NASA CR 137601

CONCEPTUAL DESIGN STUDY OF A 1985 COMMERCIAL STOL TILT ROTOR TRANSPORT

C. A. WIDDISON

J. P. MAGEE

H. R. ALEXANDER
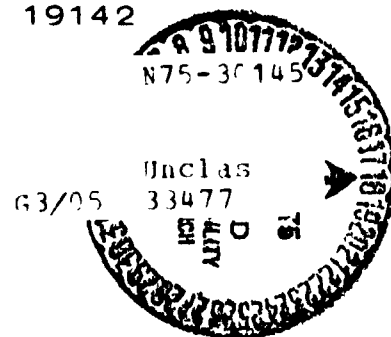
NOVEMBER 1974

D210-10873-1

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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 develop-
ment 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 improve-
ment 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/ft²)	108(22.12)	73.26(15)
WING LOADING	Kg/m^2(Lbs/ft²)	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)	8.31X106(11,142)	12.36X106(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 fóssil 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

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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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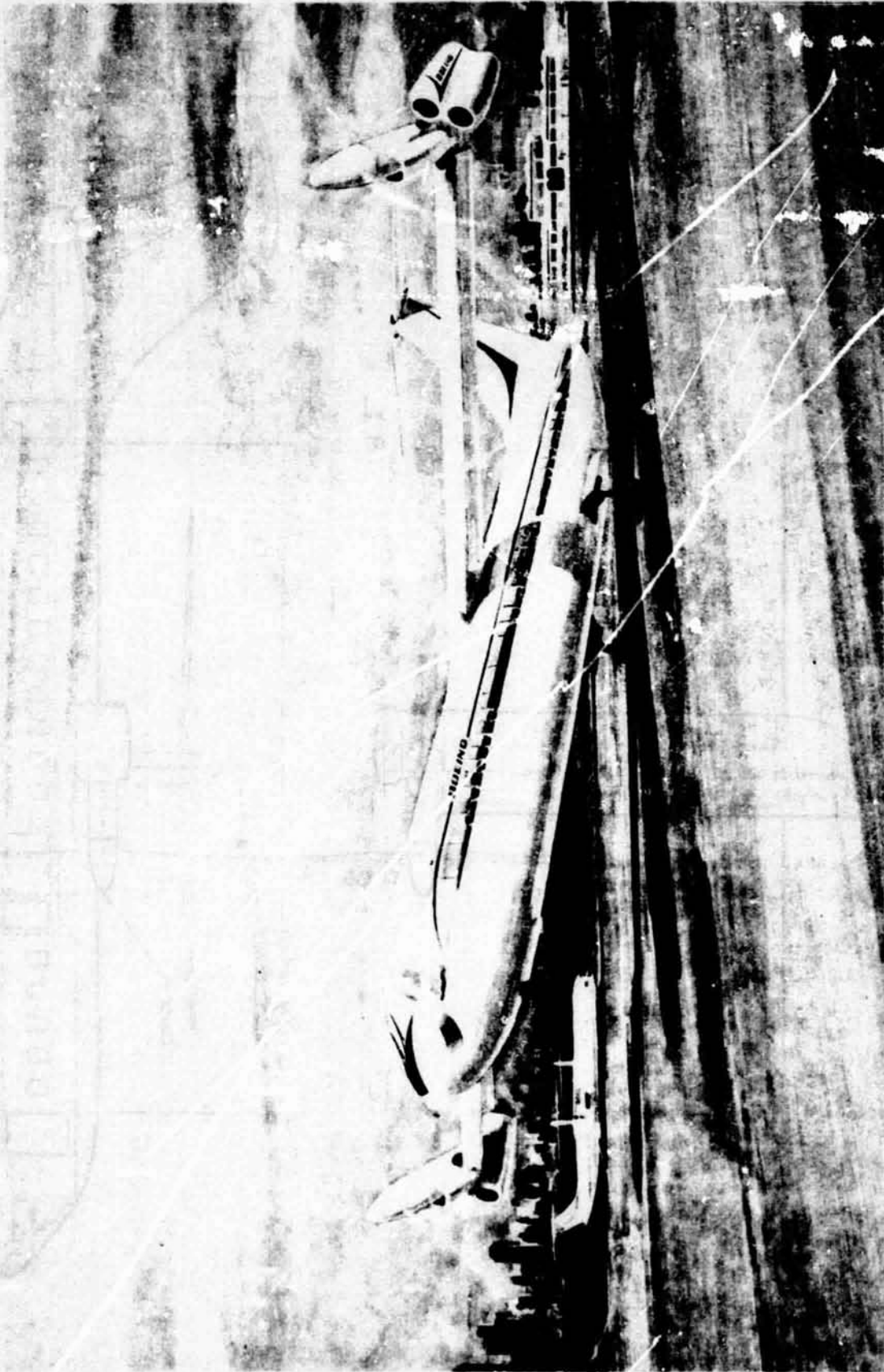
FIGURE 2.1. BOEING VERTOL 1985 100 PASSENGER STOL
TILT ROTOR COMMERCIAL TRANSPORT
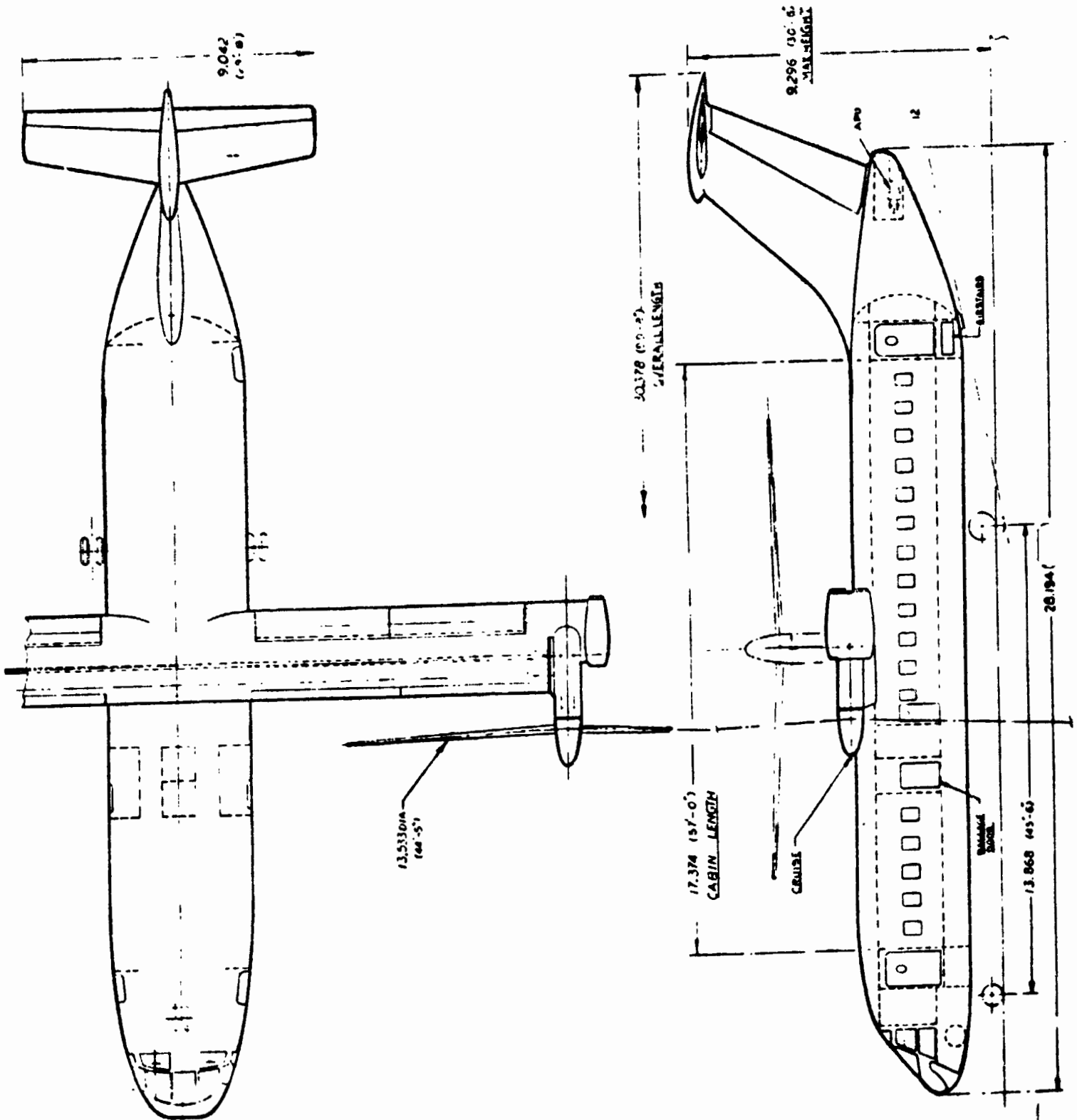
FIGURE 2.2. SIDE AND PLAN VIEWS.
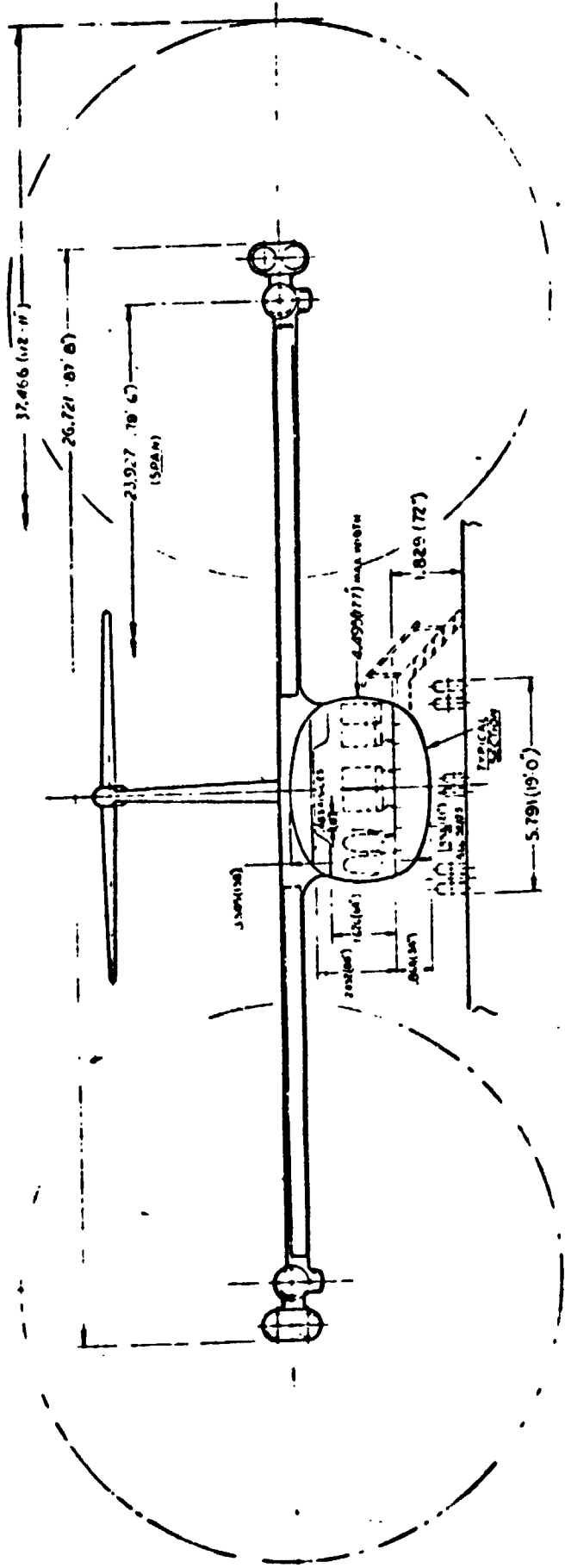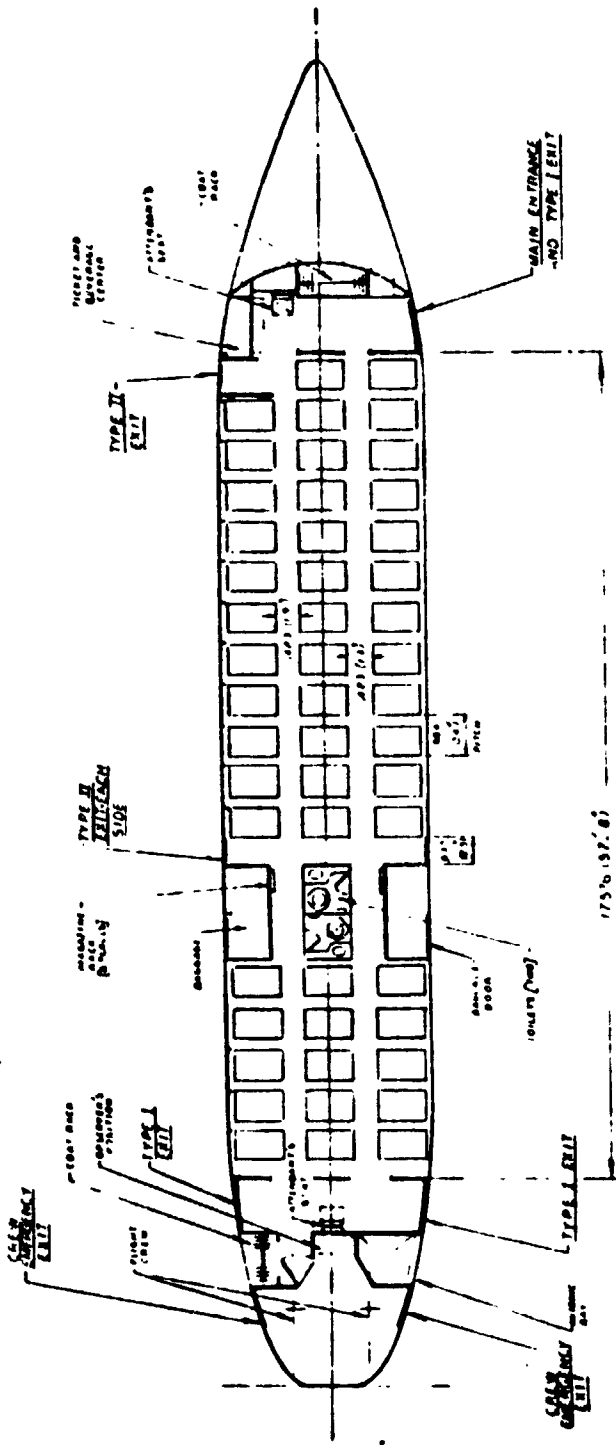
FIGURE 2.3. FRONT VIEW AND CABIN LAYOUT.

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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 per-
formance.

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 air-
foil 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

- Lighting
- Overhead Stowage Doors/Lockable
- Air Vents & Individual Lights
- Under Seat Stowage (9" x 16" x 23")

Windows

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
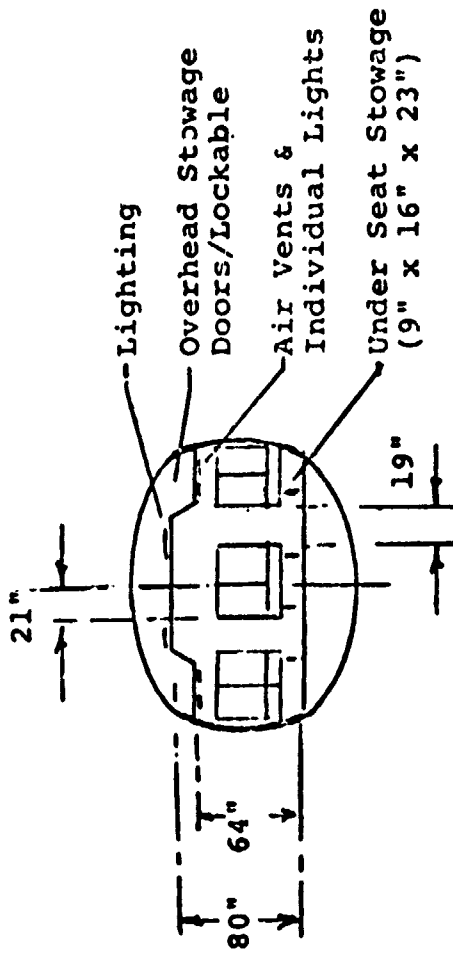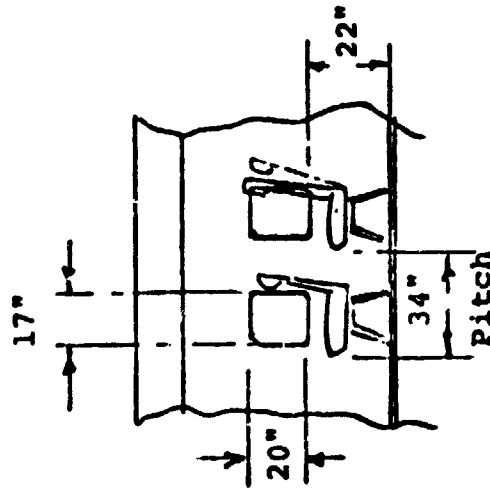
Systems

- 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

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
segmer: 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.

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TABLE 2.2. STOL MISSION PROFILE DEFINITION.

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	

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FIGURE 2.5. DESIGN SHORT HAUL MISSION.

