ROLL CONTROL

TABLE 3.9. TRANSMISSION SIZING CONDITIONS.

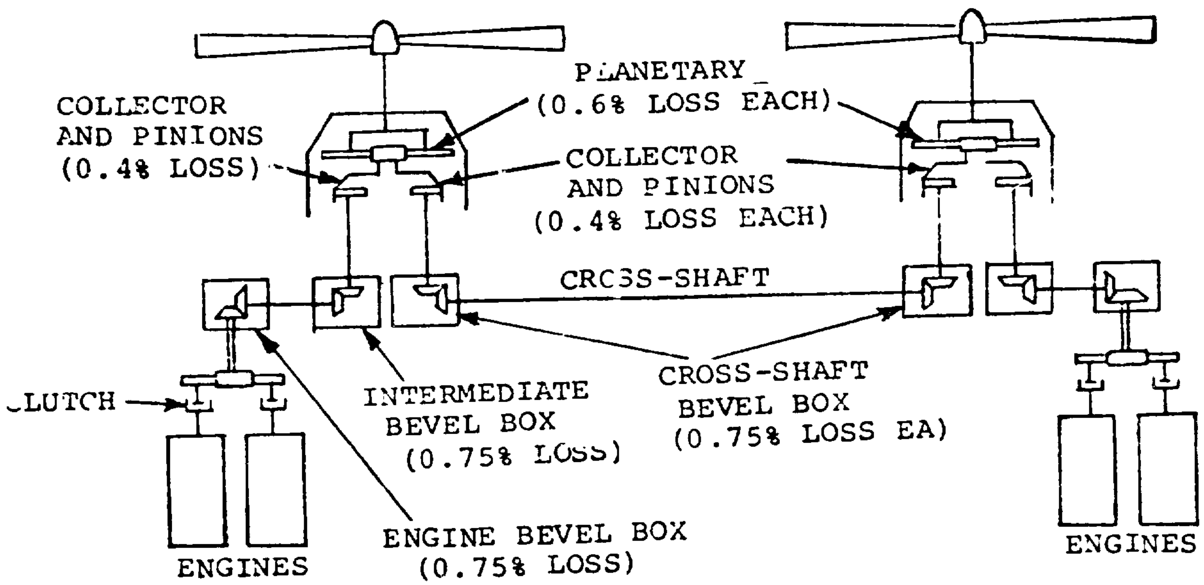

FIGURE 3.44. TRANSMISSION SCHEMATIC.

direction of takeoff. The bevel input RPM at this condition is 7,200 RPM and has a 1:1 reduction ratio. The critical mesh torque is 3,318 foot-pounds.

Rotor System

The rotor system used in the design is a hingeless soft in-plane rotor. The rotor has three blades and is 44.4 feet in diameter. The rotor solidity is 0.082 giving a blade chord of 22.9 inches. Figure 3.45 shows the characteristics of the blade. The hingeless rotor is attractive for the commercial application since it enables a simpler hub design than the other alternatives (gimballed or articulated). The rotor out-of-plane flapping excursions are low which should make passenger acceptance of the large rotor propulsion system easier. The advantage of design simplicity should favorably impact the maintenance and reliability of the aircraft.

Tilt Rotor - Fly-By-Wire Controls

The tilt rotor control system requires extensive mixing, gain and shaping changes as a function of flight condition and is, therefore, a good candidate for fly-by-wire controls. A block diagram of a possible system is shown in Figure 3.46. Each of the control inputs is converted to electrical signal using linear variable displacement transducers. Four transducers on each control provide inputs to four fly-by-wire channels. Each channel drives one of four drive actuators on each control. The main actuators are hydraulic and are dual actuators which receive command from the four drive

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

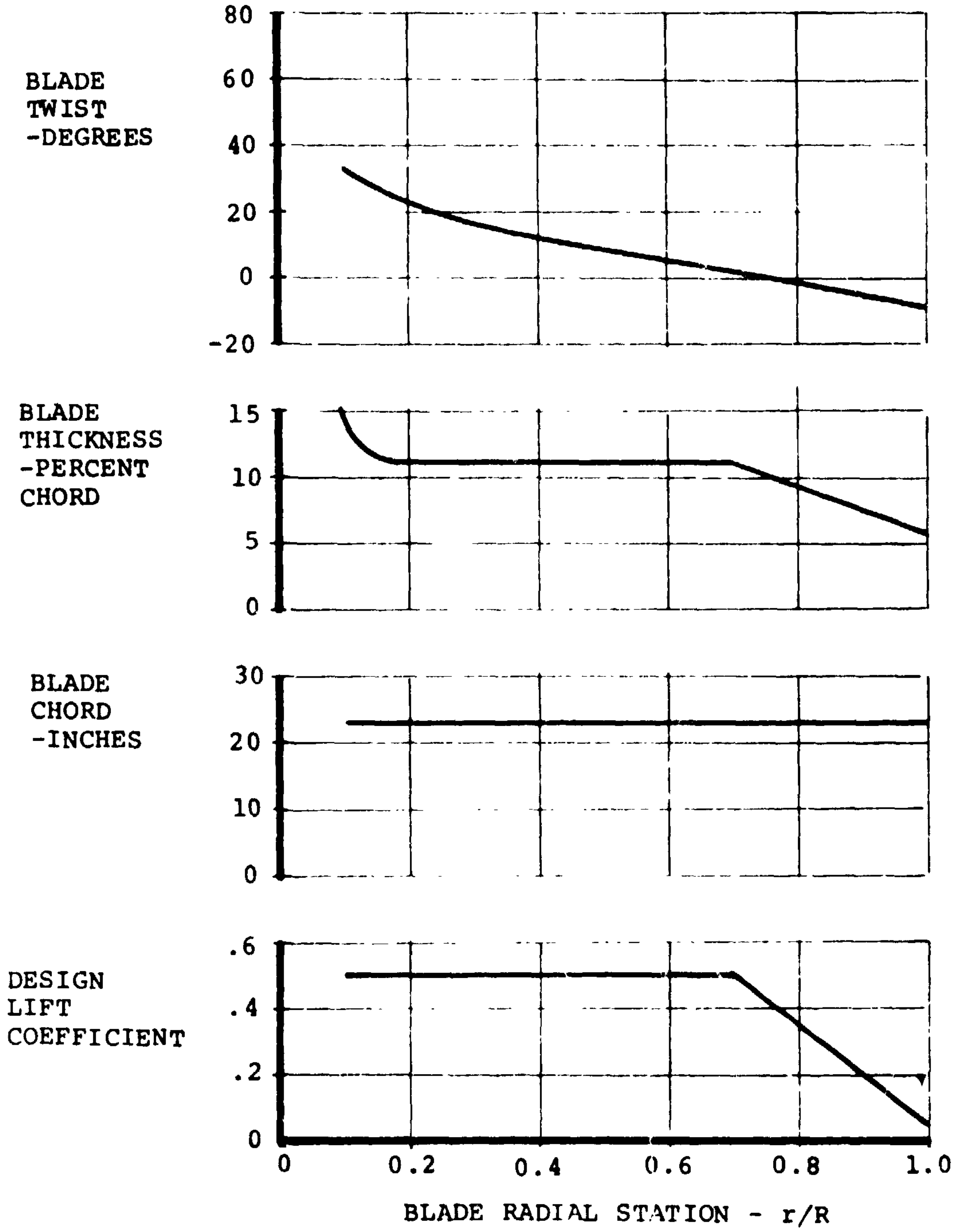

FIGURE 3.45. ROTOR BLADE GEOMETRY.

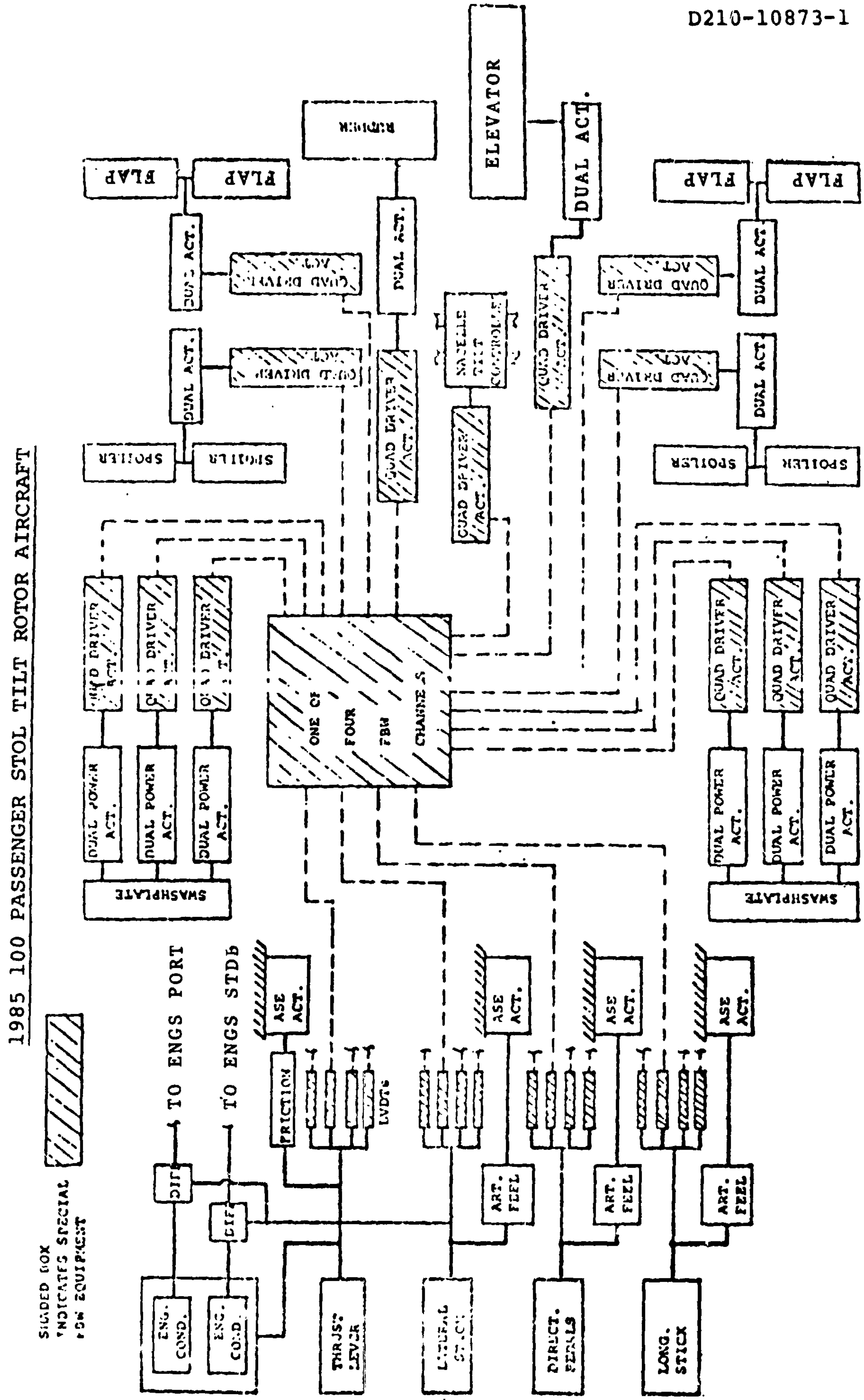

FIGURE 3.46. TILT ROTOR FLY-BY-WIRE CONTROL SYSTEM.

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actuators. The control logic for failure sensing must be designed to utilize the quadruple redundant system to be "fail operable" with any single failure and "fail safe" with double failure.

In this instance "fail operable" is intended to reflect no degradation of controls in the event of a single failure.

<u>Tilt Rotor Fuel System</u>

The fuel tanks are located in the wing. Four self-sealing integral fuel cells are used,each with a capacity of 132 gallons. Each tank contains an integral fuel pump and cross feed valving allows for fuel redistribution in flight. The system is designed for pressure refueling at 300 gallons per minute and incorporates fuel dump valves for jettison.

4.0 DESIGN DATA COMPARISONS

The purpose of this section of the report is to compare the STOL tilt rotor aircraft of this study with the baseline VTOL tilt rotor and baseline tandem helicopter defined in Reference 1. In particular, the benefits of employing a STOL operation rather than VTOL are assessed in terms of aircraft size and performance, direct operating cost, fuel economy, etc.

4.1 COMPARISON OF AIRCRAFT CHARACTERISTICS

To provide a proper basis for comparison of the VTOL and STOL tilt rotor aircraft, the STOL aircraft was designed to fly the same mission with the same payload under the same conditions that were used to define the VTOL tilt rotor except where the special requirements of STOL would conflict with them. Thus, the fuselage of the VTOL aircraft was retained as a basis for the design selection and other important parameters were varied to allow selection of the best STOL tilt rotor. The design selection process is described in detail in Appendix A of this report.

Table 4.1 contains a brief summary of the most important weight and performance parameters of the tandem helicopter, the STOL tilt rotor and the VTOL tilt rotor. The lightest of the three aircraft is the tandem rotor helicopter and the heaviest is the VTOL tilt rotor. The STOL tilt rotor aircraft is considerably lighter than the VTOL for a number of reasons. First, and most important, is the fact that

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

	STOL TILT ROTOR	VTOL TILT ROTOR
TAKE OFF GROSS WEIGHT - (LBS)	68,493	74,749
EMPTY WEIGHT (LBS)	45,023	50,068
MISSION FUEL (LBS)	3,425	4,656
INSTALLED POWER (HP SL STD)	11,142	16,579
V_{NRP} AT ALTITUDE (KTAS)	310	349
DESIGN ALTITUDE (FT)	14,000	14,000
BLOCK TIME (HOURS)	0.82	0.742

TABLE 4.1. DESIGN POINT AIRCRAFT PERFORMANCE AND WEIGHT COMPARISON.

the installed power required to allow a less than 2,000 foot takeoff run (11,142 horsepower) is considerably less than that required for vertical takeoff (16,579 horsepower). Thus, the weight of engines required for the STOL aircraft will be less than for the VTOL.

Secondly, the STOL tilt rotor has smaller rotors and empennage than the VTOL thus providing more savings in weight. In addition to the weight saving, these components have less drag than their counterparts on the VTOL tilt rotor.

Thirdly, the reduced drag, lower power and lower weight of the STOL aircraft allow for a considerably smaller fuel usage (25 percent reduction) than that of the VTOL aircraft. The net result is a weight reduction of 6,256 pounds.

The lower installed power of the STOL tilt rotor results in a significant penalty in cruise speed of 38 knots, (reduction from 349 for the VTOL down to 311 for the STOL aircraft) and also a degradation of the climb capability. However, these penalties result in a block time increase for the mission of less than 5 minutes.

Figure 4.1 is a graph of tandem helicopter gross weight versus the design maneuver load factor. The design maneuver load factor employed in sizing the tandem helicopters described in Reference 1 was 3.5 compared with 2.5 for the tilt rotor airplanes. The graph of Figure 4.1 was prepared

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

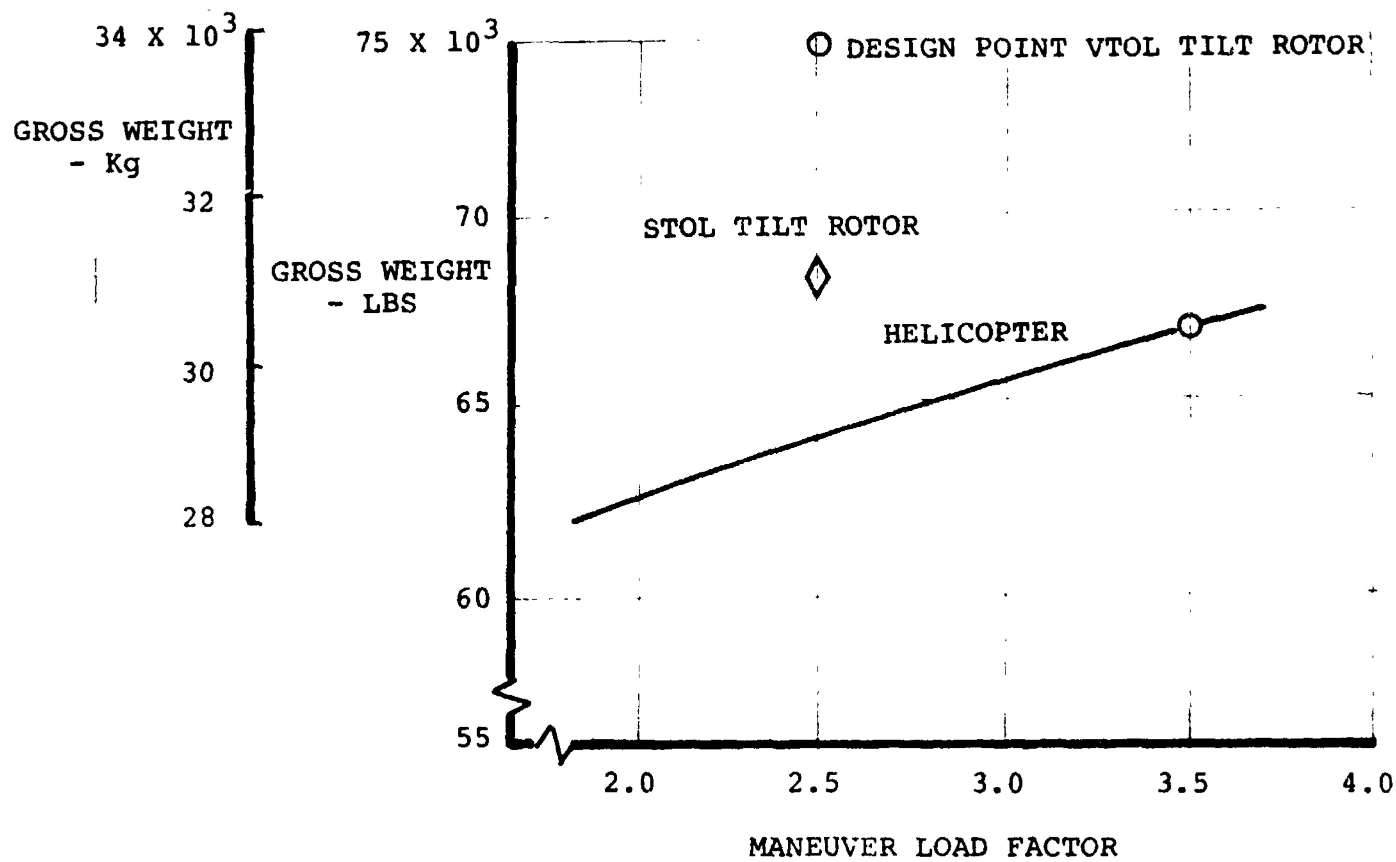


FIGURE 4.1. EFFECT OF DESIGN MANEUVER LOAD FACTOR ON AIRCRAFT SIZE.

in order to allow a comparison of the helicopter with the tilt rotor at the same maneuver load factor. Gross weight for the VTOL and STOL tilt rotors have been spotted on the graph clearly demonstrating the weights of the three different aircraft at a load factor of 2.5.

The 500 foot sideline perceived noise level of the STOL tilt rotor is higher than those of the baseline tandem helicoper and the VTOL tilt rotor, as shown in Figure 4.2. This is due to its higher rotor tipspeed and blade loading, C_T/σ (800 feet per second and 0.166 compared with 725 and .088 for the helicopter).

In Reference 1 the effect of imposing external noise constraints on the VTOL tilt rotor and tandem helicopter designs was investigated. This investigation entailed resizing both configurations for a 5 PNdB greater and a 5 PNdB lower noise level than the baseline aircraft. The effect of the external noise level on the aircraft design was used as a basis for comparison of the two concepts. Figures 4.2 through 4.7 of this report contain these comparisons with the design point STOL tilt rotor plotted in order to allow comparison with the designs of Reference 1.

Figure 4.2 shows a comparison of the aircraft design gross weight as a function of the external noise level. This clearly shows the gross weight and noise level of the STOL tilt rotor in relation to the other two baseline aircraft and their "noise derivatives".

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

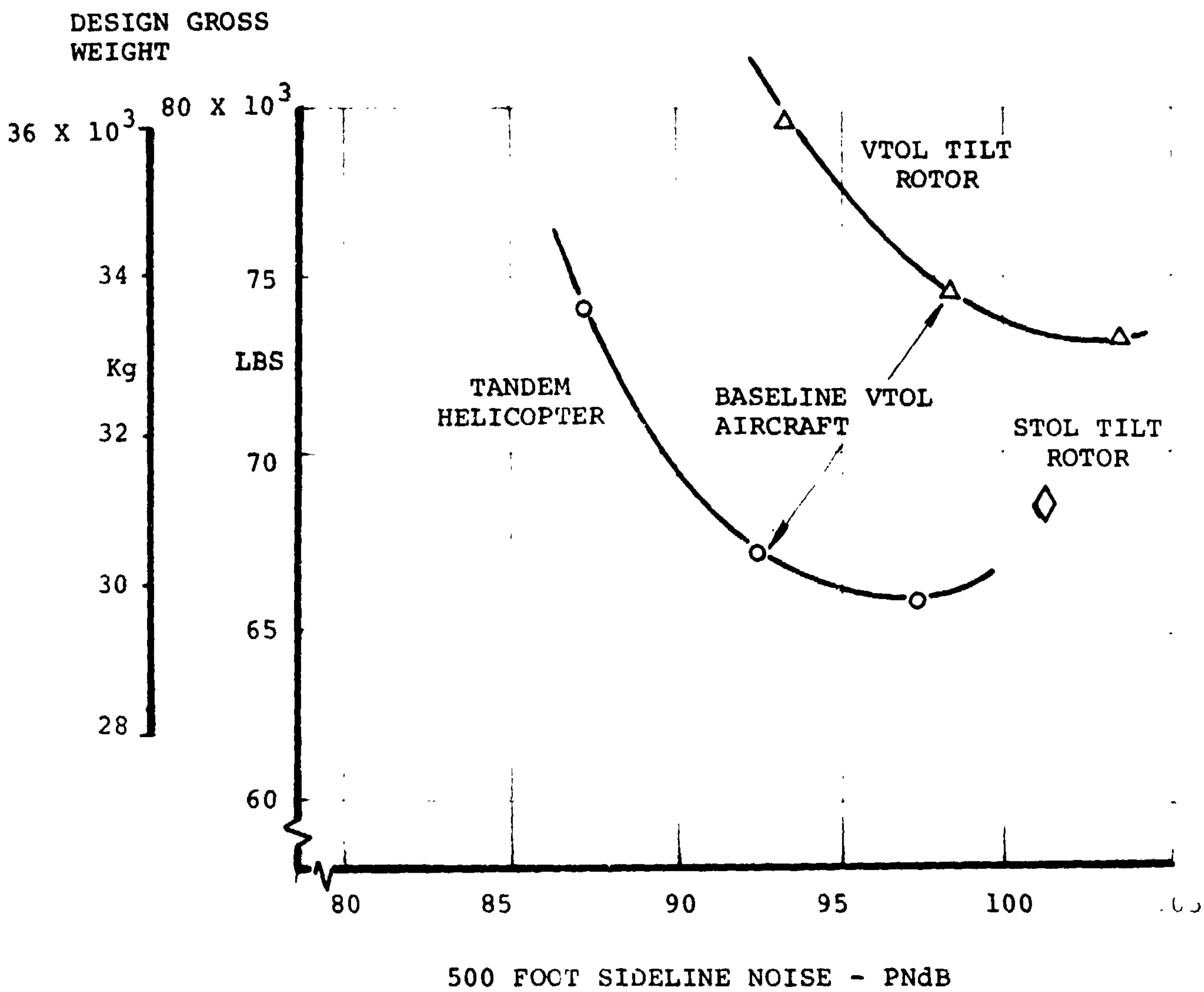


FIGURE 4.2. THE EFFECT OF EXTERNAL NOISE DESIGN CRITERIA ON AIRCRAFT DESIGN GROSS WEIGHT.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

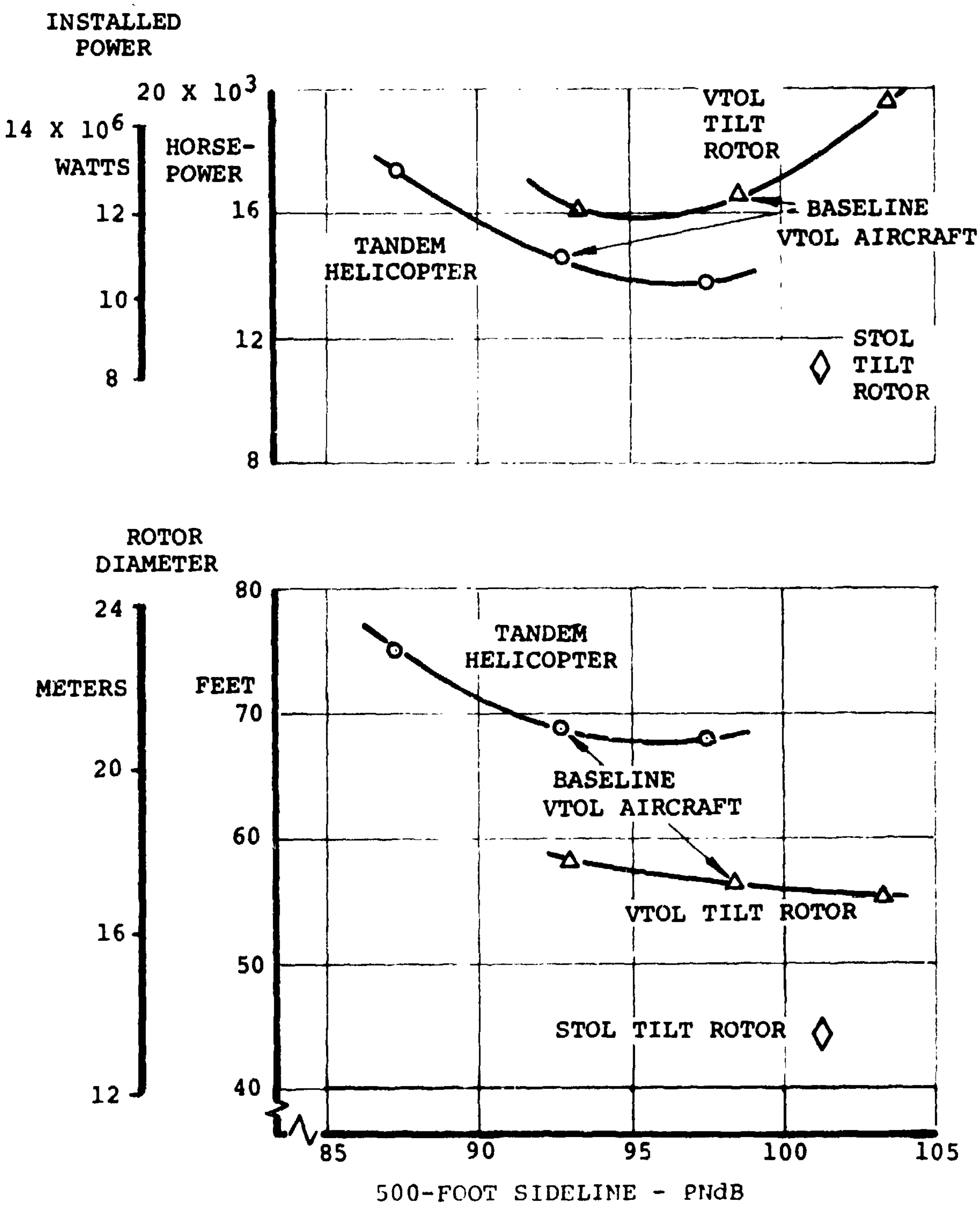


FIGURE 4.3. EFFECT OF EXTERNAL NOISE DESIGN CRITERIA ON ROTOR DIAMETER AND INSTALLED POWER

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

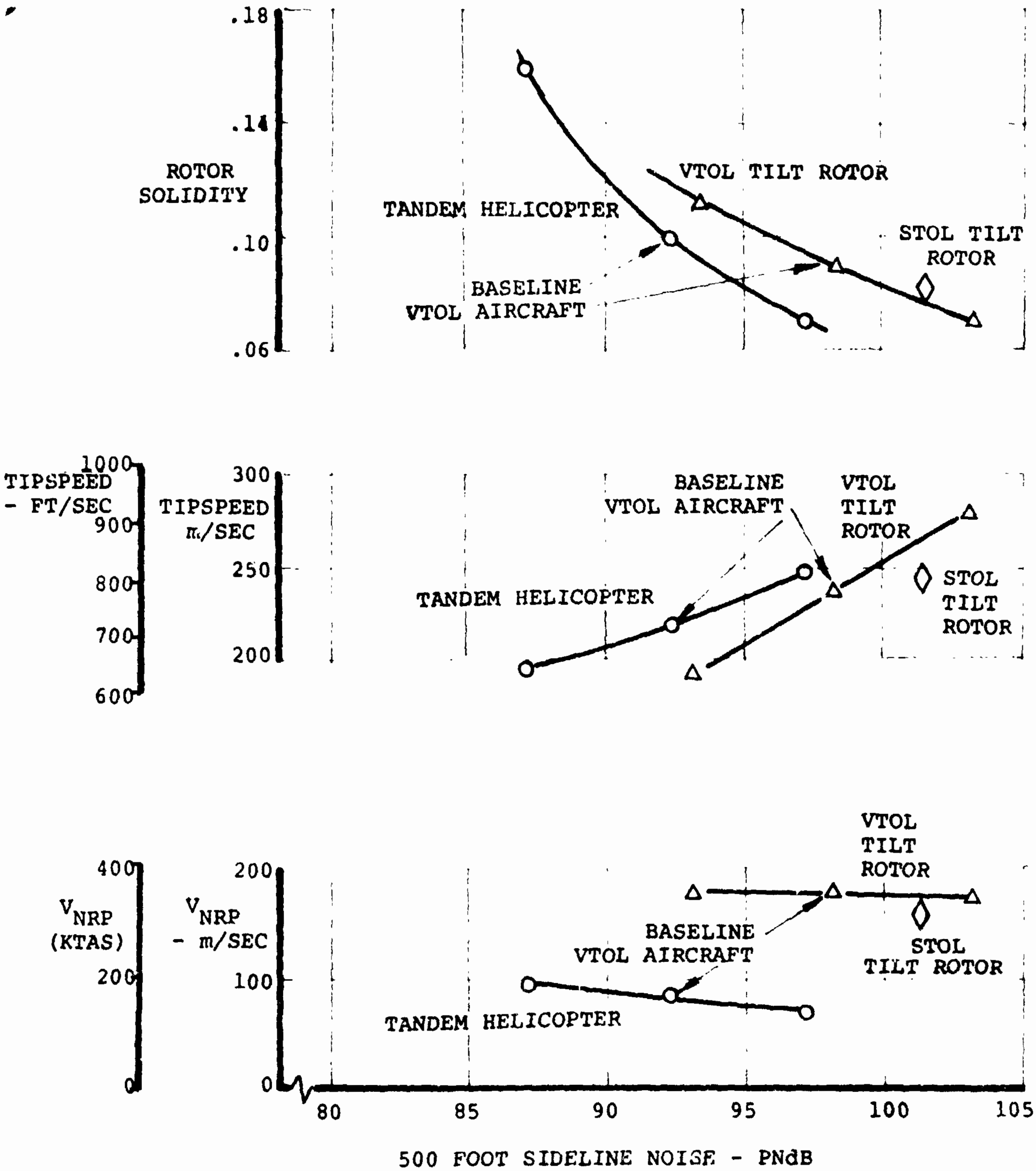


FIGURE 4.4. EFFECT OF EXTERNAL NOISE CRITERIA ON DESIGN PARAMETERS.

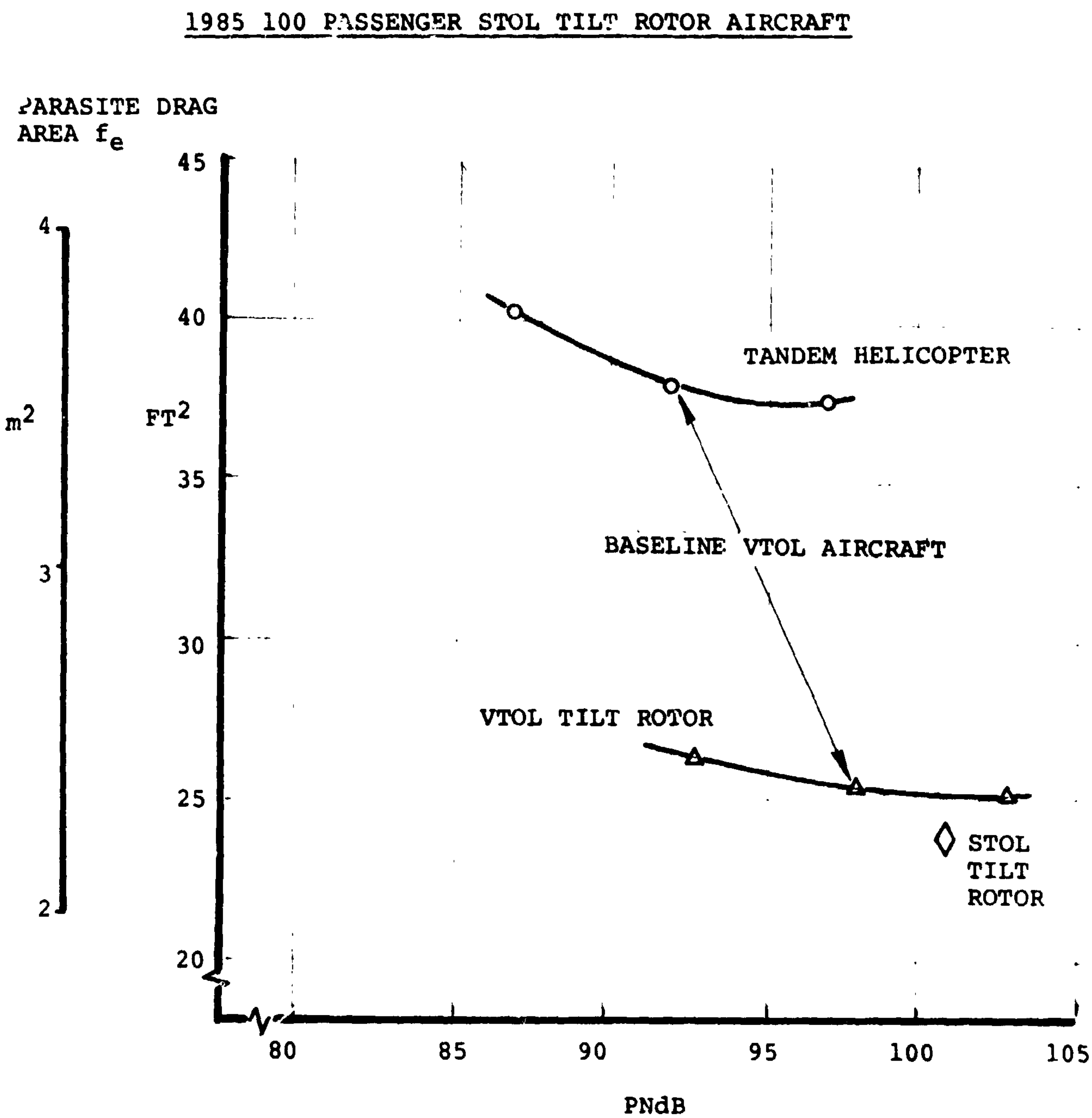


FIGURE 4.5. EFFECT OF EXTERNAL NOISE CRITERIA ON PARASITE DRAG AREA f_e.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

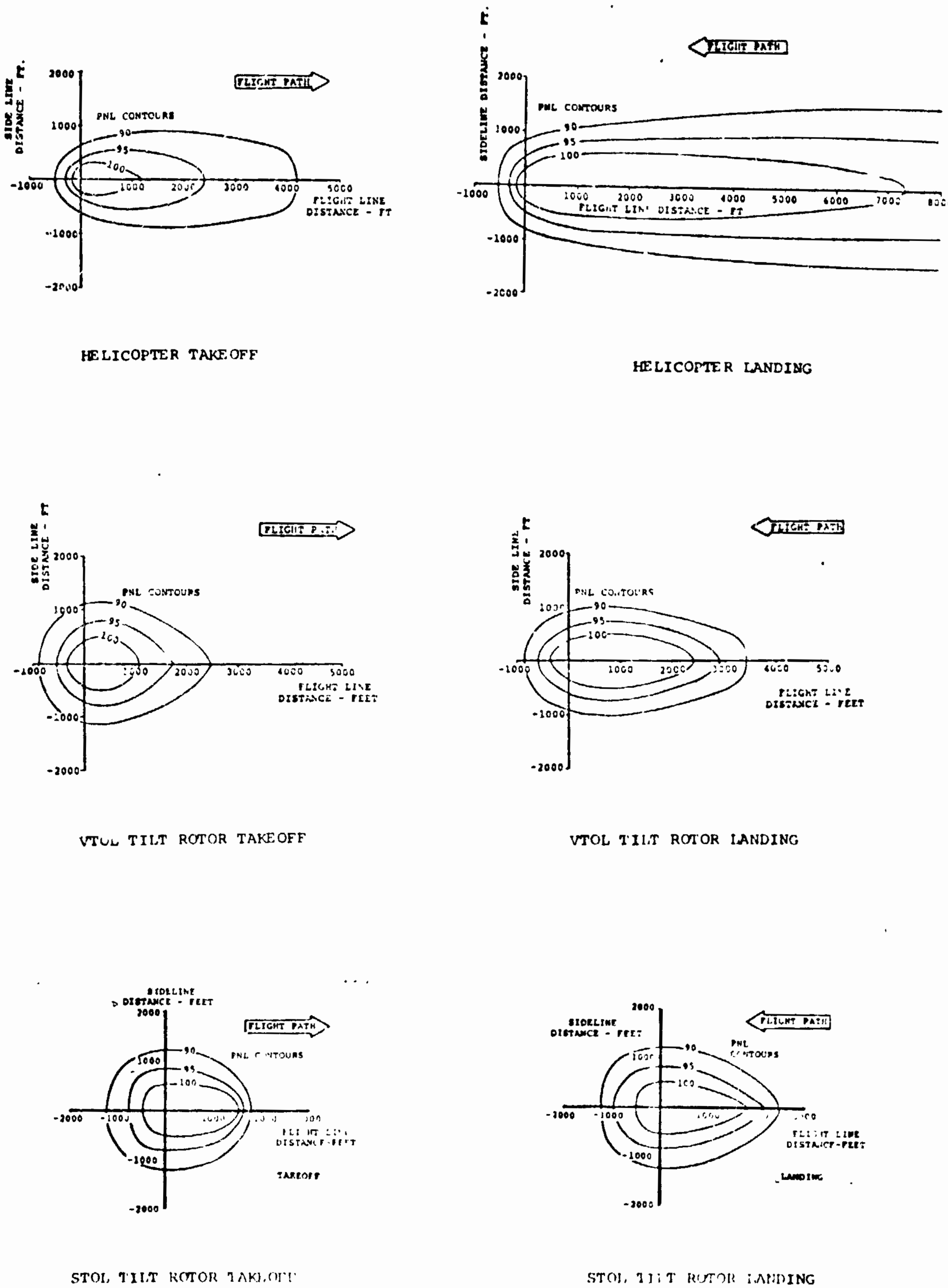


FIGURE 4.6. COMPARISON OF TAKEOFF AND LANDING - PERCEIVED NOISE LEVEL CONTOURS.