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1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

	TIME (HOURS)	DISTANCE (N.MI.)	WEIGHT (LBS)	FUEL (LBS)	V (KNOTS)	R/C (FT/MIN)
TAXI	0	0	68,493	9	–	–
TAKEOFF	.017	0	68,484	141	–	–
CLIMB	.050	0	68,343	389	153/181	3105/1551
CRUISE	.152	16.34	67,954	1605	311/313	0
DESCENT	.657	173.88	66,349	87	374/258	-4388/-2137
AIR MANEUVER	.746	200	66,262	43	143	0
DESCENT	.771	200	66,219	8	257/254	-2326/-2134
LANDING	.779	201.92	66,211	101	–	–
TAXI	.804	201.92	66,110	9	–	–
RESERVE	.820	201.92	66,101	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
TAXI	0	0	31,068	4	-	-
TAKEOFF	.017	0	31,064	64	-	-
CLIMB	.050	0	31,000	176	153/181	15.8/7.9
CRUISE	.152	30.26	30,823	728	311/313	0
DESCENT	.657	322.03	30,095	39	374/258	-22.3/-10.9
AIR MANEUVER	.746	370.40	30,056	20	143	0
DESCENT	.771	370.40	30,036	4	257/254	-11.8/-10.8
LANDING	.779	373.96	30,033	46	-	-
TAXI	.804	373.96	29,987	4	-	-
RESERVE	.820	373.96	29,983	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 mainten- ance 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).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

	$/LB $/KG AIRFRAME	HOURS UTILIZATION
	$ 90/$198	2500
	$110/$243	2500
	$ 90/$198	3500
	$110/$243	3500

DESIGN POINT AIRCRAFT

EXTENDED RANGE VERSION

DIRECT OPERATING COST

CENTS / SEAT-ML

CENTS / SEAT-Km

RANGE – STATUTE MILES

RANGE – KILOMETERS
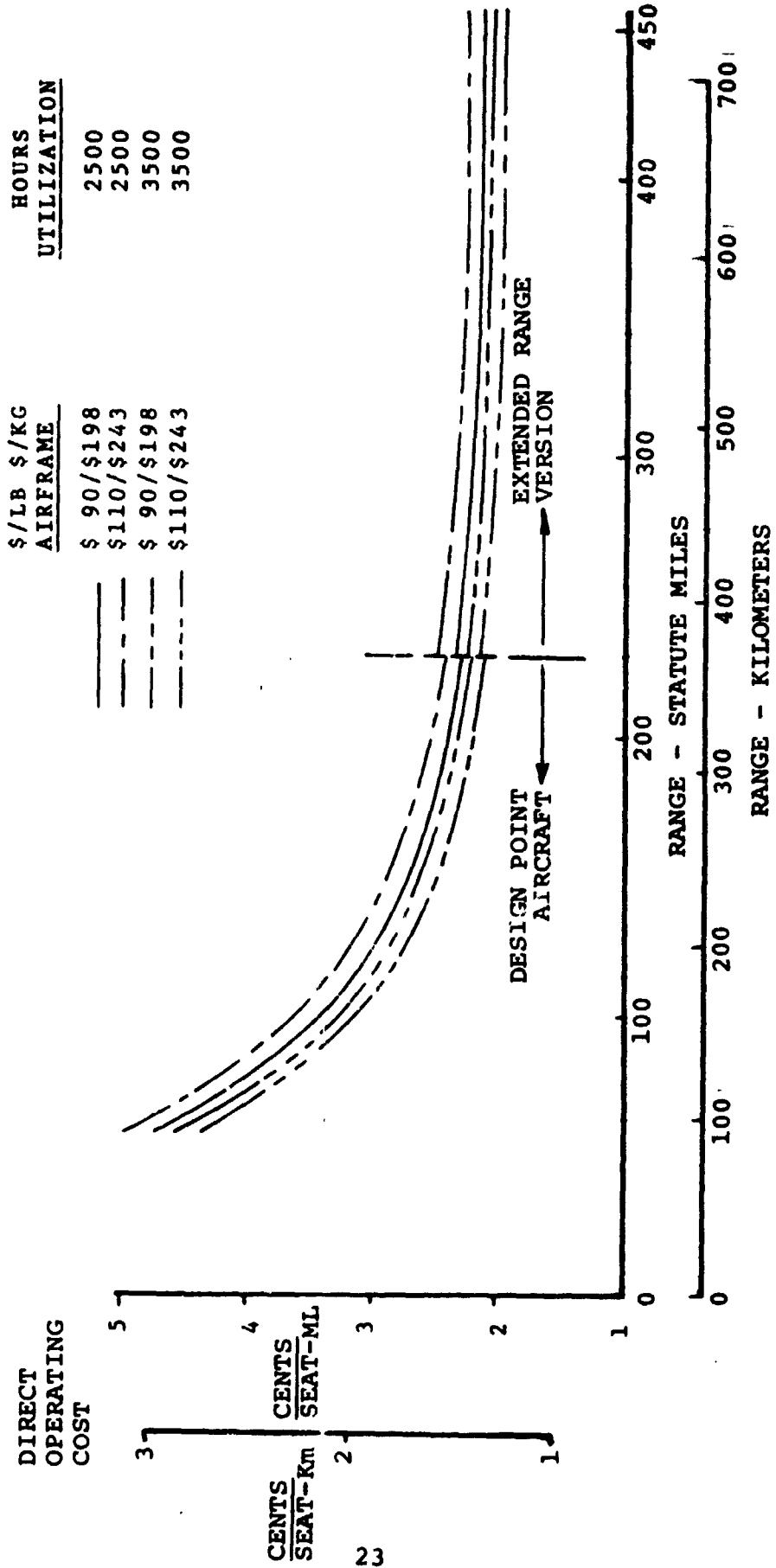
FIGURE 2.6. EFFECT OF OPERATING RANGE ON DIRECT OPERATING COST

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

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
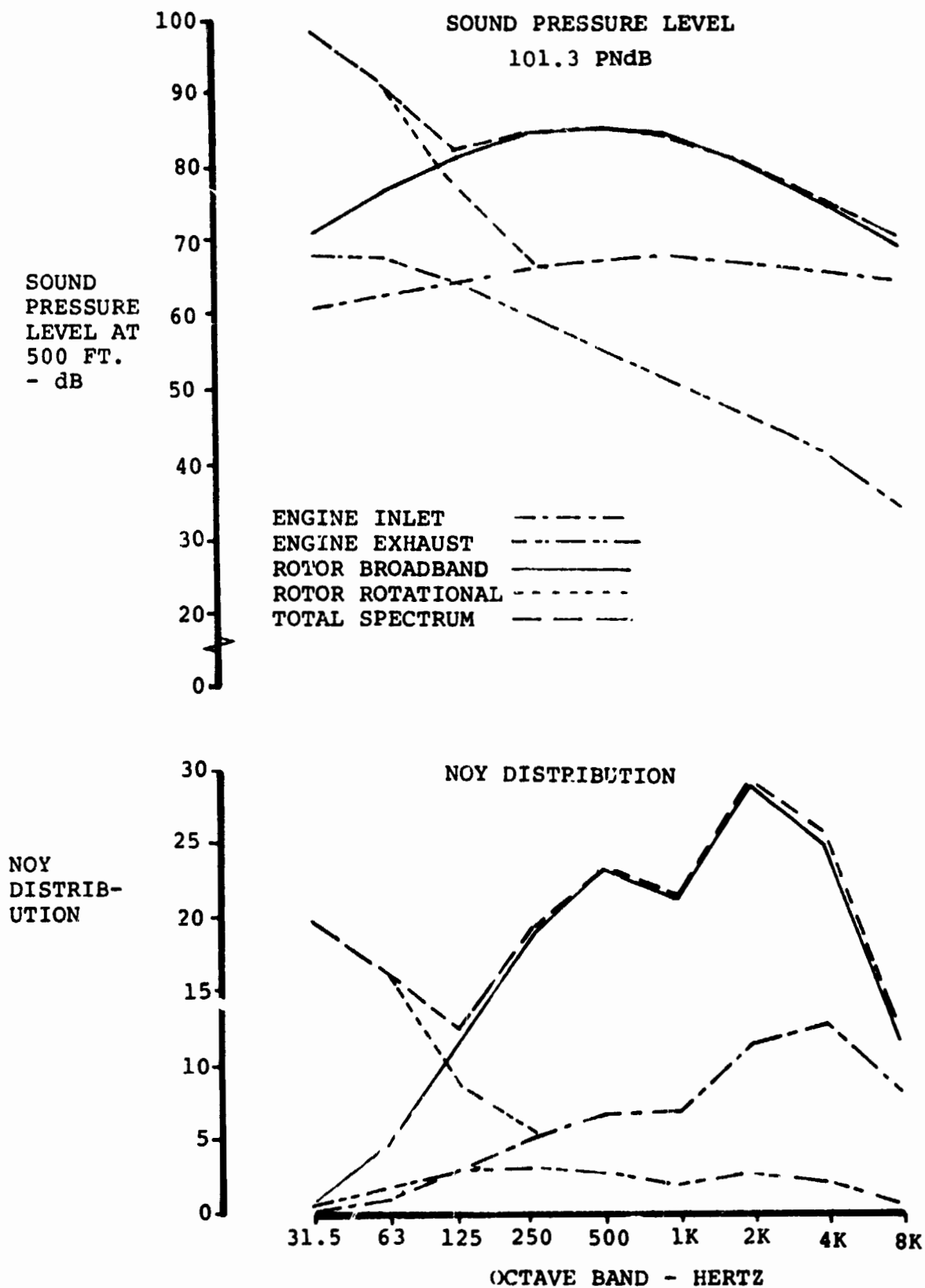
FIGURE 2.7. TAKEOFF NOISE SPECTRUM AND NOY DISTRIBUTION.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

INLET LINING ATTENUATION
DELTA SOUND PRESSURE LEVEL – db re $2 \times 10^{-5} N/m^2$

OCTAVE BAND CENTER FREQUENCY - Hz
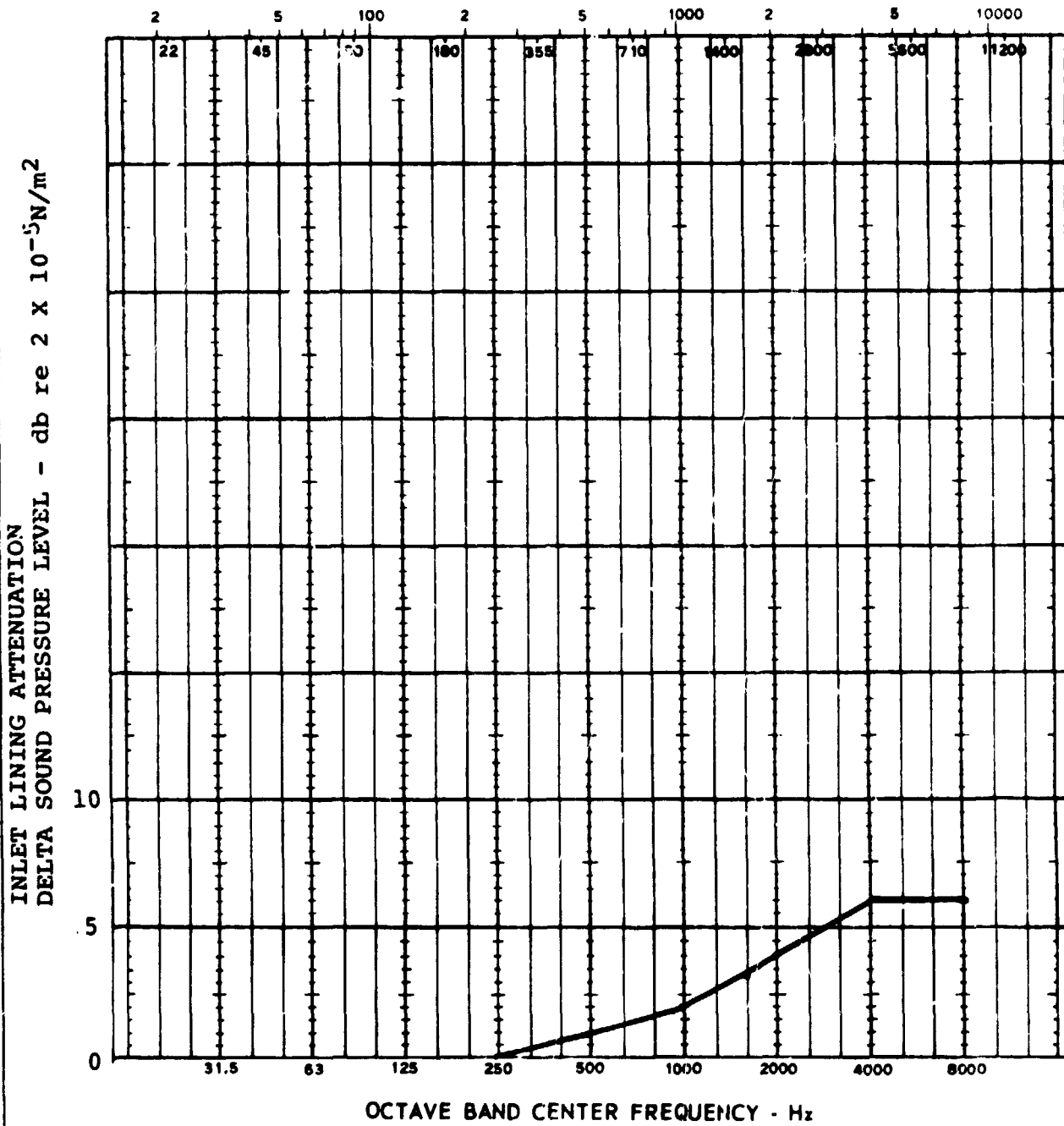
FIGURE 2.8.   ENGINE INLET NOISE SUPPRESSION

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
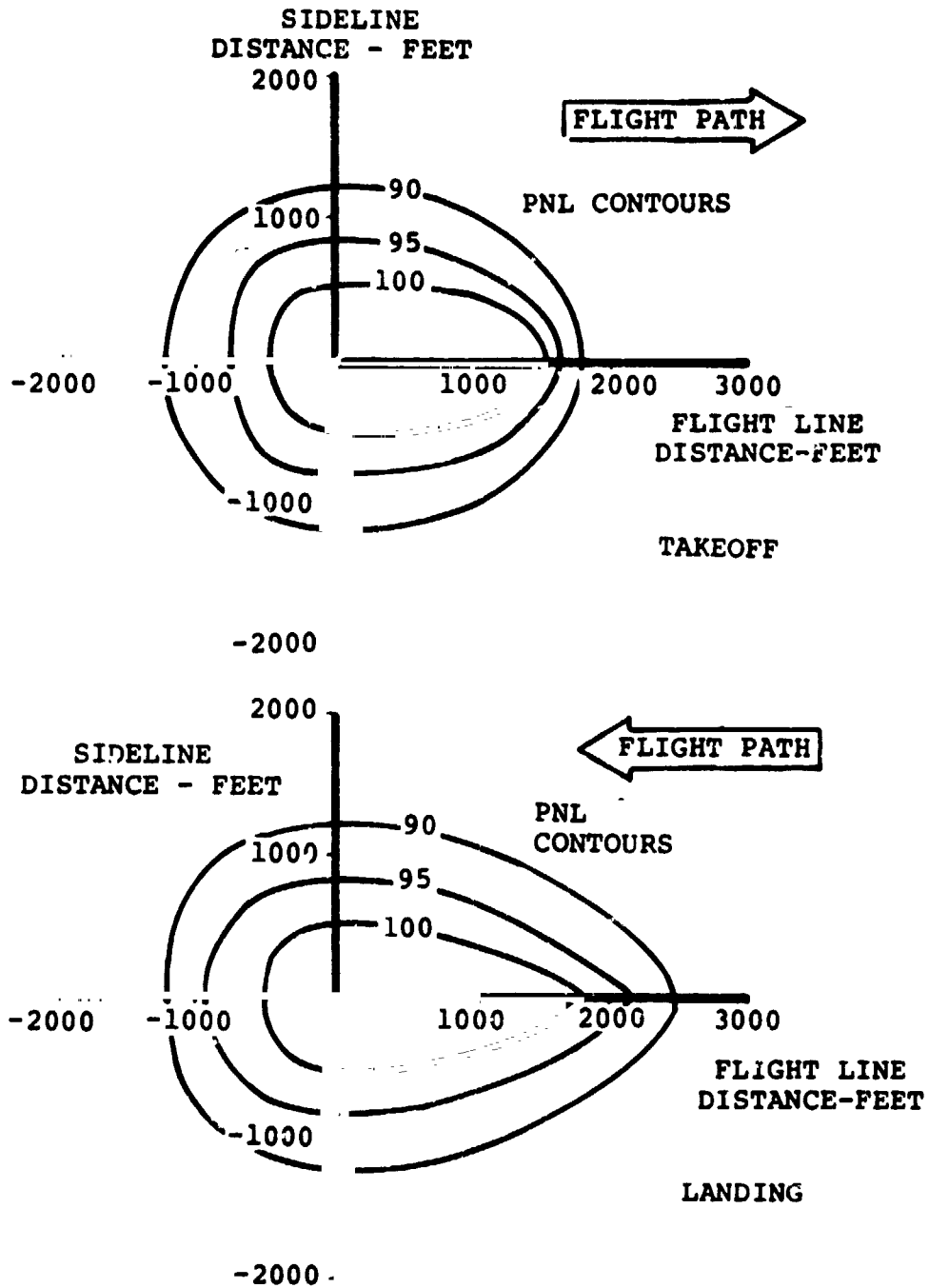
TAKEOFF