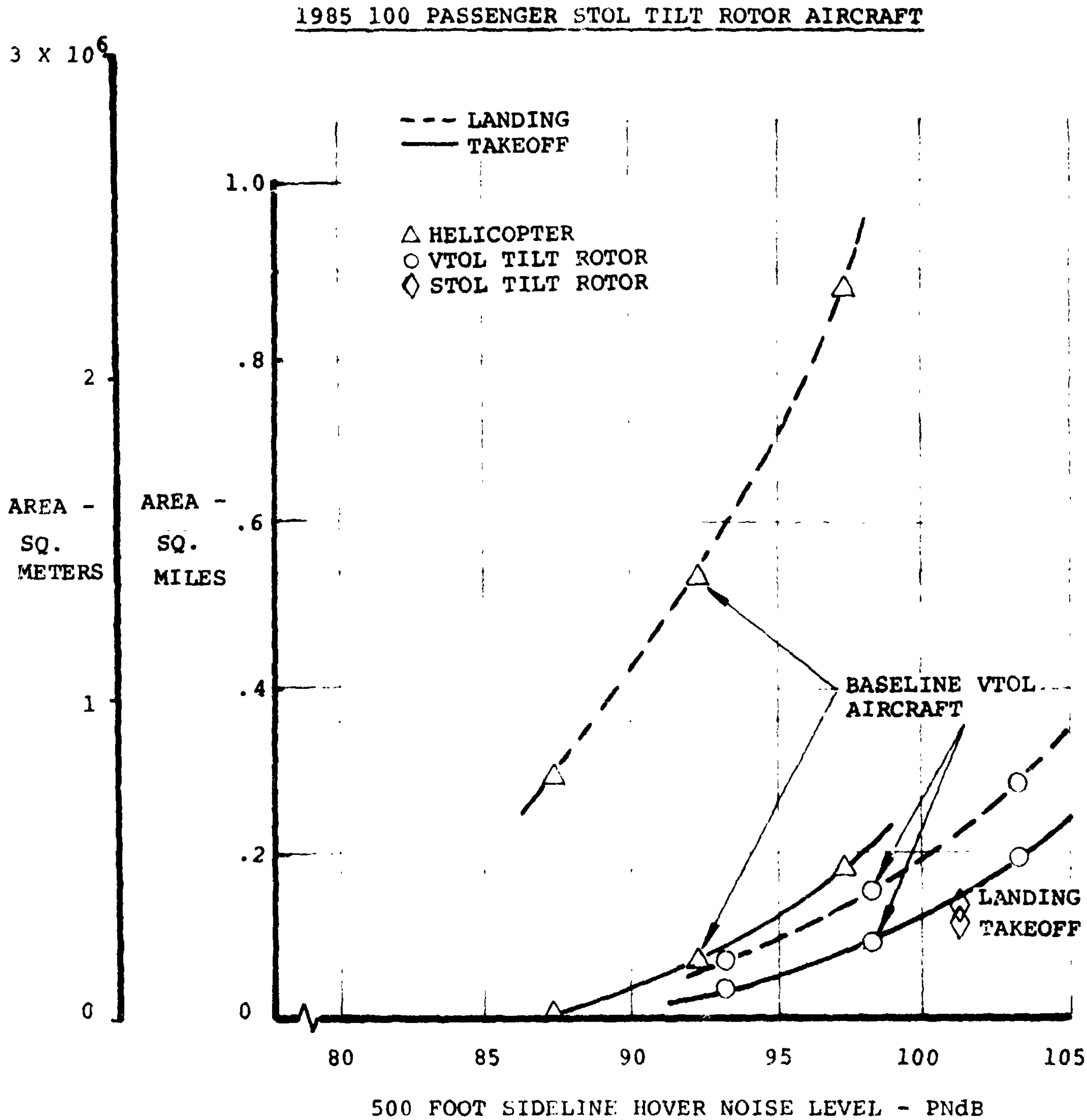


FIGURE 4.7. COMPARISON OF AREA IMPACTED BY 95 PNdB NOISE LEVEL OR GREATER DURING TAKEOFF AND LANDING.

In Figure 4.3 the installed power and rotor size are compared on the basis of external noise level. The STOL tilt rotor is seen to have a lower installed power and smaller rotor diameter than both of the other baseline aircraft. Both of these facts reflect the fact that a lower takeoff thrust is required for the STOL than for the VTOL aircraft, (static thrust/weight ratio of the STOL tilt rotor is 0.88 compared with 1.101 for the VTOL tilt rotor and 1.14 for the tandem helicopter).

The graphs of Figure 4.4 show the variation of rotor solidity, takeoff tipspeed and cruise speed at normal rated power with external perceived noise level. The solidity of the STOL tilt rotor is lower than that of both baseline aircraft and lies almost directly on the VTOL tilt rotor solidity noise trend line.

The tipspeed of the STOL tilt rotor is higher than that of either of the other baseline aircraft and is a strong influence in producing the higher external noise level of this design.

The third graph of Figure 4.4 shows the cruise speed noise trends for the VTOL tilt rotor and tandem helicopter. It is seen that the STOL tilt rotor cruise speed is only a little lower than that of the VTOL aircraft and much greater than that of the tandem helicopter. The STOL aircraft is slower than the VTOL tilt rotor due to its lower installed power. This effect is somewhat offset by the lower parasite drag level of the STOL tilt rotor.

Figure 4.5 illustrates the drag levels of the three concepts. The STOL tilt rotor is seen to have the lowest drag level. The small difference of drag between the VTOL and STOL tilt rotors is due to the smaller wings, empennage and engine and rotor nacelles.

The contours of 90, 95 and 100 PNdB of perceived noise level for takeoff and landing are shown in Figure 4.6 for the STOL tilt rotor in comparison with the VTOL tilt rotor and the tandem helicopter. The areas subjected to 90 PNdB are comparable for the tilt rotor aircraft. The area subjected to 90 PNdB by the helicopter is significantly larger, particularly for the landing maneuver.

A more accurate comparison of the area within the 95 PNdB contours during takeoff is available in Figure 4.7. The variation of the area enclosed by the 95 PNdB contour has been plotted as a function of the 500-foot sideline takeoff noise level. It is seen that the 95 PNdB contour of the STOL tilt rotor encloses a larger area (0.3 square kilometers (.115 square miles)) than those of the VTOL tilt rotor (0.23 square kilometers (0.09 square miles)) and the tandem helicopter (0.18 square kilometers (0.07 square miles)) during takeoff. The reverse is true for the landing case; 0.36 square kilometers (0.14 square miles) for the STOL tilt rotor, 0.39 square kilometers (0.15 square miles) for the VTOL tilt rotor and 1.39 square kilometers (0.535 square miles) for the tandem helicopter.

4.2 COST AND PRODUCTIVITY COMPARISONS

The variation of initial cost of the VTOL tilt rotor and tandem helicopter with external noise design criteria are shown in Figure 4.8 for two levels of airframe cost. For the same two airframe cost levels the initial cost of the STOL tilt rotor aircraft are plotted on the chart at the

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

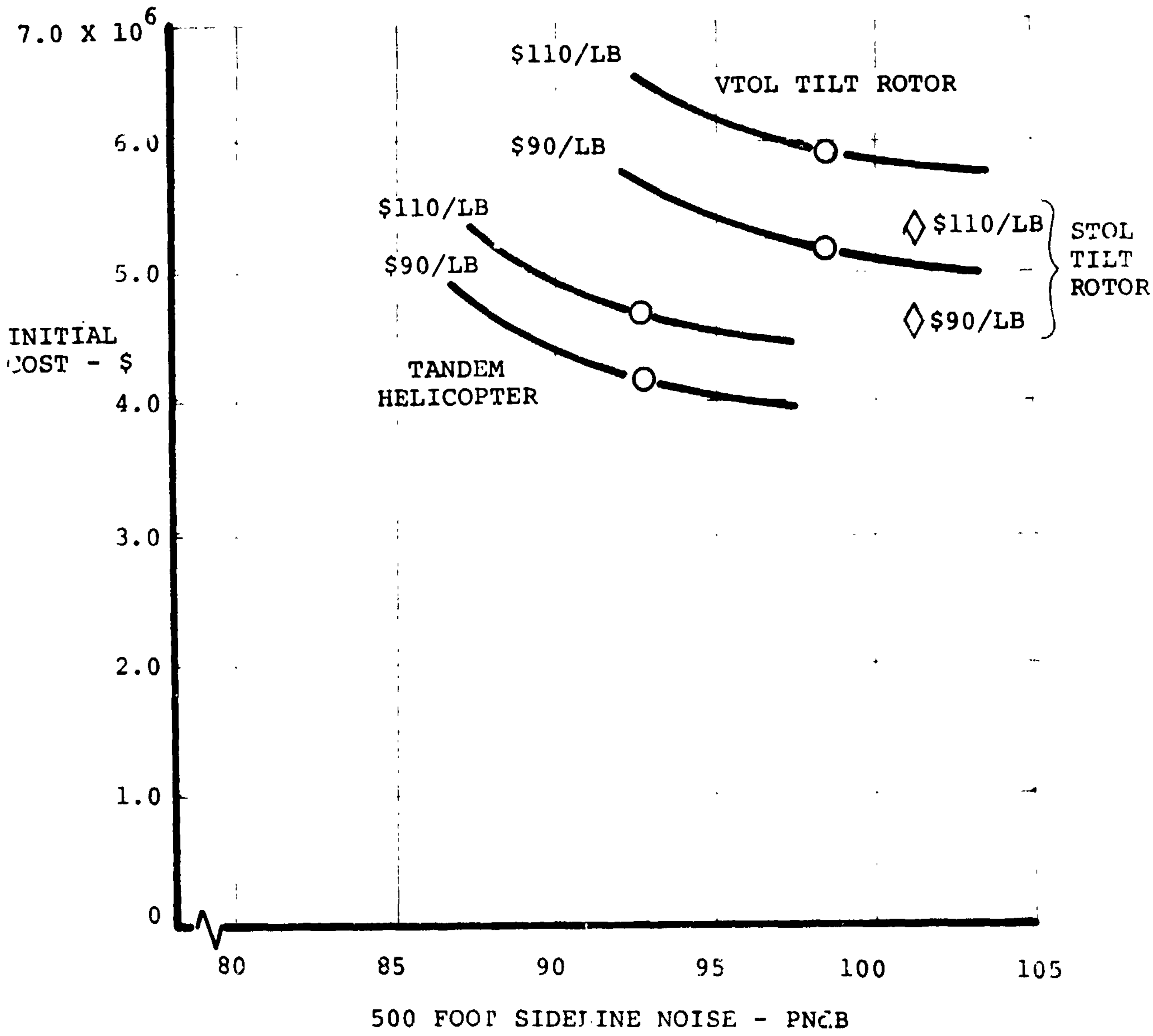


FIGURE 4.8. EFFECT OF EXTERNAL NOISE DESIGN CRITERIA ON INITIAL COST.

appropriate noise level (101.3 PNdB). The initial cost of the STOL aircraft lies roughly half way between the costs of the tandem helicopter and the VTOL tilt rotor.

In Figure 4.9 the direct operating cost of the STOL tilt rotor has been plotted on the graph showing the direct operating cost - noise trends for the VTOL tilt rotor and tandem helicopter, for two different levels of utilization. The STOL tilt rotor has a lower direct operating cost than either of the other two baseline aircraft at a given utilization.

The direct operating cost of the STOL tilt rotor is about 1.2 cents per seat mile lower than that of the helicopter, but only 0.1 cents per seat mile lower than the VTOL's cost. In each case the design points indicated on the graph were selected on the merit of minimum direct operating cost.

The speed capability of STOL tilt rotor aircraft is compared in terms of block speed variation with block distance in Figure 4.10. Because of the non-productive time consumed in terminal maneuvers and the lower speed climb portions the block speed is noticeably lower than the cruise speed for each concept. Of course, the STOL tilt rotor has a slightly lower block speed than the VTOL and a markedly higher one than the tandem helicopter.

The variation of direct operating cost with block distance is shown in Figure 4.11 in comparison with the VTOL tilt rotor and the tandem helicopter.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

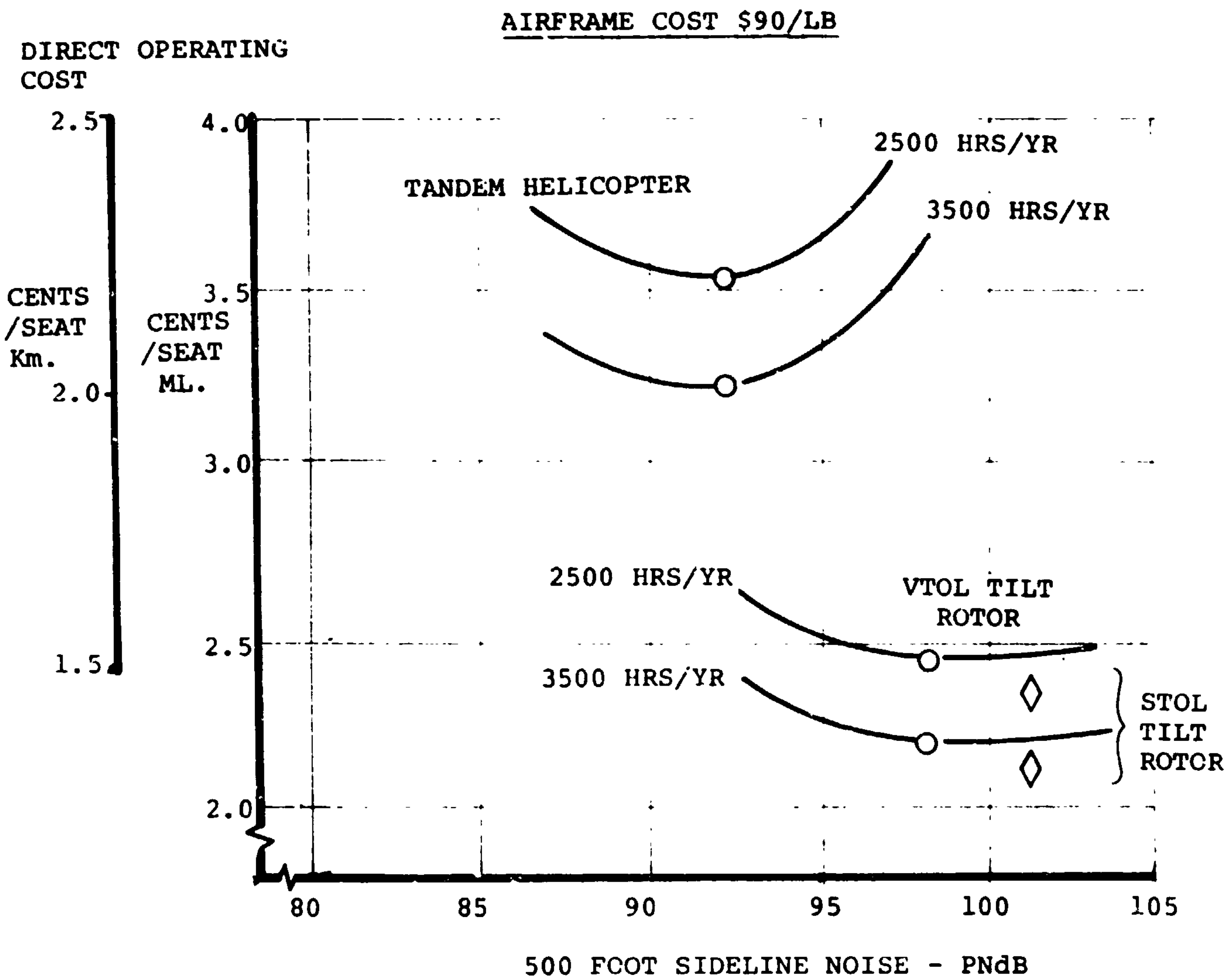


FIGURE 4.9. EFFECT OF EXTERNAL NOISE CRITERIA ON DOC AT 230 STATUTE MILES RANGE.

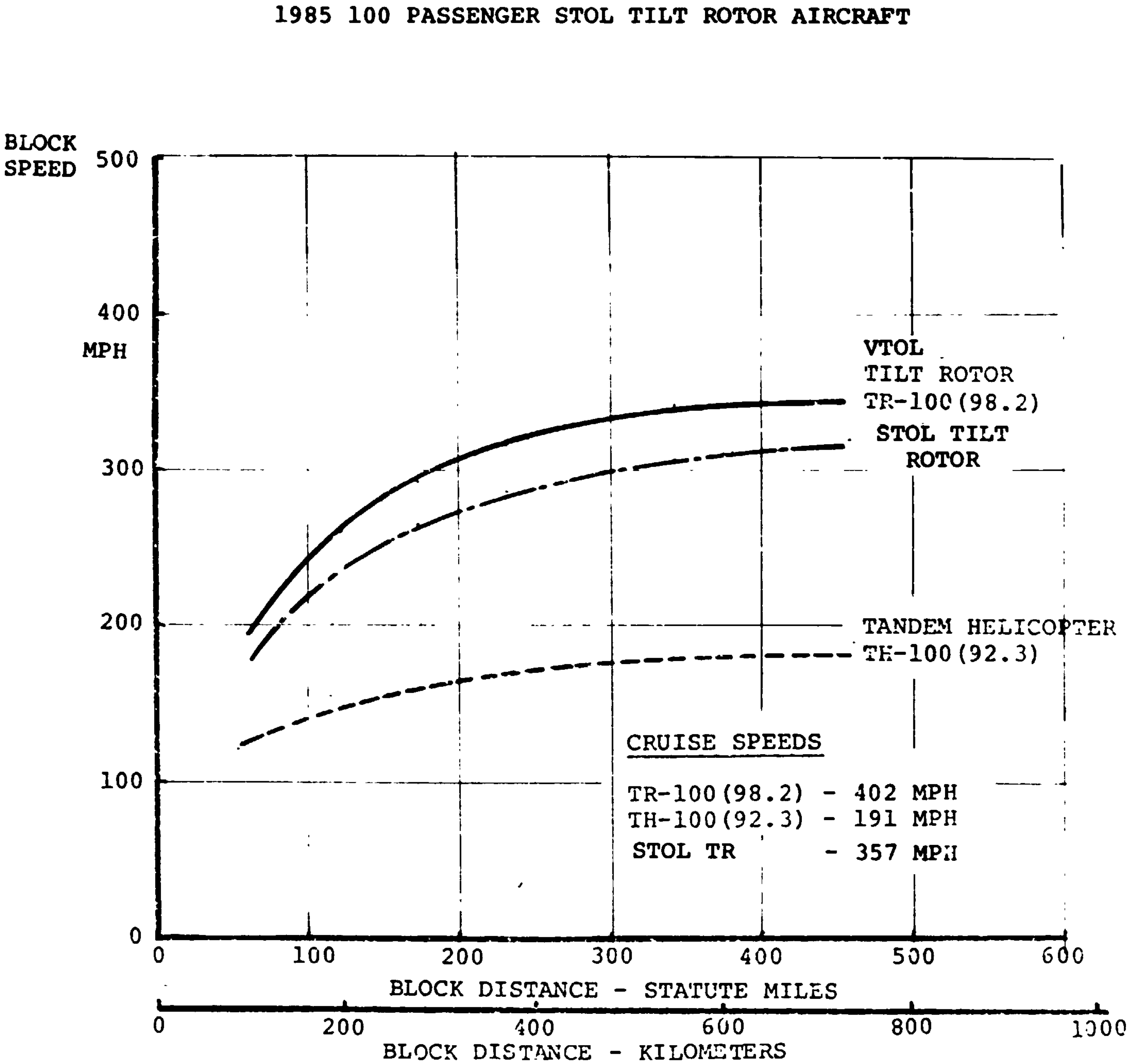


FIGURE 4.10. COMPARISON OF BASELINE DESIGN AIRCRAFT - BLOCK SPEED AND DISTANCE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

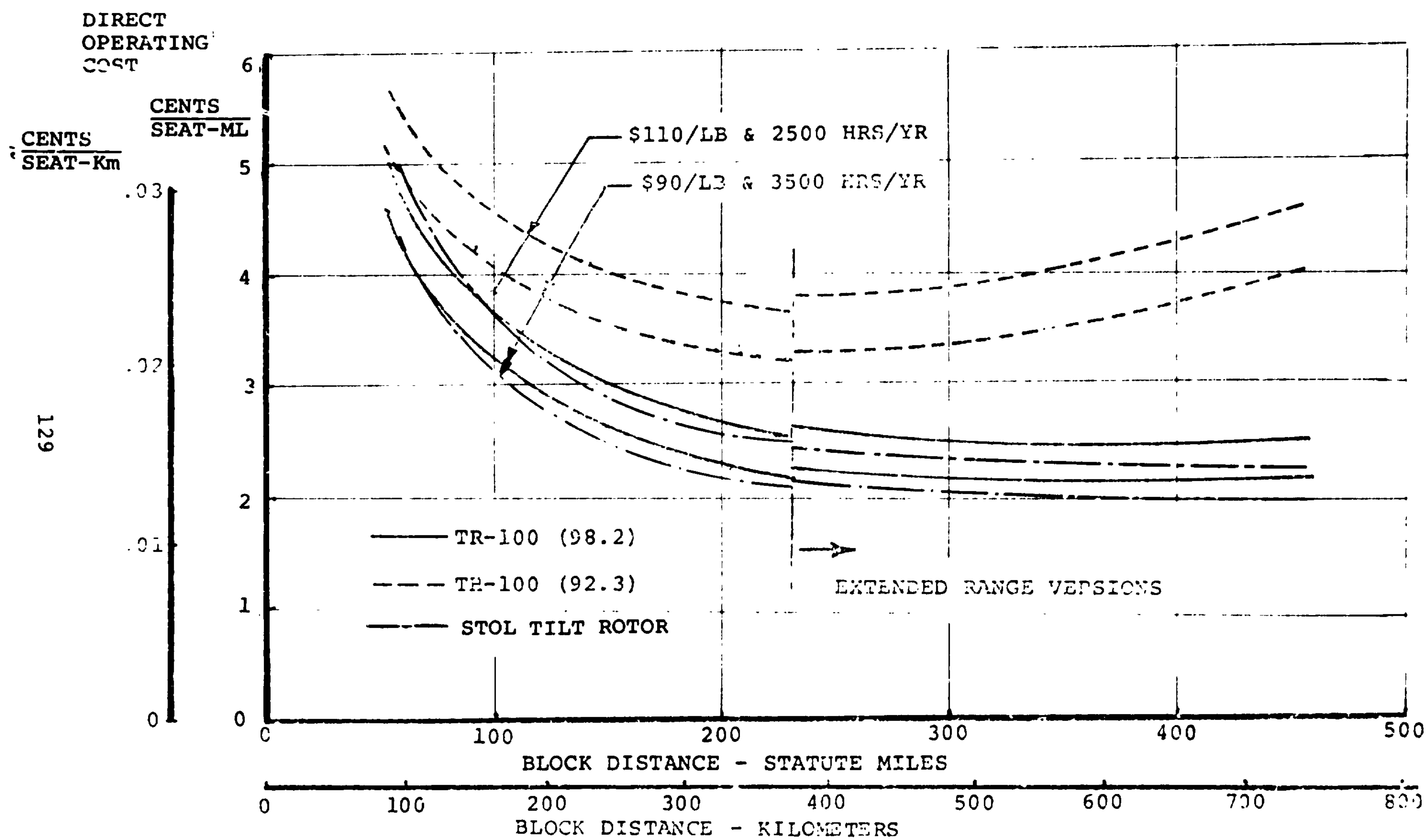


FIGURE 4.11. DIRECT OPERATING COST COMPARISON WITH TANDEM HELICOPTER AND VTOL TILT ROTOR.

Two curves are shown for each aircraft. The upper curve represents an airframe cost of $110 per pound and an annual utilization of 2,500 hours.

The lower curve represents $90 per pound airframe cost and a utilization of 3,500 hours per year. For all but block distances of less than 100 statute miles the STOL tilt rotor has a lower direct operating cost than the VTOL tilt rotor and the tandem helicopter. It is to be noted that a higher utilization leads to a lower operating cost (all other things being equal). The comparison (with respect to annual utilization) is somewhat unfair to the tilt rotor aircraft in that a higher speed aircraft can have a higher utilization than one with a lower speed (assuming equal non-productive times for maintenance, etc.) and as a result would incur a lower operating cost.

Fuel consumption as a function of cruise speed is illustrated in Figure 4.12 for each of the three concepts. The fuel consumption is expressed in passenger miles per gallon of fuel used. It can be seen that both of the tilt rotor configurations show a greater economy of fuel than does the helicopter and, in addition, fly much faster. It should also be noted that, for the design condition, the cruise altitude was optimum for each configuration. The design points indicated on the graph correspond to the maximum cruise speed with the cruise power setting and cruise RPM, and as such do not coincide with optimum fuel consumption.

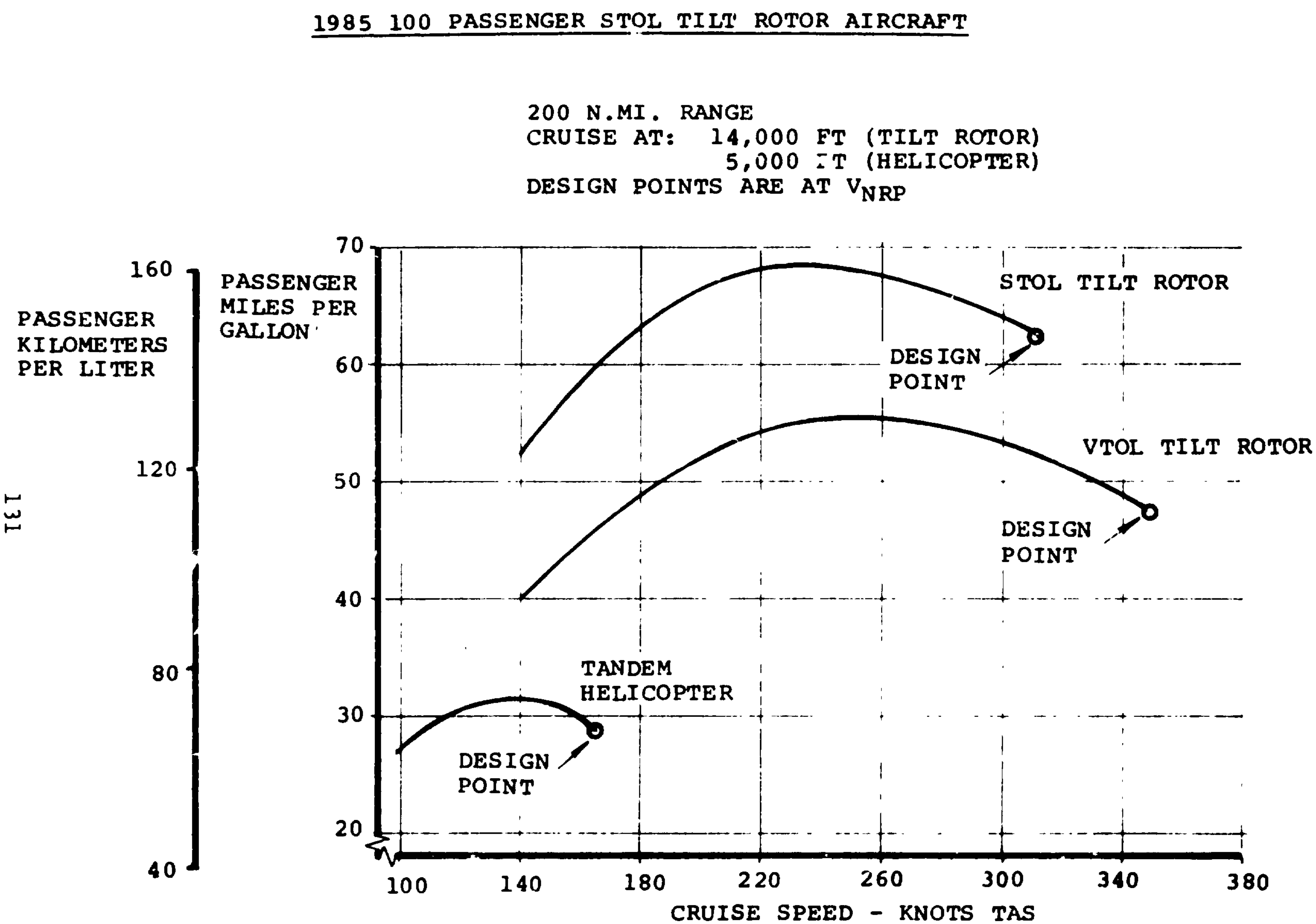


FIGURE 4.12. VARIATION OF FUEL CONSUMPTION WITH CRUISE SPEED.

In each case an improvement of about 10% can be achieved in fuel consumption by flying the cruise part of the mission at the optimum speed. This would, however, impose a higher direct operating cost and lower productivity. For any given cruise speed the STOL tilt rotor has by far the best fuel economy by a margin of at least 10 passenger miles per gallon. By accepting the sacrifice of direct operating cost and cruising at the speed for best fuel consumption an improvement from 62.5 to 68.8 passenger miles per gallon could be achieved.

A convenient measure of productivity, defined by forming the product of payload and block speed and dividing by weight empty is shown in Figure 4.13 as a function of range. On this basis the tilt rotor aircraft are almost identical in performance and surpass the helicopter by a wide margin that increases with range. The higher speed of the VTOL tilt rotor, relative to that of the STOL, is offset by its higher empty weight.

Figure 4.14 shows the fuel consumption of the STOL tilt rotor, design point VTOL tilt rotor and tandem helicopter in comparison with a wide variety of existing aircraft.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

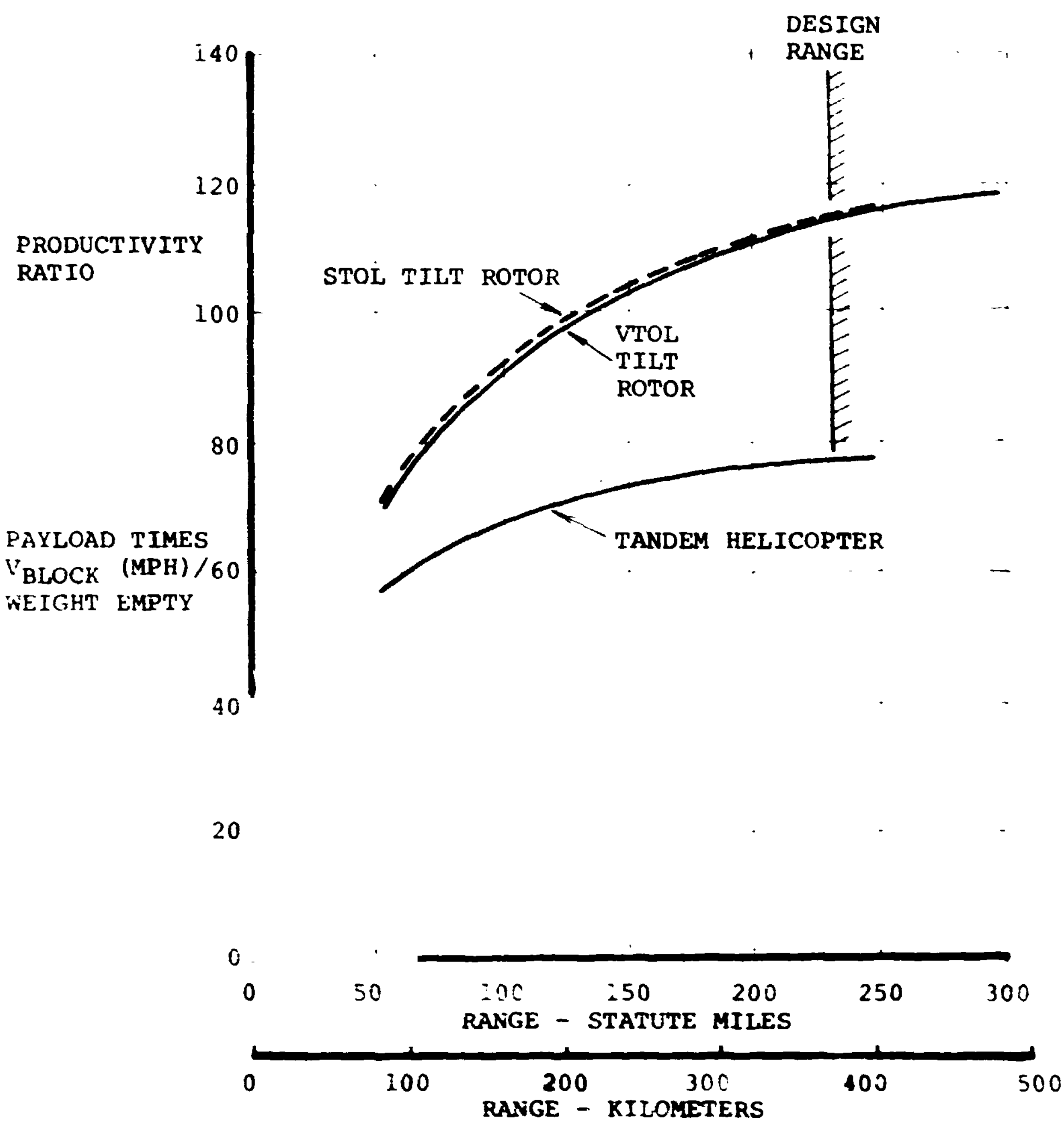


FIGURE 4.13. PRODUCTIVITY RATIO COMPARISON: TANDEM ROTOR HELICOPTER AND TILT ROTORS.

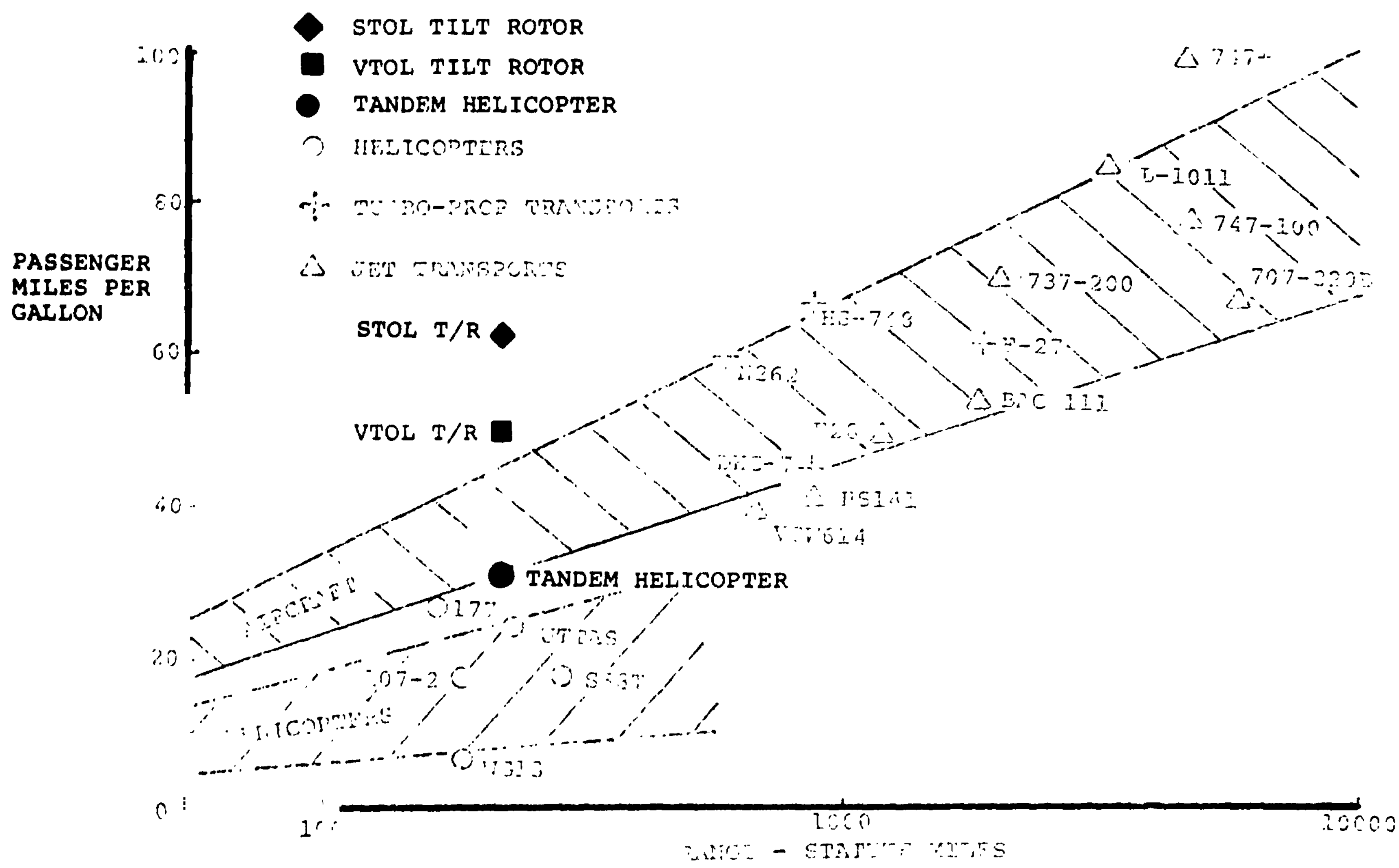


FIGURE 4.14. FUEL CONSUMPTION COMPARISON.

5.0 TECHNICAL RISK AND SIZE LIMITATIONS

Background

The STOL tilt rotor transport was defined to have the same passenger carrying capacity as the VTOL aircraft designed for the same mission in Reference 1. These aircraft, both helicopter and tilt rotor, were not found to be limited up to the 100 passenger mark set by the study guidelines. However, since this issue of size provoked much thought and discussion in the VTOL studies the arguments and decision data are briefly recapitulated here for ease of reference, along with new issues specific to the STOL configuration.

The VTOL study groundrules 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 VTOL study was fully responsive to this groundrule, which might, under some circumstances, have forced the selection of uneconomic designs. However, careful examination of technology issues did not result in the identification of any serious impediments to this maximum size aircraft. In fact, only the 100 passenger constraint was found to be more restrictive than either technological or economic considerations in both the helicopter and VTOL tilt rotor configurations. In both configurations the optimum operating costs occurred around the 100 passenger mark and there was 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 was considered 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 would fall within some set of budgetary and schedule constraints. Another factor to be considered was the credibility of the size selected and support among the technical community. It would be more difficult to generate and sustain support for a larger rather than a smaller sized development. Other issues which were identified as intruding into the area of economics were such questions as passenger density and frequency of schedule, 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 were more than the commercial carrier had at its disposal.

On the other hand an aircraft that was too small would be uneconomical to operate and would require a premium fare structure which might preclude use by the desired market. Some of these issues were not readily quantified and were in many cases outside the defined scope of the study. All of these issues have a substantially similar impact in the STOL tilt rotor configuration.

Nevertheless, economics were considered to be of such importance that the discussion of risk was expanded to include the effects of direct operating cost as well as an evaluation of the technical risks.

No identified technological problems restricted either the tandem rotor helicopter or the VTOL tilt rotor configurations to sizes less than 100 passengers, and the figures for direct operating costs strongly suggest 100 passengers or above. The same statements may be made in relation to the STOL tilt rotor.

The fundamental assumption in the evaluation of risk for both the VTOL and STOL 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 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 ensures 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 for size ranges of the same magnitude and for much larger aircraft than the 100 passenger aircraft provides a solid basis for design and development.

Development 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 rotation speeds be successfully designed?

2. Rotor System

   Does the rotor blade strength keep pace with rotor loads as size is increased?

   A positive conclusion was reached on these issues in the VTOL tilt rotor. The STOL tilt rotor which features a smaller diameter rotor and higher operating RPM than the VTOL aircraft would be included in these conclusions.

3. Rotor, Nacelle, Wing Aeroelastic Considerations

   As size is increased, do the design constraints of wing strength and frequency become more or less restrictive?

   In the case of the VTOL aircraft it was computed that provision of adequate stiffness did not incur any excessive penalties. However, STOL operation raises several new issues which are discussed below.

Each of these areas is 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

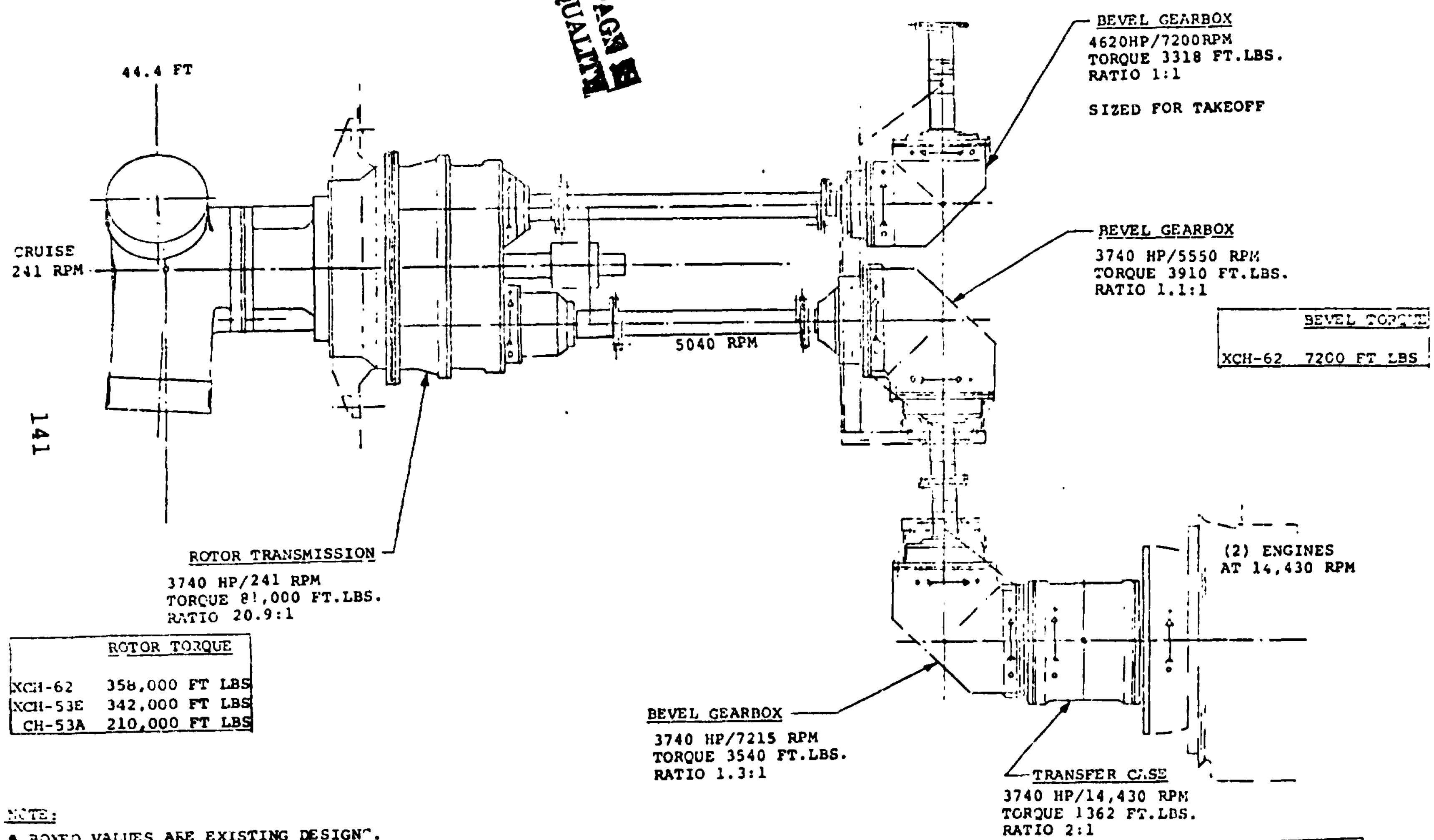
The drive train required by the 100 passenger STOL tilt rotor aircraft is shown schematically in Figure 5.1. The technical risks may be evaluated by comparing each transmission box or gear train with existing hardware.

The engine transfer case critical mesh torque is 1,362 foot-pounds. A similar spur torque mesh exists in the AH-56 transmission designed to 9,895 foot-pounds.

The largest of the bevel boxes requires the transmission of 3,910 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 81,000 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 rotor transmission requires a reduction ratio of 20.9:1. The XCH-53E main rotor transmission had a reduction ratio of 35.8:1 and the CH-53A 32.5:1. The XCH-62 reduction ratio is 51.2:1.



NOTE:

- BOXED VALUES ARE EXISTING DESIGN.

FIGURE 5.1. COMPARISON OF TRANSMISSION DESIGN TECHNOLOGY FOR STOL TILT ROTOR.

The maximum reduction ratio required for the bevel boxes is 1.3:1 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 2:1 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 cannot 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 inch/inch. These data reflect today's technology and are, therefore, reasonably conservative for the 1985 time frame.

The estimated allowable total alternating blade bending moment is 200,000 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 5.2 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% to 75% of the estimated fatigue allowable. The rotor loads e 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 a function of longitudinal stick. Figure 5.3 shows the estimated normal load factor at which endurance limit loads on the blade root occur. For speeds in excess of 223 knots the aircraft can pull its design maneuver limit with no fatigue damage and at the worst case can pull 2.1 g's before fatigue damage occurs.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

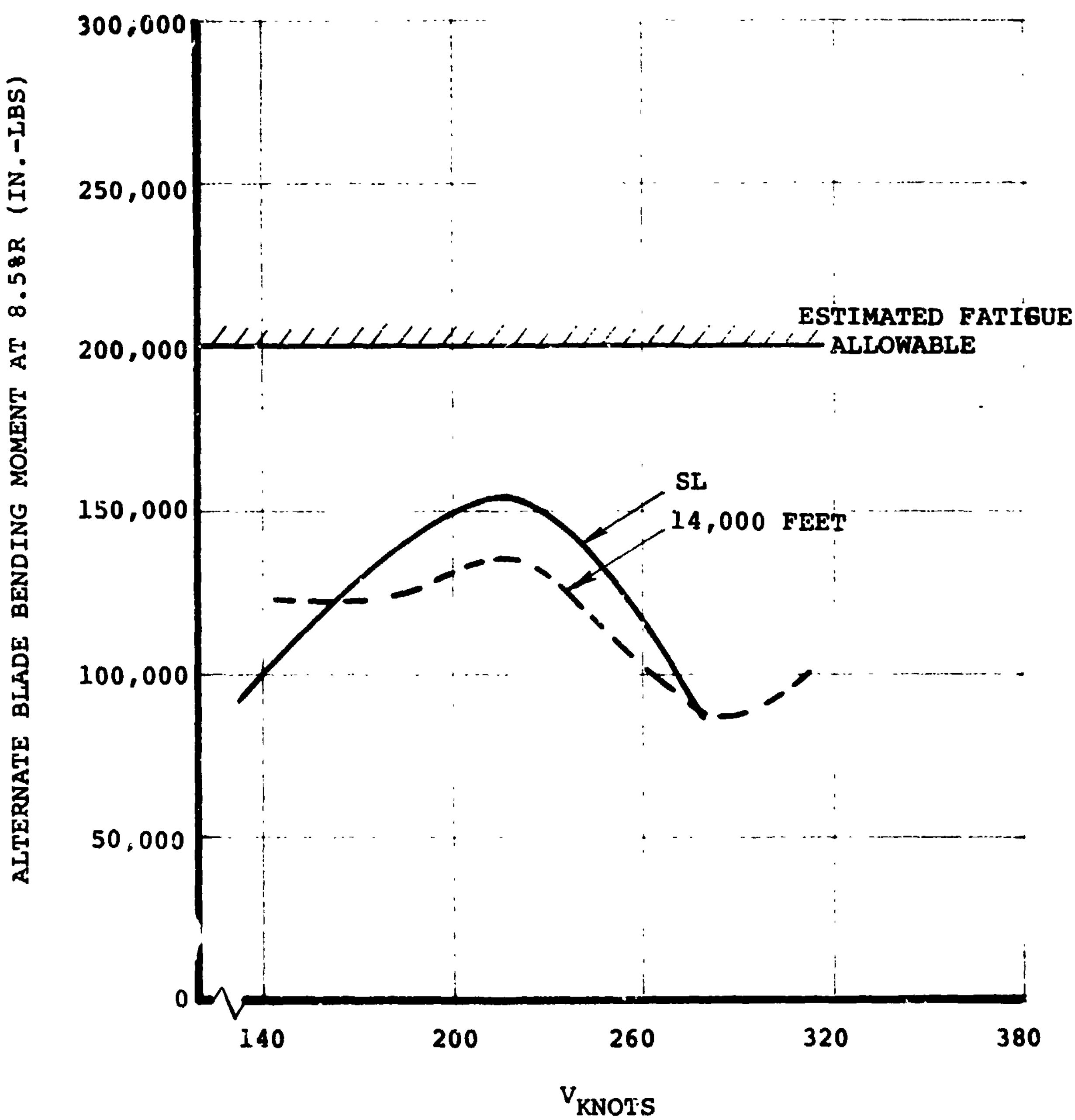


FIGURE 5.2. ALTERNATE BLADE BENDING LOADS IN CRUISE - 1g LEVEL FLIGHT.

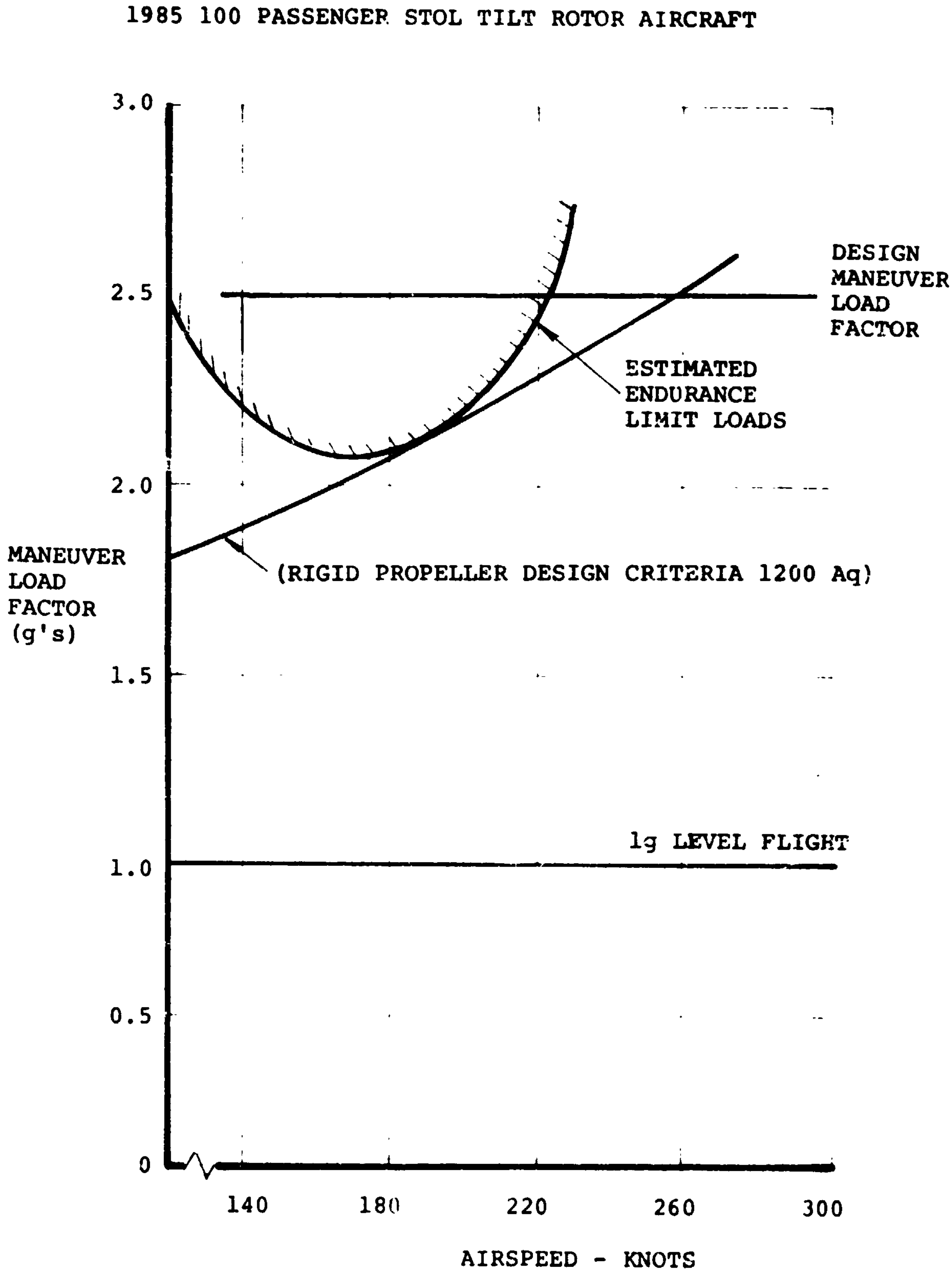


FIGURE 5.3. BLADE FATIGUE LIMITS ON MANEUVER ENVELOPE.

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 5.3 to provide a comparison.

The detailed design of the blade and the aircraft control system in takeoff and transition would be required to compute the blade fatigue life. However, the magnitude of the loads estimated in cruise and their 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 XV-3 and XV-15 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:

(i) by direct application using scaling laws, and

(ii) 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 40 by 80-foot wind tunnel. This experience is summarized in Figure 5.4 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 44.4 foot diameter rotor system selected for the STOL tilt rotor aircraft. The more general question of validation of methodology has been addressed at length in other Boeing documents (e.g., Reference 3) 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

Aeroelastic stability was recognized as a potential area of risk as aircraft size grew from levels which had been studied in depth (e.g., Boeing Vertol Model 222 and Bell Model 301). STOL operation does not introduce any new

FULL SCALE & 1/9 SCALE TESTING FOR HINGELESS
ROTOR AEROELASTIC STABILITY

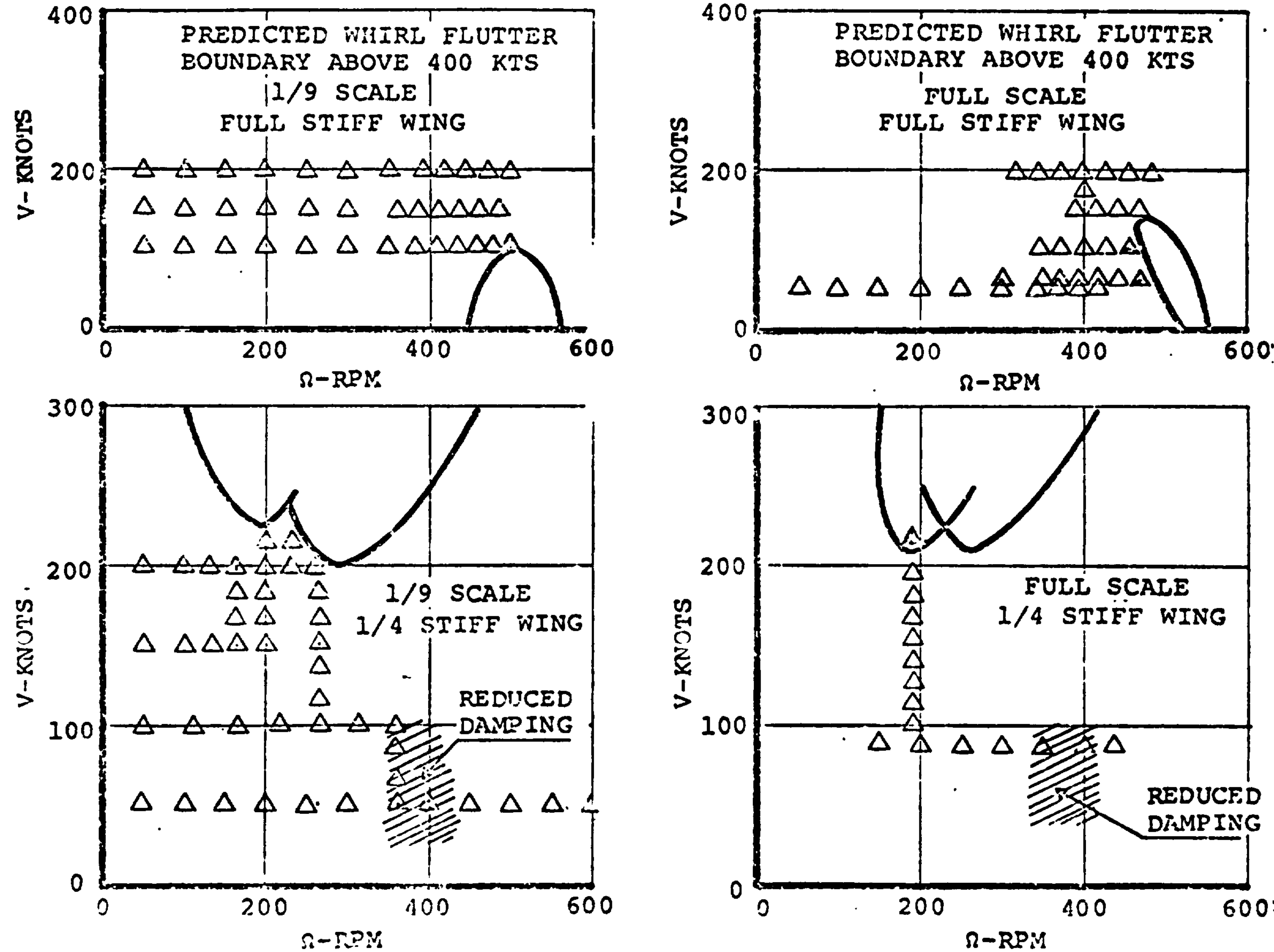


FIGURE 5.4. SCALING SUMMARY.

technological problems but the impact of the different mode of takeoff needs to be evaluated. 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 compromise the payload carrying capability of the aircraft.

For the STOL tilt rotor aircraft the hingeless rotor is grown from the 26 foot diameter size designed and tested for the Model 222 to 44.4 foot diameter, tipspeed is increased but blade per rev frequencies as a function of percentage of cruise RPM are maintained at the values selected for the Model 222 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 Model 222. 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 3).

This detailed calculation of aeroelastic behavior was made when the final design point aircraft was selected.

The approach to a satisfactory design from an aeroelastic point of view is slightly different in the STOL configuration from that adopted in the case of the VTOL tilt rotor. In the VTOL case the jump takeoff criterion for wing strength generally provides adequate beamwise bending stiffness to ensure satisfactory margins in the air resonance mode where the rotor regressive in-plane mode couples with wing beam bending. In the STOL aircraft, wing strength requirements do not necessarily provide such margins and frequency criteria based on air resonance and whirl flutter margins were established.

In establishing a wing beam bending frequency criterion the conservative assumption was made that takeoff RPM would be used with the nacelles fully down, and that a 20% margin on RPM would be maintained in this condition. This is conservative because the nacelles will be tilted as rotor speed is increased from cruise RPM to hover RPM, and the coupling between wing beam bending and rotor in-plane motion is significantly reduced as the nacelles assume a landing or takeoff attitude. This led to the selection of a wing beam frequency of 3.5 Hz and by implication a chord bending frequency of 8.75 Hertz.

Torsion stiffness was selected on the basis of providing stability up to 1.2 V_D. This led to a torsional frequency requirement of 3.7 Hertz.

Two altitudes were examined - sea level and 13,300 feet, the altitude at which the maximum V_{MO} (TAS) is encountered. The aeroelastic boundaries for these two altitudes are shown in Figure 5.5 and 5.6. No particular difficulty or unacceptable

1985 100 PASSENGER STOL TILT ROTOR AEROELASTIC BOUNDARIES - SEA LEVEL

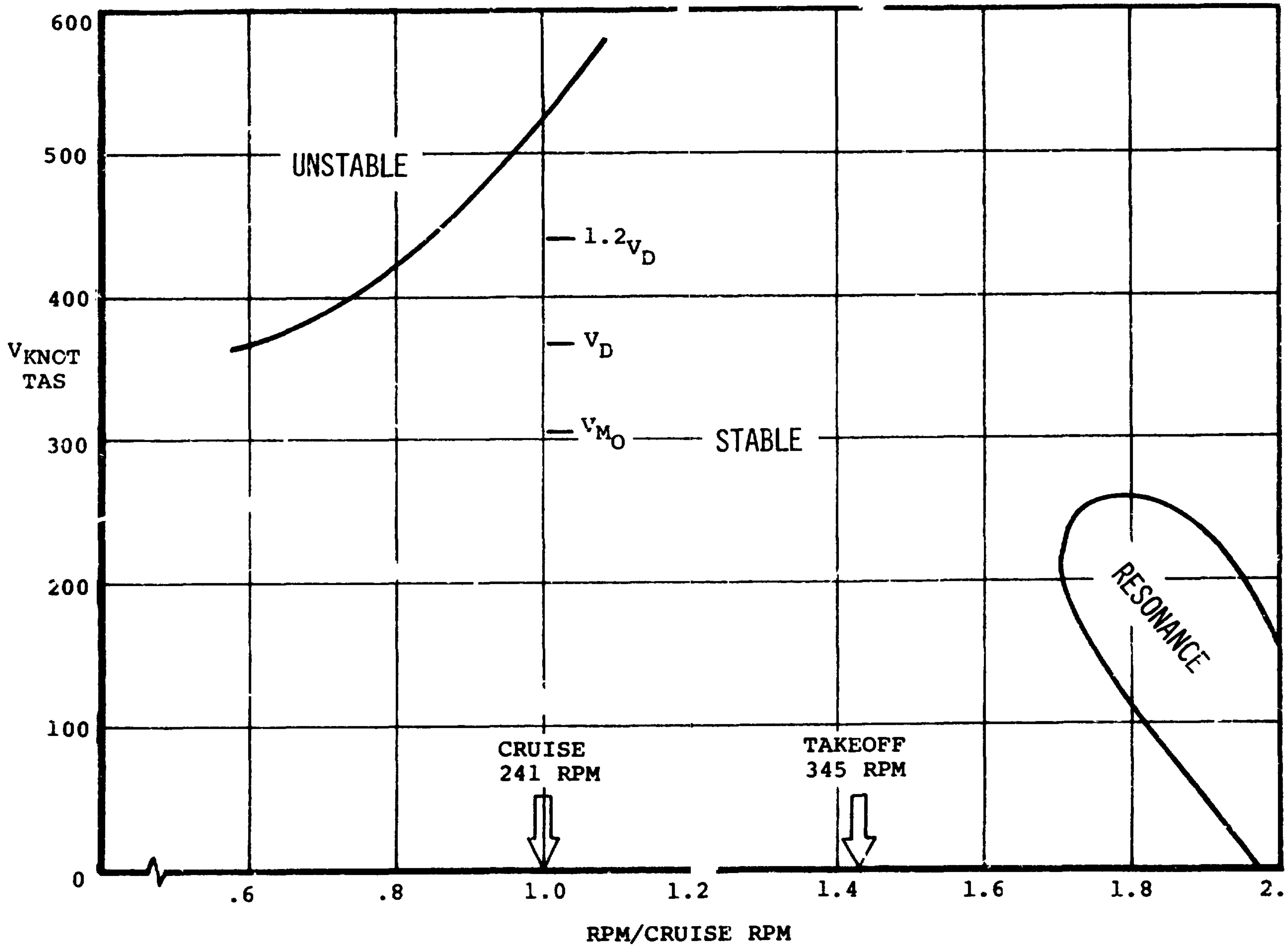


FIGURE 5.5. AEROELASTIC STABILITY BOUNDARIES FOR STOL TILT ROTOR AT SEA LEVEL.

1985 100 PASSENGER STOL TILT ROTOR AEROELASTIC
BOUNDARIES - ALTITUDE 13,300 FEET

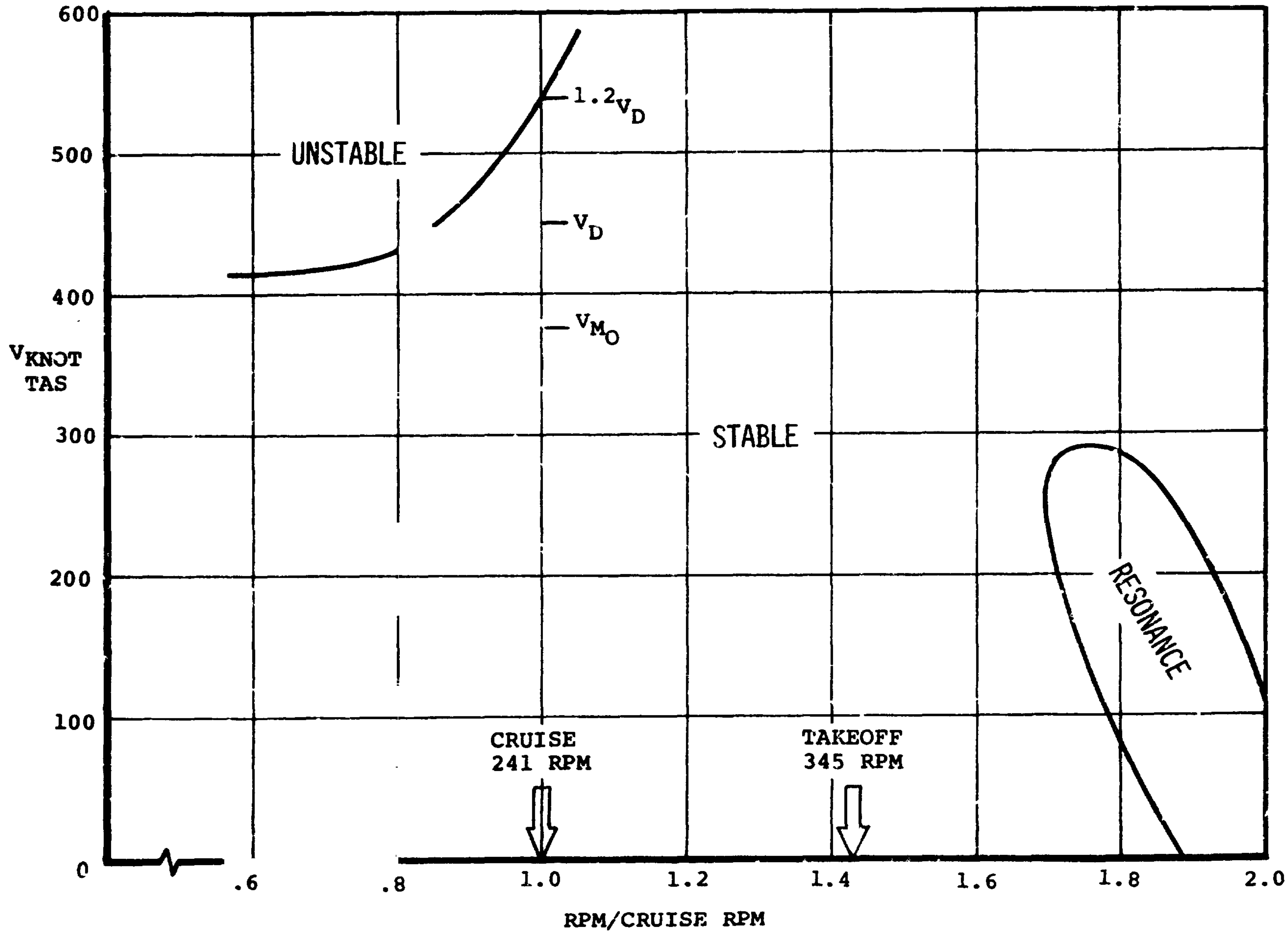


FIGURE 5.[illegible] AEROELASTIC STABILITY BOUNDARIES FOR STOL TILT ROTOR AT 13,300 FEET ALTITUDE.

weight penalty is anticipated in providing the structural stiffness implied by the above frequency criteria.

Economics

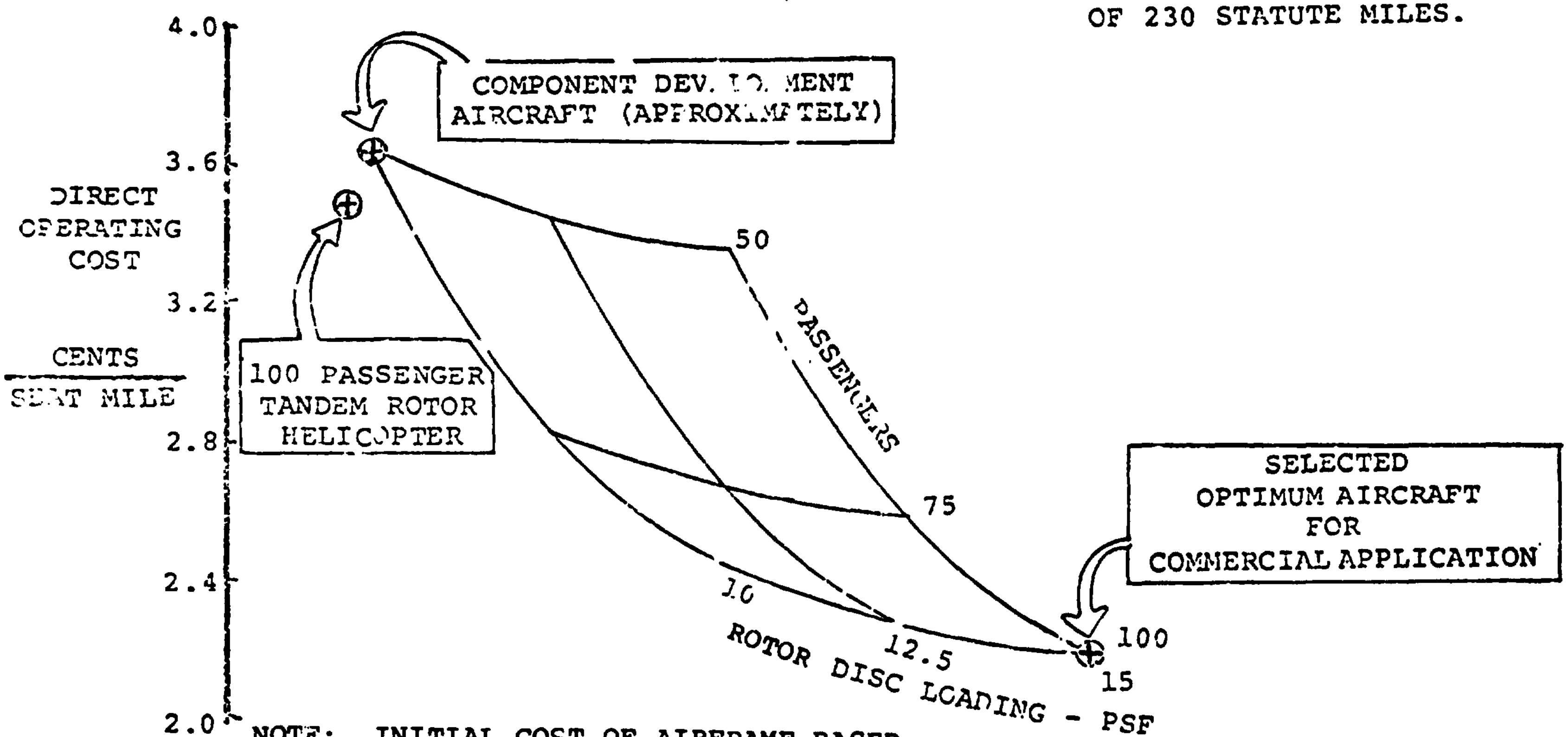
The single most important risk parameter in selecting a successful commercial vehicle is cost. The STOL aircraft passenger carrying capacity was maintained the same as for the VTOL vehicle. The issue of economics is so important that some of the arguments which led to the selection of a 100 passenger VTOL tilt rotor are recapitulated here for reference.