LANDING

FIGURE 2.9. TAKEOFF AND LANDING PNL CONTOURS.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
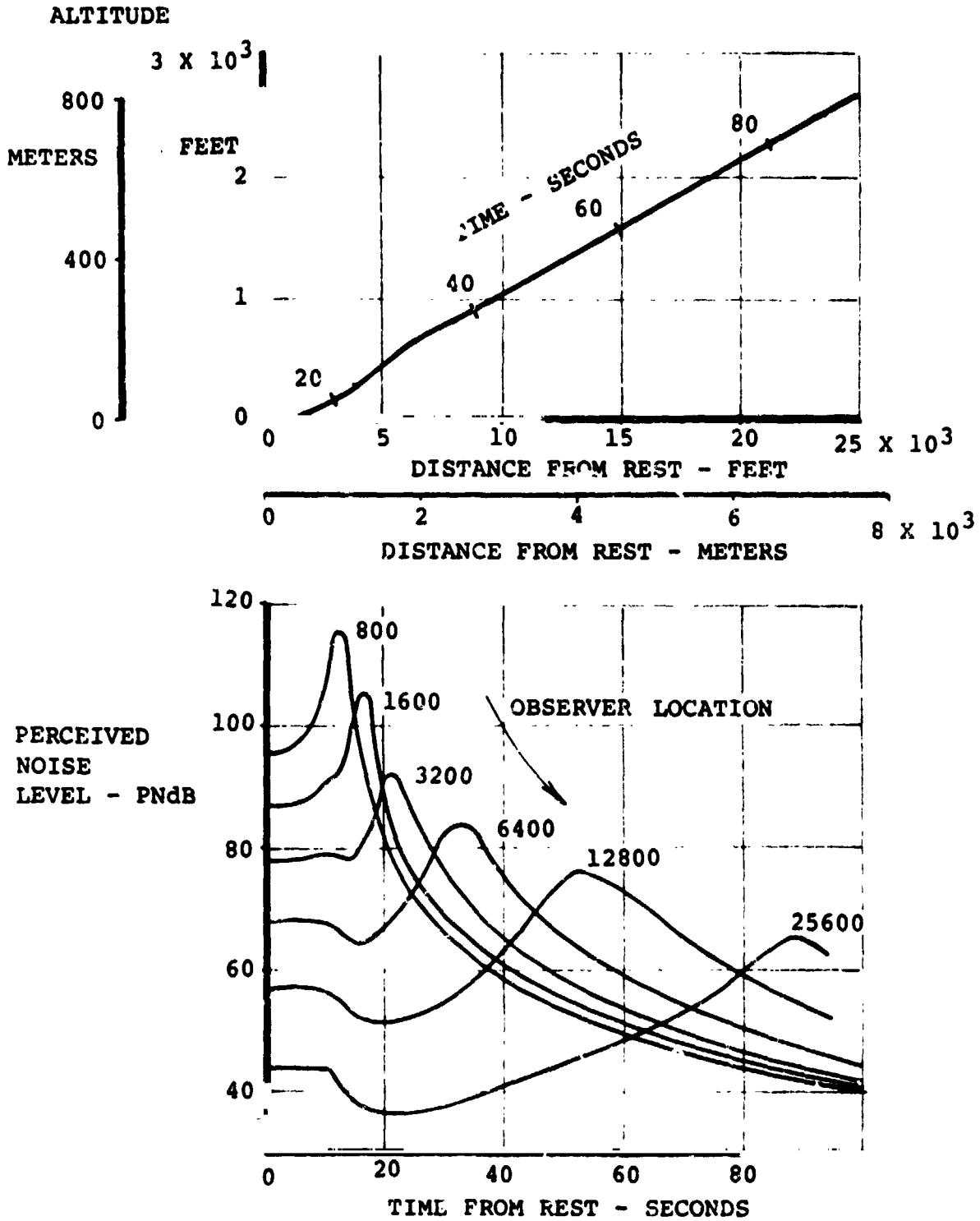
FIGURE 2.10. TAKEOFF PERCEIVED NOISE LEVEL HISTORY.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
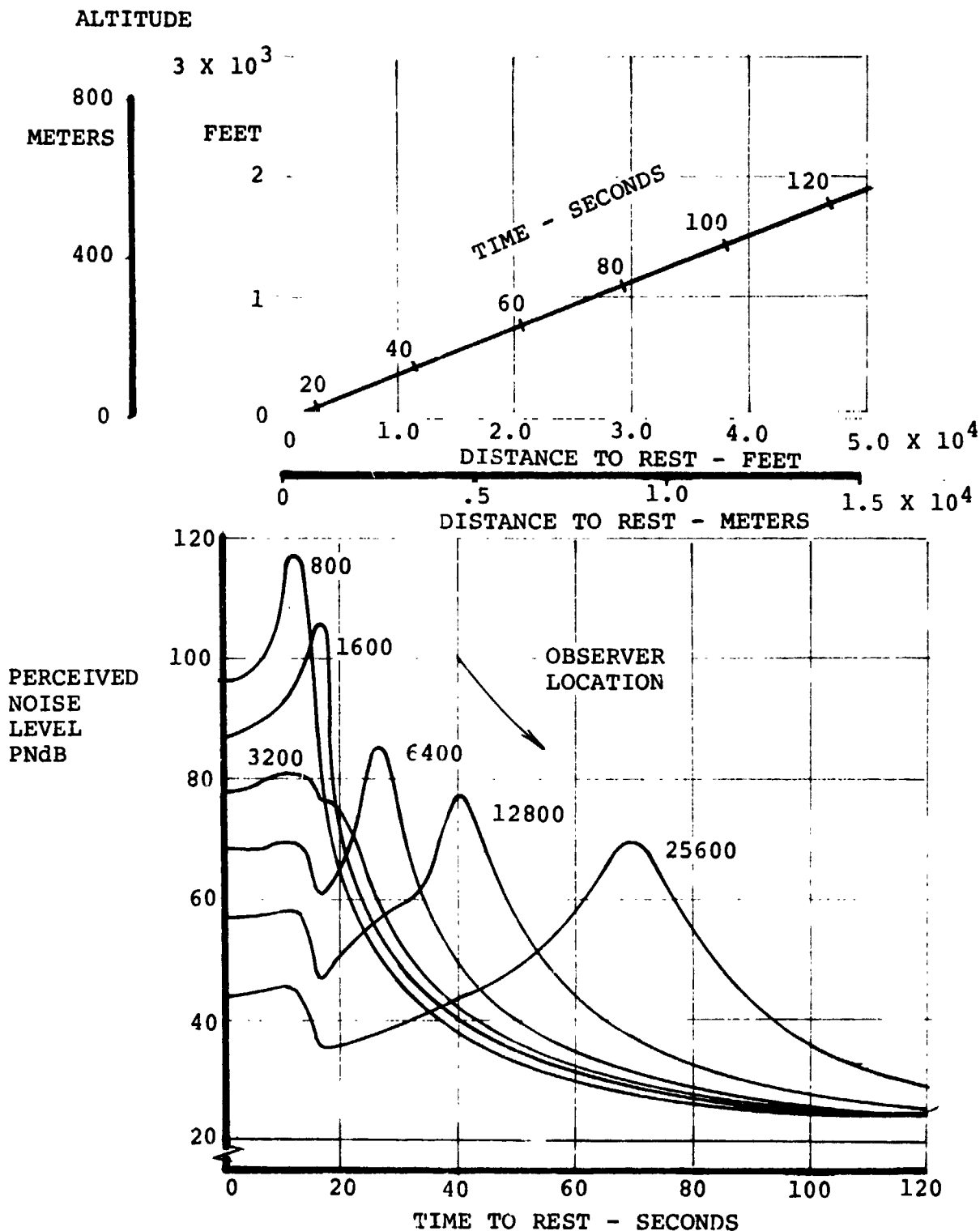
FIGURE 2.11. LANDING PERCEIVED NOISE LEVEL HISTORY.

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 (OBSTACLE HEIGHT = 35 FT)
TO OBSTACLE:

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

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

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

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, μ = .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 degra-

dation 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.

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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°

DISTANCE   DGW = 68,493 LBS/31,068 Kg
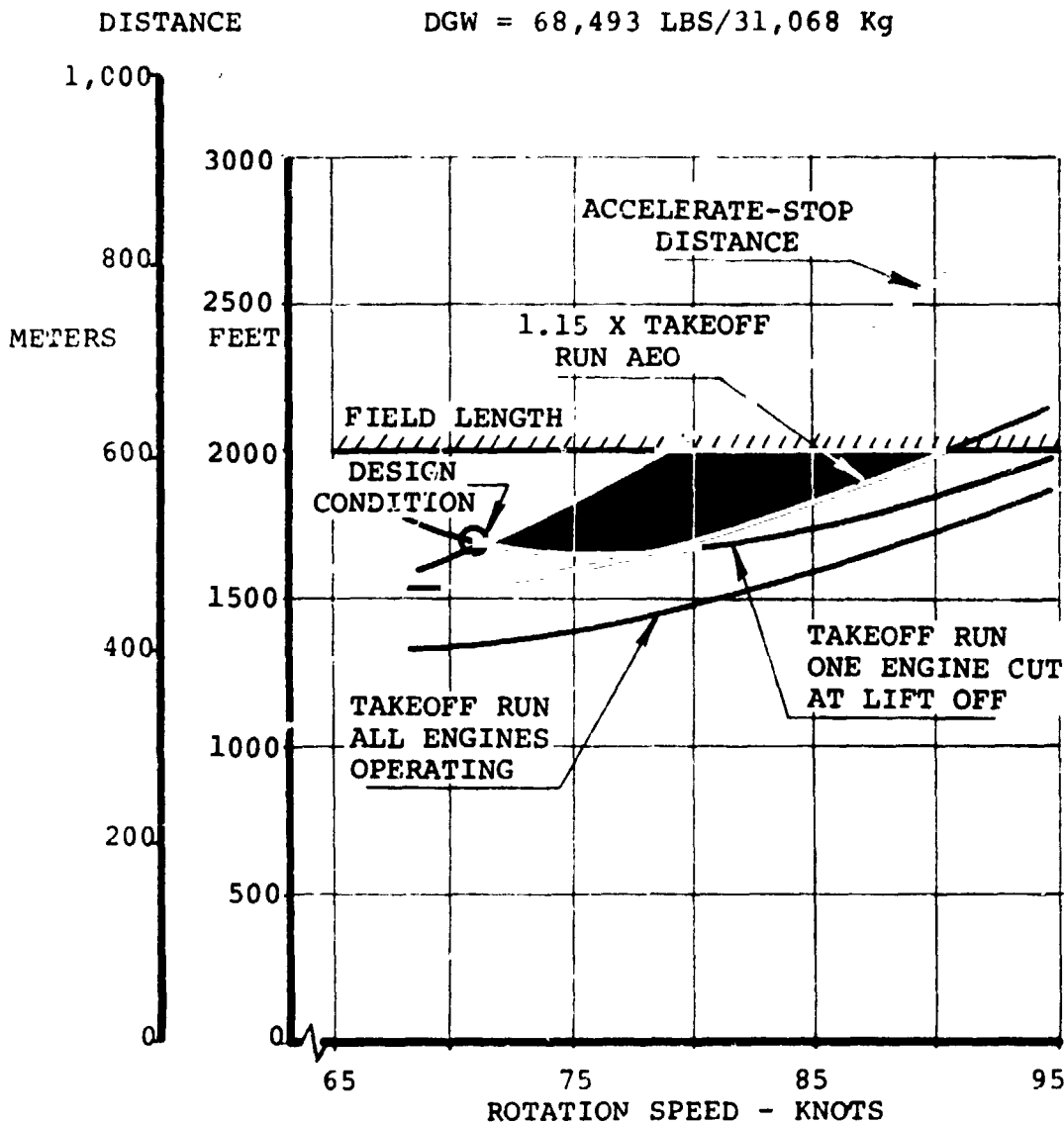
FIGURE 3.1.   EFFECT OF ROTATION SPEED ON TAKEOFF PERFORMANCE.

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1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
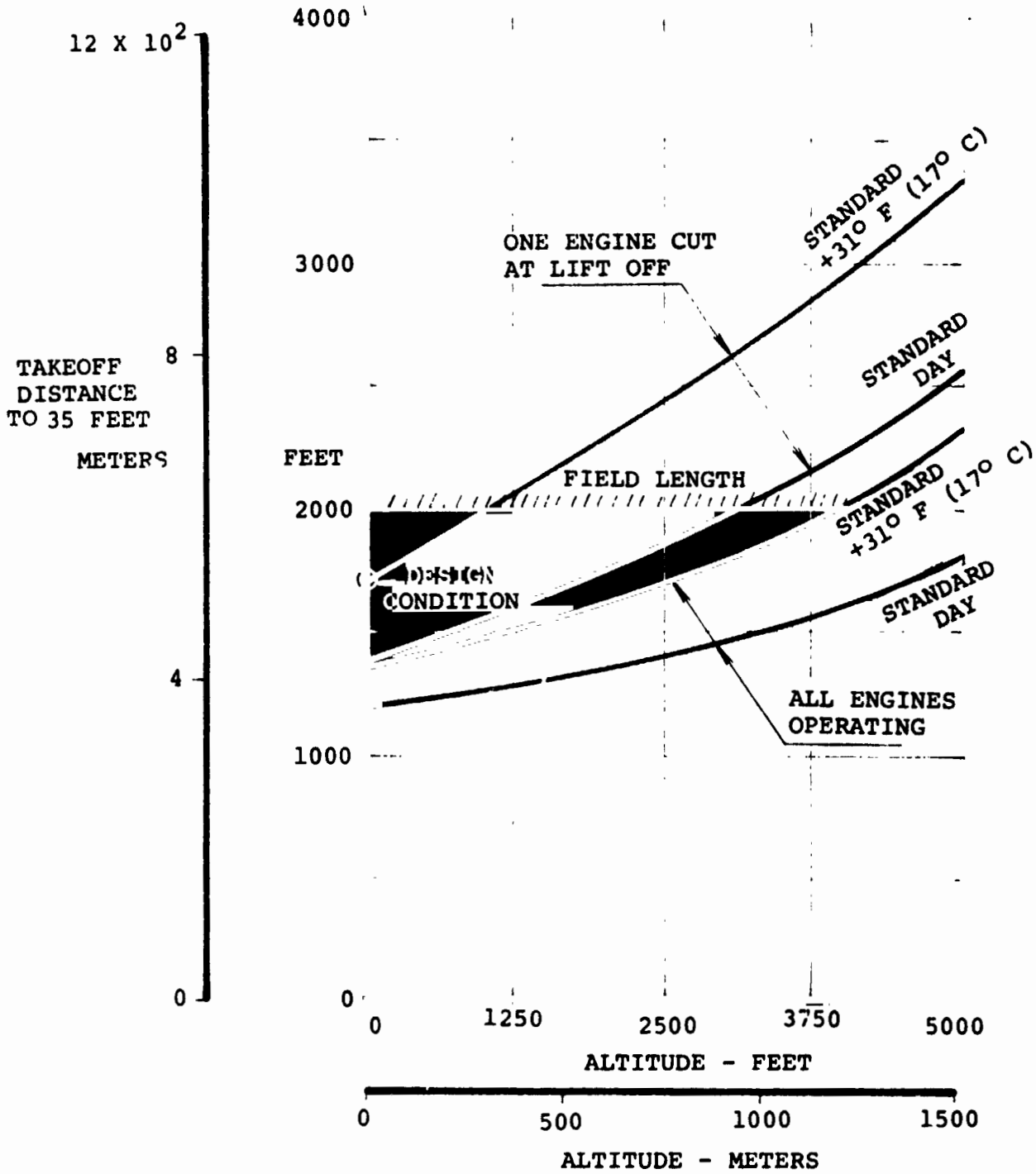
FIGURE 3.2.   EFFECT OF ALTITUDE AND AMBIENT TEMPERATURE ON
TAKEOFF PERFORMANCE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

TAKEOFF AT SL/90° F (32° C)
FLAP SETTING = 40°
KRUGER FLAP DEPLOYED
NACELLE INCIDENCE = 66°
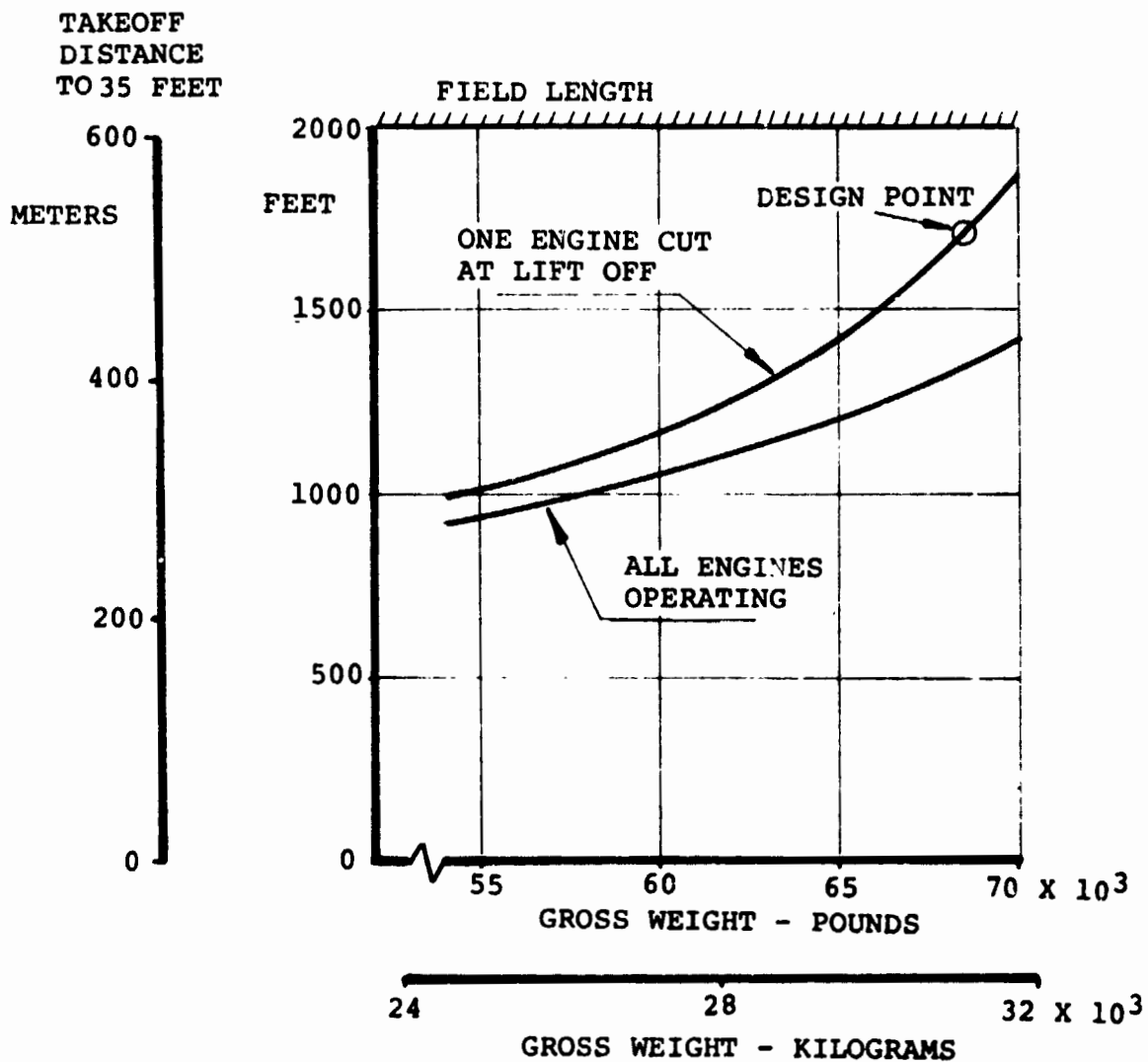
FIGURE 3.3. EFFECT OF GROSS WEIGHT ON TAKEOFF PERFORMANCE.