As the payload (i.e., number of passengers) and size of the VTOL aircraft increased, the direct operating costs decreased. This was illustrated by Figure 5.7. For example, the costs of operation per passenger mile of a 50 passenger VTOL aircraft would be 43% higher than its 100 passenger counterpart.

Since no major technology issues were identified limiting size in the study range, the optimization of vehicle cost clearly indicated that a 100 passenger vehicle (maximum allowed by the study guidelines) had to be selected. It was considered that 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 were slightly worse than the 100 passenger helicopter which was considered as almost falling within the current state-of-the-art. This

IMPACT OF PASSENGER CAPACITY ON OPERATING ECONOMICS

NOTE: INITIAL COST OF AIRFRAME BASED ON $90 PER LB AND UTILIZATION OF 2500 HOURS PER YEAR AND A RANGE OF 230 STATUTE MILES.



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 5.7. TILT ROTOR DIRECT OPERATING COSTS AS A FUNCTION OF DISC LOADING AND NUMBER OF PASSENGERS

comparison would tend to eliminate the tilt rotor from contention.

In addition, an intermediate sized vehicle would not compare favorably with conventional aircraft in terms of operating cost whereas a one hundred passenger vehicle is potentially superior as shown in Figure 5.8. This advantage is even more marked in the case of the STOL tilt rotor.

In the commercial situation, it was recognized that these economic facts required that unless compelling technical and engineering reasons were clearly identified limiting the size of the aircraft, the selected vehicle had to be of 100 passenger size if the VTOL tilt rotor concept was to realize its potential and successfully compete in the short haul market place. 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 was proposed in Reference 1.

Program Schedule

To meet a 1985 deadline for the 100 passenger transport, the program would require initiation in 1978, with laboratory work and whirl tests during 1979 and 1980. The fuselage for an intermediate sized aircraft would 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

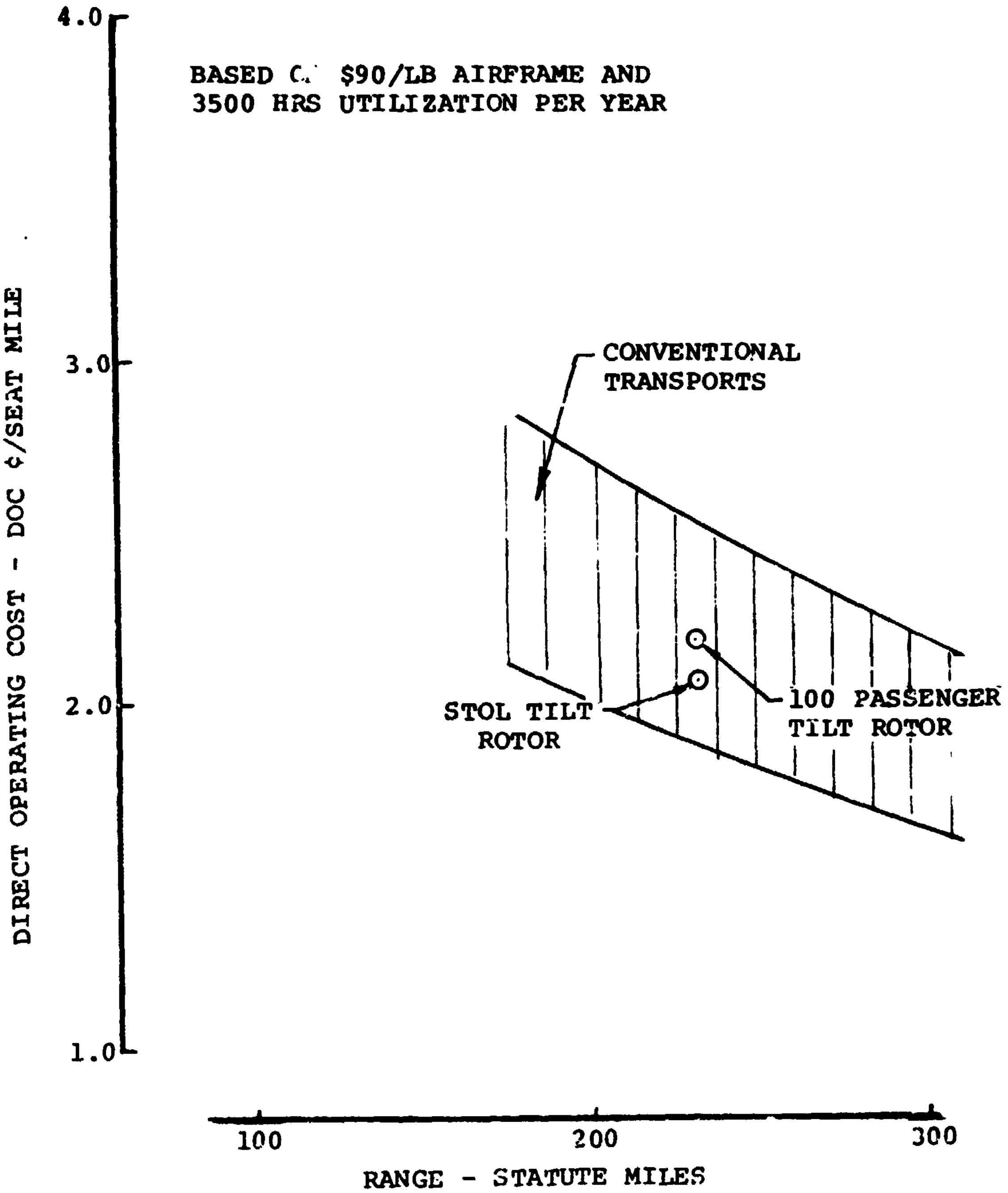


FIGURE 5.8. COMPARISON OF STOL AND VTOL TILT ROTOR AIRCRAFT DOC WITH CONVENTIONAL TRANSPORTS.

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

A 100 passenger STOL tilt rotor aircraft has been designed that can perform the same mission as the VTOL tilt rotor of Reference 1 at lower cost and fuel consumption. The price of these improvements includes a slightly lower maximum cruise speed and the loss of vertical lift capability.

In changing from VTOL to STOL the following benefits accrue:

- Initial cost is reduced from $5.15M to $4.62M.
- Fuel consumption is improved 32.1% from 47.3 to 62.5 passenger miles per gallon.
- Direct operating cost is reduced from 2.19 to 2.09 cents per seat mile.
- Gross weight, weight empty, installed power and rotor size are all significantly reduced.

The above benefits are attained at the price of the following items:

- The maximum cruise speed falls from 349 knots to 311 knots. This results in a reduction of block speed from 268 knots to 246 knots and a block time increase of less than five minutes.
- The 500 foot sideline perceived noise level is increased from 98.2 to 101.3 PNdB and the area subject to a takeoff noise level of 95 PNdB or more is increased from 0.09 to 0.11 square

miles. However, the area subject to 95 PNdB landing noise is reduced from 0.15 to 0.13 square miles.

- The installed power is sufficient to lift only 25 passengers vertically.

The change in productivity ratio when designing for STOL rather than VTOL is negligible in that reduction in speed is balanced by the reduction in empty weight.

There is no identifiable risk that can be quantified for the STOL tilt rotor though a component development program would be required to minimize development risks.

REFERENCES

1. Conceptual Design Studies of 1985 Commercial VTOL Transports That Utilize Rotors. Volume I, NASA CR 137599 and Volume II NASA CR 137600.
2. User's Manual for VASCOMP II, The V/STOL Aircraft Sizing and Performance Computer Program.
3. Wind Tunnel Tests of a Full Scale Hingeless Prop/Rotor Designed for the Boeing Vertol Model 222 Tilt Rotor Aircraft, NASA CR114-664, J.P. Magee and H.R. Alexander.

APPENDIX A

DESIGN POINT SELECTION PROCESS

The study approach, outlined in the Statement of Work, stated that the design point VTOL tilt rotor aircraft should be reevaluated as a STOL tilt rotor. To provide a meaningful comparison of STOL with VTOL this was interpreted to mean that the two aircraft should have the same capabilities in terms of payload and range. The benefits (or deficits) accruing from designing for STOL rather than VTOL would then be easily visible in terms of performance parameters and design characteristics.

In view of the desirability of achieving the same payload (100 passengers) and range (200 nautical miles) the fuselage (and its contents) of the VTOL tilt rotor was retained without modification, and the study consisted of selecting the appropriate rotors, engine power, wings and empennage.

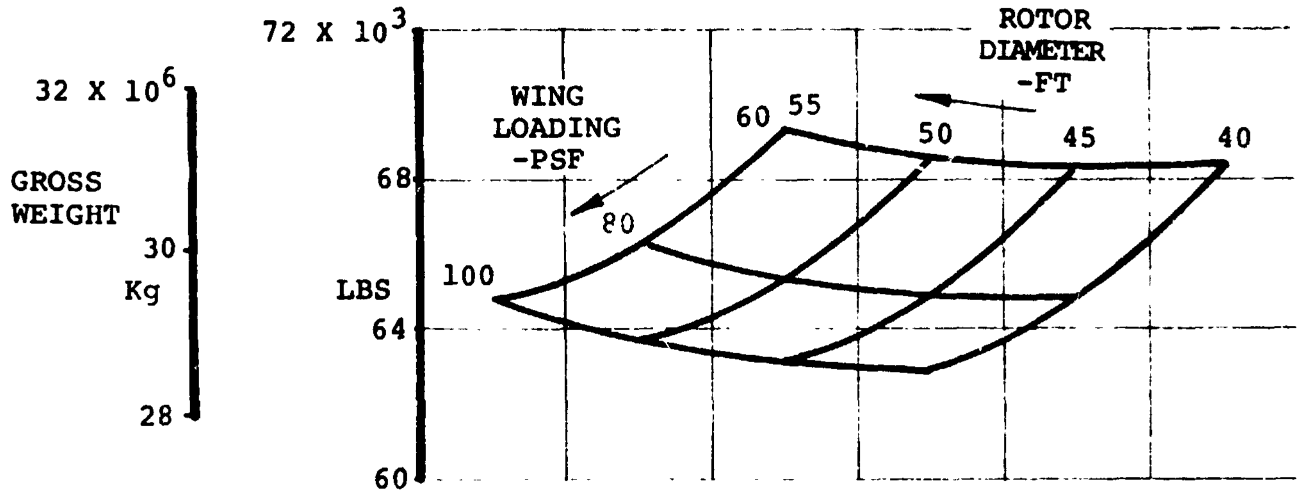
The results of the parametric studies carried out while selecting the VTOL tilt rotor are reported in Reference 1. These data form a firm basis for the initial selection of certain parameters and design conditions that are directly applicable to the selection of the STOL tilt rotor. Because of this large background of suitable data the number of parametric studies required to define the best STOL tilt rotor was greatly reduced. As in the case of the VTOL aircraft, the parametric studies were carried out using VASCOMP II, the V/STOL Aircraft Sizing and Performance Computer Program, Reference 2.

The initial parameter investigation was carried out with the baseline VTOL tilt rotor as a starting point. The most obvious parameter to change is the static thrust to weight ratio, because a smaller value of thrust is required to execute a running takeoff than would be required to lift the aircraft vertically. The other parameters exercised during the first iteration were wing loading and rotor diameter. At this stage the rotor design remained unchanged except for such changes as resulted from scaling the diameter at constant solidity. The takeoff and cruise rotor tipspeeds were held at 775 and 542.5 feet per second respectively.

The results of the sizing calculations carried out at this stage are plotted in carpet form in Figures A1 through A9. On each graph of direct operating cost versus wing loading and rotor diameter (Figures A3, A6 and A9) a selection was made of the wing loading and rotor diameter corresponding to minimum direct operating cost. Obviously, on the graphs shown, Figure A3 for example, there is no mathematical minimum of operating cost. There are, however, practical constraints that place a lower bound or limit on the cost. The limiting factors in this case was imposed by practical design considerations. A maximum wing loading

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ASPECT RATIO = 7.14
ROTOR SOLIDITY = 0.09
ROTOR TIPSPEED = 775 FT/SEC @ T.O.
STATIC THRUST/WEIGHT = 0.8



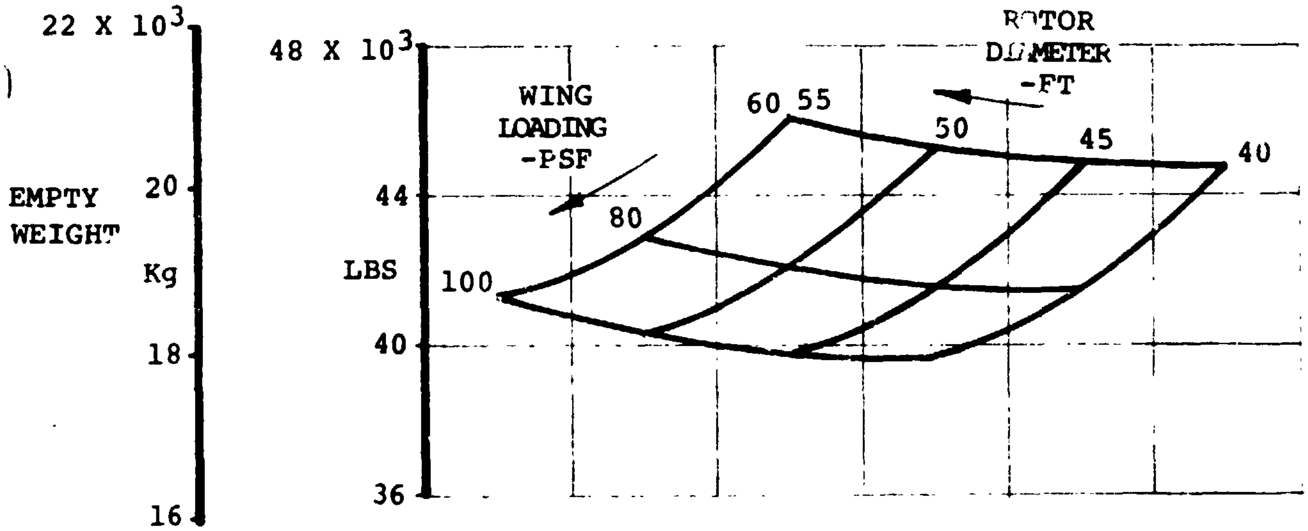
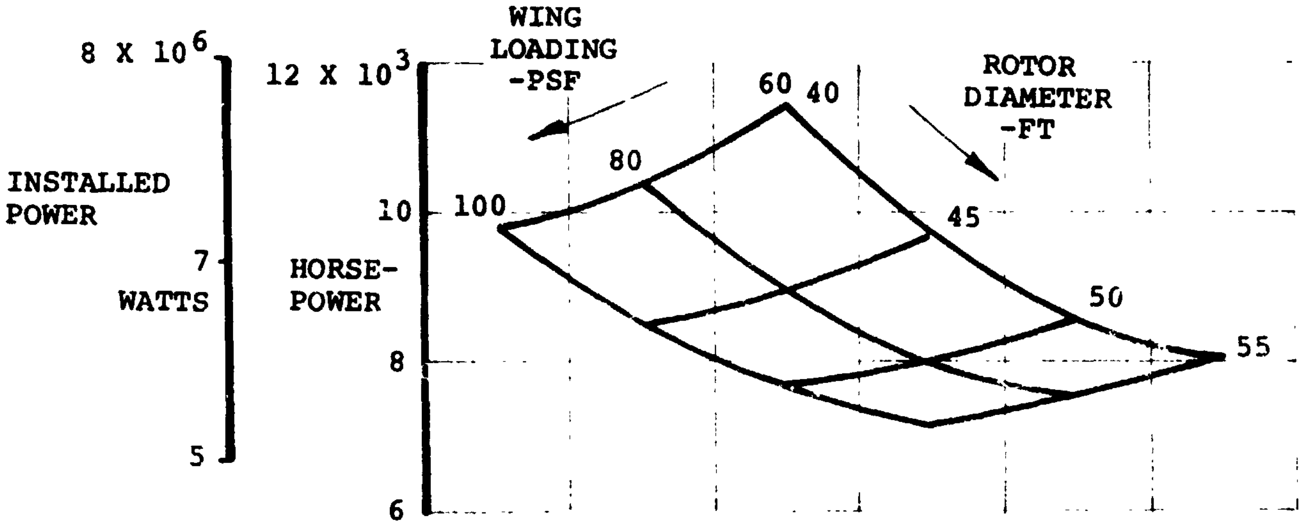


FIGURE A-1. VARIATION OF GROSS WEIGHT AND EMPTY WEIGHT WITH WING LOADING AND ROTOR DIAMETER (T/W = 0.8).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ASPECT RATIO = 7.14
ROTOR SOLIDITY = 0.09
ROTOR TIPSPEED = 775 FT/SEC @ T.O.
STATIC THRUST/WEIGHT = 0.8



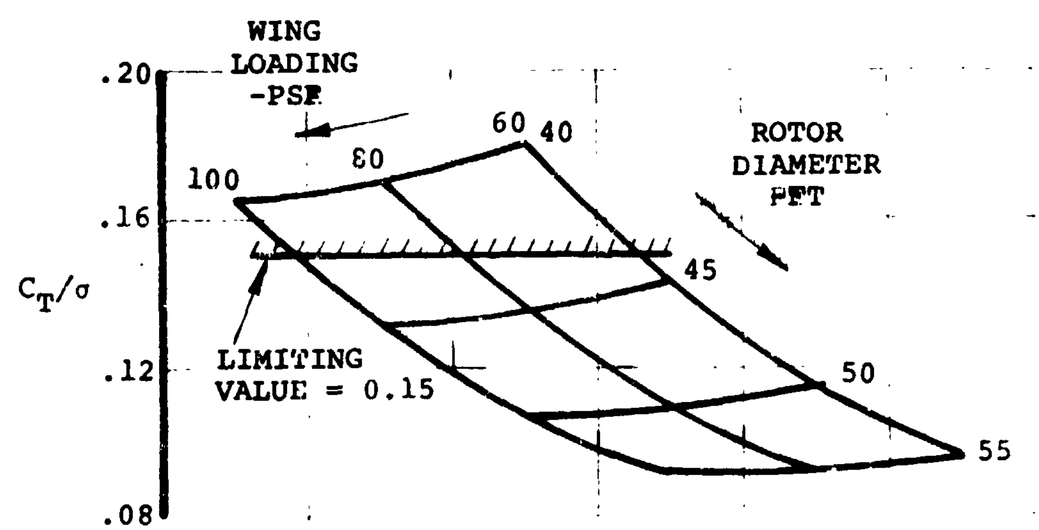


FIGURE A-2. VARIATION OF INSTALLED POWER AND ROTOR LOADING WITH WING LOADING AND ROTOR DIAMETER (T/W = 0.8).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ASPECT RATIO = 7.14
ROTOR SOLIDITY = 0.09
ROTOR TIPSPEED = 775 FT/SEC @ T.O.
STATIC THRUST/WEIGHT = 0.8

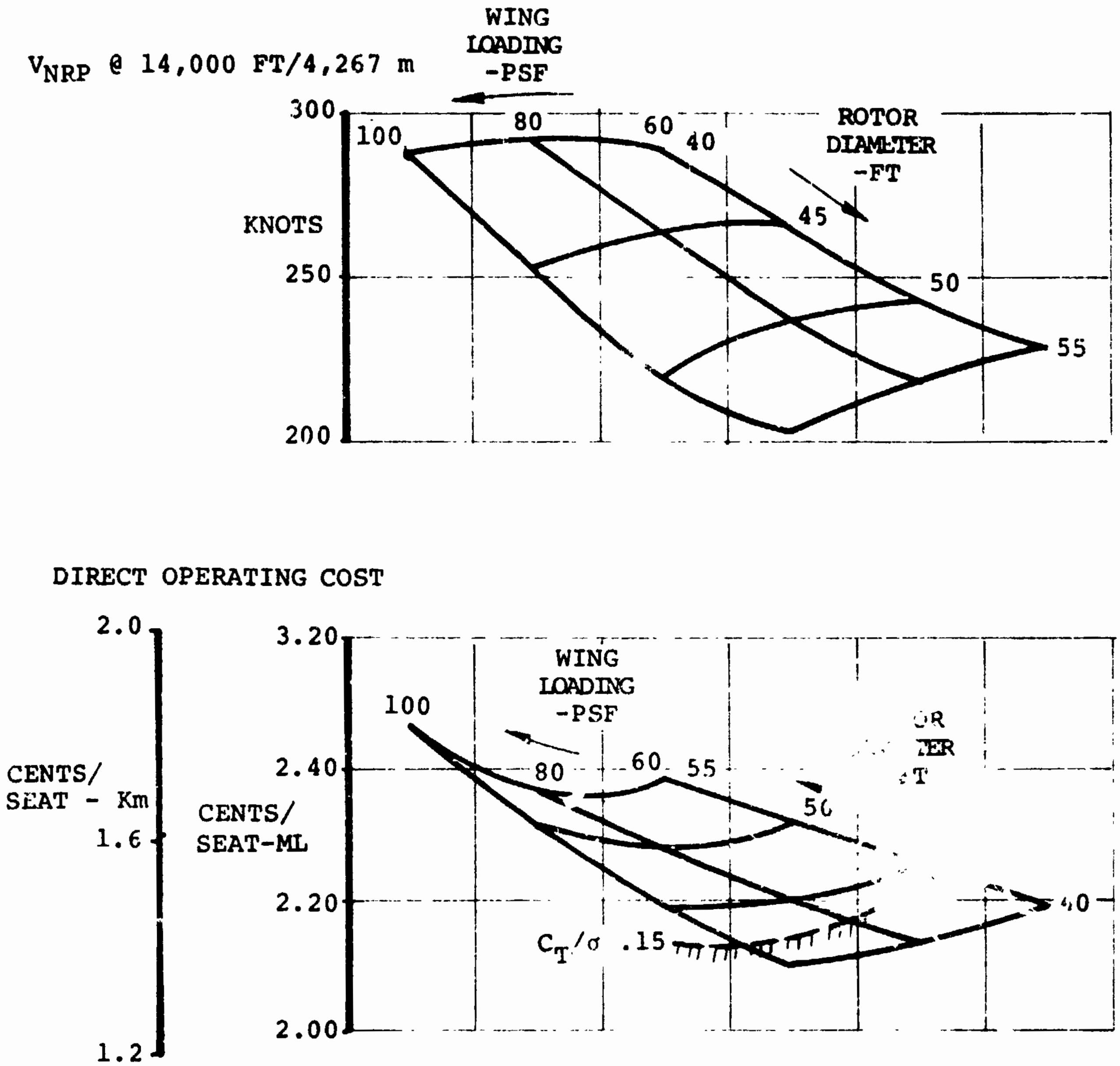
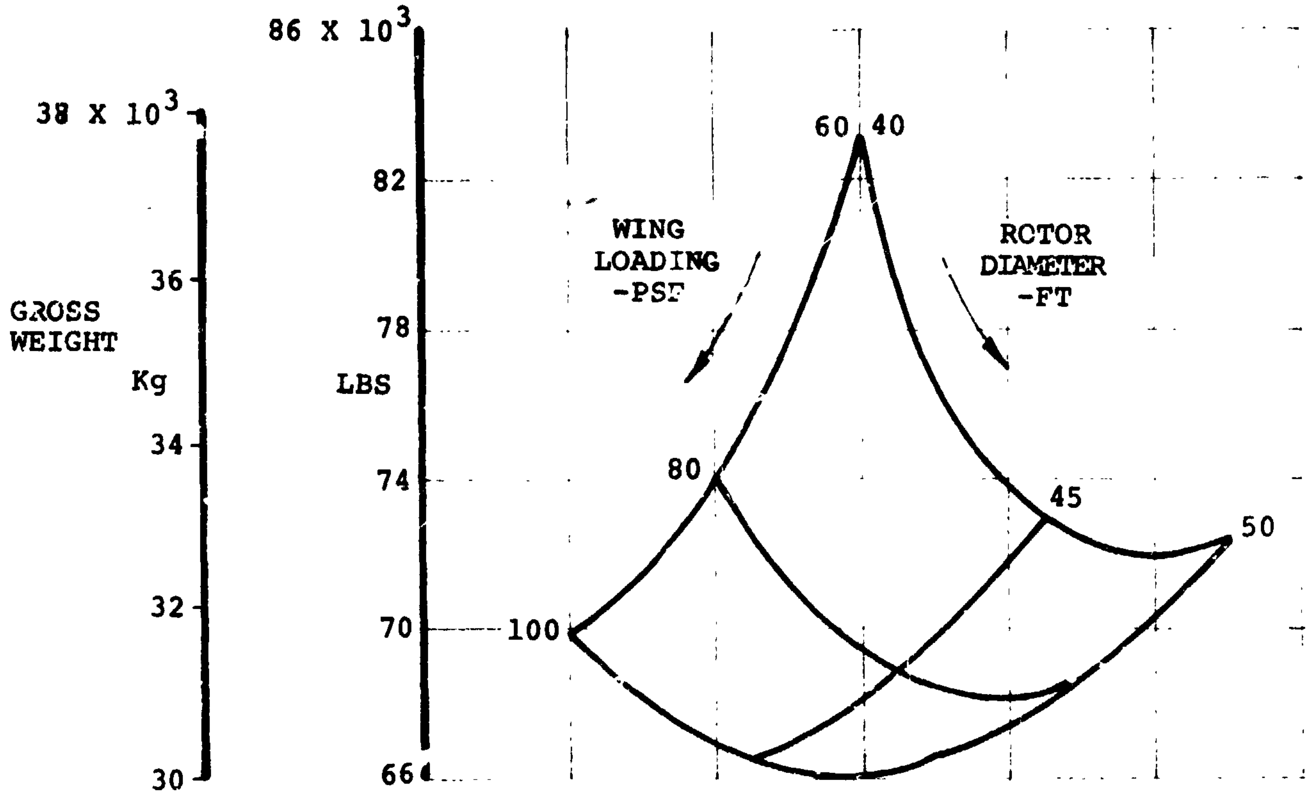


FIGURE A-3. VARIATION OF MAXIMUM CRUISE SPEED AND DIRECT OPERATING COST WITH WING LOADING AND ROTOR DIAMETER (T/W = 0.8).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ASPECT RATIO = 7.14
ROTOR SOLIDITY = 0.09
ROTOR TIPSPEED = 775 FT/SEC @ T.O.
STATIC THRUST/WEIGHT = 0.95



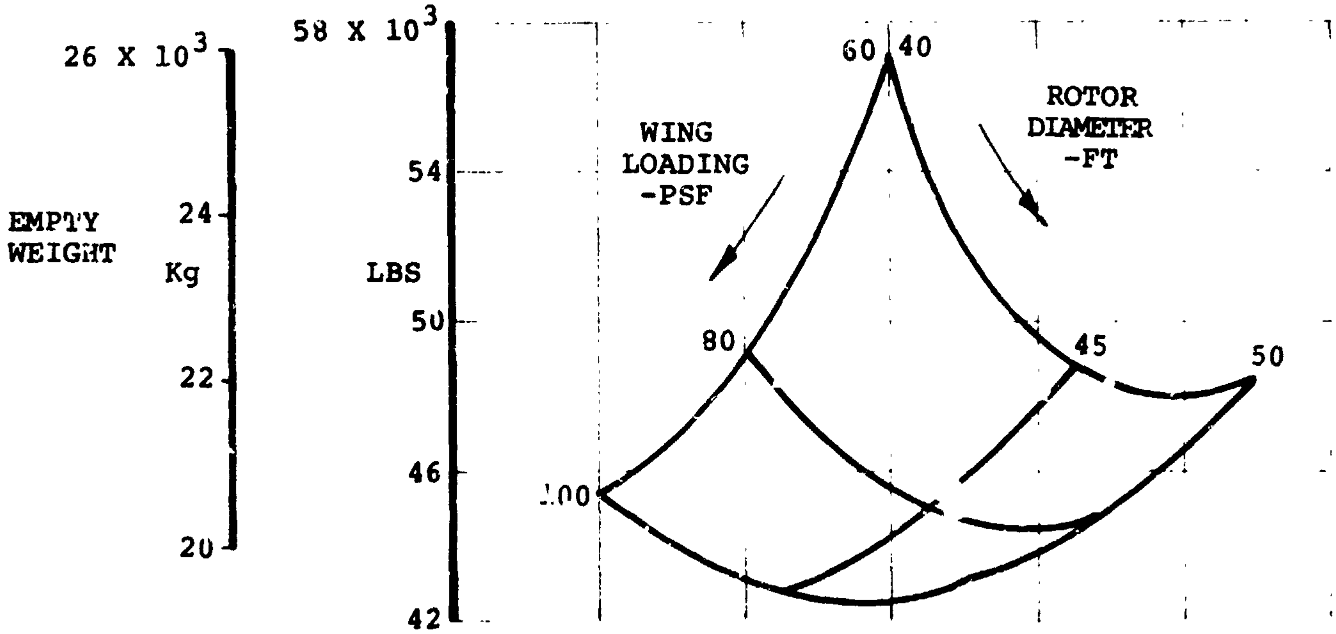
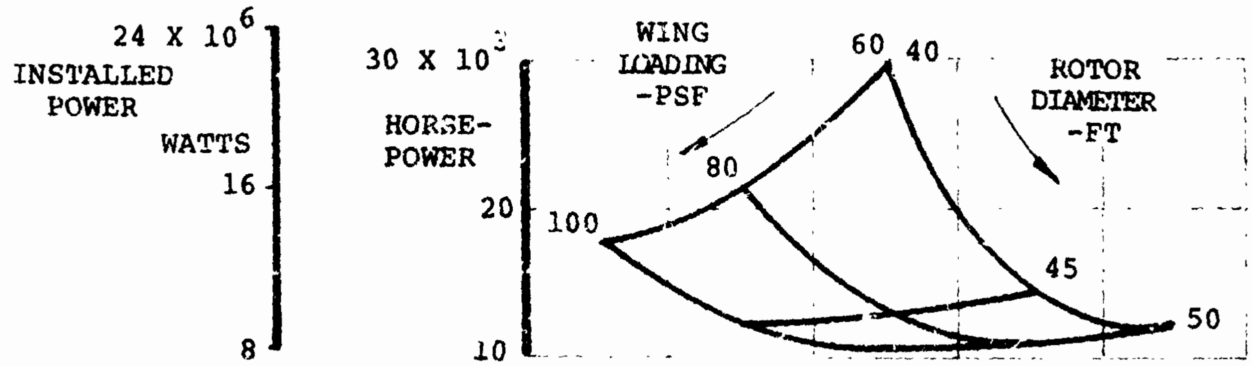


FIGURE A-4. VARIATION OF GROSS WEIGHT AND EMPTY WEIGHT WITH WING LOADING AND ROTOR DIAMETER (T/W = 0.95).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ASPECT RATIO = 7.14
ROTOR SOLIDITY = 0.09
ROTOR TIPSPEED = 775 FT/SEC @ T.O.
STATIC THRUST/WEIGHT = 0.95



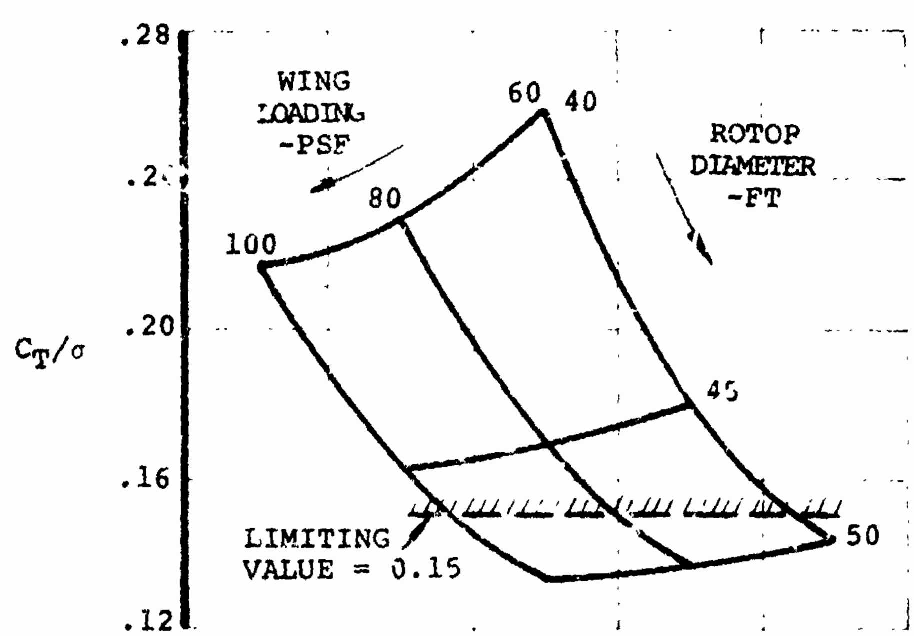


FIGURE A-5. VARIATION OF INSTALLED POWER AND ROTOR LOADING WITH WING LOADING AND ROTOR DIAMETER (T/W = 0.95).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ASPECT RATIO = 7.14
ROTOR SOLIDITY = 0.09
ROTOR TIPSPEED = 775 FT/SEC @ T.O.
STATIC THRUST/WEIGHT = 0.95

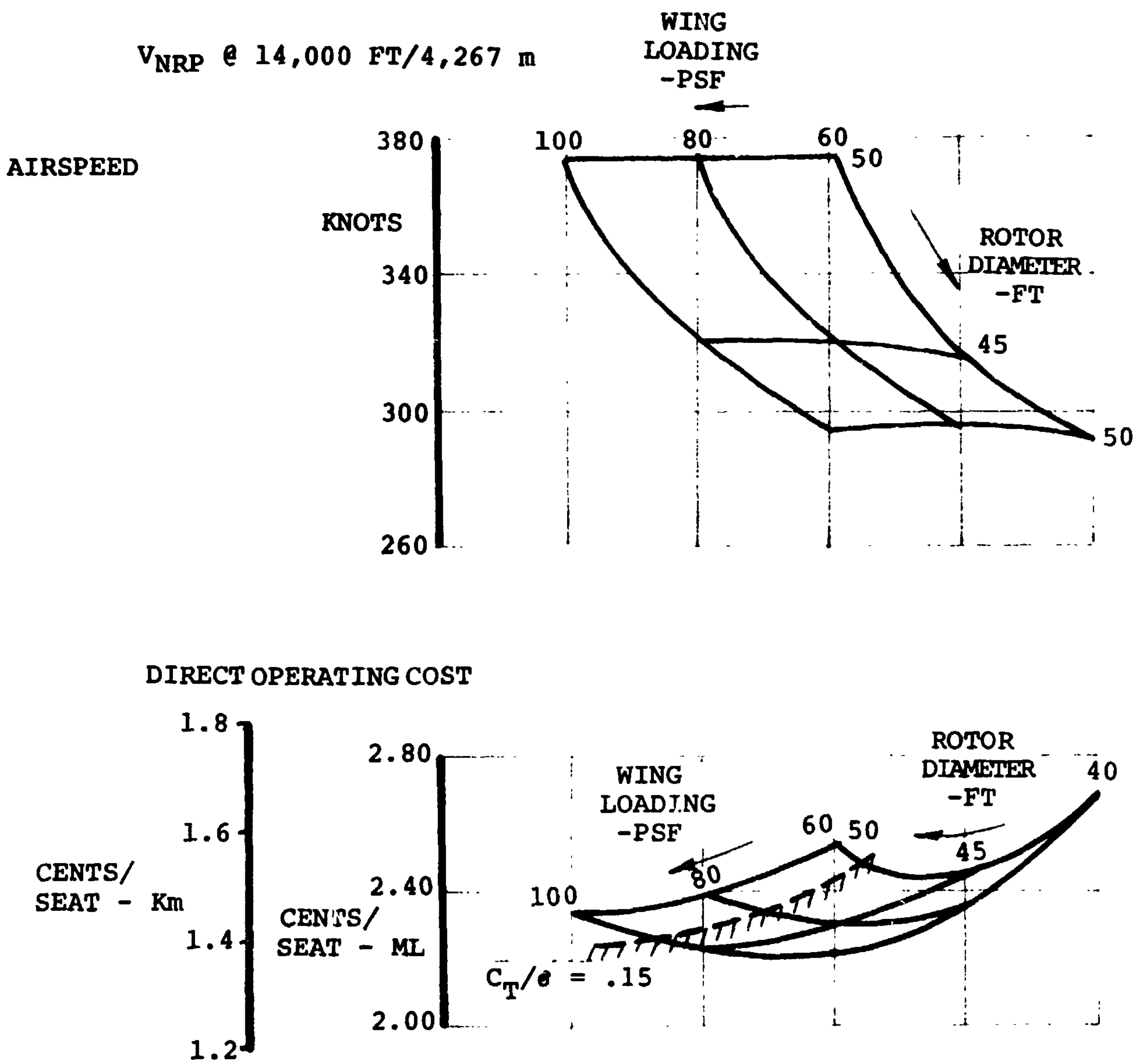


FIGURE A-6. VARIATION OF MAXIMUM CRUISE SPEED AND DIRECT OPERATING COST WITH WING LOADING AND ROTOR DIAMETER (T/W = 0.95).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ASPECT RATIO = 7.14
ROTOR SOLIDITY = 0.09
ROTOR TIPSPEED = 775 FT/SEC @ T.O.
STATIC THRUST/WEIGHT = 1.05