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

TAKEOFF
DISTANCE
TO 35 FEET

ONE ENGINE CUT
AT LIFT OFF

FIELD LENGTH

DESIGN CONDITION

ALL ENGINES
OPERATING

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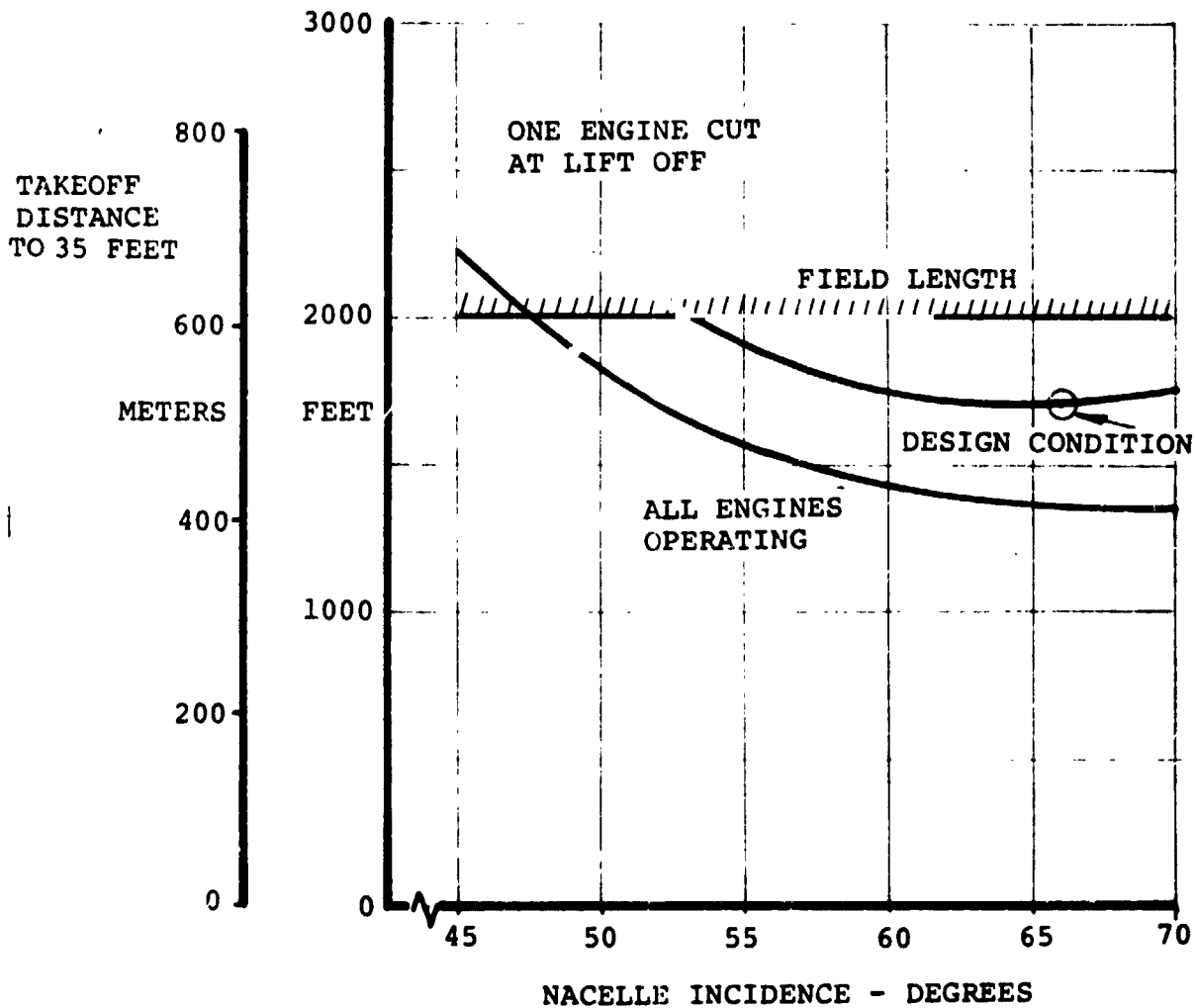
FIGURE 3.4.   EFFECT OF NACELLE INCIDENCE SETTING ON
TAKEOFF DISTANCE.

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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/?1,068 Kg
NACELLE INCIDENCE = 66°
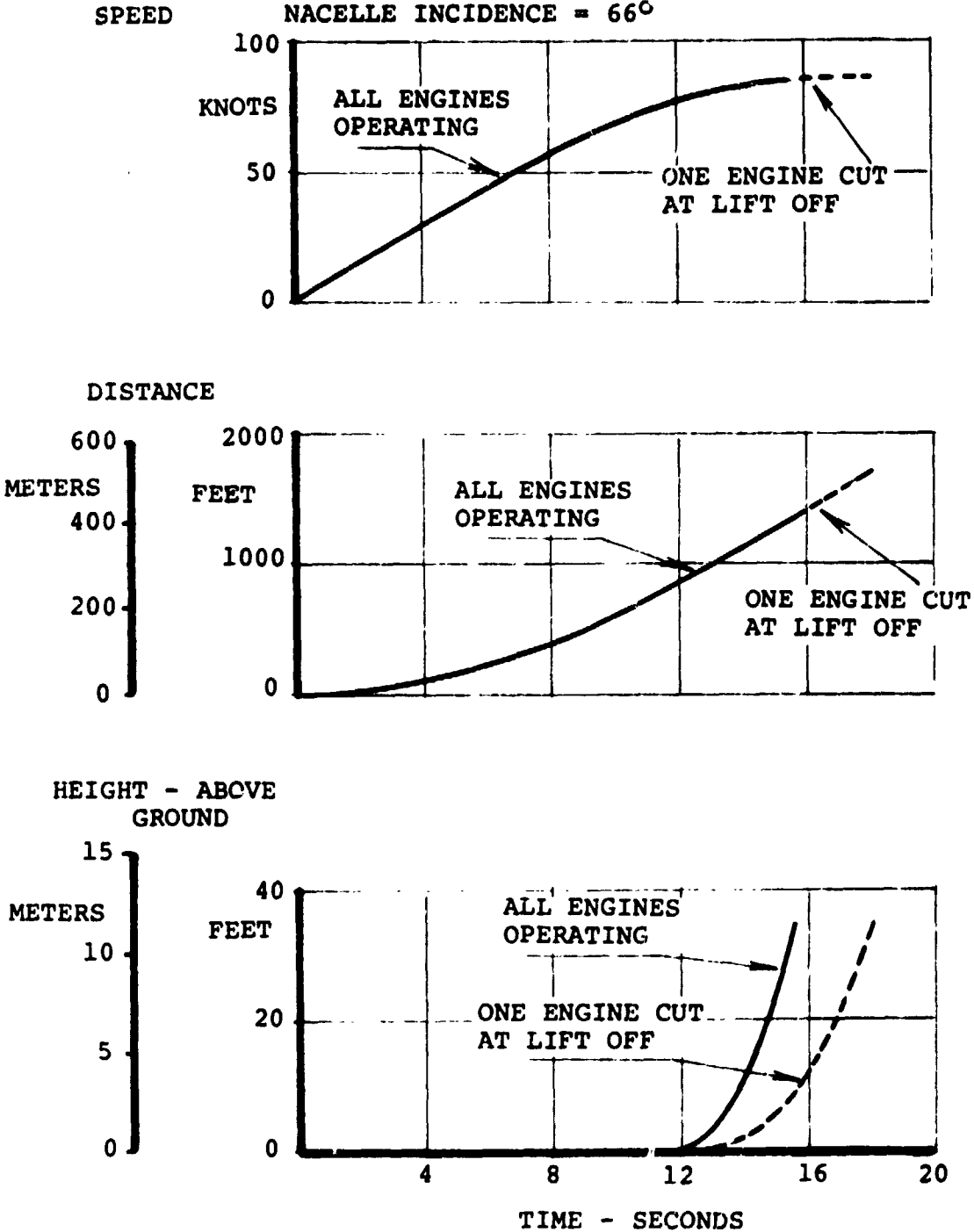
FIGURE 3.5.   TAKEOFF TIME HISTORY.

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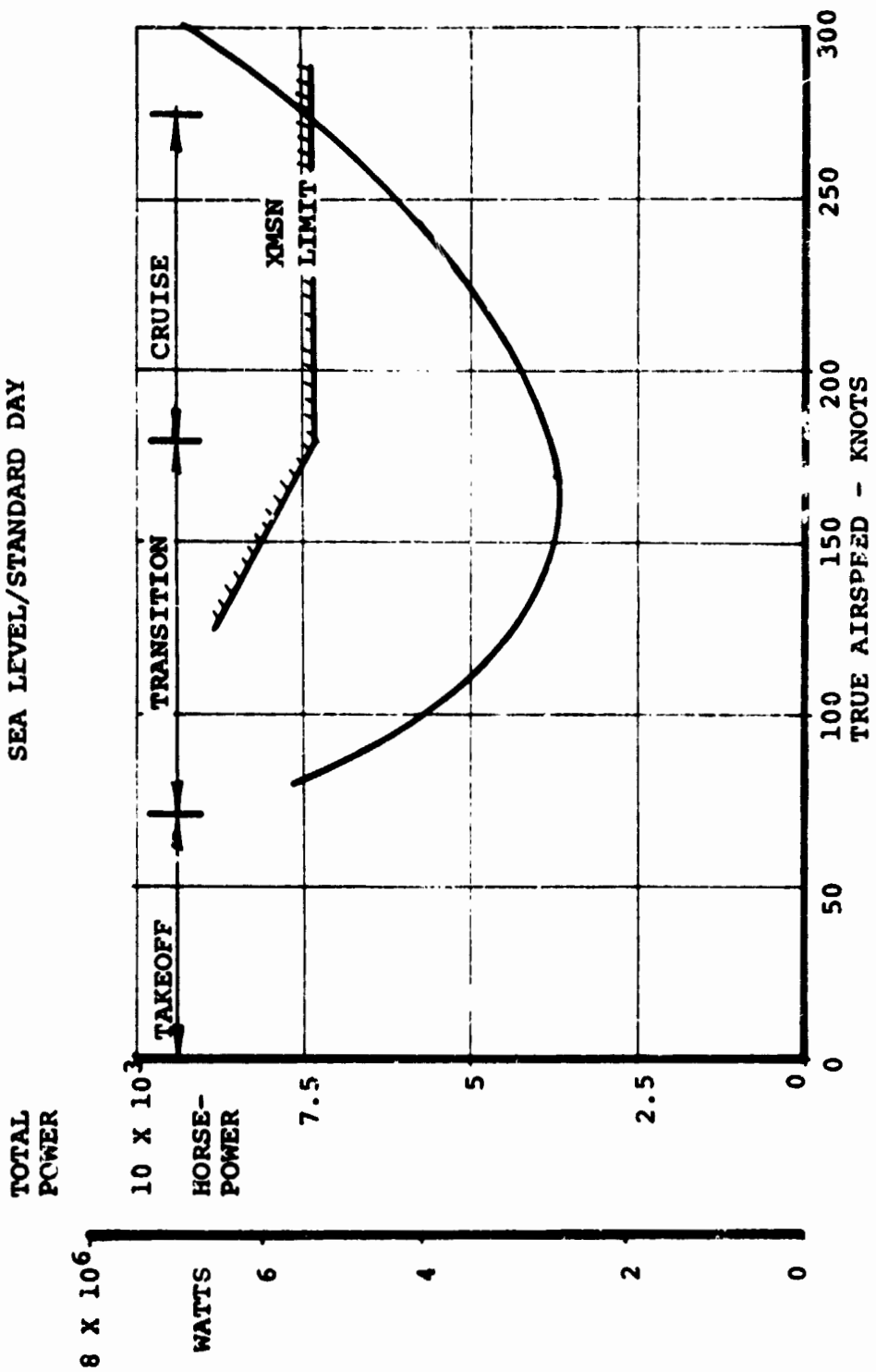
FIGURE 3.6.   POWER REQUIRED FOR EQUILIBRIUM FLIGHT IN TRANSITION AND CRUISE.

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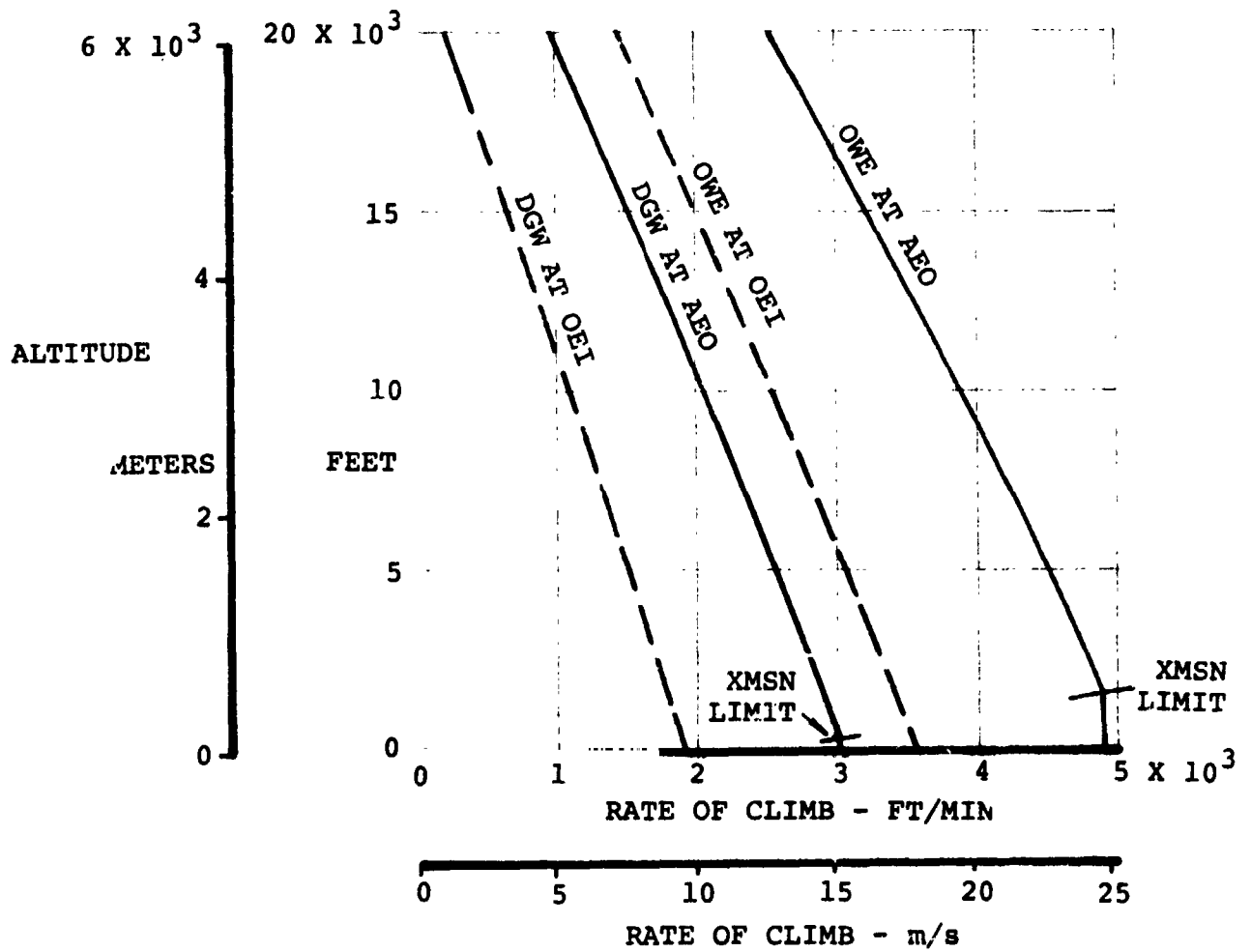
FIGURE 3.7. CLIMB PERFORMANCE - ALL ENGINES OPERATING
AND ONE ENGINE INOPERATIVE.

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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 alt.tude 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 Cruise Performance

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).