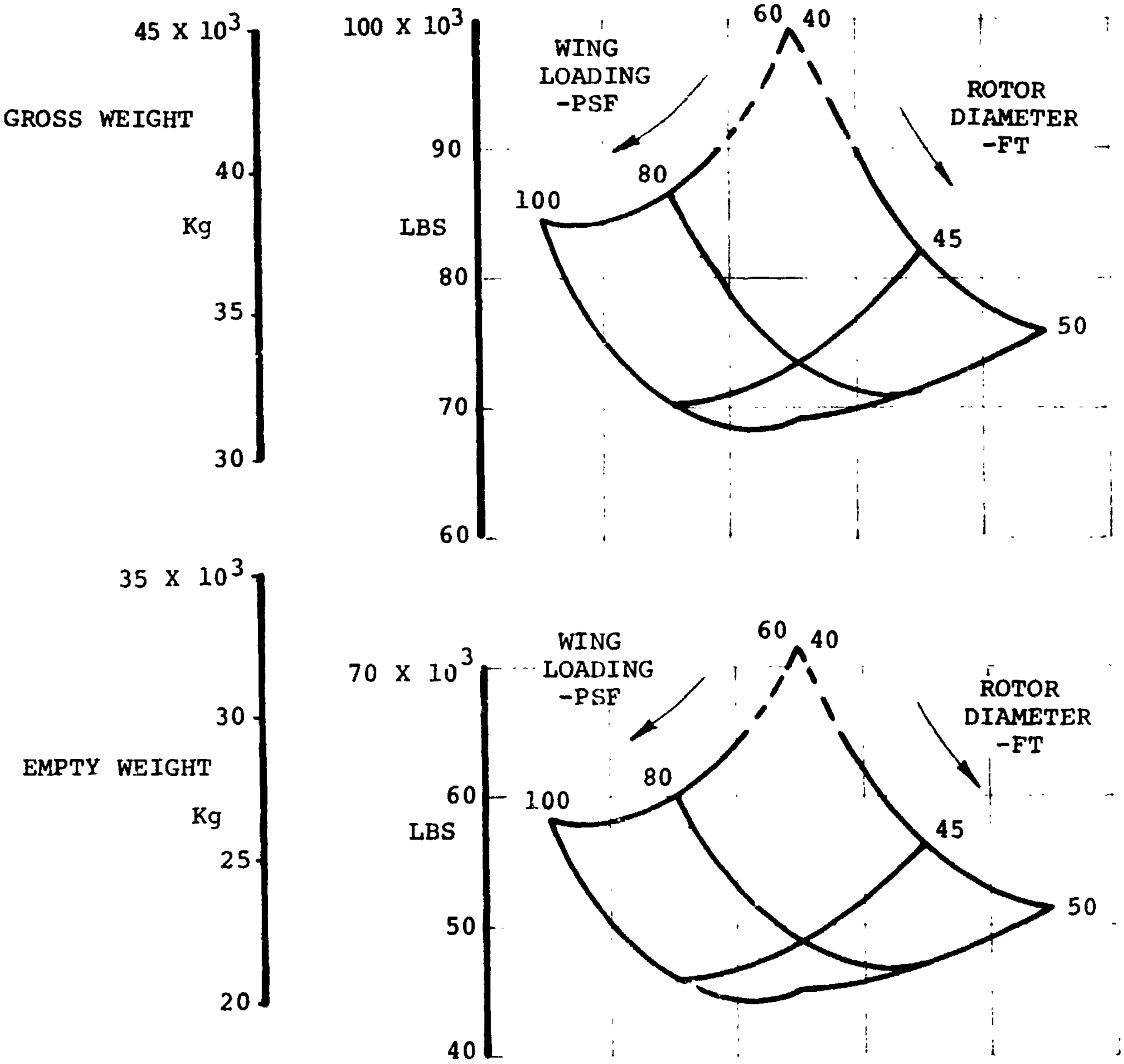


FIGURE A-7. VARIATION OF GROSS WEIGHT AND EMPTY WEIGHT WITH WING LOADING AND ROTOR DIAMETER (T/W = 1.05).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ASPECT RATIO = 7.14
ROTOR SOLIDITY = 0.09
ROTOR TIPSPEED = 775 FT/SEC @ T.O.
STATIC THRUST/WEIGHT = 1.05

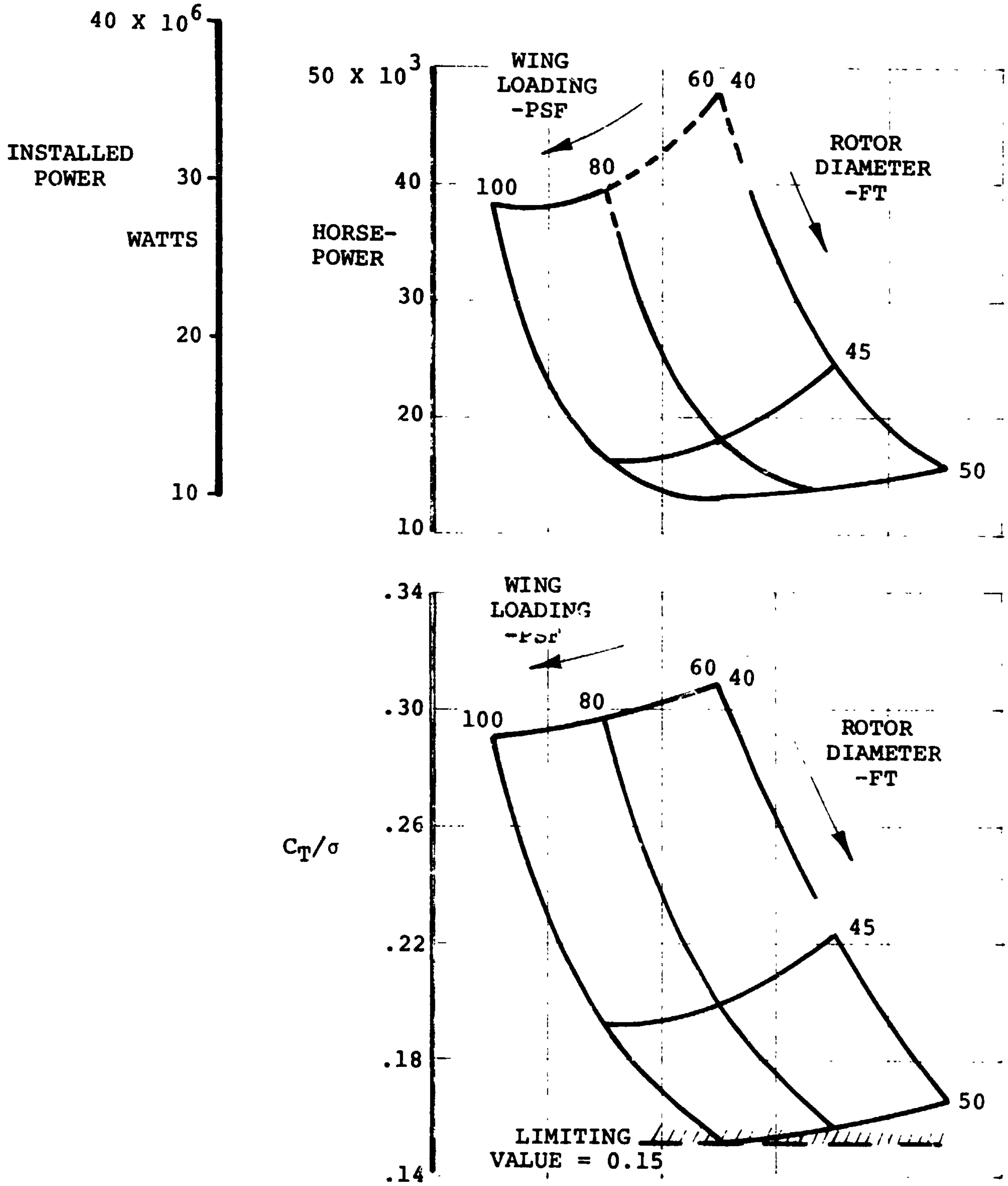


FIGURE A-8. VARIATION OF INSTALLED POWER AND ROTOR LOADING WITH WING LOADING AND ROTOR DIAMETER (T/W = 1.05).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ASPECT RATIO = 7.14
ROTOR SOLIDITY = 0.09
ROTOR TIPSPEED = 775 FT/SEC @ T.O.
STATIC THRUST/WEIGHT = 1.05

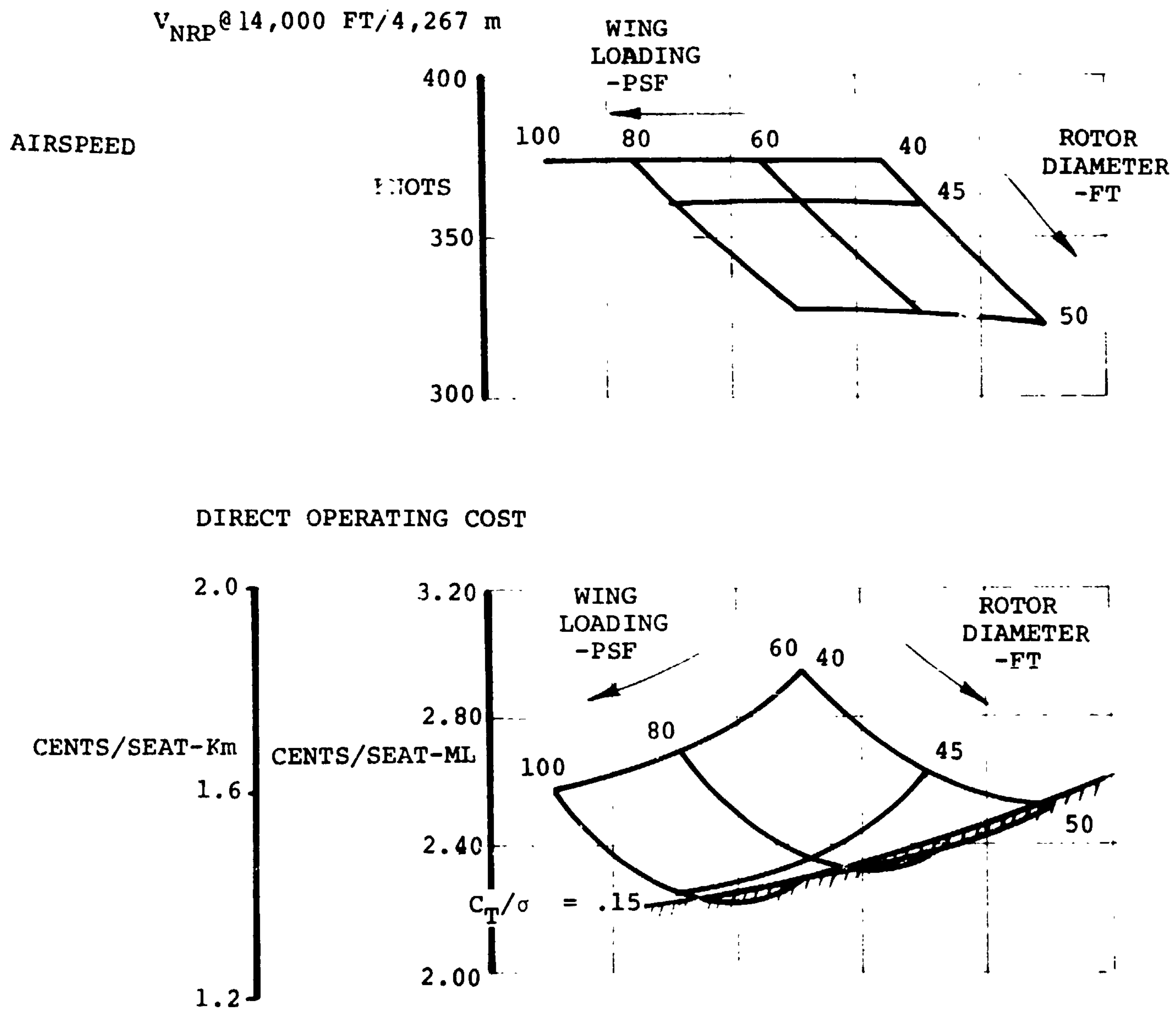


FIGURE A-9. VARIATION OF MAXIMUM CRUISE SPEED AND DIRECT OPERATING COST WITH WING LOADING AND ROTOR DIAMETER (T/W = 1.05).

of 100 pounds per square foot has been specified for several reasons. First, higher values of wing loading lead to an unacceptably large stall speed (and, therefore, too high a speed at the end of transition without flaps). For a maximum lift coefficient of 1.31 an end of transition speed of 180 knots allows a 20 percent margin above stall speed. Other factors adversely affected by high wing loading are the takeoff and landing performance and the weight of the wing structure.

The other limiting factor at this stage of the selection process was a value of blade loading, C_T/σ of 0.15. This was an estimated practical limit based on the loading calculations carried out in Reference 1.

The direct operating cost minima (which all coincided with the wing loading limit) and the corresponding rotor diameters were then plotted as a function of static thrust to weight ratio, Figure A10. The static thrust to weight ratio was then selected to correspond with the minimum direct operating cost and the corresponding rotor diameter was read off at the same thrust to weight. The maximum wing loading and the optimum values or rotor diameter and thrust to weight ratio were retained in subsequent tradeoffs.

At this stage a preliminary check of the takeoff and landing performance was made for the aircraft defined by the parameters selected. The results indicated that the takeoff performance was marginal in that field lengths of slightly

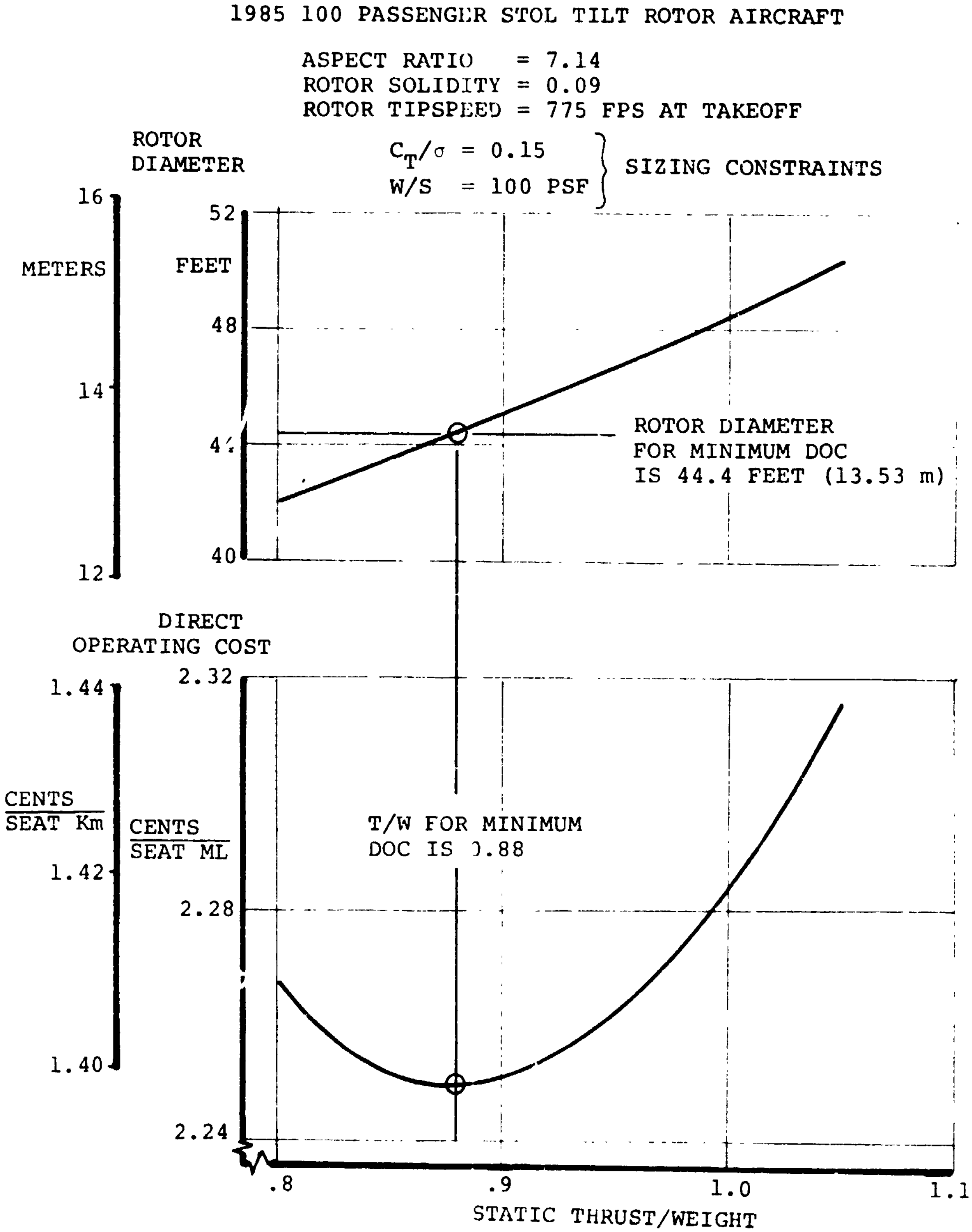


FIGURE A-10. VARIATION OF ROTOR DIAMETER AND MINIMUM DIRECT OPERATING COST WITH STATIC THRUST/WEIGHT RATIO.

greater than the required 2,000 feet were required. However, the process of optimizing the aircraft parameters had not been completed and detail geometry related to the takeoff configuration had not been defined, so a considerable improvement in takeoff performance could be expected. Consequently, the optimum values of diameter, wing loading and static thrust to weight ratio were retained for the next iteration in selection of the optimum aircraft.

The second part of the design selection process involved the isolation of the best combination of rotor solidity, takeoff tipspeed and wing aspect ratio. The results of this parametric study are plotted, in Figures A11 through A22, in carpet form. All important design variables are plotted as a function of wing aspect ratio and rotor tipspeed at a fixed value of rotor solidity. From the graphs of direct operating cost the aspect ratio and tipspeed corresponding to minimum direct operating cost were selected. In each case the minimum cost corresponded to an aspect ratio of 9. Values of aspect ratio greater than nine were not considered for a number of reasons. Although no calculations were made at the time, engineering judgement based on previous experience indicated that higher values of aspect ratio would result in too small a value of wing chord. The result of this would be that the wing box structure would be very small and, in order to retain sufficient static strength in the wing structure and to

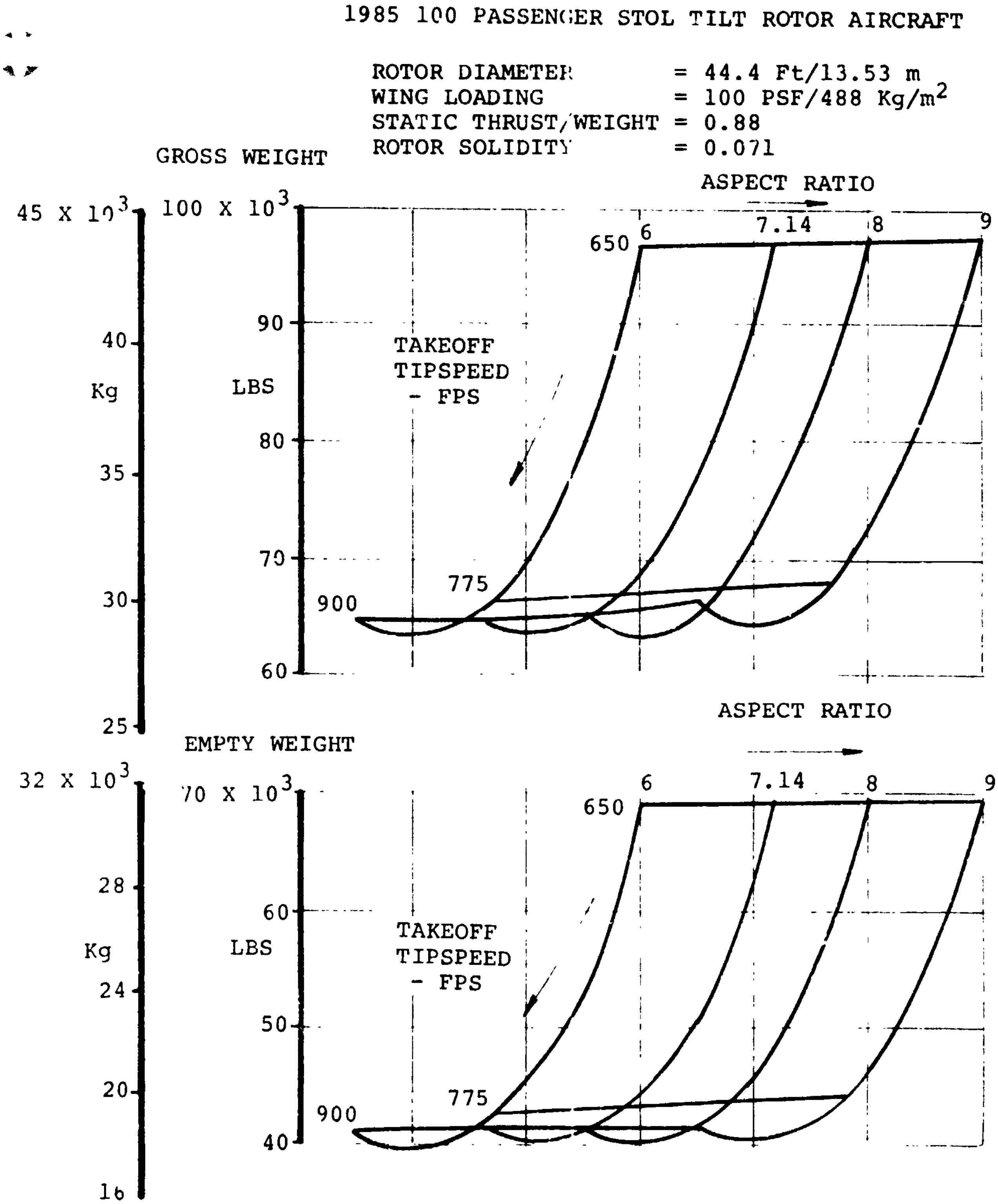
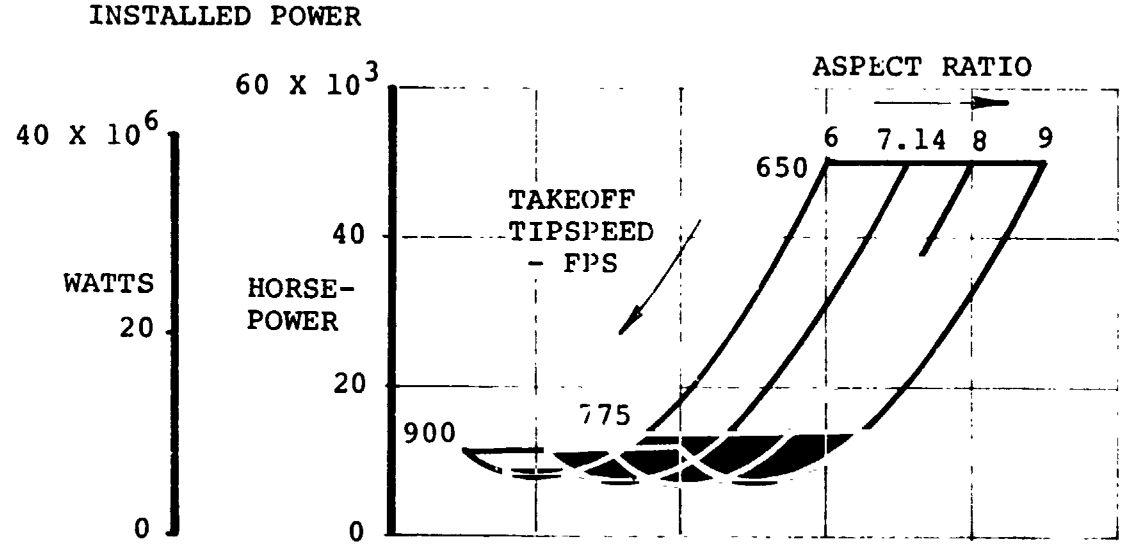


FIGURE A-11. VARIATION OF GROSS WEIGHT AND EMPTY WEIGHT WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.071).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ROTOR DIAMETER	=	44.4 FT/13.53 m
WING LOADING	=	100 PSF/488 Kg/m^2
STATIC THRUST/WEIGHT	=	0.88
ROTOR SOLIDITY	=	0.071



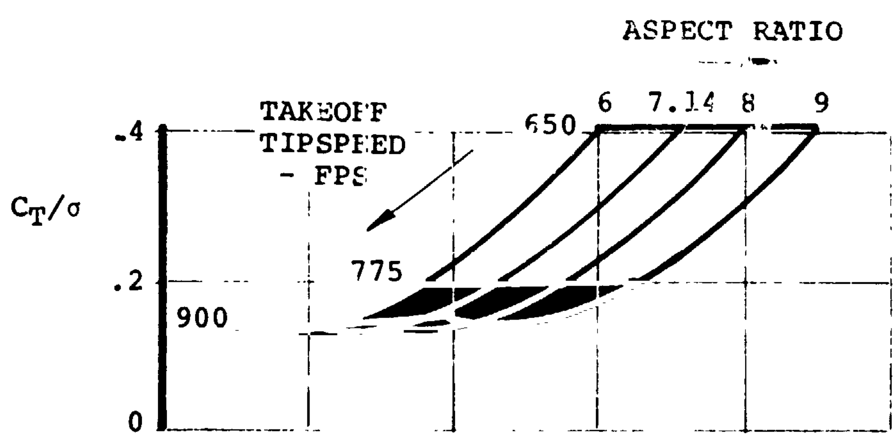


FIGURE A-12. VARIATION OF INSTALLED POWER AND ROTOR LOADING WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.071).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ROTOR DIAMETER = 44.4 Ft/13.53m
WING LOADING = 100 PSF/488 Kg/m^2
STATIC THRUST/WEIGHT = 0.88
ROTOR SOLIDITY = 0.071

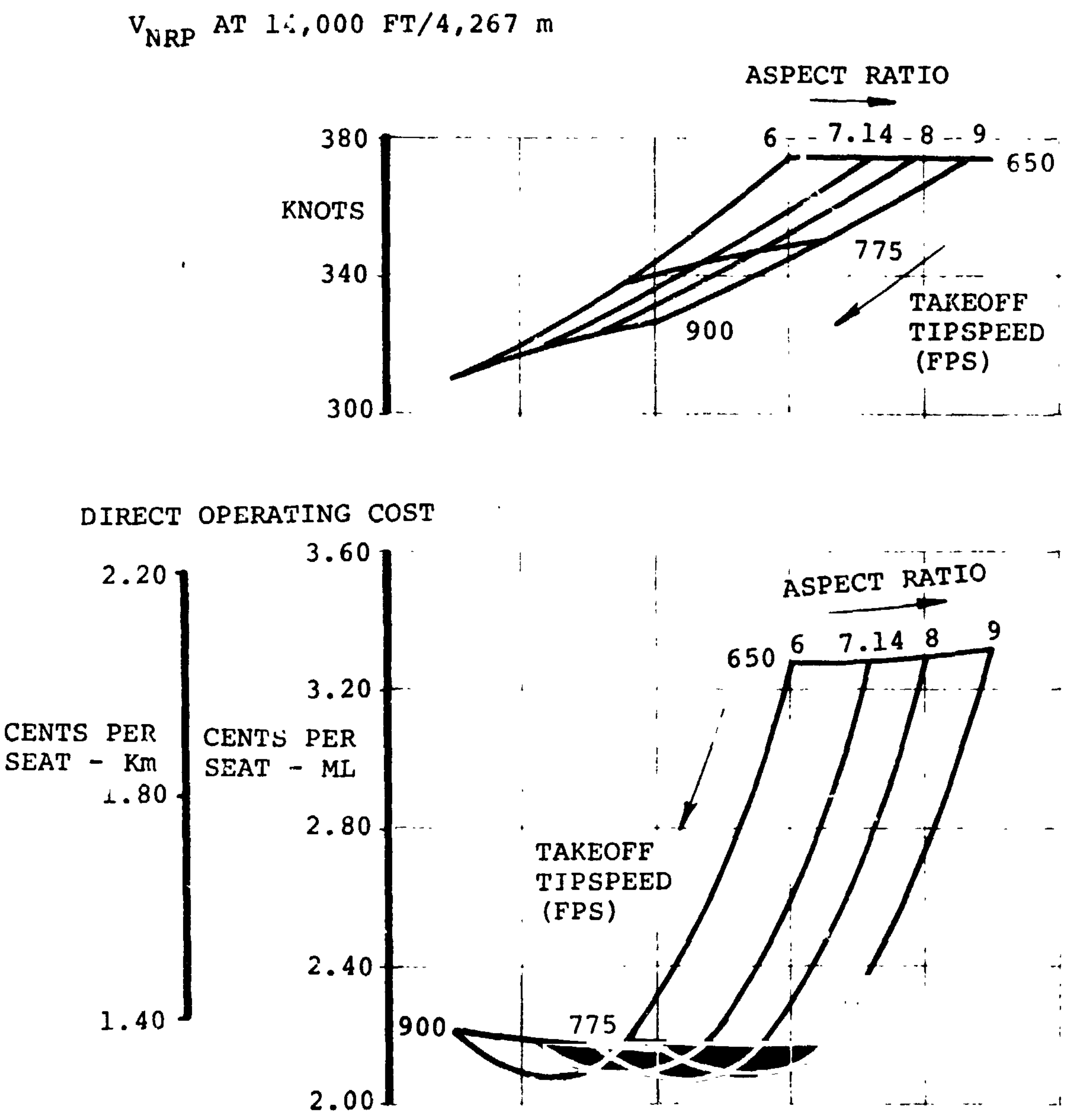


FIGURE A-13. VARIATION OF MAXIMUM CRUISE SPEED AND DIRECT OPERATING COST WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = .071).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ROTOR DIAMETER = 44.4 FT/13.53 M
WING LOADING = 100 PSF/488 Kg/m^2
STATIC THRUST/WEIGHT = 0.88
ROTOR SOLIDITY = 0.09

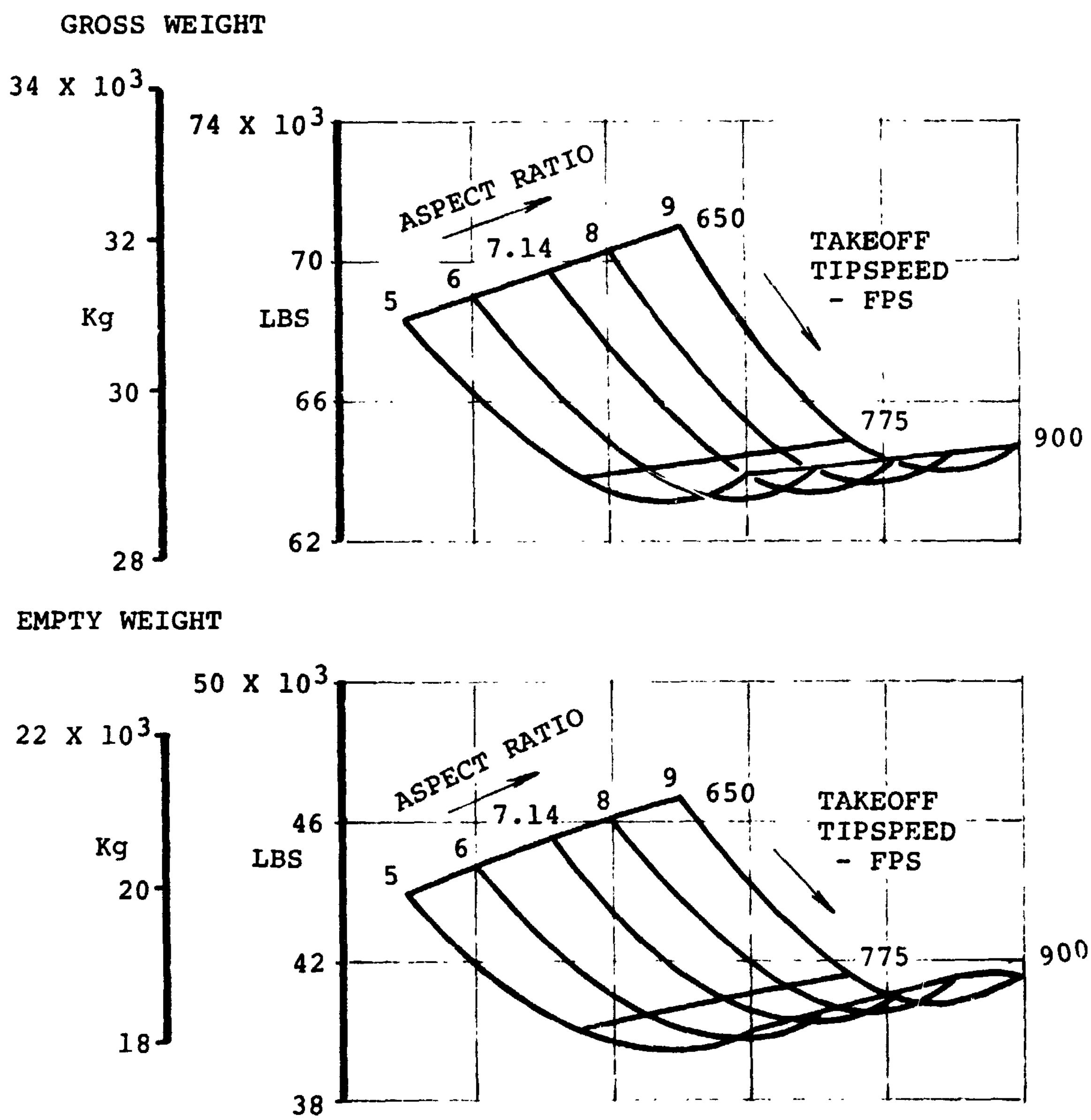


FIGURE A-14. VARIATION OF GROSS WEIGHT AND EMPTY WEIGHT WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.09).

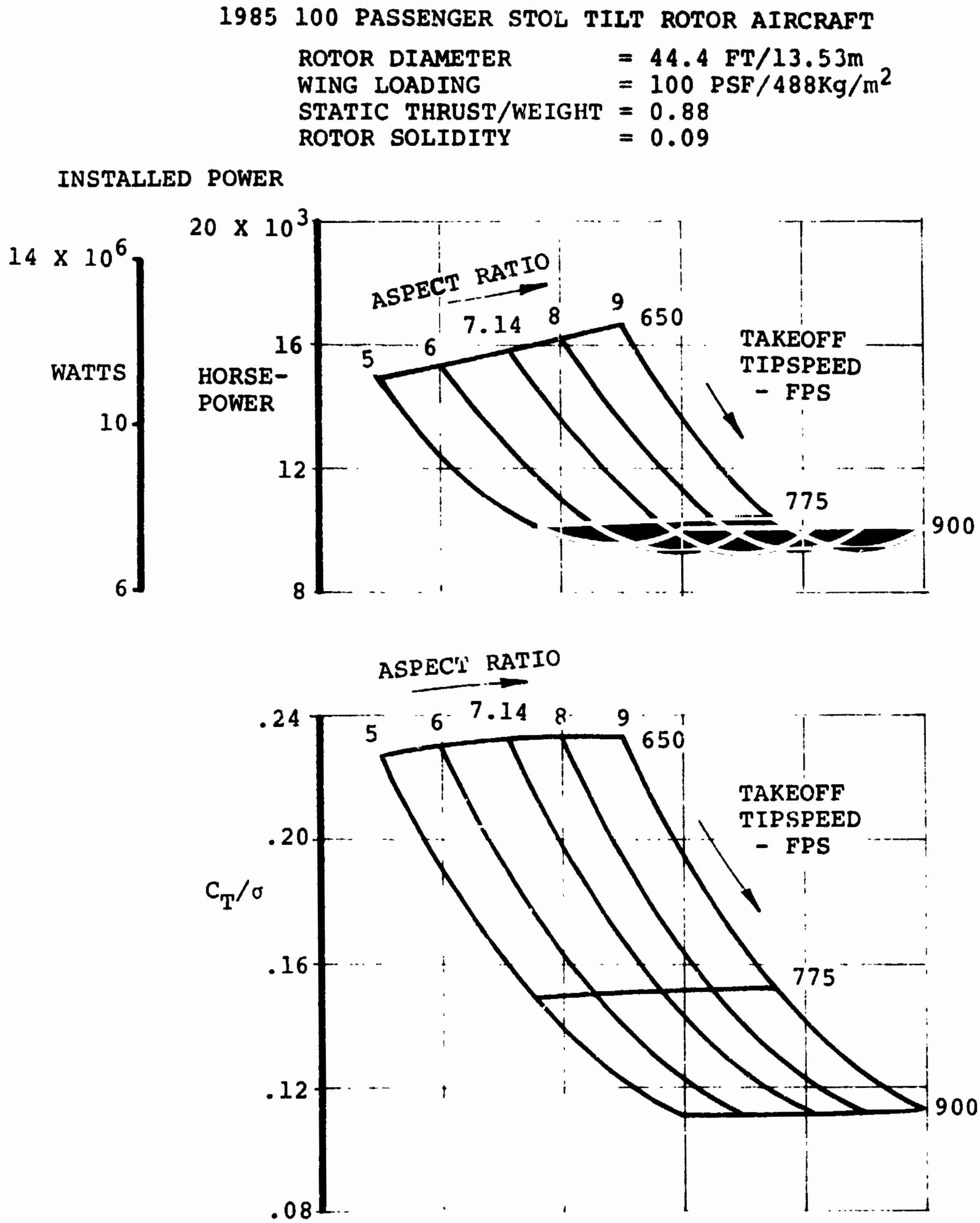


FIGURE A-15. VARIATION OF INSTALLED POWER AND ROTOR LOADING WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.09).

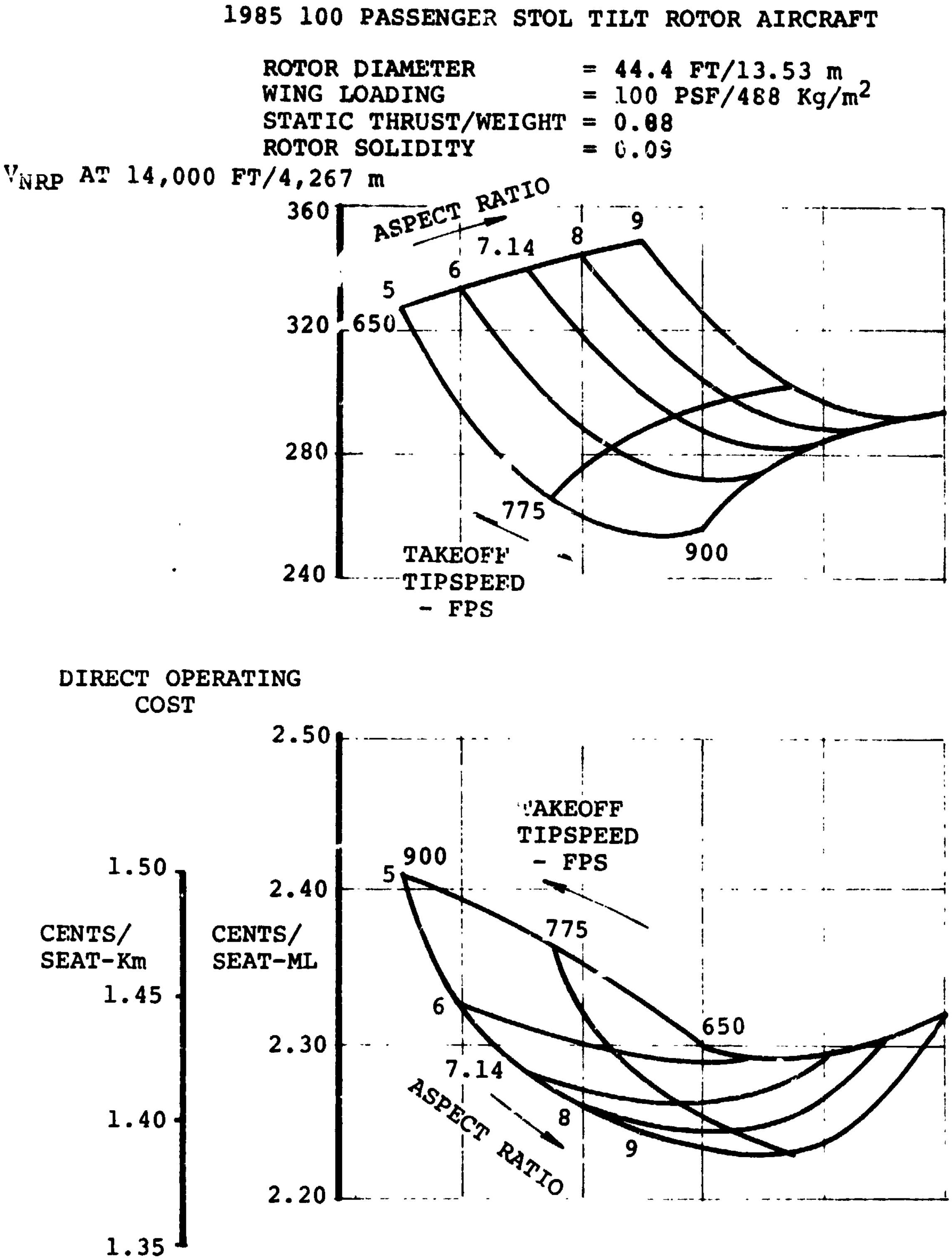


FIGURE A-16. VARIATION OF MAXIMUM CRUISE SPEED AND DIRECT OPERATING COST WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.09).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ROTOR DIAMETER = 44.4 FT/13.53 m
WING LOADING = 100 PSF/488 Kg/m^2
STATIC THRUST/WEIGHT = 0.88
ROTOR SOLIDITY = 0.105

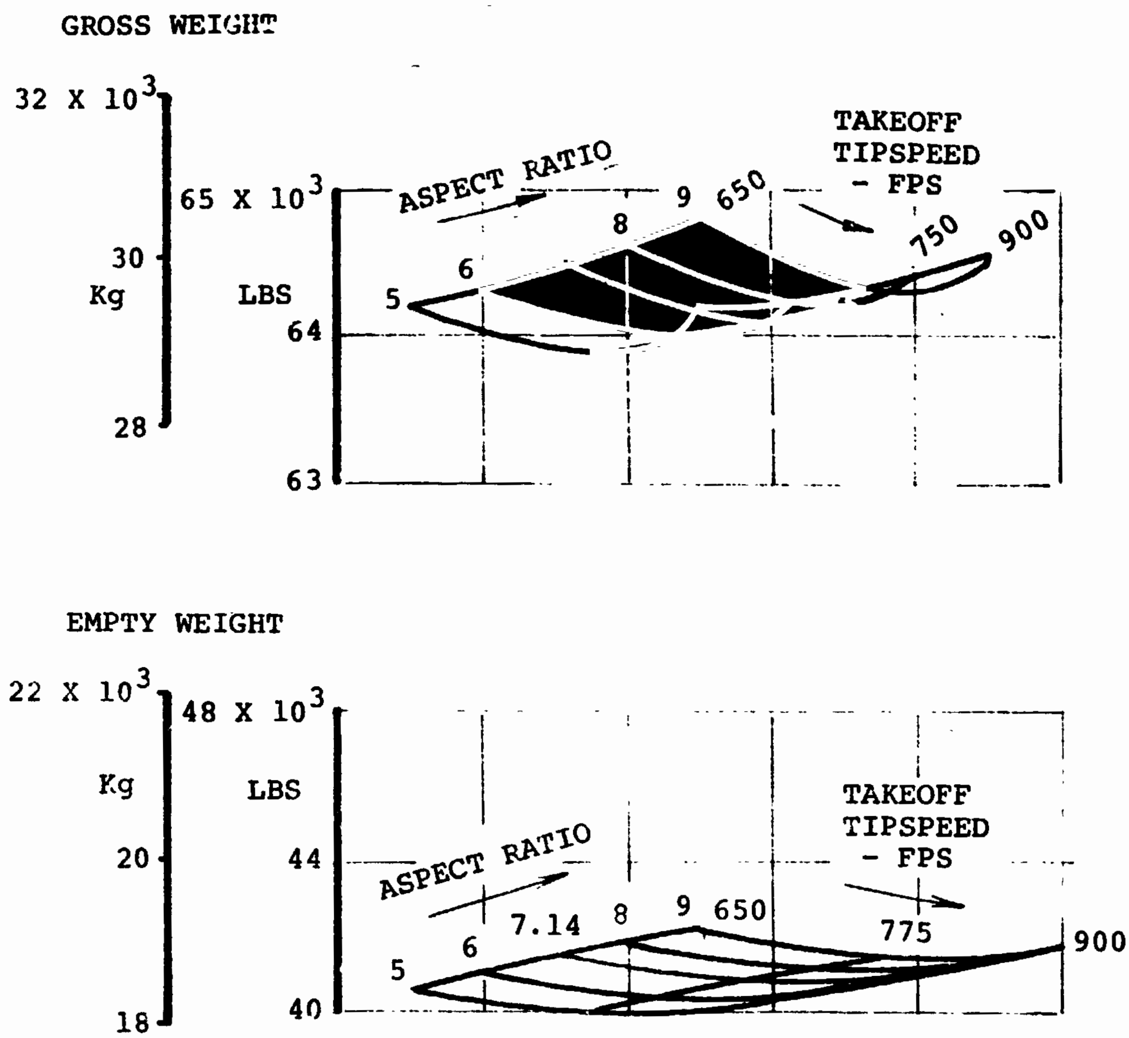


FIGURE A-17. VARIATION OF GROSS WEIGHT AND EMPTY WEIGHT WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.105).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ROTOR DIAMETER = 44.4 FT/13.53 m
WING LOADING = 100 PSF/488 Kg/m^2
STATIC THRUST/WEIGHT = 0.88
ROTOR SOLIDITY = 0.105

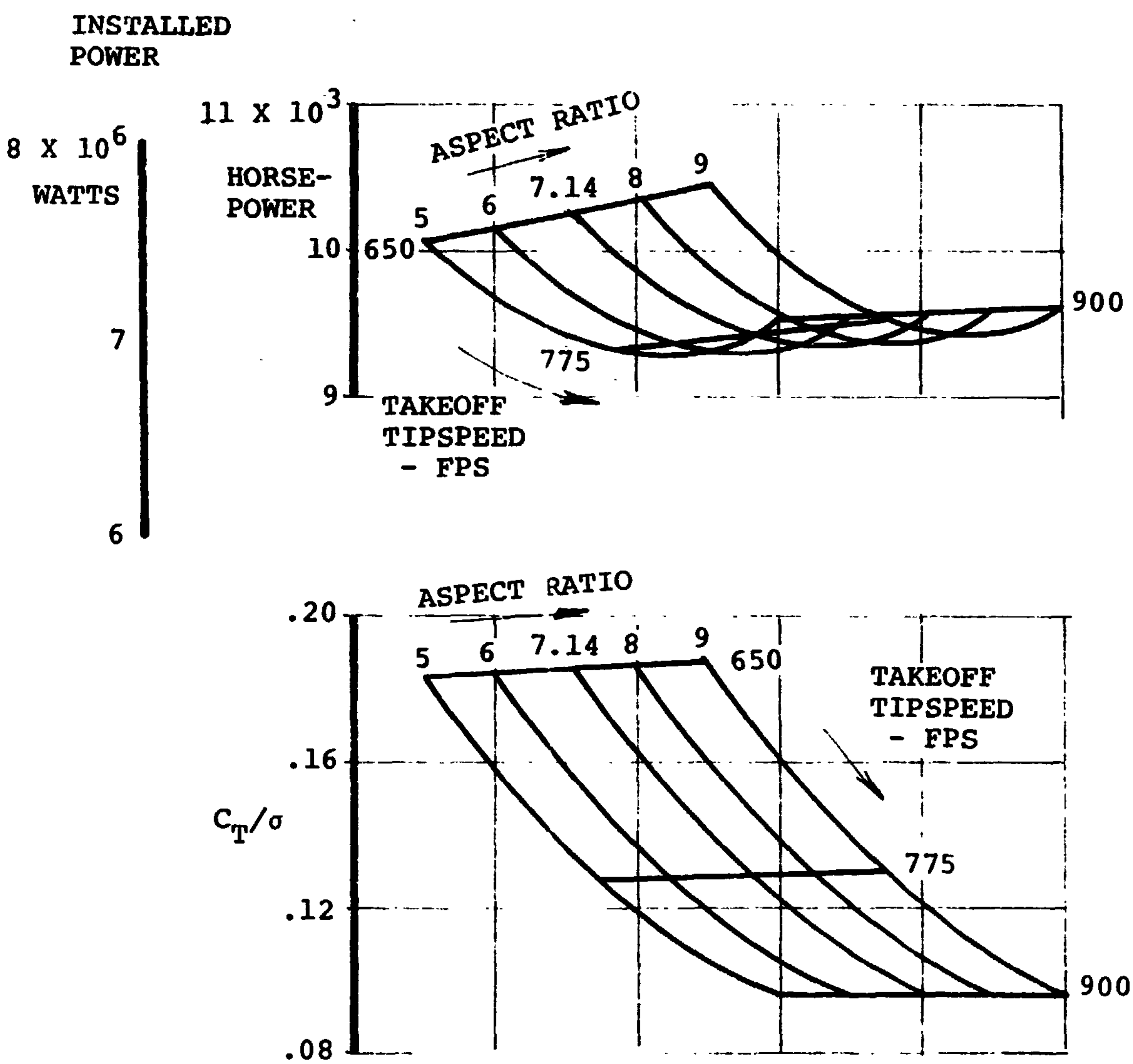
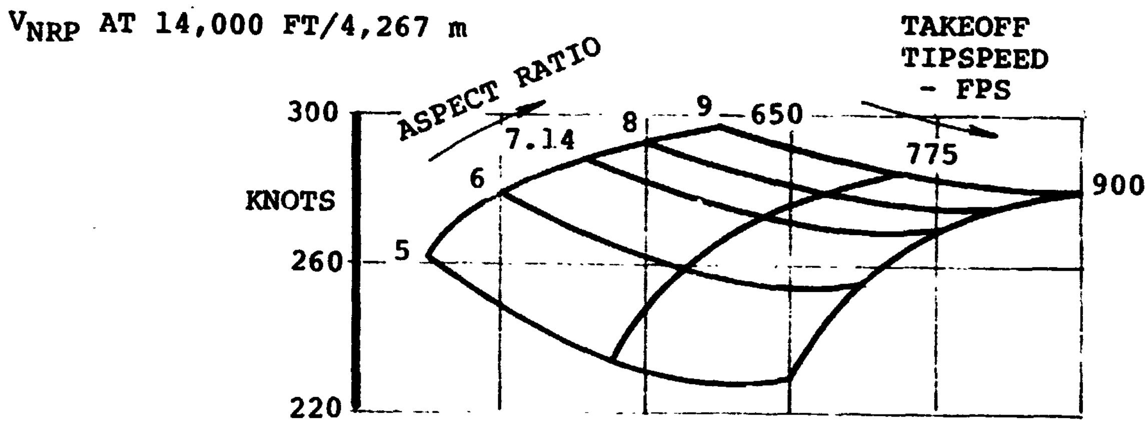


FIGURE A-18. VARIATION OF INSTALLED POWER AND ROTOR LOADING WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.105).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ROTOR DIAMETER = 44.4 FT/1,353 m
WING LOADING = 100 PSF/488 Kg/m
STATIC THRUST/WEIGHT = 0.88
ROTOR SOLIDITY = 0.105



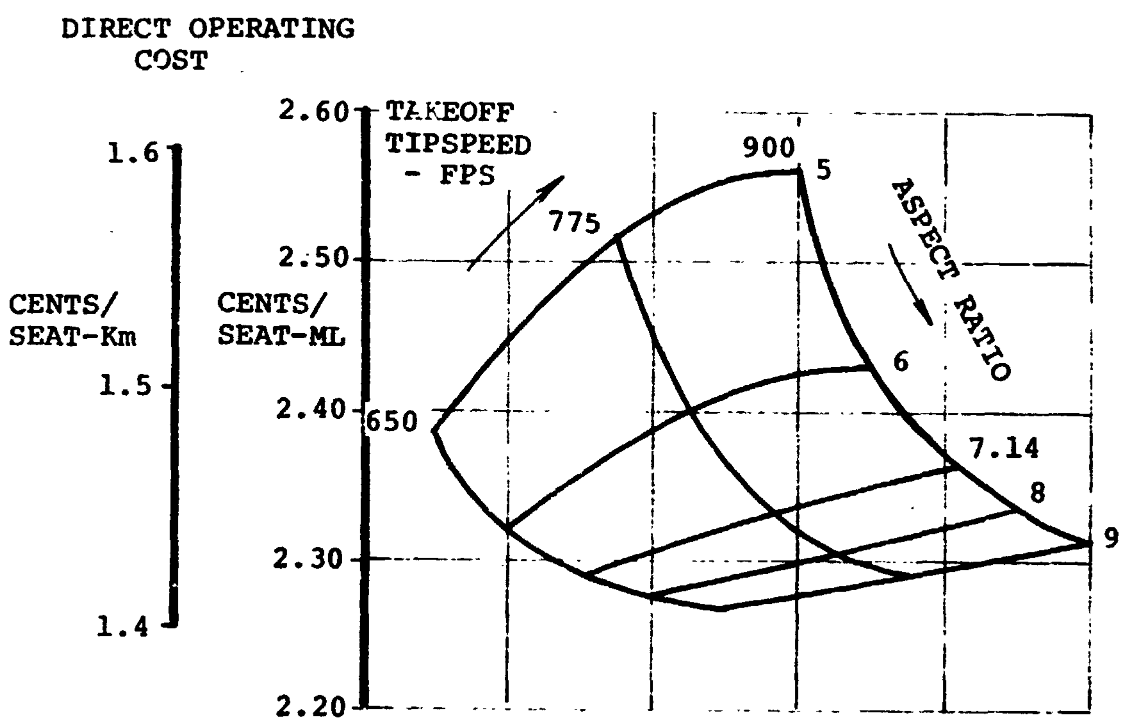


FIGURE A-19. VARIATION OF MAXIMUM CRUISE SPEED AND DIRECT OPERATING COST WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.105).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ROTOR DIAMETER = 44.4 FT/13.53 m
WING LOADING = 100 PSF/488 Kg/m^2
STATIC THRUST/WEIGHT = 0.88
ROTOR SOLIDITY = 0.12

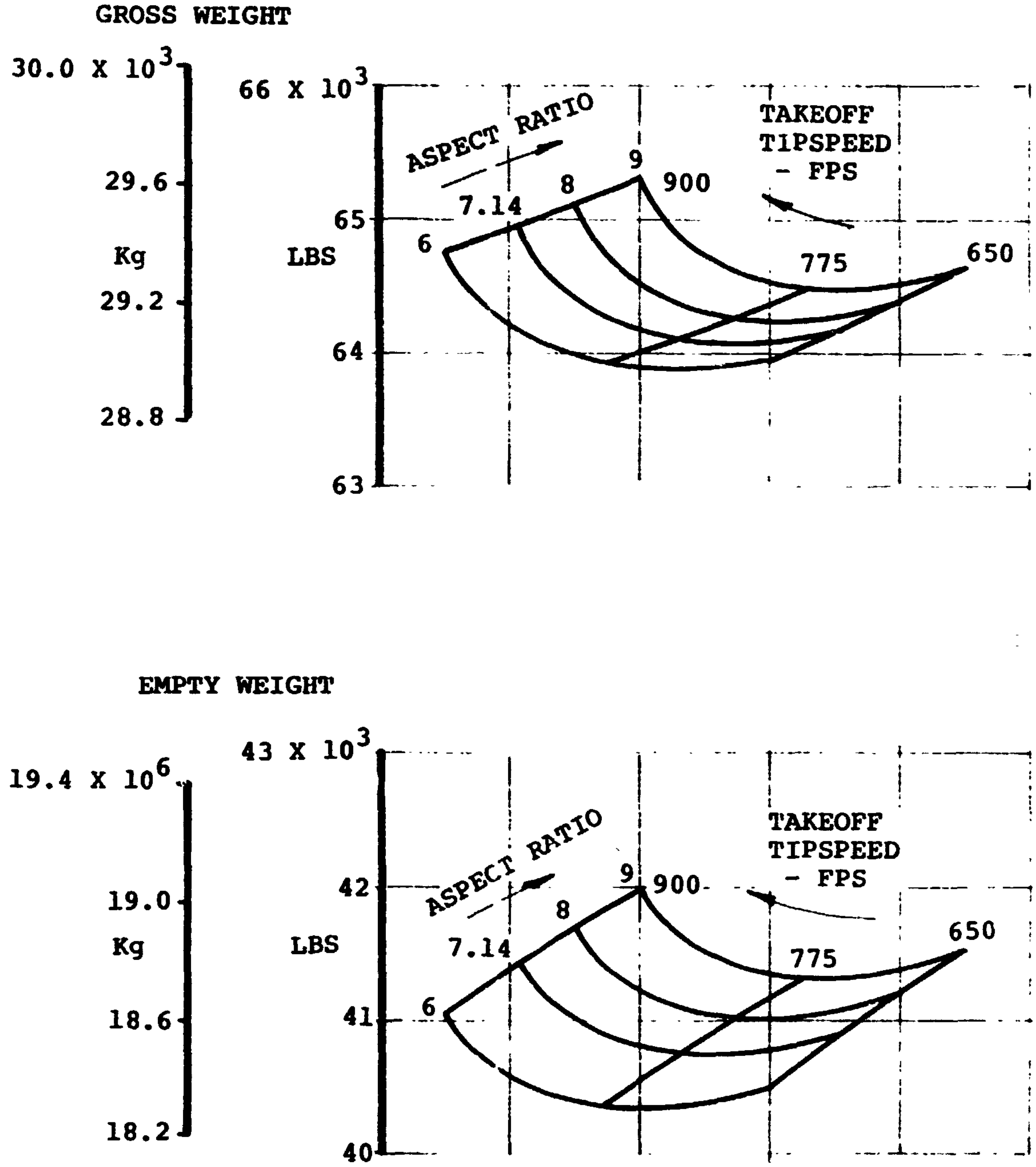


FIGURE A-20. VARIATION OF GROSS WEIGHT AND EMPTY WEIGHT WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.12).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ROTOR DIAMETER = 44.4 FT/13.53 m
WING LOADING = 100 PSF/488 Kg/m^2
STATIC THRUST/WEIGHT = 0.88
ROTOR SOLIDITY = 0.12

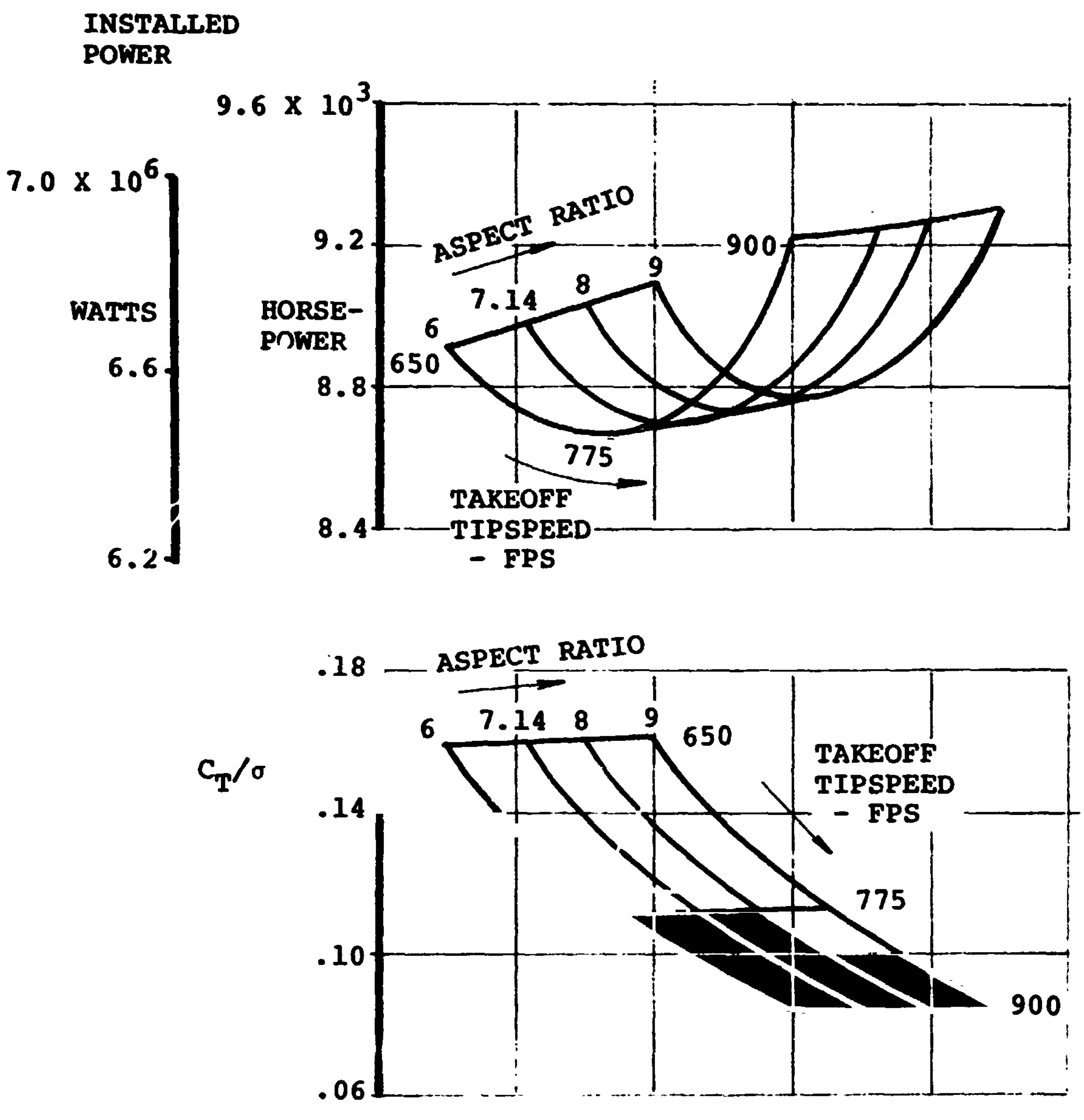


FIGURE A-21. VARIATION OF INSTALLED POWER AND ROTOR LOADING WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.12).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ROTOR DIAMETER	=	44.4 FT/13.53 m
WING LOADING	=	100 PSF/488 Kg/m^2
STATIC THRUST/WEIGHT	=	0.88
ROTOR SOLIDITY	=	0.09

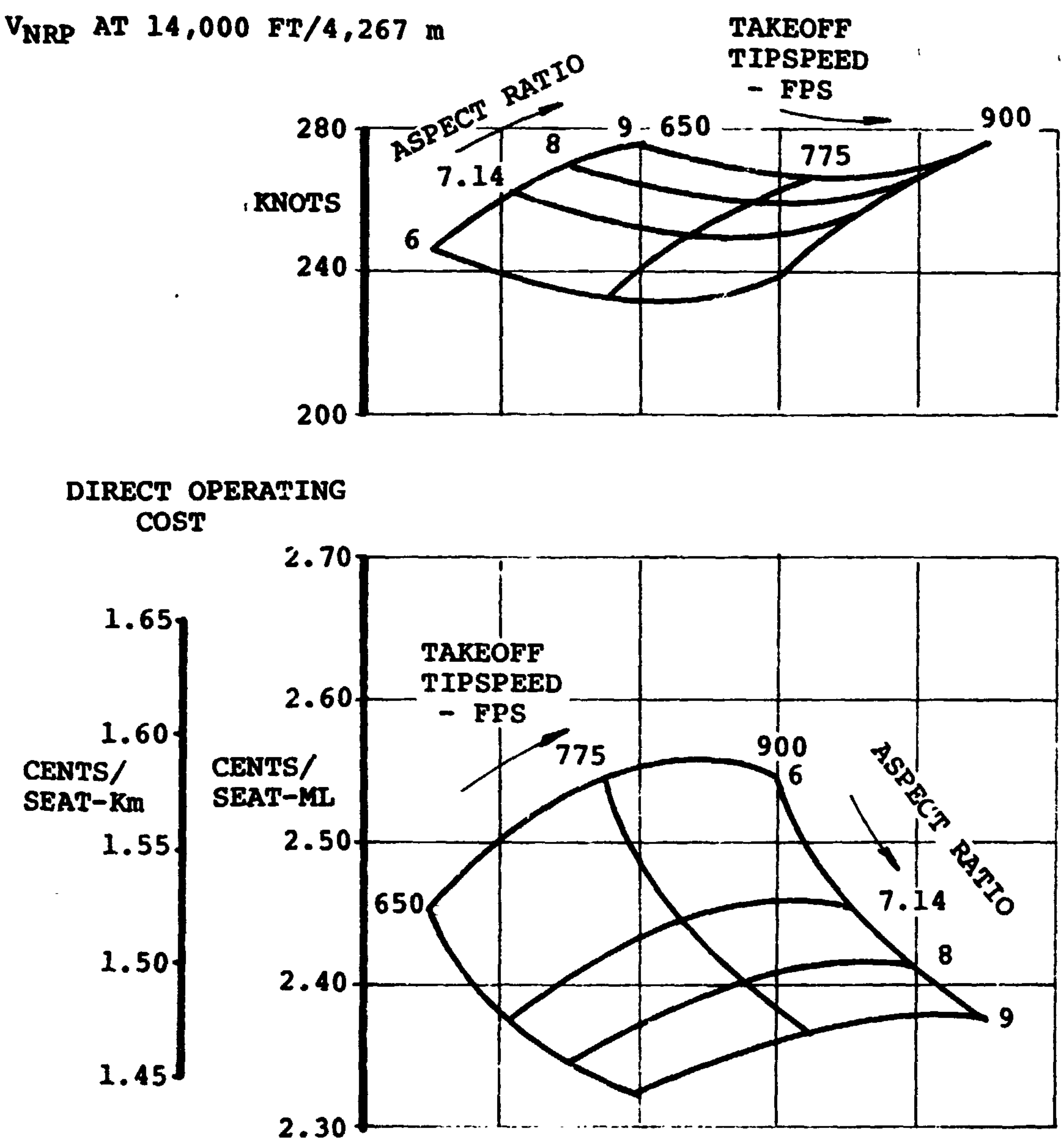


FIGURE A-22. VARIATION OF MAXIMUM CRUISE SPEED AND DIRECT OPERATING COST WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.12).

avoid aeroelastic instabilities associated with wing torsion and bending, weight penalties would become excessive. The direct operating cost and the appropriate tipspeeds were then plotted versus rotor solidity, Figure A23, and the minimum of this line was identified. Thus the aspect ratio, rotor tipspeed and rotor solidity were selected.

The rotor performance data used during the parametric studies was based on that of the rotor of the baseline VTOL tilt rotor described in Reference 1. The data were adapted for the second parametric study by scaling according to rotor solidity.

Having defined the rotor solidity, the rotor design was refined by investigating the effect of changing the twist of the blades. The twist was set so that the blades were not stalled at stations outboard of 10 percent of the blade radius when producing maximum static thrust at the start of the takeoff run. The resulting blade geometry is shown in Figure 3.45 in Section 3.4 of this report.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

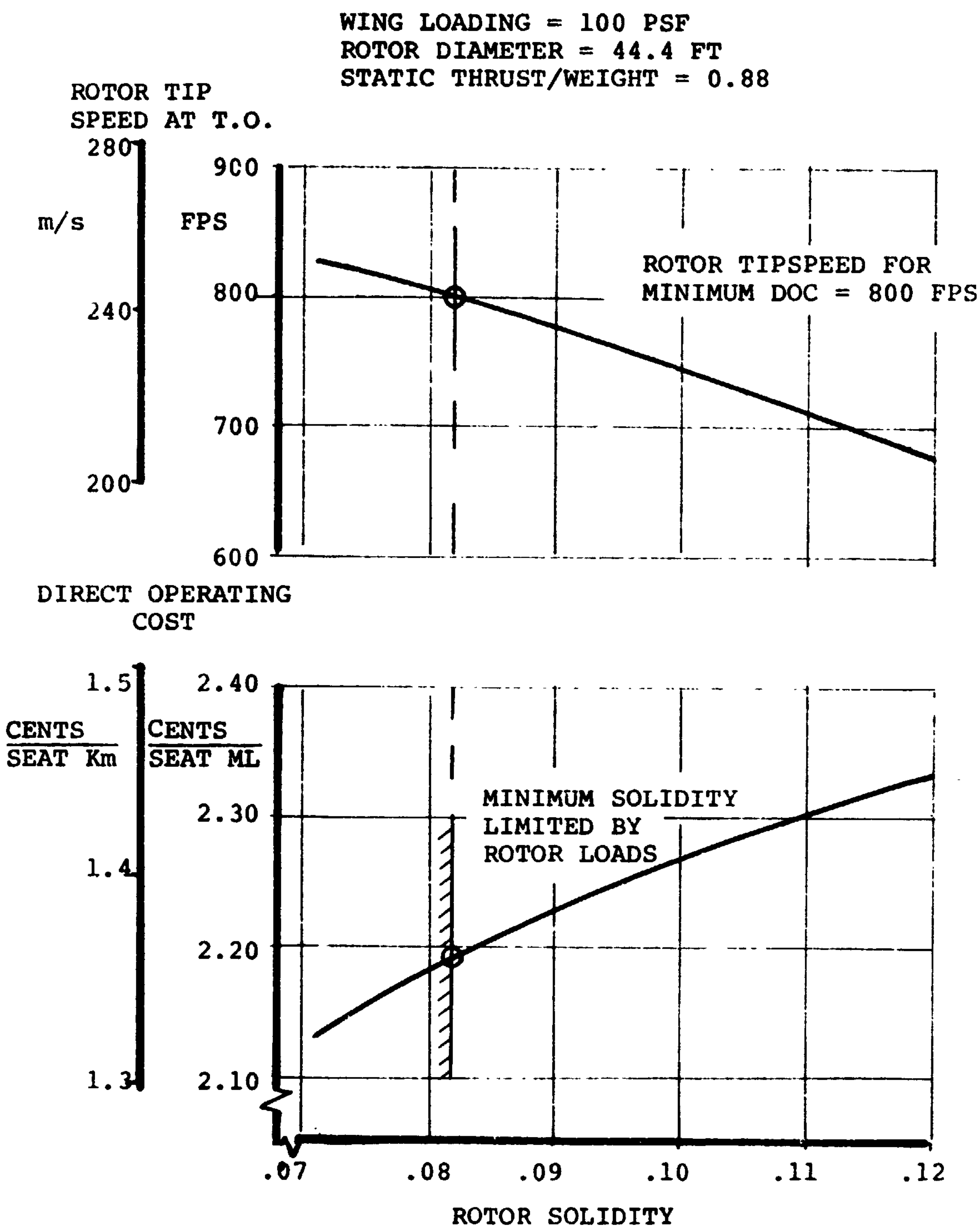


FIGURE A-23. DESIGN POINT SELECTION. VARIATION OF ROTOR TIPSPEED AND MINIMUM DIRECT OPERATING COST WITH ROTOR SOLIDITY.

APPENDIX B

COSTING METHOD

FLYAWAY COSTS

THE AIRFRAME COSTS WERE CALCULATED USING FACTORS OF \$90.00 AND \$110.00 PER POUND OF AIRFRAME. THE AIRFRAME WEIGHT WAS ARRIVED AT AS FOLLOWS:

AIRFRAME WEIGHT = WEIGHT EMPTY - $(W_R + W_{DS} + W_{EN} + W_{AV})$

WHERE:

W_R = WEIGHT OF ROTORS

W_{DS} = WEIGHT OF DRIVE SYSTEM

W_{EN} = WEIGHT OF ENGINES

W_{AV} = WEIGHT OF AVIONICS

IN THE EQUATIONS USED FOR CALCULATING AIRFRAME MAINTENANCE LABOR COSTS, WHICH USE AIRFRAME WEIGHT, THE WEIGHT OF AVIONICS WAS INCLUDED IN THE AIRFRAME SINCE THE AIA METHOD DOES NOT MAKE PROVISION FOR CALCULATING AVIONICS MAINTENANCE COST AS A SEPARATE ITEM.

OTHER MAJOR SYSTEMS COSTS WERE CALCULATED AS SHOWN BELOW:

COST OF DYNAMIC SYSTEM = \$80.00 $(W_{DR} + W_R)$

COST OF ALL ENGINES = E_N (\$280 $SHP^{.785}$)

WHERE:

E_N = NUMBER OF ENGINES

HP = STATIC SHP AT SL/STD FOR ONE ENGINE

COST OF AVIONICS = \$250,000.00

OPERATING COSTS

DIRECT OPERATING COSTS WERE DEVELOPED USING THE AEROSPACE INDUSTRIES ASSOCIATION'S (AIA) "STANDARD METHOD OF ESTIMATING DIRECT OPERATING COSTS OF TURBINE POWERED VTOL TRANSPORT AIRCRAFT" DATED 1968, MODIFIED AS DIRECTED BY NASA, AS FOLLOWS:

CREW COSTS

$\$/FH = .067\ (\text{GROSS WEIGHT}/1000) + 134$

ENGINE MAINTENANCE COSTS

LABOR $\$/FH = 0.65\ (\text{AIA COSTS})$

MATERIAL $\$/FH = 0.65\ (\text{AIA COSTS})$

MAINTENANCE BURDEN

$$\$/FH = 1.5\ (DL_{AF} + DL_{EN} + DL_{DS})$$

WHERE:

DL_{AF} = DIRECT LABOR COSTS FOR AIRFRAME MAINTENANCE

DL_{EN} = DIRECT LABOR COSTS FOR ENGINE MAINTENANCE

DL_{DS} = DIRECT LABOR COSTS FOR DYNAMIC SYSTEM MAINTENANCE

TABLE B1 LISTS FACTORS USED IN CALCULATING THE DIRECT OPERATING COSTS.