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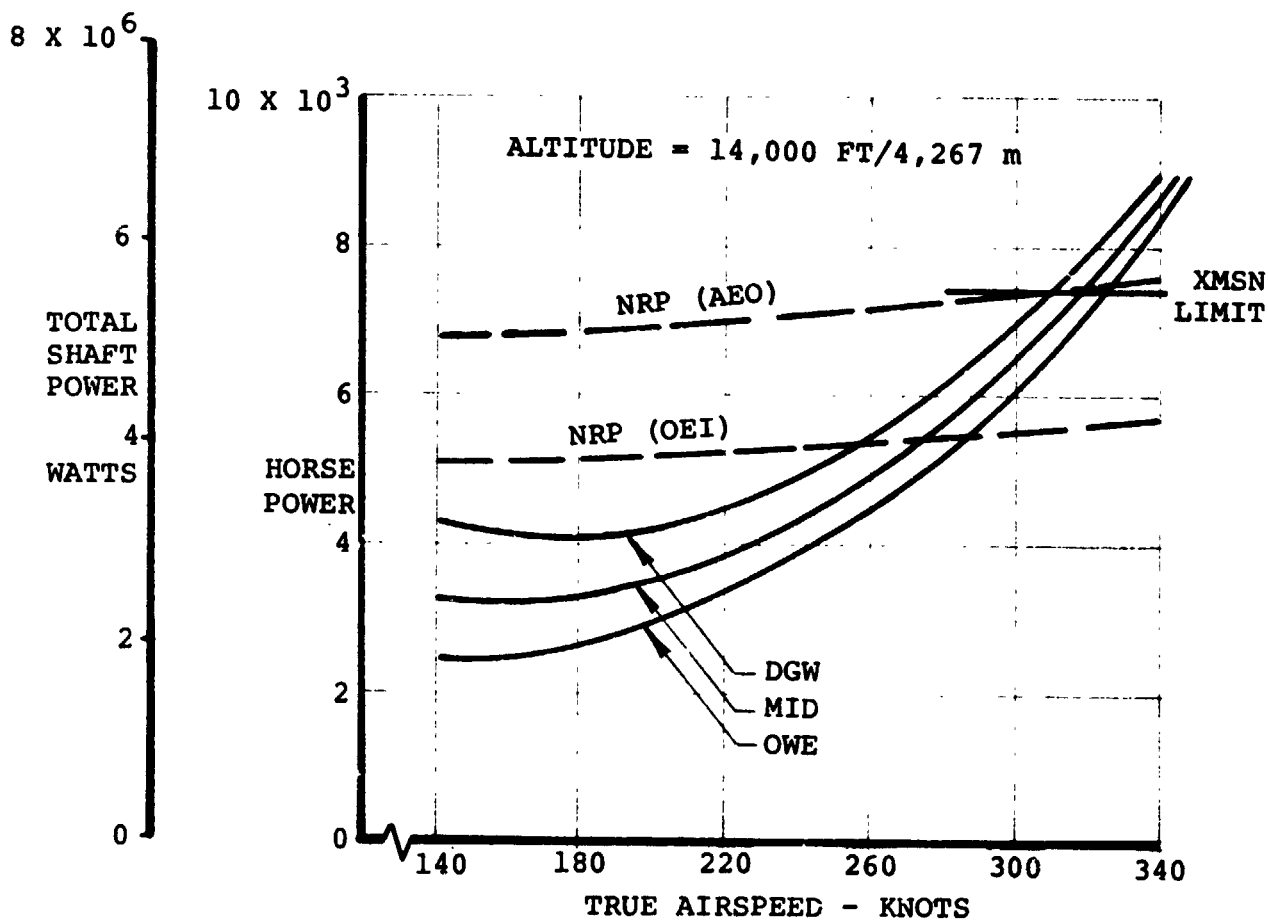
FIGURE 3.9.   CRUISE PERFORMANCE - POWER REQUIRED/
                AVAILABLE. (14,000FT.)

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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

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

STANDARD DAY - CRUISE RPM

DGW = 68,493 LBS/31,068 Kg
OWE = 47,068 LBS/21,350 Kg
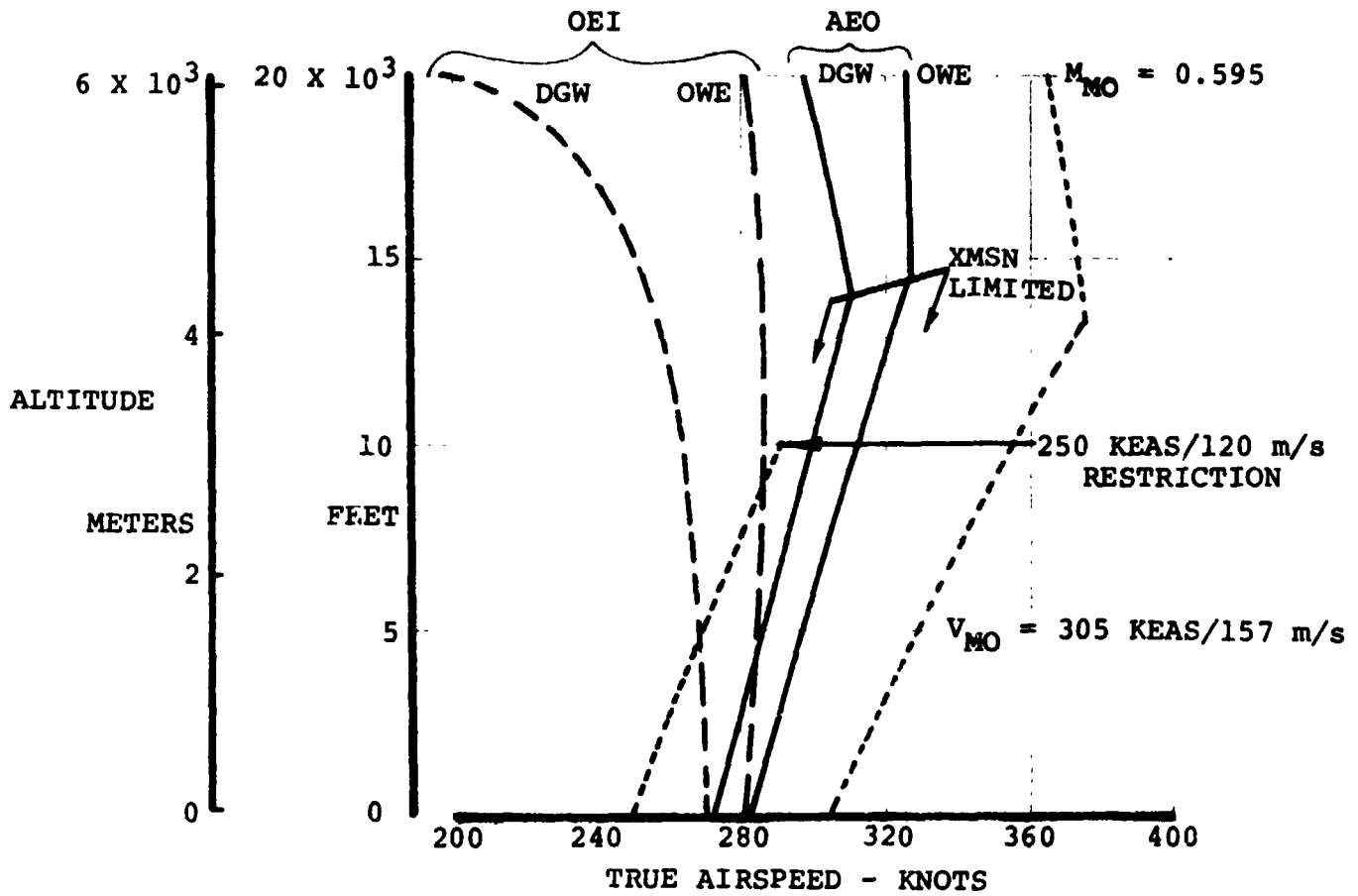
FIGURE 3.10. LEVEL FLIGHT CRUISE SPEED ENVELOPES.

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
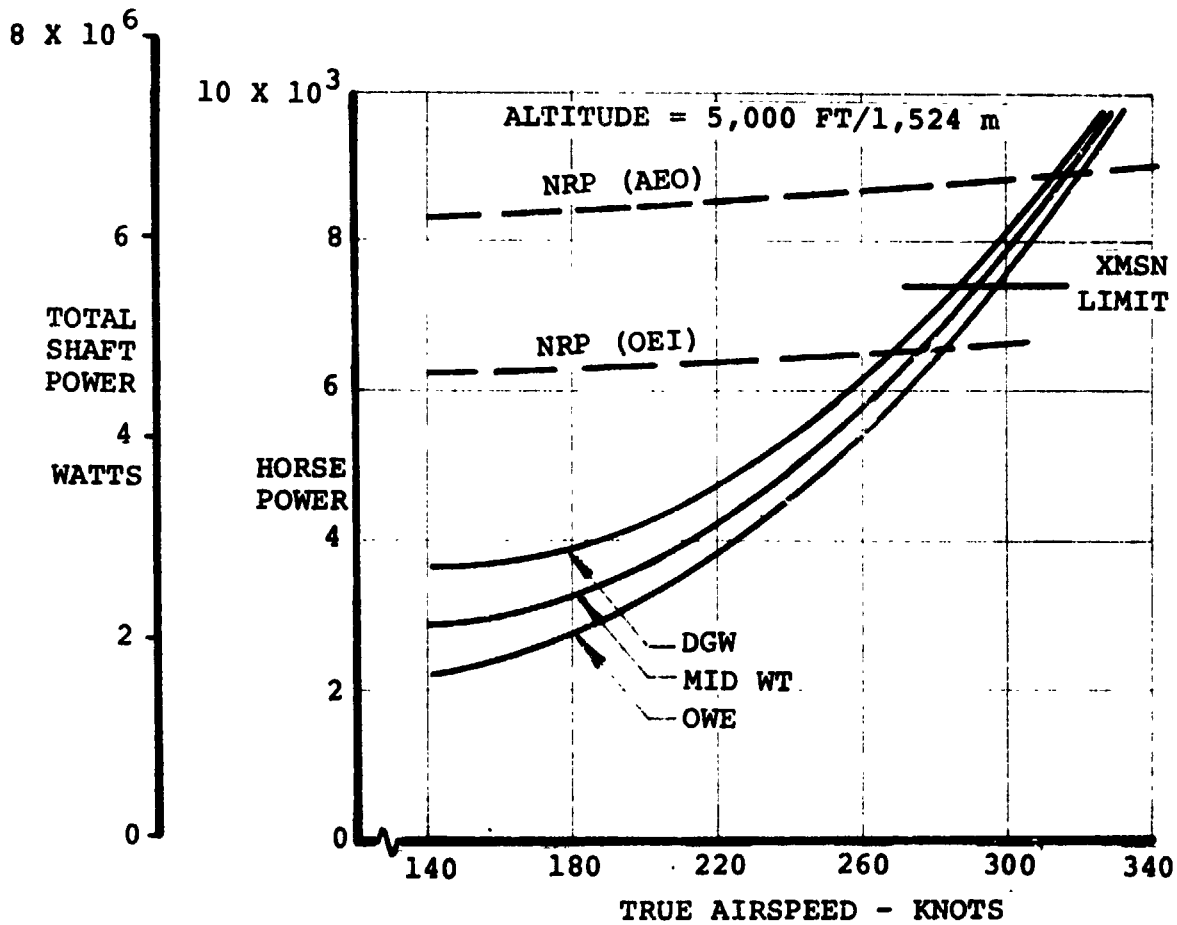MID WT = 58,000 LBS/26,309 Kg
OWE = 47,068 LBS/21,350 Kg
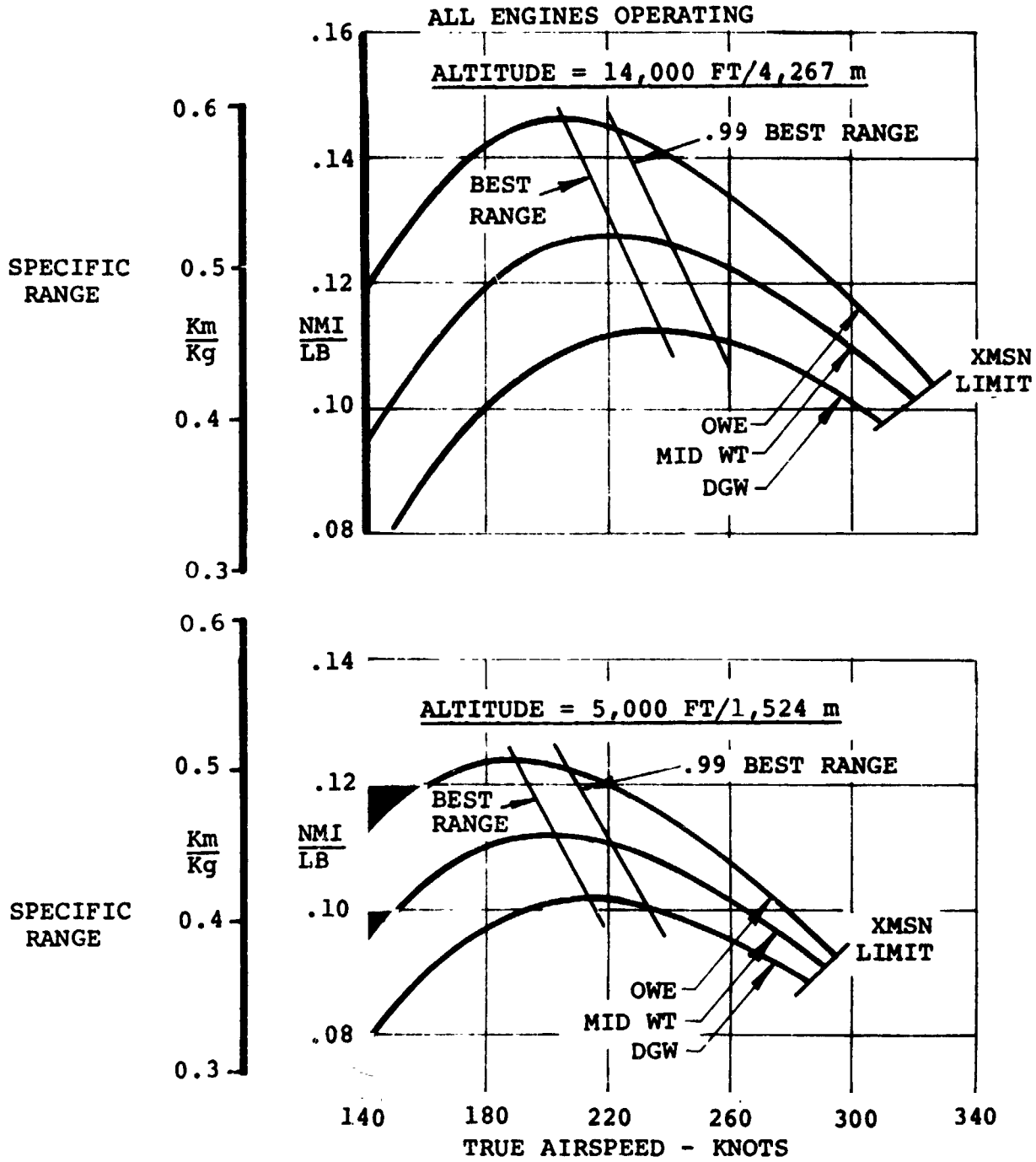
FIGURE 3.11.   CRUISE PERFORMANCE - SPECIFIC RANGE. (AEO)

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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

ONE ENGINE INOPERATIVE
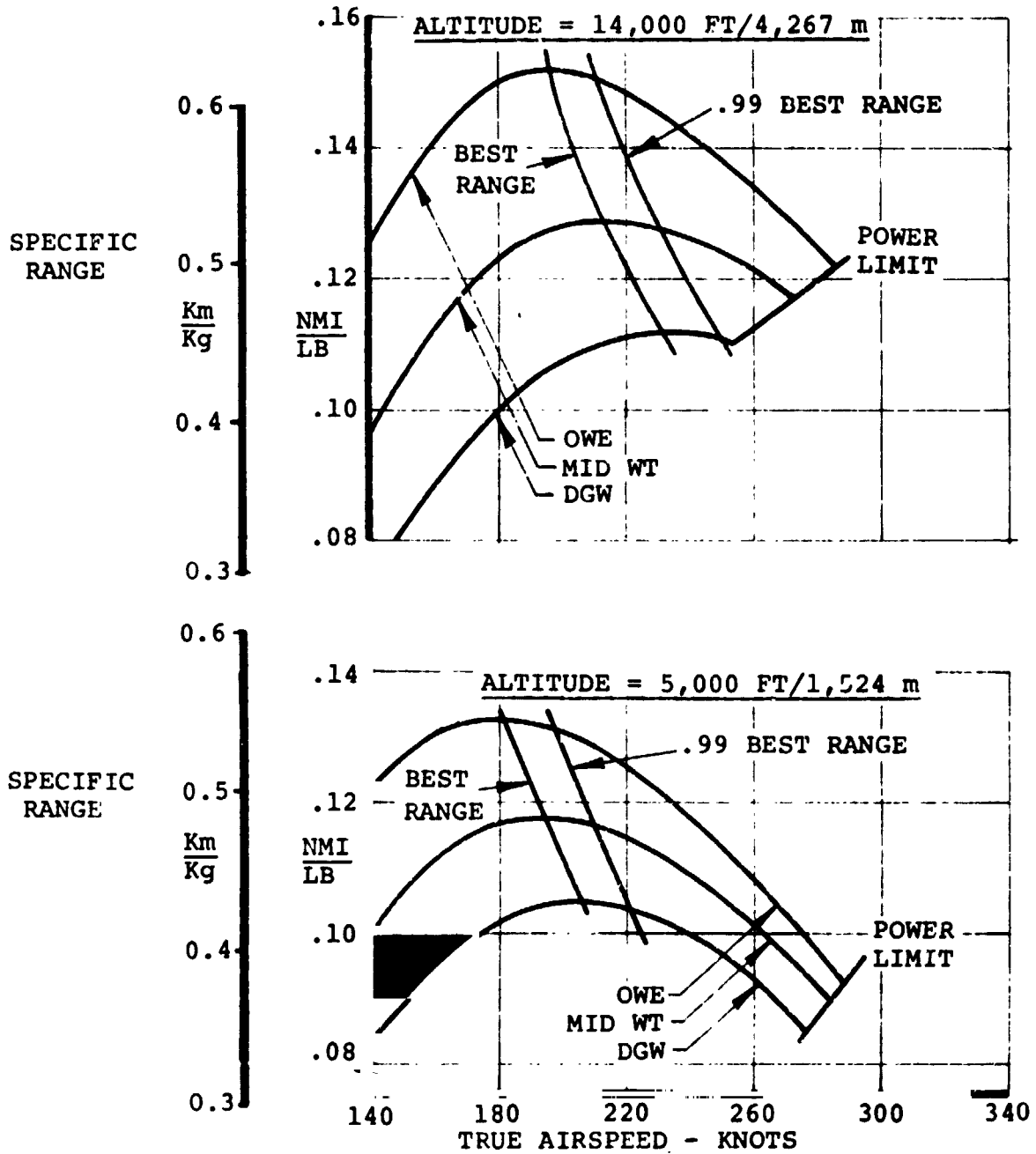
FIGURE 3.12.   CRUISE PERFORMANCE - SPECIFIC RANGE. (OEI)