FACTOR	UNITS	VALUE	SOURCE
FUEL	$/LB	0.02	NASA
OIL	$/LB	1.24	NASA
NONREVENUE FACTOR	%	2.00	AIA
LABOR RATE	$/HR	6.00	NASA
ENGINE TBO	HRS	4500	NASA
DYNAMIC SYSTEM TBO	HRS	3000	NASA
DEPRECIATION PERIOD	YRS	12	NASA
SPARES			
AIRFRAME	%	8	AIA
ENGINES	%	40	AIA
DYNAMIC SYSTEM	%	25	AIA
UTILIZATION	HRS/YR	2500 & 3500	NASA
INSURANCE RATE	%	2.00	NASA
AIRWAYS DISTANCE FACTOR	ND	1.00	NASA

TABLE B1. COMPONENTS OF DIRECT OPERATING COST.

APPENDIX C

WEIGHTS

The weight of this configuration is summarized in Table C1.

Weight trade studies leading to the selection of this configuration was accomplished using the computerized weight prediction program. VASCOMP sized and weighed the tilt rotor configurations. This sizing program includes a weights subroutine which automatically computes subsystem weight changes resulting from variations in the configuration size, flight envelope, payload, etc. They provide a consistent method for rapidly estimating the aircraft's operational weight empty and gross weight. The program divides the weight empty into three groups: propulsion, structures and flight controls. Weight trends are programmed for each group which compute their respective weights. These are then combined with weight input values of fixed useful load, fixed equipment and payload to determine the weight of the fuel available for a given gross weight and payload. The weight input values are determined from specific mission requirements and/or specified equipment lists. A flow chart for the weight trend subroutine is shown in Figure C1.

The weight trends were developed at Vertol from statistical and semianalytical data of existing aircraft. They combine geometric, design and structural parameters into an accurate weight prediction tool. Examples of the weight trends for

	KILOGRAMS	POUNDS
WING	2397.7	5286
ROTOR	1877.4	4139
TAIL	520.3	1147
SURFACES	520.3	1147
ROTOR		
BODY	3889.5	8575
BASIC		
SECONDARY		
ALIGHTING GEAR GROUP	1242.8	2740
ENGINE SECTION	288.9	637
PROPULSION GROUP	3000.9	6616
ENGINE INST'L	796.1	1755
EXHAUST SYSTEM *		
COOLING		
CONTROLS *		
STARTING *		
PROPELLER INST'L	*246.8	*544
LUBRICATING		
FUEL	76.2	168
DRIVE	1881.9	4149
FLIGHT CONTROLS	1567.2	3455
AUX. POWER PLANT	288.5	636
INSTRUMENTS	191.9	423
HYDR. & PNEUMATIC	308.4	680
ELECTRICAL GROUP	423.7	934
AVIONICS GROUP	293.9	648
ARMAMENT GROUP	3273.6	7217
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	20431.0	45043
CREW	299.4	660
TRAPPED LIQUIDS	52.2	115
ENGINE OIL	59.9	132
CARGO ACCOMMODATIONS	68.0	150
EMERGENCY EQUIPMENT	23.6	52
PASSNEGER ACCOMMO	415.5	916
PASSENGERS (100)	8164.6	18000
FUEL	1553.5	3425
GROSS WEIGHT	31067.7	6849[illegible]

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TABLE C-1. STOL TILT ROTOR WEIGHT SUMMARY.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

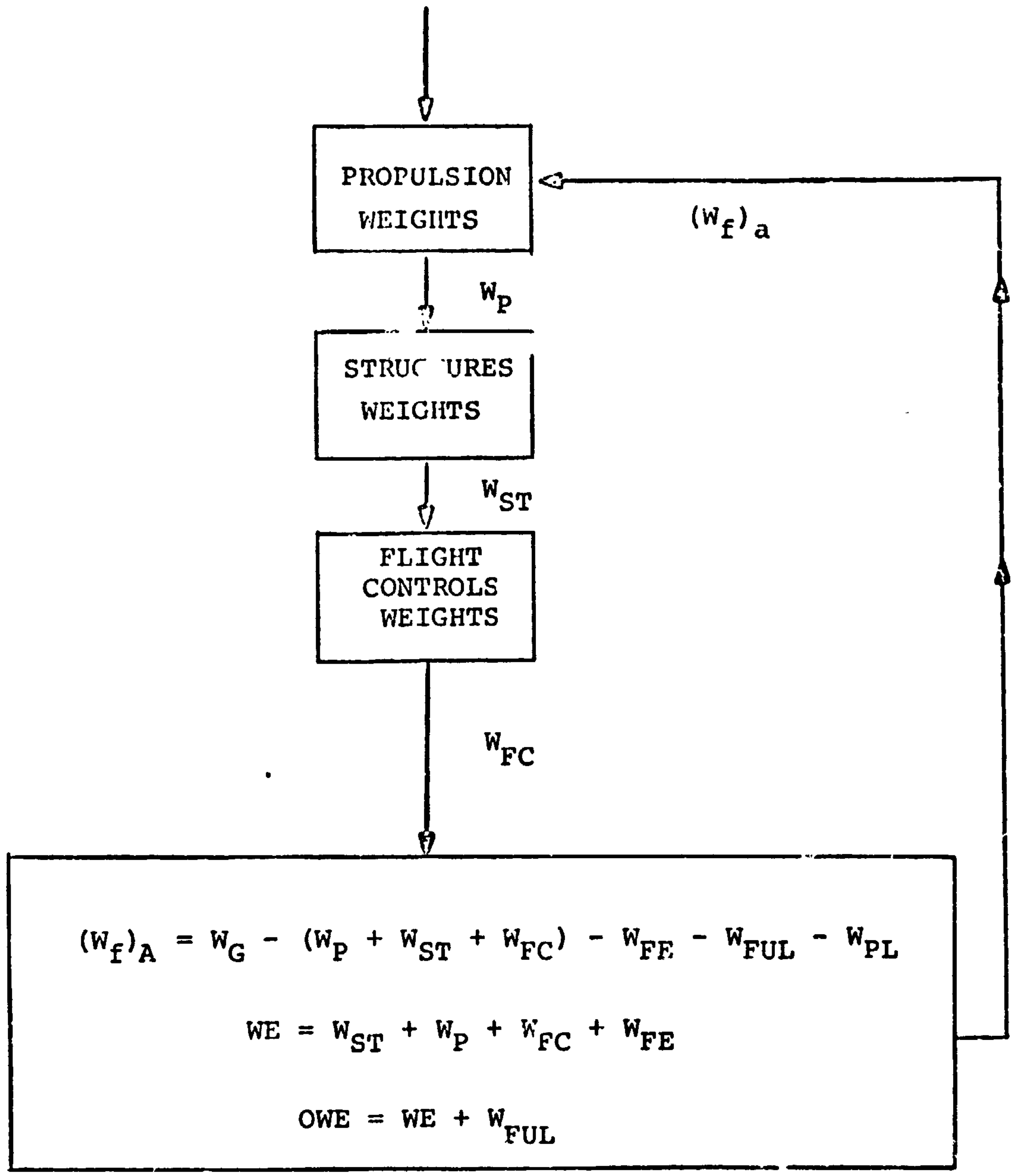


FIGURE C-1. WEIGHT TRENDS SUBROUTINE, FLOW CHART.

the major weight groups are presented in Figures C2 through C8.

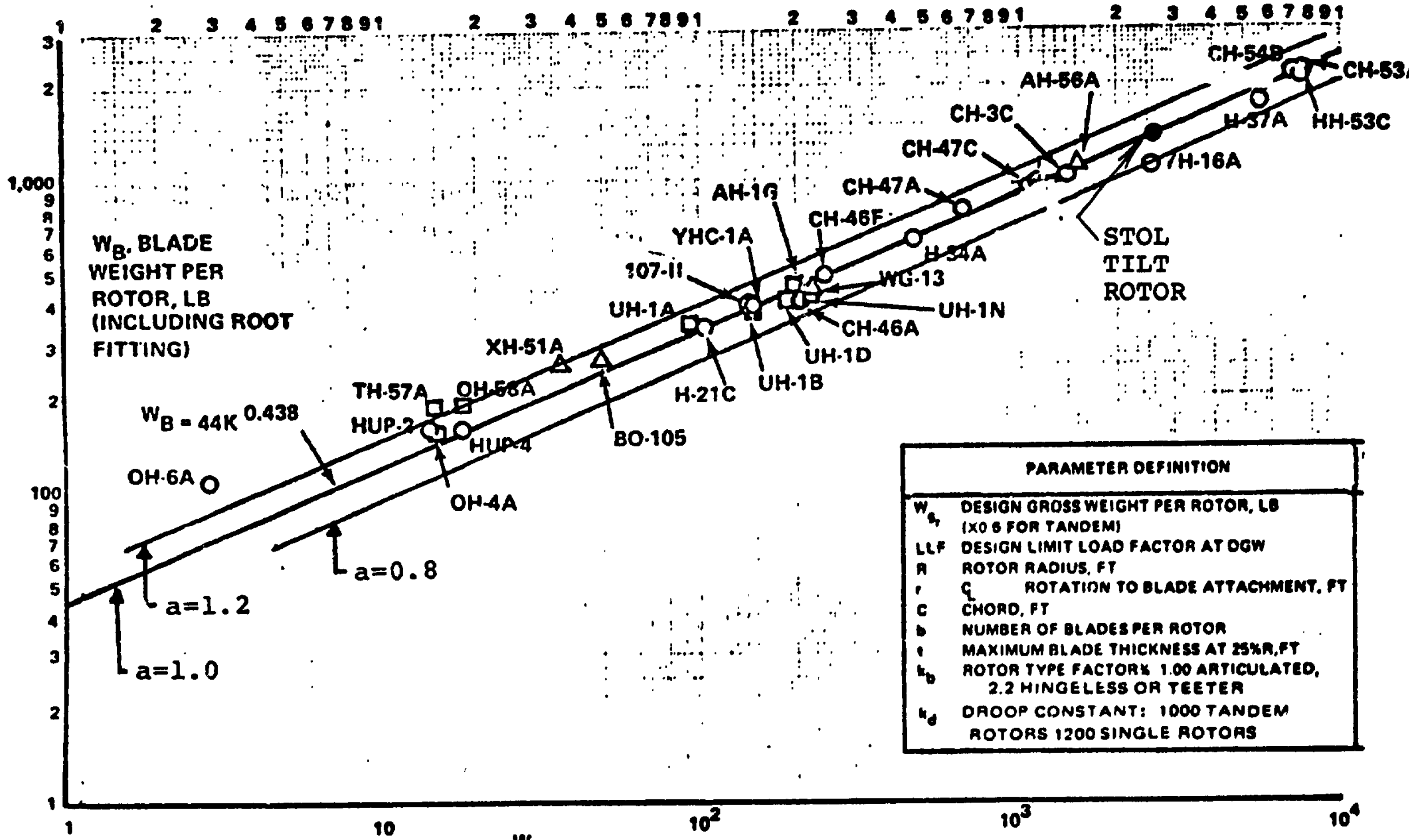
The trends include sufficient design paramteres to account for the major design features associated with each of the study configurations. In order to provide comparisons of the design points with the statistical data the assumptions for weight reduction due to advanced composite materials have been removed in Figures C2 through C8.

The flight control trend, for example, is divided into six groups, which ensures that a weight allowance is included for all the major control items and special features. It includes:

- o Cockpit controls
- o Rotor controls
- o Fixed-wing controls (includes type and number of control surfaces)
- o Systems and hydraulics
- o Tilt mechanism (includes tilting nacelle or wing mechanism)
- o SAS and mixing (integrates airplane and helicopter controls).

The rotor group weight trends, Figures C2 and C3, include parameters which considers number of rotors and blades, type of system (rigid, articulated, teetering, etc.) tipspeed, etc.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT



$$K = \frac{W_{g_R}}{10^4}\ LLF\ \frac{R^2}{100}\ \frac{(R-r)}{10}\ b\,C\,k_b\ \left[\frac{R^{1.6}}{k_d\,t}\right]$$

(USE [TERM] IF GREATER THAN 1.00

FIGURE C-2. ROTOR BLADE WEIGHT TREND.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

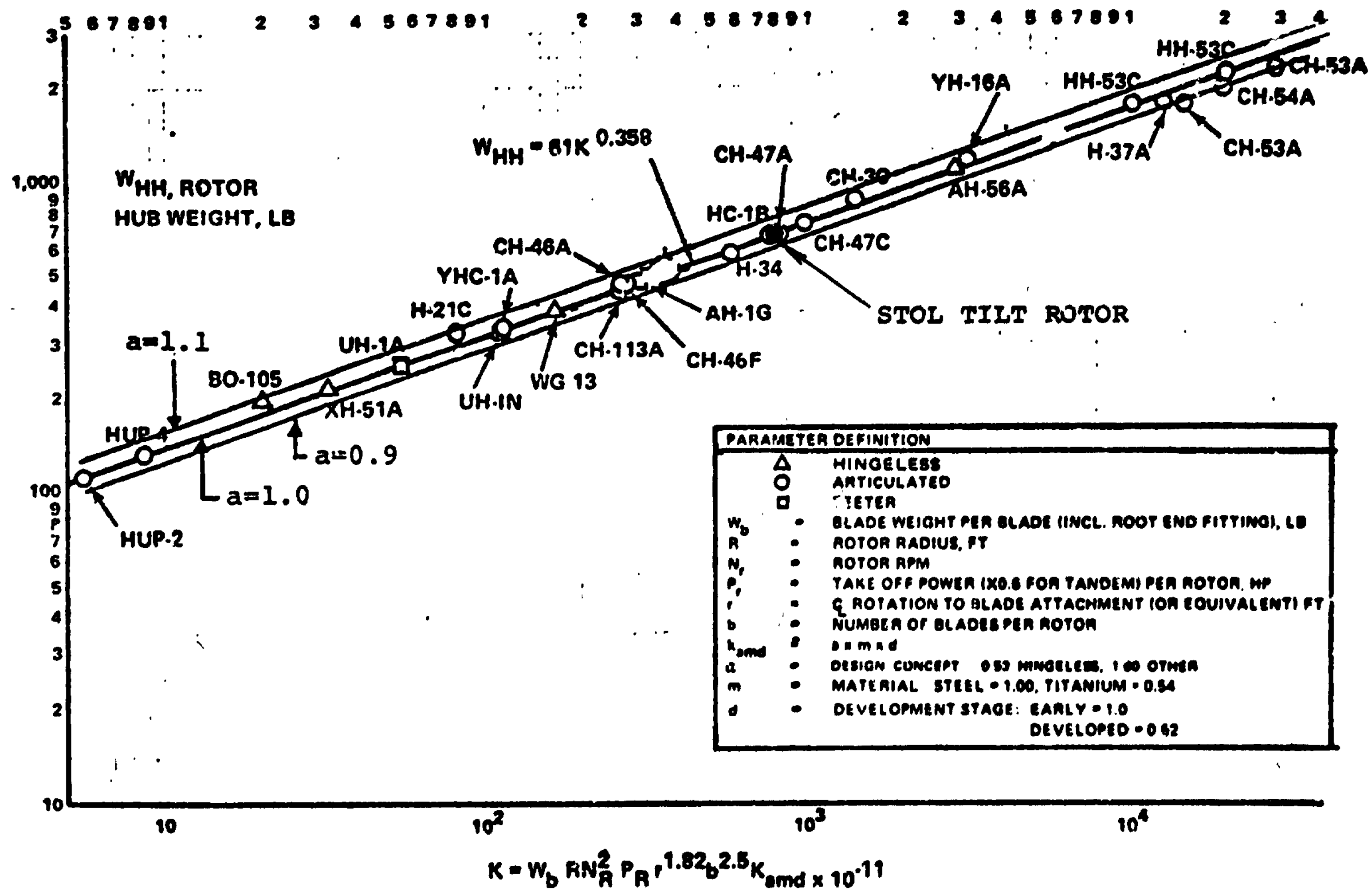


FIGURE C-3. ROTOR HUB AND HINGE WEIGHT TREND.

Weight trends shown in Figure C4 were used to predict the weight of configurations designed around airplane type fuselage structure as in the case of the tilt rotor aircraft.

Drive system weight is determined by multiplying the constant (K) by a simple torque expression as indicated by the overall drive system weight trend shown in Figure C5. Determination of the constant is the end result of a detailed box-by-box analysis of the drive system configuration. The semi-analytical method calculates the weight of each gear set. It includes the effects of Hertz stress, gear ratio, bearing support, number of gears in a stage, and external or structural supports. The drive shafting weight is determined independent of the box weight and includes parameters which consider the number of shaft sections and transmitted torque.

Wing and tail weight trends are shown in Figures C6 to C8. The trend constants "K" are primary inputs to the computer programs. Selection of the constants depend on the type of aircraft being configured - helicopter, compound, tilt rotor, etc., material, and level of technology. Peculiar design loads and stiffness requirements and special design features such as folding rotor blades, tilting nacelles, shrouded tail fans, etc. are studied individually and inputed as a variation of the constant or included as a direct weight input in the incremental group weight section of the VASCOMP weights input form.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

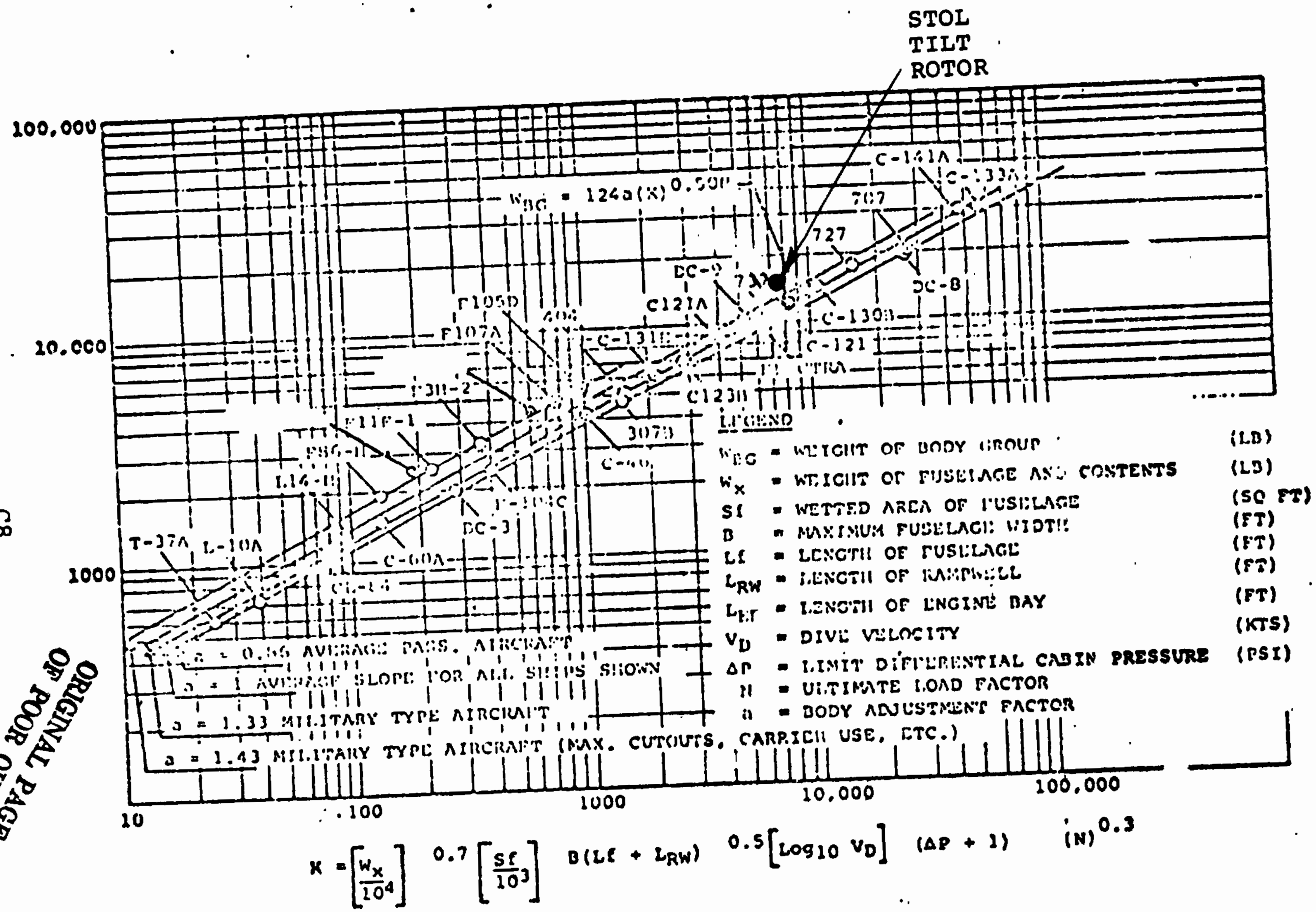


FIGURE C-4. AIRPLANE BODY GROUP WEIGHT TREND.

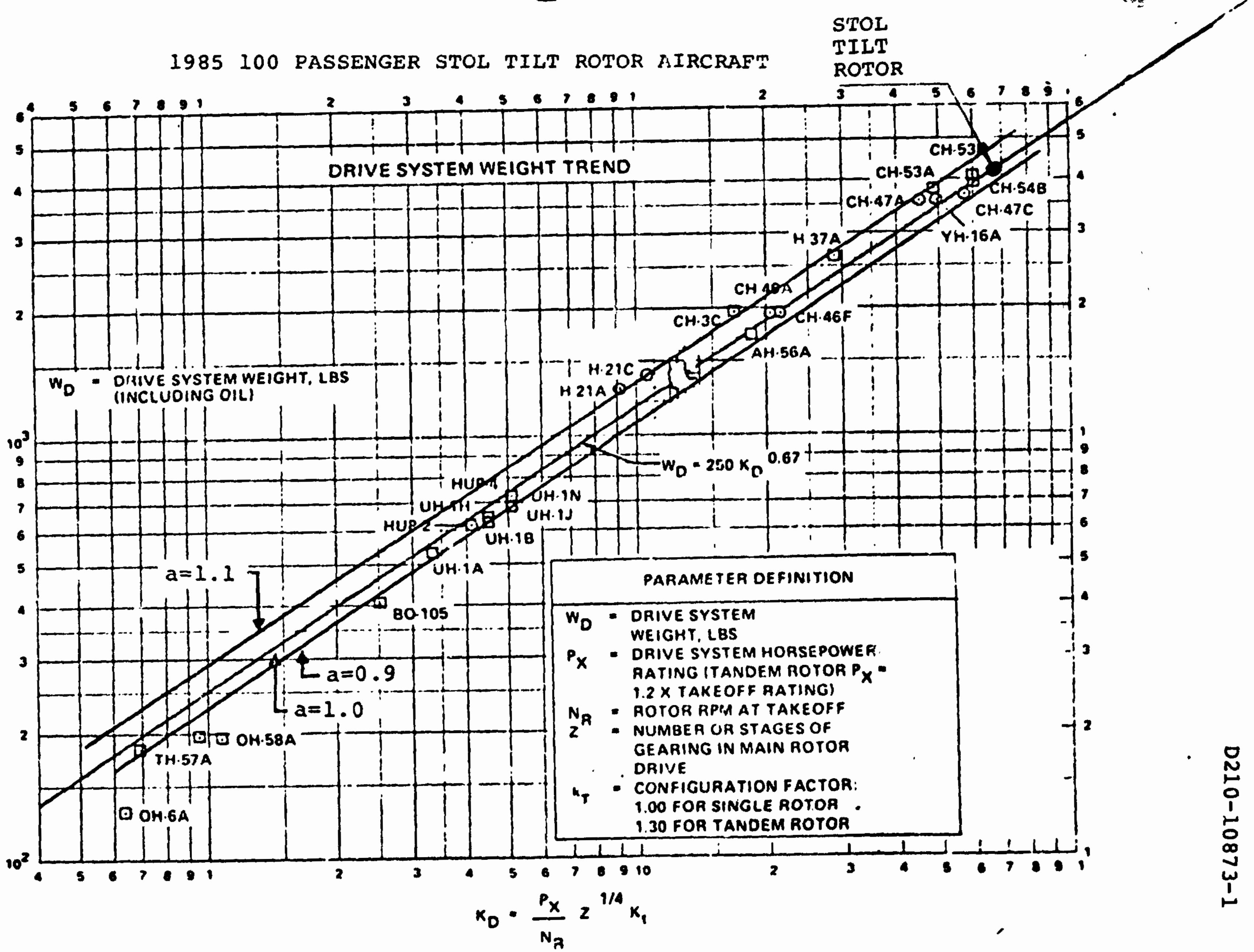


FIGURE C-5. DRIVE SYSTEM WEIGHT TREND - PRIMARY AND AUXILIARY.

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1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

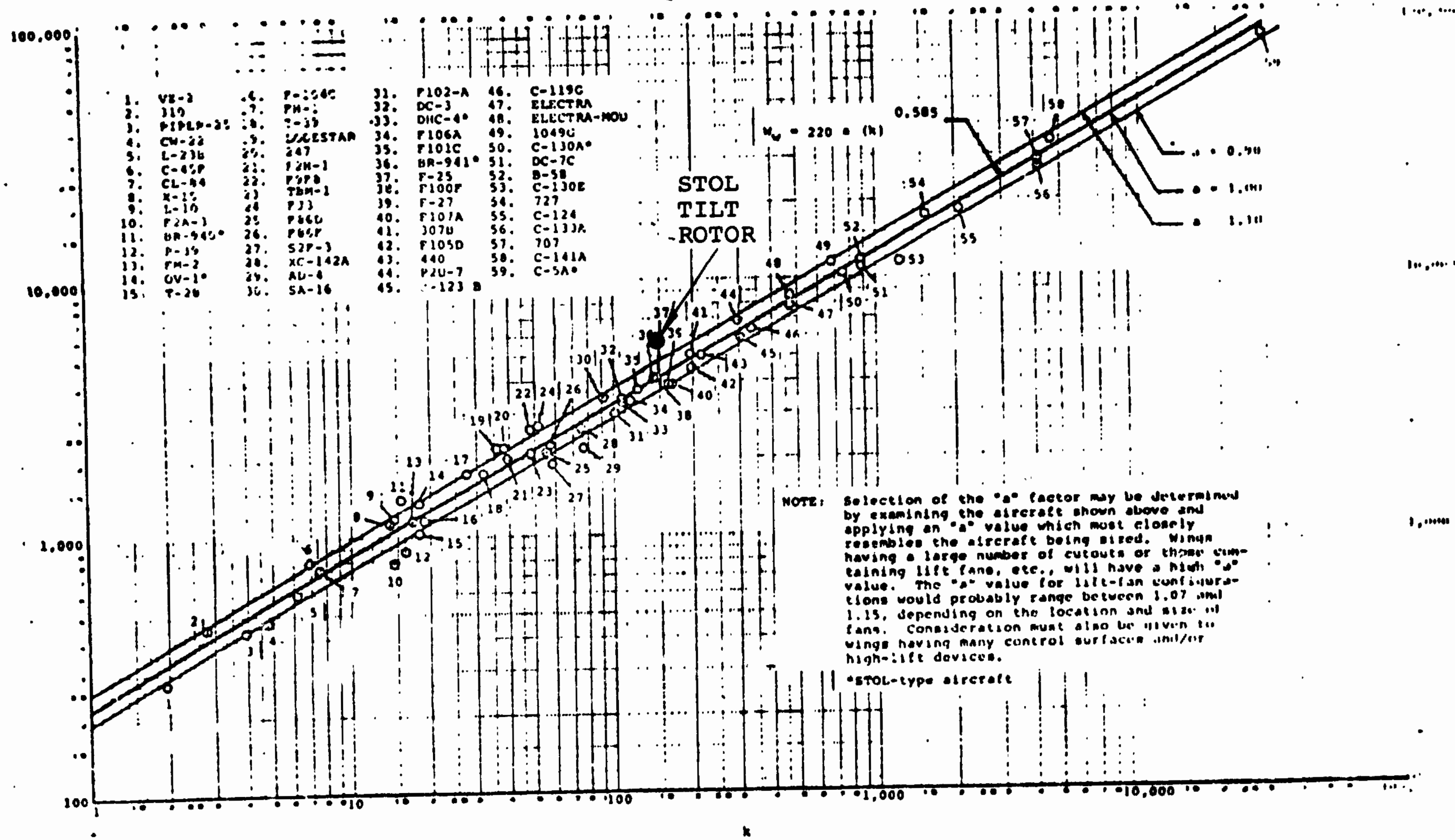

FIGURE C-6. WING WEIGHT TREND.

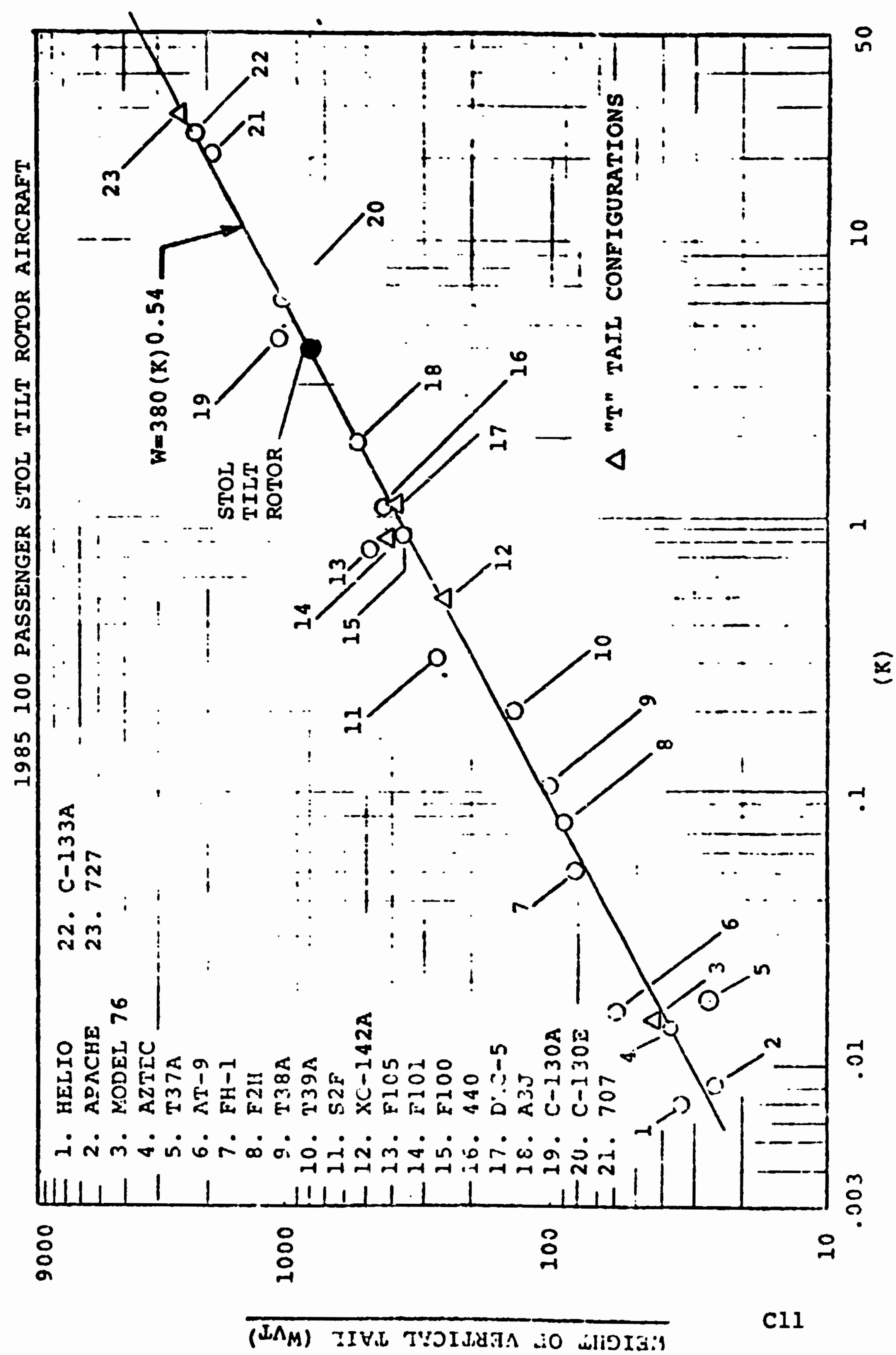

FIGURE C-7. VERTICAL TAIL WEIGHT TREND.

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1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

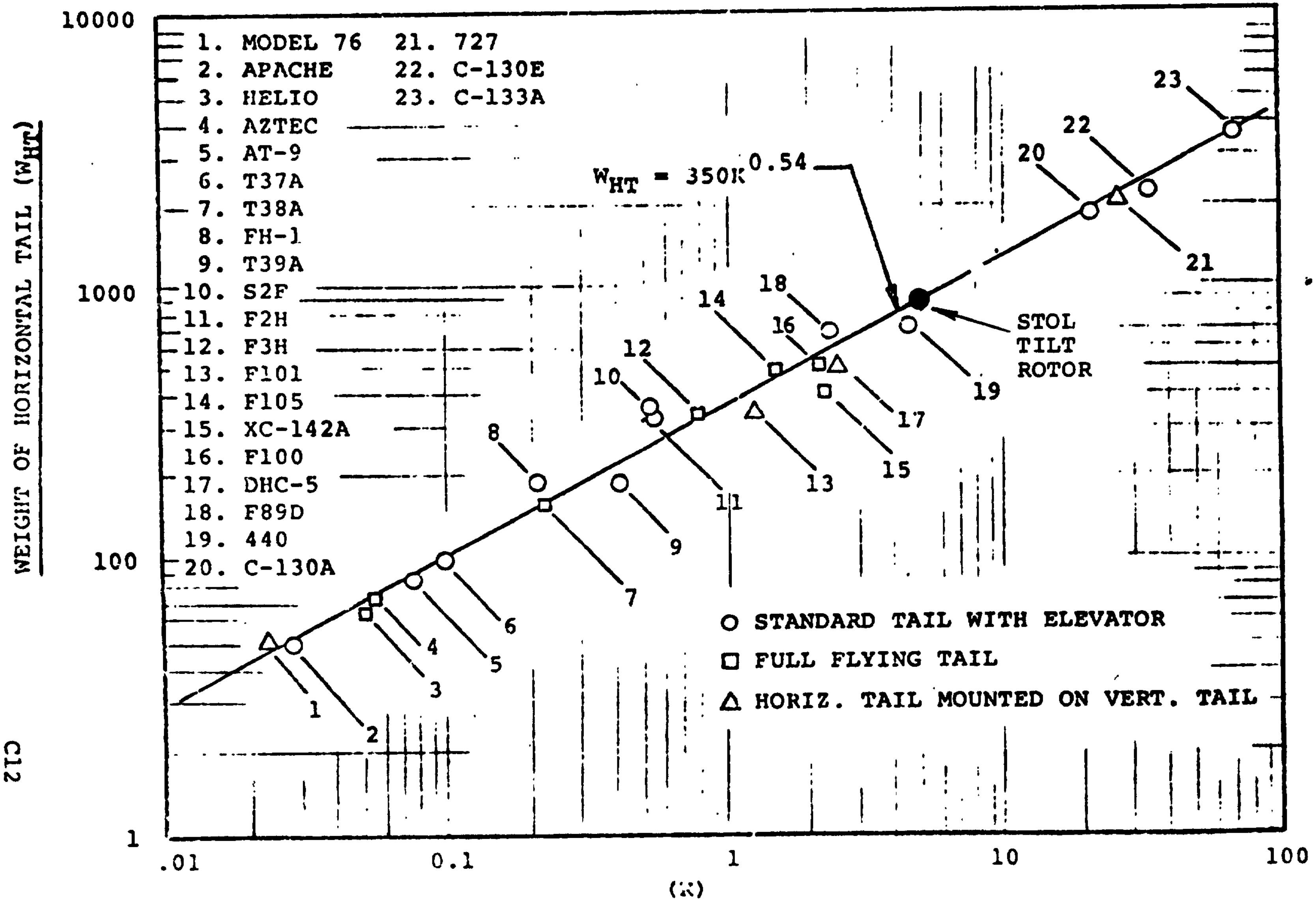

FIGURE C-8. HORIZONTAL TAIL WEIGHT TREND

The detailed design of the Model 222 Tilt Rotor Research Aircraft provide the data bank used for selecting the weight constants. In order to show substantiation of these weights, the tilt rotor configuration was evaluated, using refined prediction methods and parameters. These substantiating calculations are included on Pages C29 through C40.