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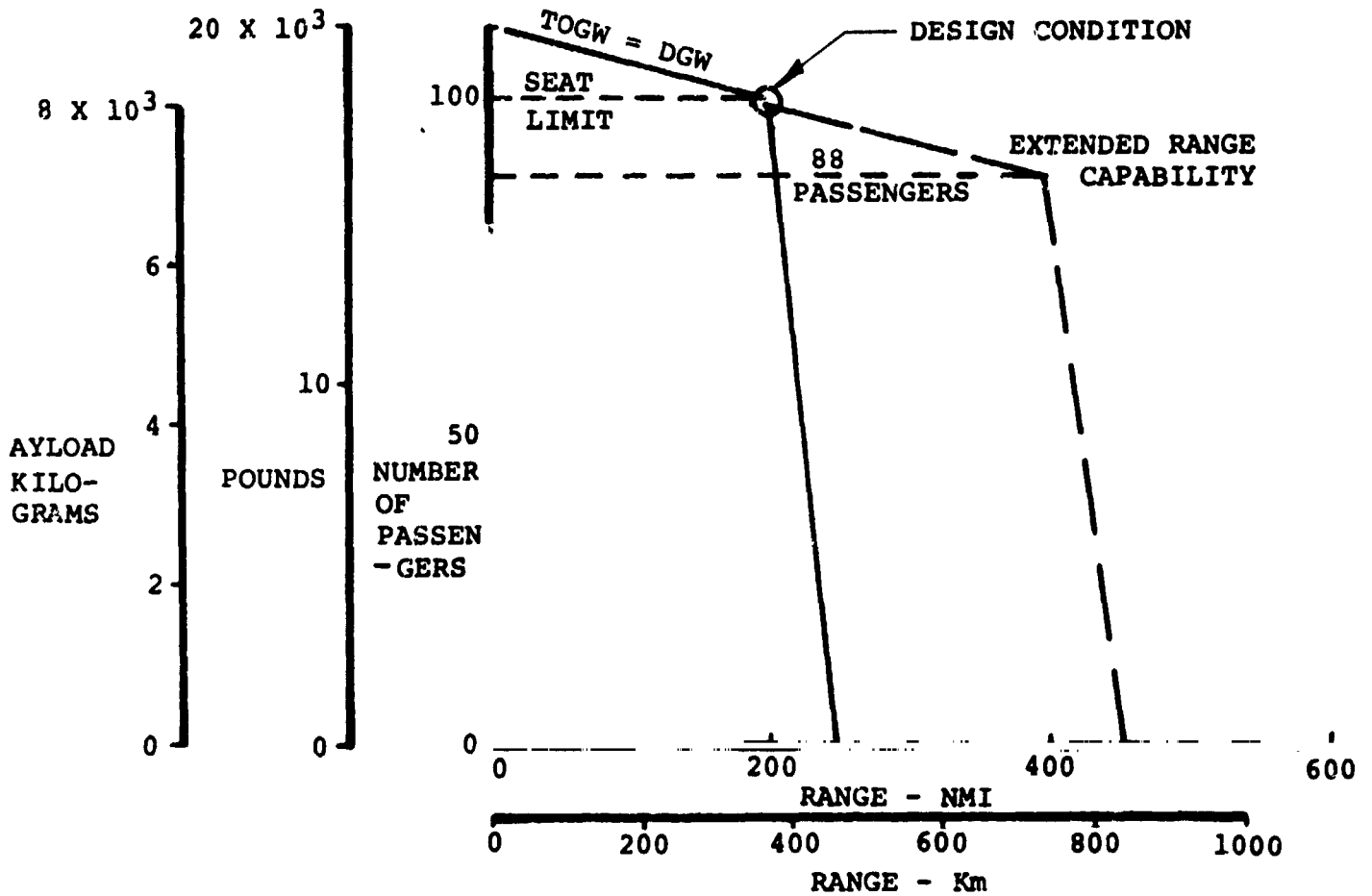
FIGURE 3.13.   PAYLOAD RANGE CAPABILITY (ALL ENGINES
OPERATING).

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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 temper-
ature 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
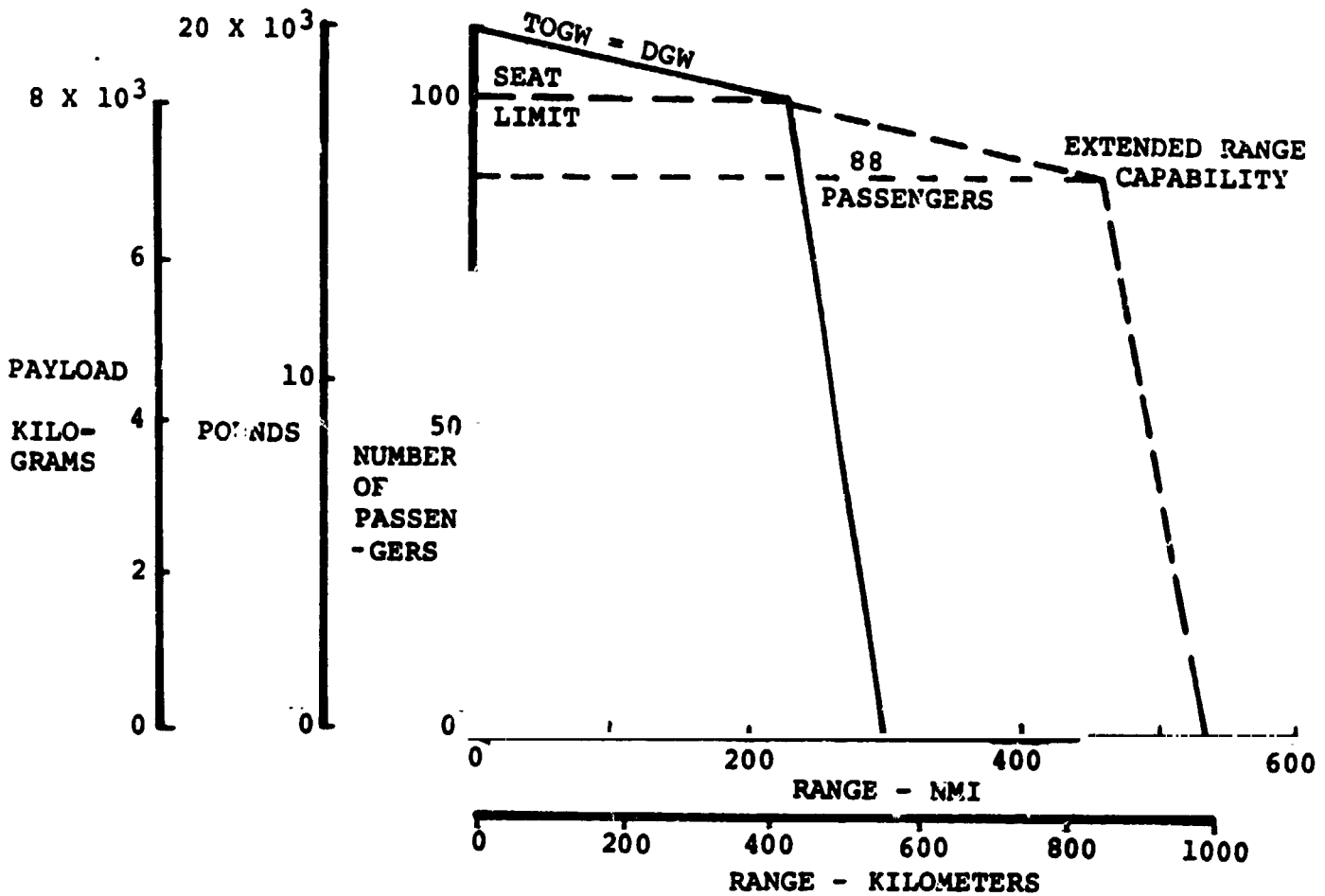
FIGURE 3.14.   PAYLOAD - RANGE CAPABILITY - OEI.

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1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
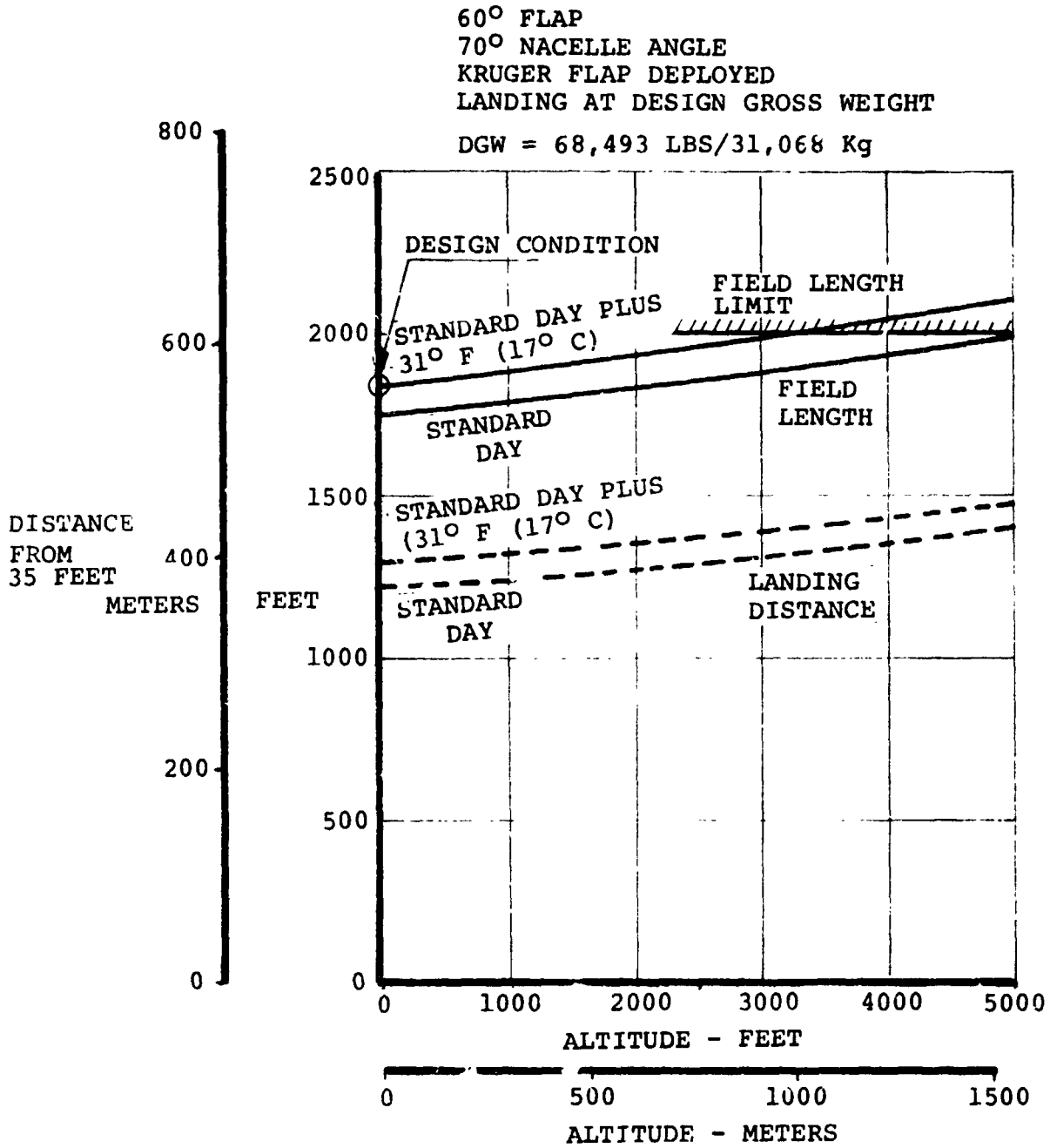
FIGURE 3.15.   EFFECT OF ALTITUDE AND AMBIENT TEMPERATURE ON LANDING
              PERFORMANCE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

60° FLAP
70° NACELLE ANGLE
KRUGER FLAP DEPLOYED
LANDING AT S.L./90° F (32° C)
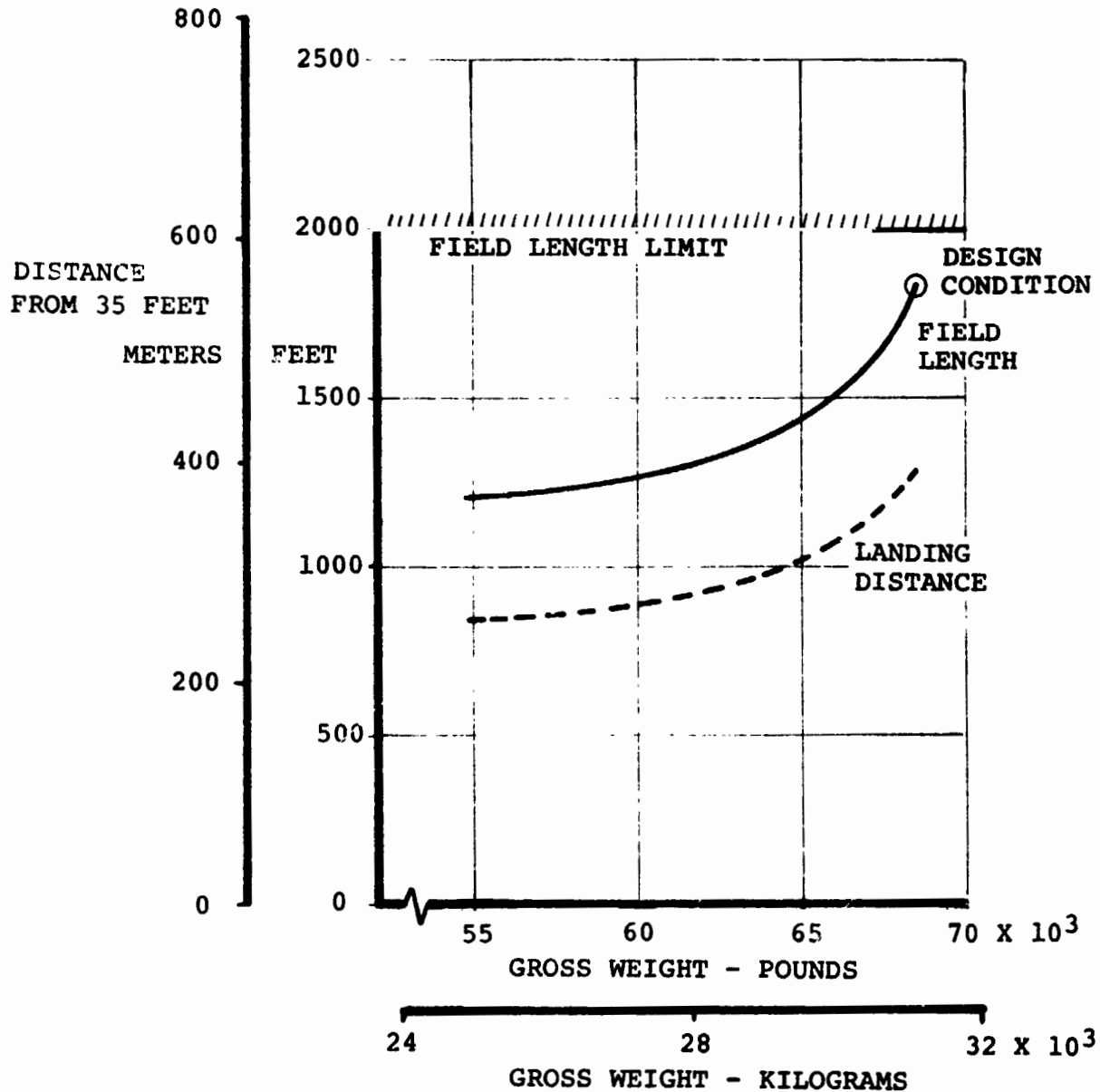
FIGURE 3.16. EFFECT OF GROSS WEIGHT ON LANDING PERFORMANCE.

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)
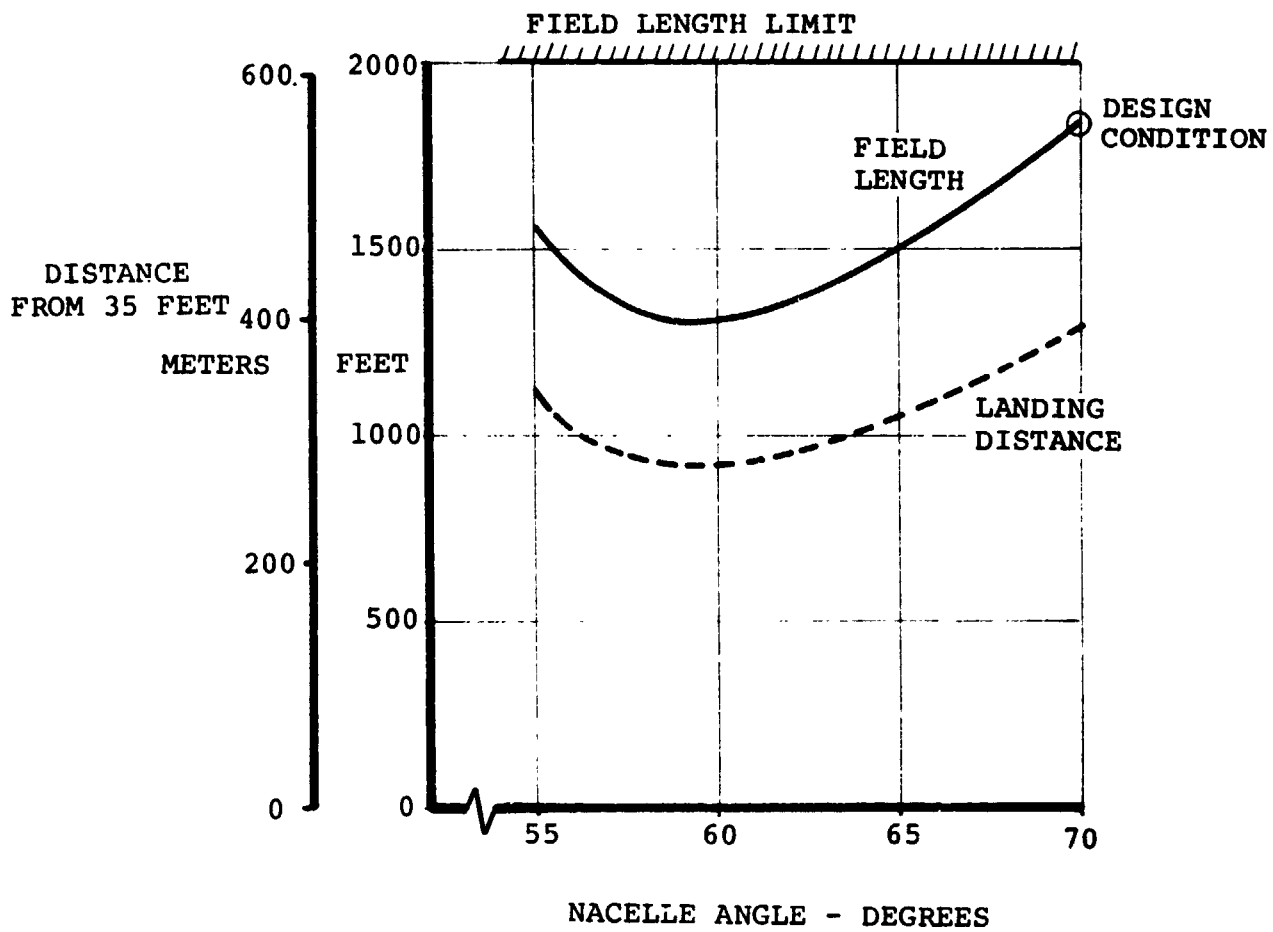
FIGURE 3.17. EFFECT OF NACELLE ANGLE ON LANDING PERFORMANCE.

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 perfor-
mance, 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 cn 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	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 OF
MERIT

$$\frac{.707 \ C_T^{3/2}}{C_p}$$

POWER
COEFFICIENT

$P/\rho A V_T^3$

THRUST COEFFICIENT - $T/\rho A V_T^2$
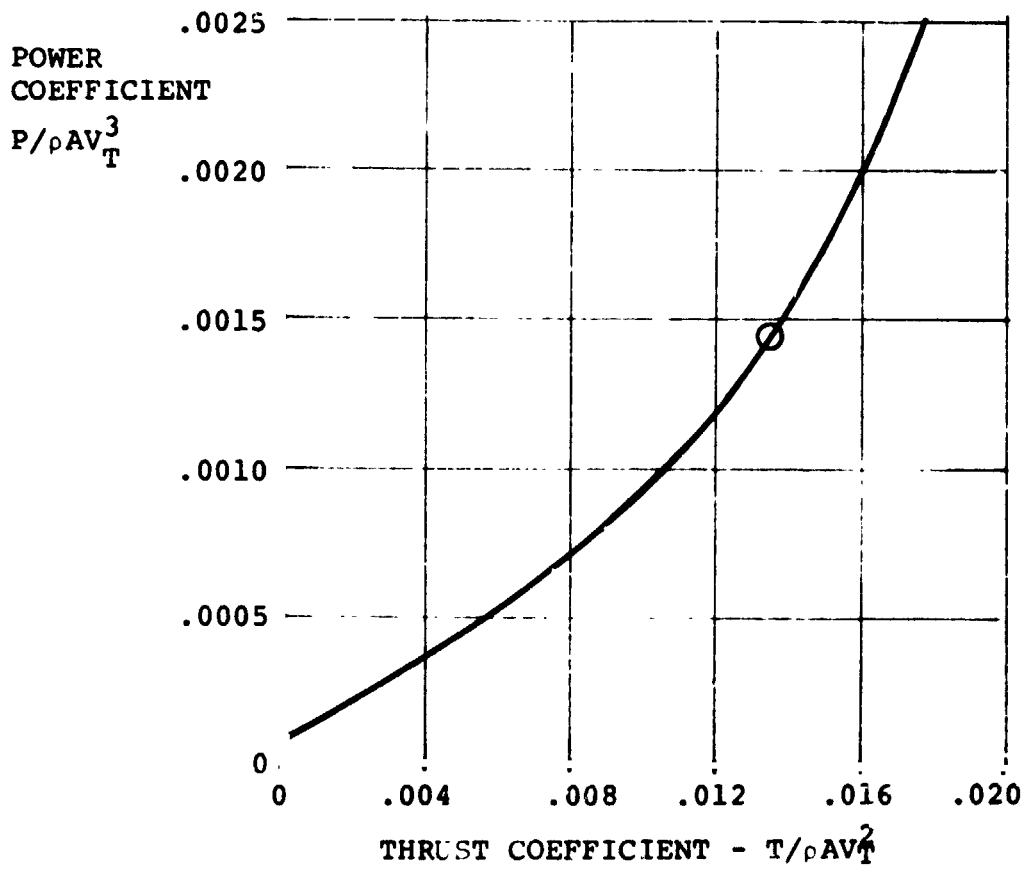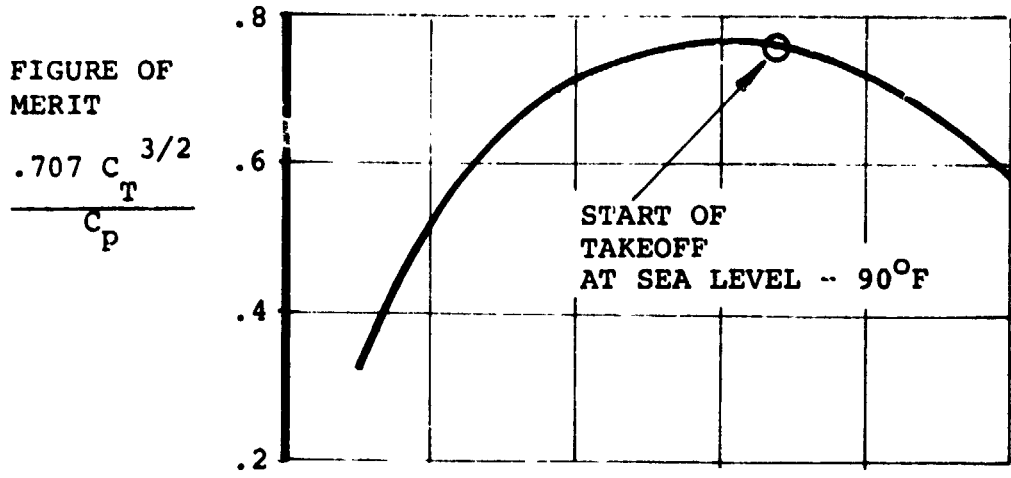
FIGURE 3.19. STATIC ROTOR PERFORMANCE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
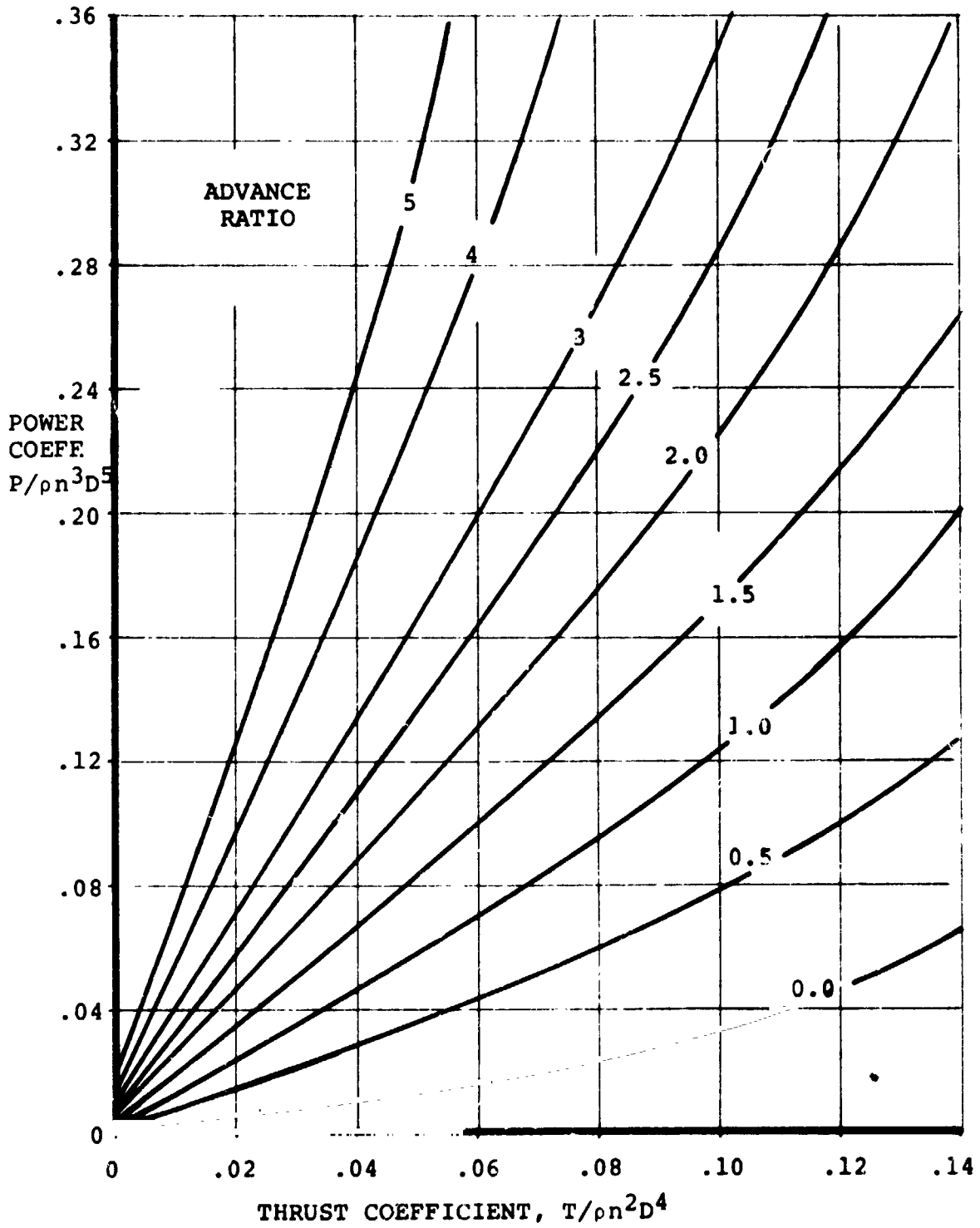
FIGURE 3.20. CRUISE PERFORMANCE OF ROTOR.

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.

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 att .dant plus gear weighs

140 pounds.

5. The aircraft shall be equipped with an APU to

meet the needs of starting, gr und air condi-

tioning and heating.

6. The aircraft designs are to be based on a 1985

operational time period. The Contractor shall

assume t.ae 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 .... be as provided

in Table 3.4, Tilt Roto. Weights Guideli. .

Fly-By-Wire Control Systems

Fly-by-wire control systems are permitted. .ntrol 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
                        TRIPLE
CREW SEATS: CABIN CREW
                        FLIGHT CREW
	
	16 LB/PASSENGER
	16 LB/PASSENGER
	16 LB/CREW MEMBER
	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 instru-
mentation, electrical, electronics and auxiliary power unit
installation has been assumed to be the uninstalled weight
and an addition 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 commer-
cial 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 recommen-
ded 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

D210-10873-1

| | 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 % MAC WATER LINE	13.14 M(517.4 IN.) 3.96 M(156.0 IN.)	13.13 M(517.1 IN.) 31.2 3.43 M(135.1 IN.)	13.03 M(513.0 IN.) 23.3 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

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
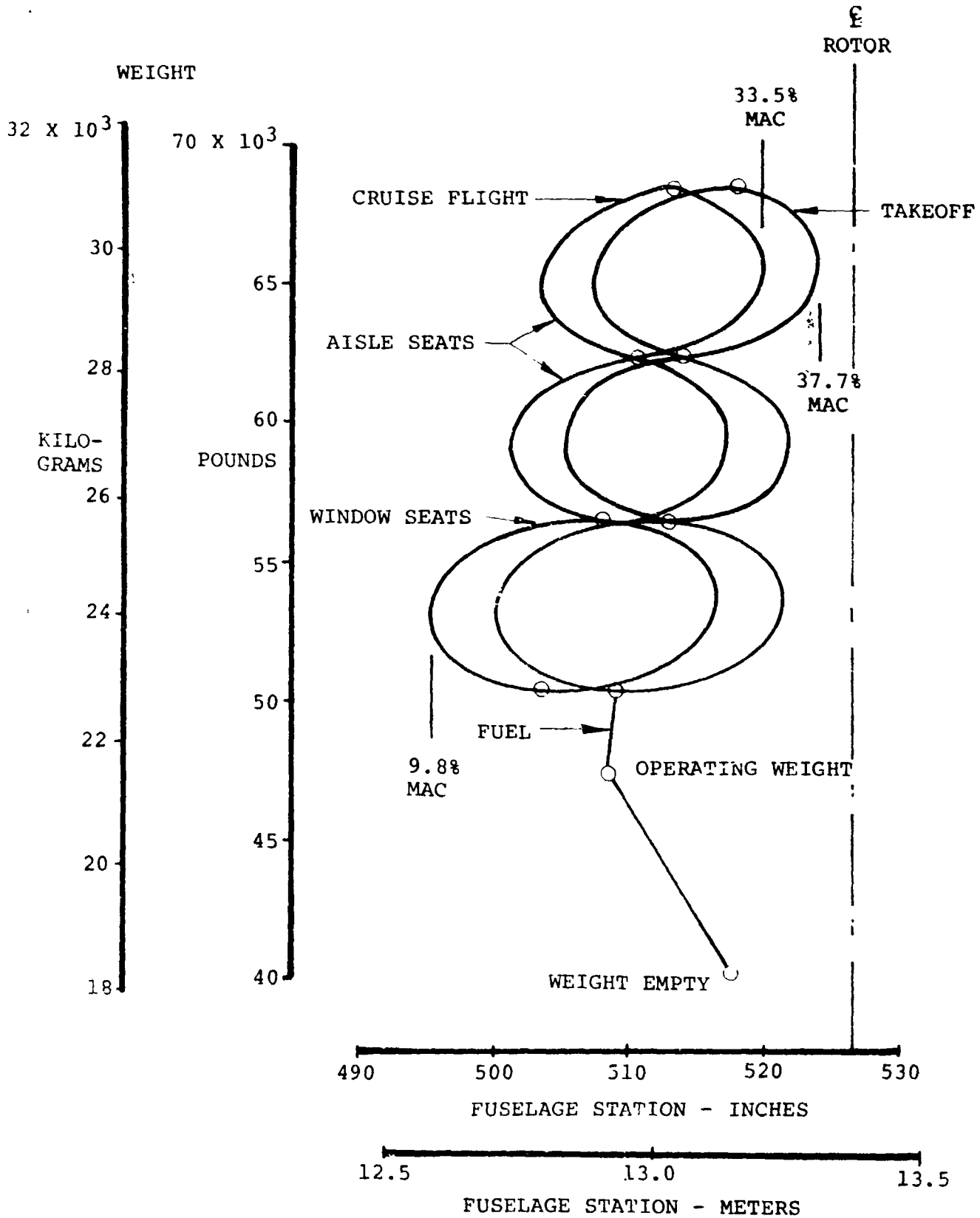
FIGURE 3.21.   CENTER OF GRAVITY ENVELOPE.

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1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
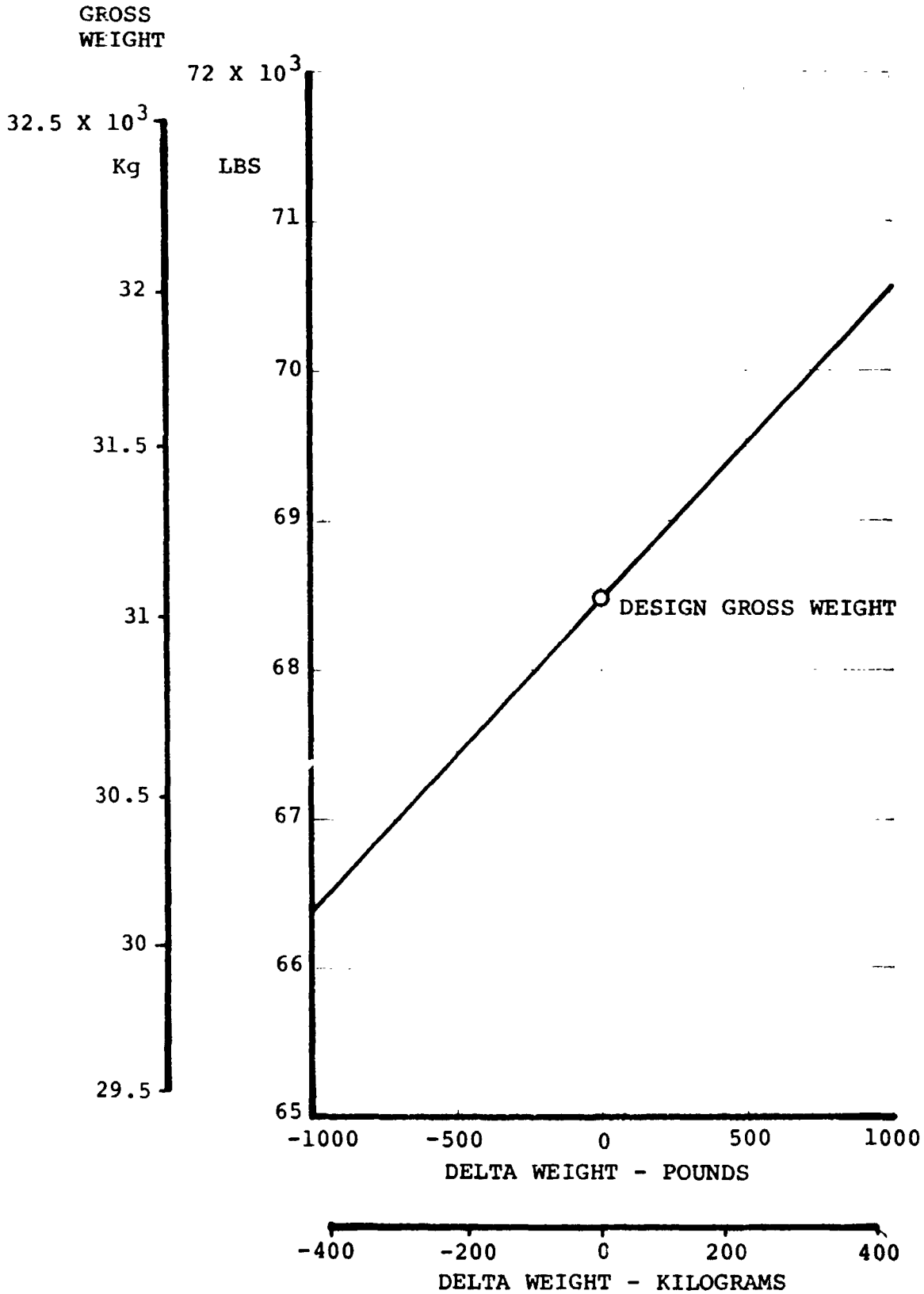
FIGURE 3.22. WEIGHT GROWTH AT CONSTANT PERFORMANCE AND STRENGTH.