The computerized weight prediction programs were based on the assumptions discussed below.

Limit Load Factor

The limit load factors at mission gross weights are:

Tilt rotor, 2.5 from FAR, Part 25.

$$L.L.F. = 2.1 + \frac{24{,}000}{W + 10{,}000}$$

"L.L.F. SHALL BE NO LOWER THAN 2.5, BUT NEED NOT BE HIGHER THAN 3.8".

W = MAXIMUM DESIGN GROSS WEIGHT

Advance Materials for 1985 Operational Time Period

From the Study Guidelines, Paragraph 4.5, the following is quoted;"The Contractor shall assume that the airframe structural weight will be reduced approximately 25% by the use of composite materials". Boeing Vertol has chosen to distribute this weight as follows:

	REDUCTION (%)
WING	30.2
TAIL	30.2
BODY	30.2
LANDING GEAR	0
ENGINE SECTION	30.2
EQUIVALENT TOTAL	25.0

Wing

The wing weight of the tilt rotor was based on that of the Boeing Vertol Model 222 tilt rotor aircraft. This wing was designed by Grumman Aircraft Company under direction of Boeing Vertol and the weights calculated in detail. Adjustments have been made for advanced materials. A comparison of the design point wing weight, with no composite material assumptions, with the weight trend curves is shown in Figure C6.

Rotor

The rotor has titanium hubs and root end fittings and fiberglass blades. The Model 222 rotor is used as a basis for the tilt rotor with adjustments made for titanium hub and root end fittings in lieu of steel.

Tail Surfaces

Tail surface weights were based on trends using statistical data from similar aircraft. Adjustments have been made for advanced materials.

Body

Weights of the bodies were based on trends using statistical data of other aircraft. Adjustments have been made for advanced materials. The body is pressurized for 5,000 feet at 14,000 feet.

Alighting Gear

The alighting gear is retractable, designed for a sink speed of 5 feet per second. A value of 4% gross weight has been selected.

Engine Section

The engine section was based on that of the Model 222, 52% of the engine weight and has been adjusted for the use of advanced materials.

Engines

The engine weights were based on rubberized LTC-4V-1 engines at 0.1575 pounds per horsepower.

Engine Installation

Engine installation consists of exhaust systems, propeller spinners, engine controls, starting system, and engine lubrication system. The input values for the computers are in terms of engine weight, 31% of the engine weight based on Model 222.

Fuel System

The fuel system weights for the tilt rotor were based on the Boeing 737-200. Tanks are in the wing.

Weight inputs in the computer are in the form of pounds per pound of fuel.

Drive System

The tilt rotor drive system weights were based on that of the Vertol Model 222.

Flight Controls

The flight control weights were based on Vertol Model 222, and were reduced for fly-by-wire systems. The reductions as applied to these systems are:

Cockpit	29%
Rotor Upper Controls	0
Rotor System Controls	20%
Airplane Type Controls	20%
SAS	0
POD Tilting Mechanism	13%

Fixed Equipment

The fixed equipment weights were based primarily on the Boeing 737-200 adjusted in some areas for weights quoted in the "Study Guidelines". Table C2 summarizes those used in these studies.

APU, Instruments, Electronics and Electrical

Paragraph 4.9 of the "Study Guidelines" quotes of weight of 1,200 pounds for these items, not including electrical generation. In comparing this to the Boeing 737-200, Boeing Vertol has assumed that this 1,200 pounds is an uninstalled weight.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

	737-200 88 PASSENGERS		100 PASSENGERS	
	Kg	(Lbs)	Kg	(Lbs)
APU	139.7	308	544.2	1200*
Instruments	146.5	323		
Electronics	127.0	280		
Electrical	322.9	712		
Installation for Above	597.3	1317	440.8	972
Electrical Generation	167.3	369	167.3	369
Hydraulics & Pneumatics	391.8	864	308.4	680
Flight Deck Accommodations	266.2	587	257.6	568
Passenger Accommodations	2829.5	6239	2737.9	6037
Cargo Accommodations	278.0	613	145.1	320
Emergency Accommodations	164.6	363	132.4	292
Air Conditioning	539.7	1190	612.2	1350
Anti-Icing	96.1	212	254.0	560
TOTAL FIXED EQUIPMENT	6066.6	13,377	5599.4	12,348

*Quoted From Study Guideline

TABLE C-2 . FIXED EQUIPMENT WEIGHTS

WEIGHT

BOEING 737-200	TOTAL	EQUIPMENT	INSTALLATION
APU	830	308	522
INSTRUMENTS	552	323	229
ELECTRONICS	846	280	566
ELECTRICAL	712	712	---
TOTALS	2,940	1,623	1,317

Using the above, it can be determined that the installation weight is 81% of the uninstalled weight.

$$\frac{1317}{1623} = 0.81$$

By applying this factor to the 1,200 pounds, the installation weight is 972 pounds. This and the electrical generation weights are shown separately in Table C2.

A growth factor (assuming constant performance and strength) has been established at 2.1. The curve in Figure 3.22 (in Section 3.2 of this report) shows the weight growth effect of this aircraft. If the 972 pounds of installation weight were not included the gross weight would decrease from 68,493 pounds to 66,452 pounds.

Hydraulics and Pneumatics

The hydraulics and pneumatics weights were established as 680 pounds based on the Boeing 737-200.

Furnishings and Equipment

Furnishings and equipment consist of Flight Deck Accommodations, Passenger Accommodations, Cargo Accommodations and Emergency

Equipment, and are listed in Tables C3 through C6. They are based primarily on those of the Boeing 737-200, adjusted in certain areas to agree with weights quoted in the "Study Guidelines".

Air Conditioning

The air conditioning system, including pressurization is based on 13.5 pounds perpassenger.

Anti-Icing

Anti-icing weights are based on:

737-200 = 0.25% Gross Weight

CH-46 = 0.50% Gross Weight

0.75%

Useful Load

The useful load weights (not including fuel) are shown in Table C7. They are based primarily on the Boeing 737-200, adjusted in certain areas for weights quoted in the "Study Guidelines".

Weight Substantiation

The weights leading to the selection of the configurations shown in this study were derived by using the computerized VASCOMP sizing and weight prediction program, Reference 2.

Substantiation of these weights using weight prediction methods developed and improved by Boeing Vertol Weights Unit, are presented in this section of the report. The group weight for each of the major components is shown (adjusted to account

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

	737-200 88 PASSENGERS		100 PASSENGERS	
	Kg	(Lbs)	Kg	(Lbs)
Seats and Belts	58.5	129	49.9	110
Pilot-Copilot 40* X 2				80
Observer 30 X 1				30
Instrument Boards	47.6	105	47.6	105
Control Stands	31.7	70	31.7	70
Sound-Proofing	44.4	98	44.4	98
Lining	28.1	62	28.1	62
Manuals	2.3	5	2.3	5
Windshield Wiper	4.1	9	4.1	9
Rain Repellent System	10.0	22	10.0	22
Misc. Equipment	(10.6)	(23)	(10.6)	(23)
Sun Visor	2.3	5	2.3	5
Mirror	.5	1	.5	1
Foot Rests	.9	2	.9	2
Waste Containers	1.4	3	1.4	3
Ash Trays & Cup Holders	1.4	3	1.4	3
Stowage and Holders	3.2	7	3.2	7
Overhead Drain Tube	.9	2	.9	2
Lighting	15.4	34	15.4	34
Wiring, Etc.	13.6	30	13.6	30
(*Quote From Study Outline)				
TOTAL FLIGHT DECK ACCOMMODATIONS	266.3	587	257.7	568

TABLE C-3 . FLIGHT DECK ACCOMMODATIONS.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

	737-200 88 PASSENGERS		100 PASSENGERS	
	Kg	(Lbs)	Kg	(Lbs)
Seats and Belts	(1036.3)	(2285)	(740.1)	(1632)
Passengers 16 Lbs Each*	1010.0	2227	725.6	1600*
Attendants 16 Lbs Each*	26.3	58	14.5	32*
Lavatories 300 Lbs Each*	205.4	453	272.1	600*
Stowage	(206.8)	(456)	(233.5)	(515)
Overhead	138.3	305	158.7	350
Magazine	3.6	8	3.6	8
Coat Racks	33.6	74	36.3	80
Food Trays	4.5	10	4.5	10
Under Seat	26.8	59	30.4	67
Soundproofing	311.1	686	353.7	780
Lining	448.5	989	510.2	1125
Floor Covering	134.2	296	154.2	340
Beverage Service	192.3	424	218.6	482
Attendants Panels	9.5	21	9.1	20
Partitions	40.4	89	40.8	90
Window Shades	24.9	55	27.2	60
Lowered Ceiling	59.0	130	----	----
Wash and Drinking Facilities	30.4	67	30.4	67
Signs and Markings	.9	2	.9	2
Lighting	110.2	243	127.0	280
Safety Straps	1.8	4	1.8	4
Finishing Panels	17.7	39	18.1	40
TOTAL PASSENGER ACCOMMODATIONS	2829.4	6,239	2737.7	6,037

*Quote From Study Outline

TABLE C-4. PASSENGER ACCOMMODATIONS.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

	737-200 88 PASSENGERS		100 PASSENGERS	
	Kg	(Lbs)	Kg	(Lbs)
Baggage Compartments	-----	-----	36.3	80
Insulation	60.8	134	36.3	80
Lining	112.0	247	72.6	160
Tie-Down	8.6	19	----	---
Nets	21.3	47	----	---
Partitions	33.1	73	----	---
Warm Air Ducts	7.7	17	----	---
Attachments	34.5	76	----	---
TOTAL CARGO ACCOMMODATIONS	278.0	613	145.2	320

TABLE C-5. CARGO ACCOMMODATIONS.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

	737-200 88 PASSENGERS		100 PASSENGERS	
	Kg	(Lbs)	Kg	(Lbs)
Oxygen System	(59.9)	(132)	(66.7)	(147)
Passengers	43.1	95	49.9	110
Crew	16.8	37	16.8	37
Fire and Smoke Protection	(52.2)	(115)	(47.2)	(104)
Detection	26.3	58	26.3	58
Extinguishing	20.9	46	20.9	46
Viewers - Cargo Compartment & Gear Downlock	5.0	11	----	---
Escape Provisions	(34.0)	(75)	----	---
Slides	29.5	65	----	---
Ropes	4.5	10	----	---
Hand Fire Extinguishers	14.1	31	14.1	31
First Aid	2.7	6	2.7	6
Axes	1.8	4	1.8	4
TOTAL EMERGENCY EQUIPMENT	164.7	363	132.5	292

TABLE C-6 . EMERGENCY EQUIPMENT.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

	737-200 88 PASSENGERS		100 PASSENGERS	
	Kg	(Lbs)	Kg	(Lbs)
Flight Crew	154.2	340	172.3	380*
Flight Attendants	176.9	390	127.0	280*
Crew Baggage	56.7	125	56.7	125
Brief Cases & Navigational Equipment	11.3	25	11.3	25
Unusable Fuel	52.2	115	52.2	115
Oil	59.9	132	59.9	132
Emergency Equipment	(84.9)	(187)	(23.6)	(52)
Oxygen	16.3	36	16.3	36
Escape Slides	59.9	132	----	--
Fire Axe	1.4	3	----	--
Oranasal Masks	2.3	5	2.3	5
Smoke Goggles	.5	1	.5	1
Hand Megaphones	4.5	10	4.5	10
Passenger Accommodations	(664.0)	(1464)	(415.4)	(916)
Water	81.2	179	90.7	200
Toilet Chemicals	22.7	50	22.7	50
Beverage	77.6	171	90.7	200*
Serving Trays	5.4	12	6.3	14
Galley Structure	272.1	600	----	--
Galley Service Equipment	103.4	228	103.4	228
Passenger Service Equipment	101.6	224	101.6	224
Passengers	7183.7	15,840	8163.4	18,000*
TOTAL USEFUL LOAD (NOT INCLUDING FUEL)	8443.8	18,618	9081.7	20,025

*Quoted From Study Outline

TABLE C-7 . USEFUL LOAD.

BOEING VERTOL COMPANY

WEIGHT SUMMARY - PRELIMINARY DESIGN
MIL-STD-1374

D210-10873-1

	COMPUTER (POUNDS)	TREND SUBSTAN-TIATION (POUNDS)	
WING	5286	5249	
ROTOR	4139	4114	
TAIL	1147	1146	
SURFACES	1147	1146	
ROTOR			
BODY	8575	8497	
BASIC			
SECONDARY			
ALIGHTING GEAR GROUP	2740	2740	
ENGINE SECTION	637	637	
PROPULSION GROUP	6616	6668	
ENGINE INST'L	1755	1755	
EXHAUST SYSTEM *			
COOLING			
CONTROLS *			
STARTING *			
PROPELLER INST'L	*544	544	
LUBRICATING *			
FUEL	168	168	
DRIVE	4149	4201	
FLIGHT CONTROLS	3455	3431	
AUX. POWER PLANT	636	636	
INSTRUMENTS	423	423	
HYDR. & PNEUMATIC	680	680	
ELECTRICAL GROUP	934	934	
AVIONICS GROUP	648	648	
ARMAMENT GROUP			
FURN. & EQUIP. GROUP	7217	7217	
ACCOM. FOR PERSON.			
MISC. EQUIPMENT			
FURNISHINGS			
EMERG. EQUIPMENT			
AIR CONDITIONING	1350	1350	
ANTI-ICING GROUP	560	560	
LOAD AND HANDLING GP.			
WEIGHT EMPTY	45043	44930	
CREW			
TRAPPED LIQUIDS			
ENGINE OIL			
FUEL			
GROSS WEIGHT			

REV.

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FORM 26391 (2)

TABLE C-8. TILT ROTOR WEIGHT SUBSTANTIATION.

for the reduced weight due to advanced composites) on the appropriate weight trend graph.

The pitch and yaw radius of gyration trends are shown in Figures C9 and C10 respectively. These trends also indicate the values for the STOL tilt rotor aircraft, again with the proper adjustment for composite materials.

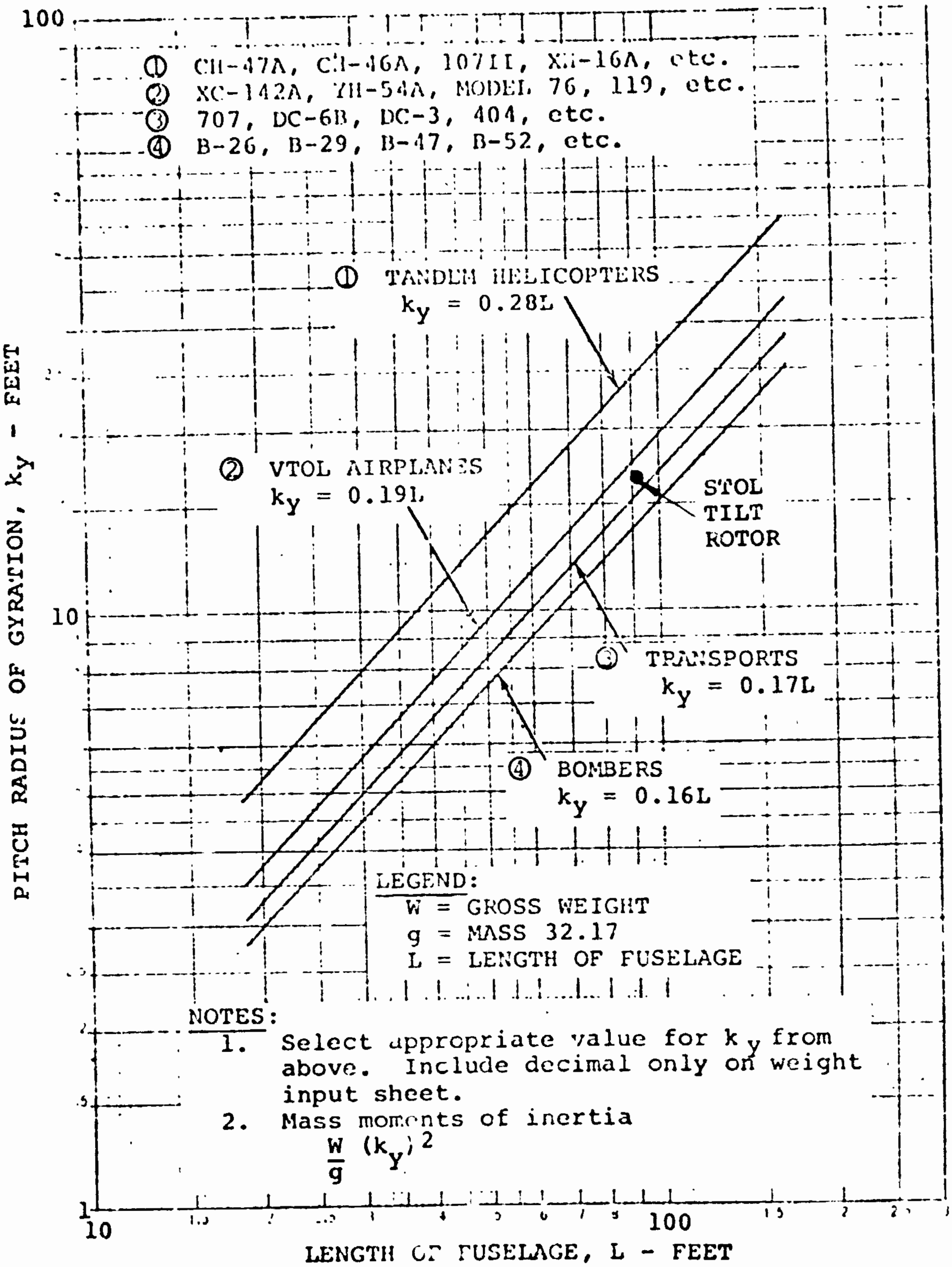
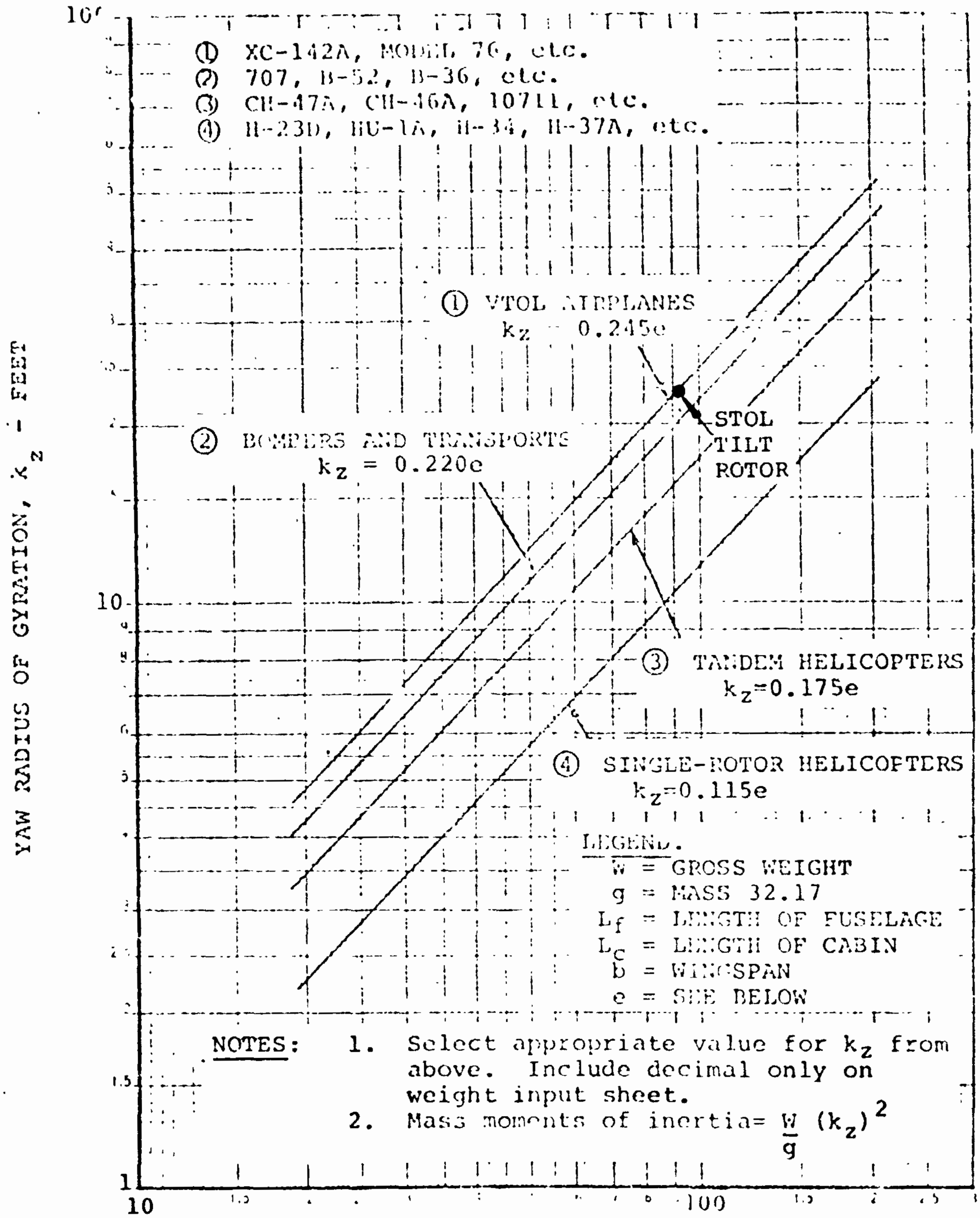


FIGURE C-9. RADIUS OF GYRATION TREND - PITCH.



$$e = \frac{b + L_f}{2} \text{ (AIRPLANES)} \qquad e = L_f + L_c \text{ (HELICOPTERS)}$$

FIGURE C-10. RADIUS OF GYRATION TREND - YAW.

DESIGN POINT TILT ROTOR WEIGHT SUBSTANTIATION

WING (REFERENCE - WEIGHT TREND CURVE, FIGURE C6)

5,249 LBS

$$W = (K)^{0.585}$$

$$K = \left(\frac{R_M \; W_{XW}}{10^4}\right)\left(\frac{S_W}{10^2}\right)\left(\text{Log}_{10} \frac{b}{B}\right)\sqrt{\frac{1 + \lambda}{2K_r}} \sqrt{N} \left(\text{Log}_{10} \; V_D\right) \left(\text{Log}_{10} \; AR\right)$$

WHERE:

W_W = WEIGHT OF WING

R_M = RELIEF TERM

W_{XW} = GROSS WEIGHT LESS ITEMS AT CENTER OF LIFT = 46,557 LBS

GROSS WEIGHT	68,493
ROTORS	-2,939
WING	-5,286
ENGINE SECTION	-637
PROPULSION	-7,918
ROTOR CONTROLS	-888
TILT MECHANISM	-596
FUEL	-3,425
TRAPPED LIQUIDS	-115
OIL	-132
TOTAL W_{XW}	46,557

S_W = PLANFORM WING AREA = 684.9 FT^2

b = WING SPAN = 78.5 FT

B = MAXIMUM FUSELAGE WIDTH = 14.8 FT

λ = WING TAPER RATIO = 1.0

K_r = WING ROOT THICKNESS, PERCENT CHORD = .21

N = ULTIMATE LOAD FACTOR = 3.75

V_D = DIVE VELOCITY = 375 KNOTS

AR = ASPECT RATIO = 9.0

$$K = \left(\frac{(.6)(46,557)}{10^4}\right)\left(\frac{684.9}{10^2}\right)\left(\mathrm{Log}_{10}\ \frac{78.5}{14.8}\right)\sqrt{\frac{1+1}{2(.21)}}$$

$$\sqrt{3.75}\quad\left(\mathrm{Log}_{10}\ 375\right)\quad\left(\mathrm{Log}_{10}\ 9.0\right)$$

$$= (2.79)\ (6.85)\ (.72)\ (2.18)\ (1.94)\ (2.57)\ (.95) = 142.1$$

$$W_W = 377.2\ (142.1)^{0.585} = 6916$$

ADD: WING-POD ATTACHMENTS

.036 (POD WEIGHT)

.036 (11,720) = + 422

7338

REDUCE FOR COMPOSITES

0.302 (6916) = - 2089

5249

ROTORS

4,114 LBS

BLADES (REFERENCE - WEIGHT TREND CURVE, FIGURE C2)

$$W_B = 44\ K^{0.438}$$

$$K = \left(\frac{W_g}{10^4}\right)\left(LLF\right)\left(\frac{R^2}{100}\right)\left(\frac{R-r}{10}\right)\left(bCK_b\right)\left[\frac{R^{1.6}}{K_d t}\right]$$

$$\left[\frac{R^{1.6}}{K_d t}\right] = 1.0 \text{ OR GREATER}$$

WHERE:

W_B = BLADE WEIGHT PER ROTOR

W_g = DESIGN GROSS WEIGHT = 60,000 LBS

LLF = LIMIT LOAD FACTOR = 3.5

R = ROTOR RADIUS = 22.2 FT

r = ℄ ROTATION TO BLADE ATTACHMENT = 1.66 FT

b = NUMBER OF BLADES PER ROTOR = 3

C = BLADE CHORD = 1.91 FT

K_b = ROTOR TYPE FACTOR - HINGELESS = 2.2

K_d = DROOP CONSTANT - TANDEM = 1,000

t = BLADE THICKNESS AT 25% CHORD = .23 FT

$$K = \left(\frac{60{,}000}{10^4}\right)(3.5)\left[\frac{(22.2)^2}{100}\right]\left(\frac{22.2-1.66}{10}\right)(3)(1.91)(2.2)\left[\frac{(22.2)^{1.6}}{(1000)(.23)}\right]$$

$$= (6.0)(3.5)(4.9)(2.05)(12.61)(1.0)$$

$$= 2660$$

$W_B = 44(2660)^{0.438} = 1,392$ LBS PER ROTOR

HUB (REFERENCE - WEIGHT TREND CURVE, FIGURE C3)

$W_H = 61\ K^{0.358}$

$K = (W_b)(R)(N_R)^2\ (P_R)(r)^{1.82}(b)^{2.5}(K_{amd})(10)^{-11}$

WHERE:

W_H = HUB WEIGHT PER ROTOR

W_b = BLADE WEIGHT = 464.0 LBS EACH

R = ROTOR RADIUS = 22.2 FT

N_R = ROTOR RPM = 333

P_R = ROTOR HORSEPOWER PER ROTOR = 11,142 X 0.55 = 6,128 HP

r = ℄ ROTATION TO BLADE ATTACHMENT = 1.66 FT

b = NUMBER OF BLADES PER ROTOR = 3

K_{amd} = a X m X d = 0.53 X .54 X 1.0 = .29

$K = (464.0)(22.2)(333)^2(6128)(1.66)^{1.82}(3)^{2.5}(.29)(10)^{-11}$

$= 791$

$W = 61(791)^{0.358} = 665$ LBS PER ROTOR

TOTAL ROTOR WEIGHT

BLADES = 1392

HUB = 665

2057 X 2 = 4,114 LBS

HORIZONTAL TAIL (REFERENCE - WEIGHT TREND CURVE, FIGURE C8)

618 LBS

$$W_{HT} = 350 \ (K)^{0.54}$$

$$K = (F_H) \left(\frac{S_H}{10^2}\right) \left(\frac{\text{Log} \quad V_D}{T_{MA} X t}\right)$$

$$F_H = \left(\frac{W_g}{10^4}\right)\left(\frac{K_y}{10}\right)\left(\frac{b_H}{10}\right) \left(\frac{1 + 2\lambda}{1 + \lambda}\right) \quad (K_{TL})$$

WHERE:

W_{HT} = WEIGHT OF HORIZONTAL TAIL

S_H = TAIL PLAN AREA = 171.0 FT

V_D = DIVE VELOCITY = 375 KNOTS

T_{MA} = TAIL MOMENT ARM = 51.0 FT

t = ROOT THICKNESS = 0.72 FT

W_g = DESIGN GROSS WEIGHT = 68,493 LBS

K_y = PITCH RADIUS OF GYRATION = 16.65 FT (FIGURE C9)

b_H = TAIL SPAN = 29.7 FT

λ = TAPER RATIO = .625

K_{TL} = TAIL LOAD FACTOR = 1.0

$$F_H = \left(\frac{68,493}{10^4}\right) \left(\frac{16.65}{10}\right) \left(\frac{29.7}{10}\right) \left(\frac{1 + 1.25}{1 + .625}\right) \left(1.0\right)$$

$$= 46.89$$

$$K = (46.89) \left[\frac{(171.0)}{10^2}\right] \left(\frac{\text{Log}_{10} \ 375}{51.0 \ X \ 0.72}\right)$$

$$= 5.61$$

$W_{HT} = 350\ (5.61)^{.54} \qquad = \quad 888$

REDUCTION FOR COMPOSITE

$.302 \times 888 \qquad = \quad \underline{-270}$

HORIZONTAL TAIL WEIGHT $\qquad = \quad 618$

VERTICAL TAIL (REFERENCE WEIGHT TREND CURVE, FIGURE C7)

528 LBS

$$W_{VT} = 360\ (K)^{0.54}$$

$$K = \left[(FV) + \frac{aF_H}{2b_V} \right] \left(\frac{S_V}{10^2} \right) \left(\frac{\text{Log}_{10}\ V_D}{T_{MA} X t} \right)$$

$$F_V = \left(\frac{W_g}{10^4} \right) \left(\frac{K_Z}{10} \right) \left(\frac{b_V}{10} \right) \left(\frac{1 + 2\lambda}{1 + \lambda} \right)$$

WHERE:

W_{VT} = WEIGHT OF VERTICAL TAIL

a = HEIGHT OF HORIZONTAL TAIL ABOVE ROOT CHORD = 29.7 FT

b_V = TAIL SPAN = 29.7 FT

S_V = TAIL AREA = 185.0 FT^2

V_D = DIVE VELOCITY = 375 KNOTS

T_{MA} = TAIL MOMENT ARM = 42.05 FT

t = ROOT THICKNESS = 1.579 FT

W_g = DESIGN GROSS WEIGHT = 68,493 LBS

K_Z = YAW RADIUS OF GYRATION = 22.0 FT (FIGURE C10)

λ = TAPER RATIO = 0.543

$$F_V = \left(\frac{68{,}493}{10^4}\right)\left(\frac{22.0}{10}\right)\left(\frac{29.7}{10}\right)\left(\frac{1 + 2 \times .543}{1 + .543}\right)$$

$$= 31.78$$

$$K = \left[\left(31.78\right)+\left(\frac{29.7 \times 46.89}{2 \times 29.7}\right)\right]\left(\frac{185.0}{10^2}\right)\left(\frac{\text{Log}_{10}\ 375}{42.05 \times 1.579}\right)$$

$$= (55.23)(1.85)(.039)$$

$$= 3.96$$

$$W_{VT} = 360\ (3.96)^{0.54} \qquad = \quad 757$$

REDUCTION FOR COMPOSITE

.302 X 757 = $\underline{-229}$

VERTICAL TAIL WEIGHT = 528

<u>BODY</u> (REFERENCE - WEIGHT TREND CURVE, FIGURE C4)

8,497 LBS

$$W_B = 126.0\ (K)^{0.508}$$

$$K = \left(\frac{W_X}{10^4}\right)^{0.7}\left(\frac{S_f}{10^3}\right)(B)(L_f + L_{RW})^{0.5}(\text{Log}_{10}\ V_D)$$

$$(\Delta_p + 1)^{0.2}\ (N)^{0.3}$$

WHERE:

W_B = WEIGHT OF BODY

W_{XB} = WEIGHT OF FUSELAGE AND CONTENTS = 46,557 LBS

GROSS WEIGHT = 68,493 LBS

LESS

ROTORS		-2,939
WING		-5,286
ENGINE SECTION		-637
PROPULSION		-7,918
ROTOR CONTROLS		-888
TILT MECHANISM		-596
FUEL		-3,425
TRAPPED LIQUIDS		-115
OIL		-132
TOTAL WEIGHT	=	46,557 LBS

S_f = WETTED AREA = 3,464 FT^2

B = BODY WIDTH = 14.8 FT

L_f = LENGTH OF FUSELAGE = 92.5 FT

L_{RW} = LENGTH OF RAMP WELL = 0 FT

V_D = DIVE VELOCITY = 375 KNOTS

Δ_p = LIMIT DIFFERENTIAL CABIN PRESSURE = 3.13 PSI

N = ULTIMATE LOAD FACTOR = 3.75

$$K = \left(\frac{46,557}{10^4}\right)^{0.7}\left(\frac{3464}{10^3}\right)^{0.5} (14.8)(92.5 + 0)^{0.5} (\mathrm{Log}_{10}\ 375)$$

$$(3.13 + 1)^{0.2}\ (3.75)^{0.3}$$

$= (2.93)(3.46)(14.8)(9.62)(2.57)(1.33)(1.49)$

$= 7351$

$W_B = 126(7351)^{0.508}$ = 11,600

REDUCE FOR COMPOSITE

.302 X 11,600 = -3,503

8,097

ADD AIR STAIRS (1) +400

8,497

LANDING GEAR (4% DESIGN GROSS WEIGHT) 2,740 LBS

0.04 (68,493) = 2,740

ENGINE SECTION (52% ENGINE WEIGHT) 637 LBS

0.52 (1755) = 913

REDUCE FOR COMPOSITE

.302 (913) = -276

ENGINE SECTION WEIGHT = 637 LBS

ENGINES 1,755 LBS

ENGINE INSTALLATION (31% ENGINE WEIGHT) 544 LBS

(0.31) (1755) = 544

FUEL SYSTEM (4.9% FUEL) 168 LBS

(0.049) (3425) = 168

DRIVE SYSTEM (REFERENCE - WEIGHT TREND CURVE, FIGURE C5)

4,201 LBS

$$W_{DS} = 250 \ (K)^{0.67}$$

$$K = \left(\frac{HPX1.1}{RPM}\right) \ (Z)^{0.25} \ (K_T)$$

WHERE:

W_{DS} = WEIGHT OF DRIVE SYSTEM

HP = TOTAL HORSEPOWER = 11,142

RPM = ROTOR DESIGN RPM = 333

Z = NUMBER OF STAGES IN MAIN DRIVE = 4

K_T = CONFIGURATION FACTOR = 1.3

$$K = \left(\frac{11{,}142 \times 1.1}{333}\right) (4)^{0.25} (1.3) = 67.46$$

$$W_{DS} = 250(67.46)^{0.67} = 4{,}201$$

FLIGHT CONTROLS (FLY-BY-WIRE) 3,431 LBS

Cockpit

$$W_{CC} = 26 \left(\frac{W_g}{10^4}\right)^{0.41}$$

WHERE:

W_{CC} = WEIGHT OF COCKPIT CONTROLS

W_g = DESIGN GROSS WEIGHT = 68,493 LBS

$$W_{CC} = 26 \left(\frac{68{,}493}{10^3}\right)^{0.41} = 147$$

REDUCE FOR FLY-BY-WIRE

0.29(147) = -43

TOTAL COCKPIT CONTROLS = 104

Rotor Controls

$$W_{RC} = 0.30\ (W_R)$$

WHERE:

W_{RC} = WEIGHT OF ROTOR CONTROLS

W_R = WEIGHT OF ROTOR = 4,114

W_{RC} = 0.30 (4,114)

= 1234

System Controls

$$W_{SC} = 45 \left(\frac{W_R}{100}\right)^{0.84}$$

WHERE:

W_{SC} = WEIGHT OF SYSTEM CONTROLS

W_R = WEIGHT OF ROTOR = 4,114 LBS

$$W_{SC} = 41 \left(\frac{4114}{100}\right)^{0.84} = 931$$

REDUCE FOR FLY-BY-WIRE

0.20 X 931 = -186

TOTAL SYSTEM CONTROLS = 745

Airplane Controls

W_{AC} = 0.011 (W_g)

WHERE:

W_{AC} = WEIGHT OF AIRPLANE CONTROL SYSTEM

W_g = DESIGN GROSS WEIGHT = 68,493 LBS

W_{AC} = 0.011(68,493) = 753

REDUCE FOR FLY-BY-WIRE

0.2 (753) = -151

TOTAL AIRPLANE CONTROLS = 602

SAS - ESTIMATED WEIGHT = 150 LBS

Tilt Mechanism

W_{TM} = 0.010 (W_g)

WHERE:

W_{TM} = WEIGHT OF POD TILTING MECHANISM

W_g = DESIGN GROSS WEIGHT = 68,943 LBS

W_{TM} = 0.010 (68,493) = 685

REDUCE FOR FLY-BY-WIRE

0.13 (685) = -89

TOTAL TILT MECHAMISM 596

SUMMARY OF FLIGHT CONTROLS WEIGHT

COCKPIT	=	104
ROTOR	=	1,234
SYSTEM	=	745
AIRPLANE	=	602
SAS	=	150
TILT MECHANISM	=	596
TOTAL		3,431

FIXED EQUIPMENT (REFERENCE TABLE C2) 12,348 LBS