74

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 '00 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.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

GROSS WEIGHT = 68,493 LBS/31,068 Kg
TRANSITION AT SL/STD
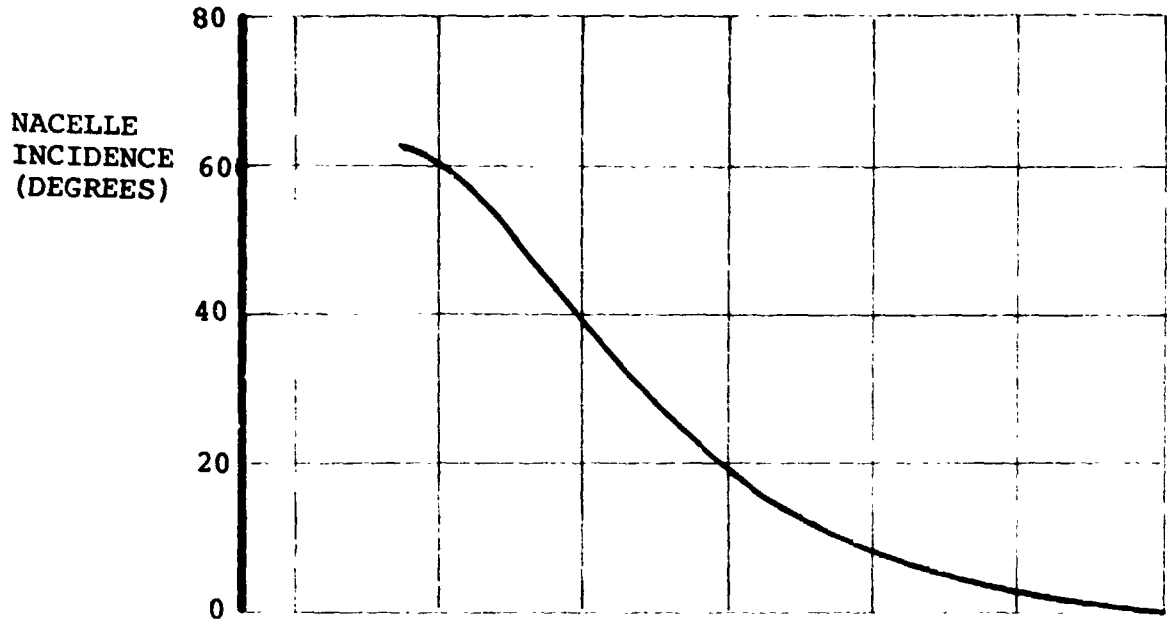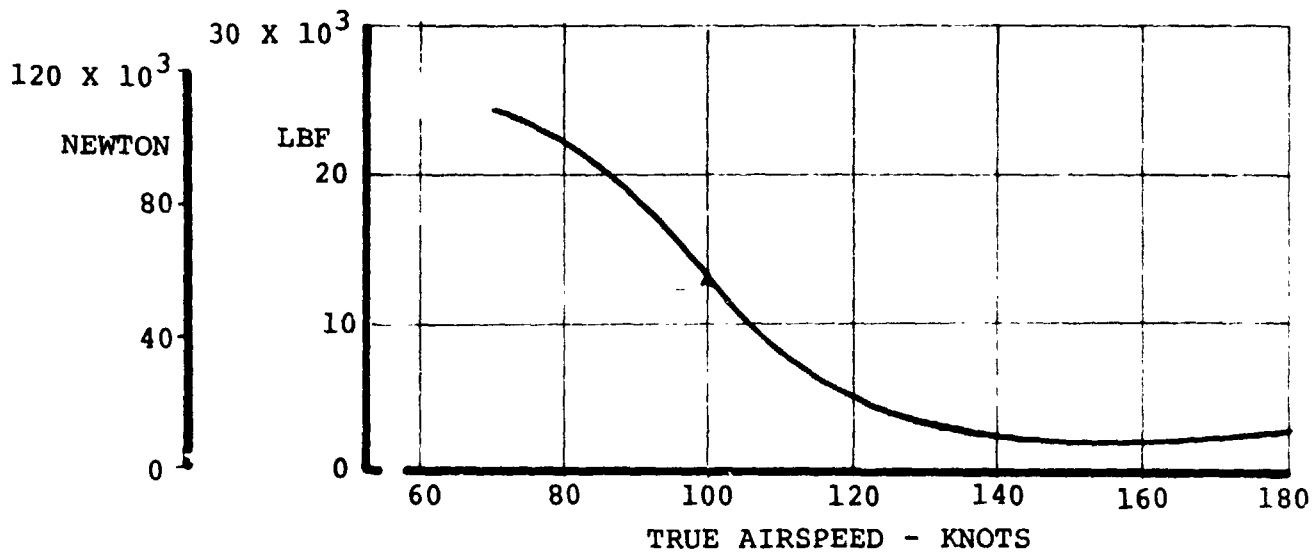
FIGURE 3.23. TYPICAL TRIM CHARACTERISTICS IN TRANSITION -
THRUST AND NACELLE INCIDENCE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

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

TRANSITION AT SL/STD
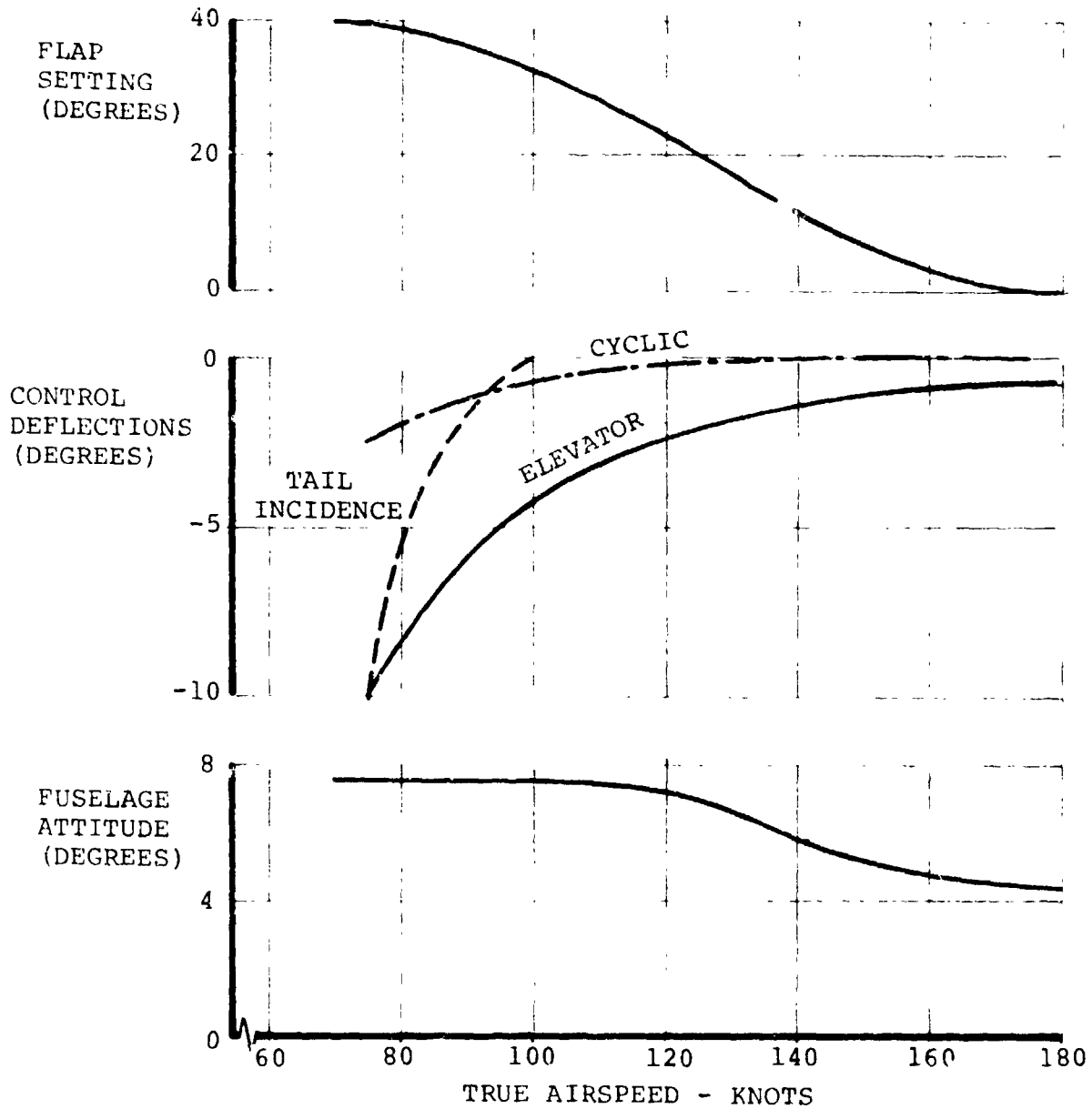
FIGURE 3.24. TYPICAL TRIM CHARACTERISTICS IN TRANSITION - CONTROL DEFLECTIONS AND FUSELAGE ATTITUDE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

GROSS WEIGHT = 68,493 LBS/31,068 Kg
TRANSITION AT SL/STD
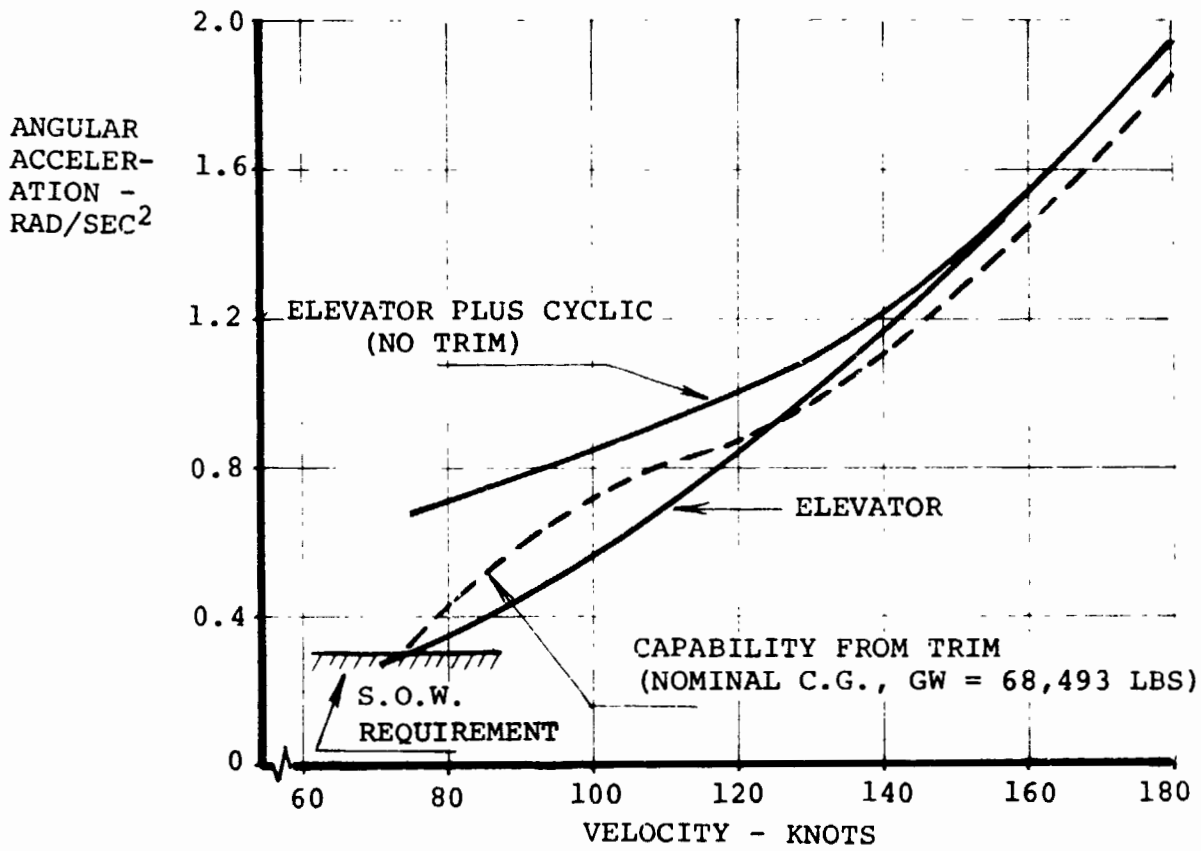
FIGURE 3.25.   PITCH ANGULAR ACCELERATION CAPABILITY IN
TRANSITION.

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1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

GROSS WEIGHT = 68,4º3 LBS/31,068 KG
TRANSITION AT SL/STD

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°

DIFFERENTIAL
COLLECTIVE -
DEGREES ±

NACELLE INCIDENCE - DEGREES

DIFFERENTIAL
LONGITUDINAL
CYCLIC -
DEGREES ±

NACELLE INCIDENCE - DEGREES

YAW ANGULAR
ACCELERATION
CAPABILITY
- RAD/SEC^2

RUDDER PLUS DIFF. CYCLIC
COLLECTIVE

S.O.W.
REQUIREMENT

RUDDER

TRUE AIRSPEED - KNOTS
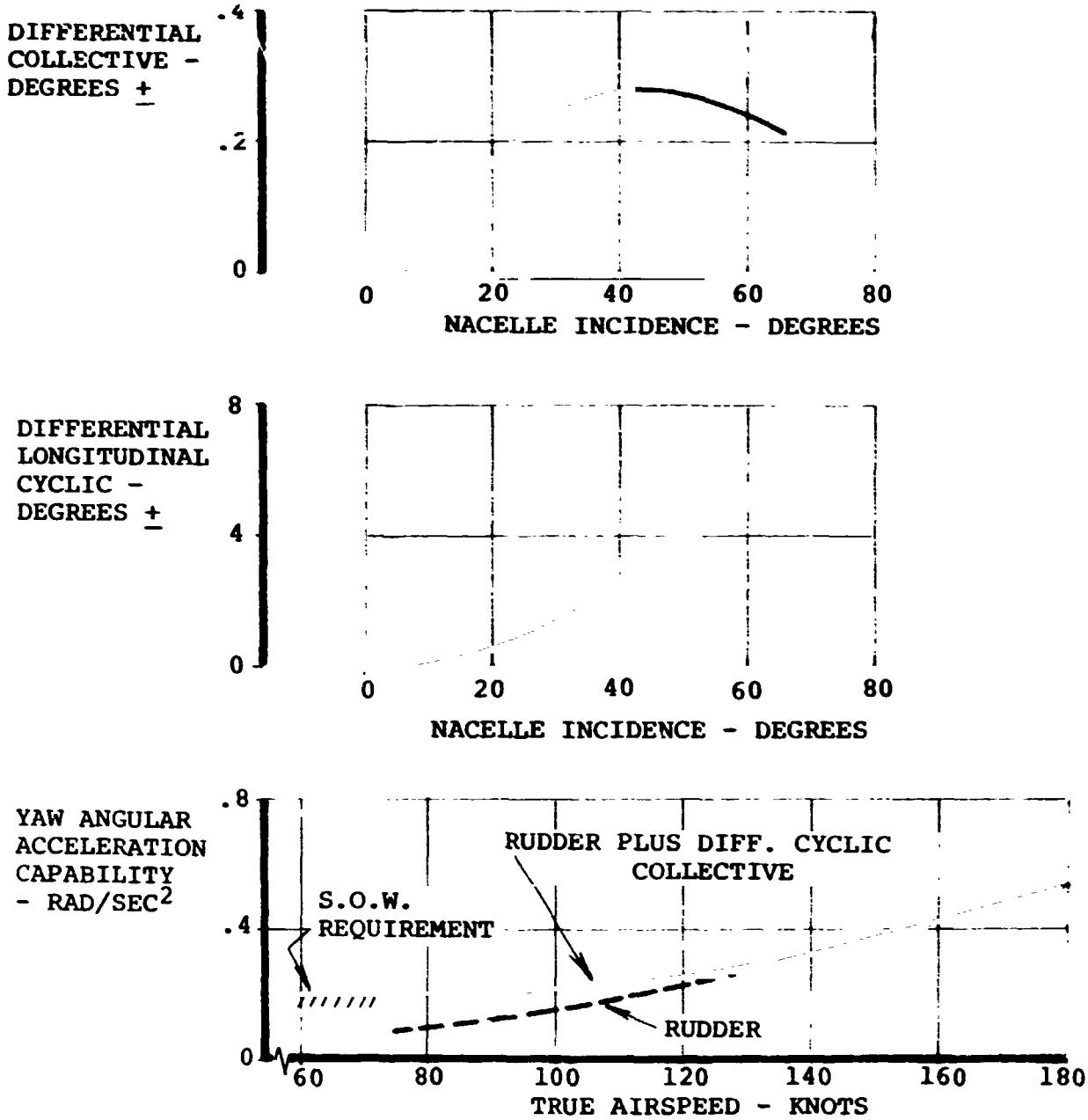
FIGURE 3.27. YAW CONTROL IN TRANSITION.

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.

1985 100 PASSENGER STOL TILT ROTOR AIRCRA..

CRUISE MODE
NACELLE INCIDENCE = 0
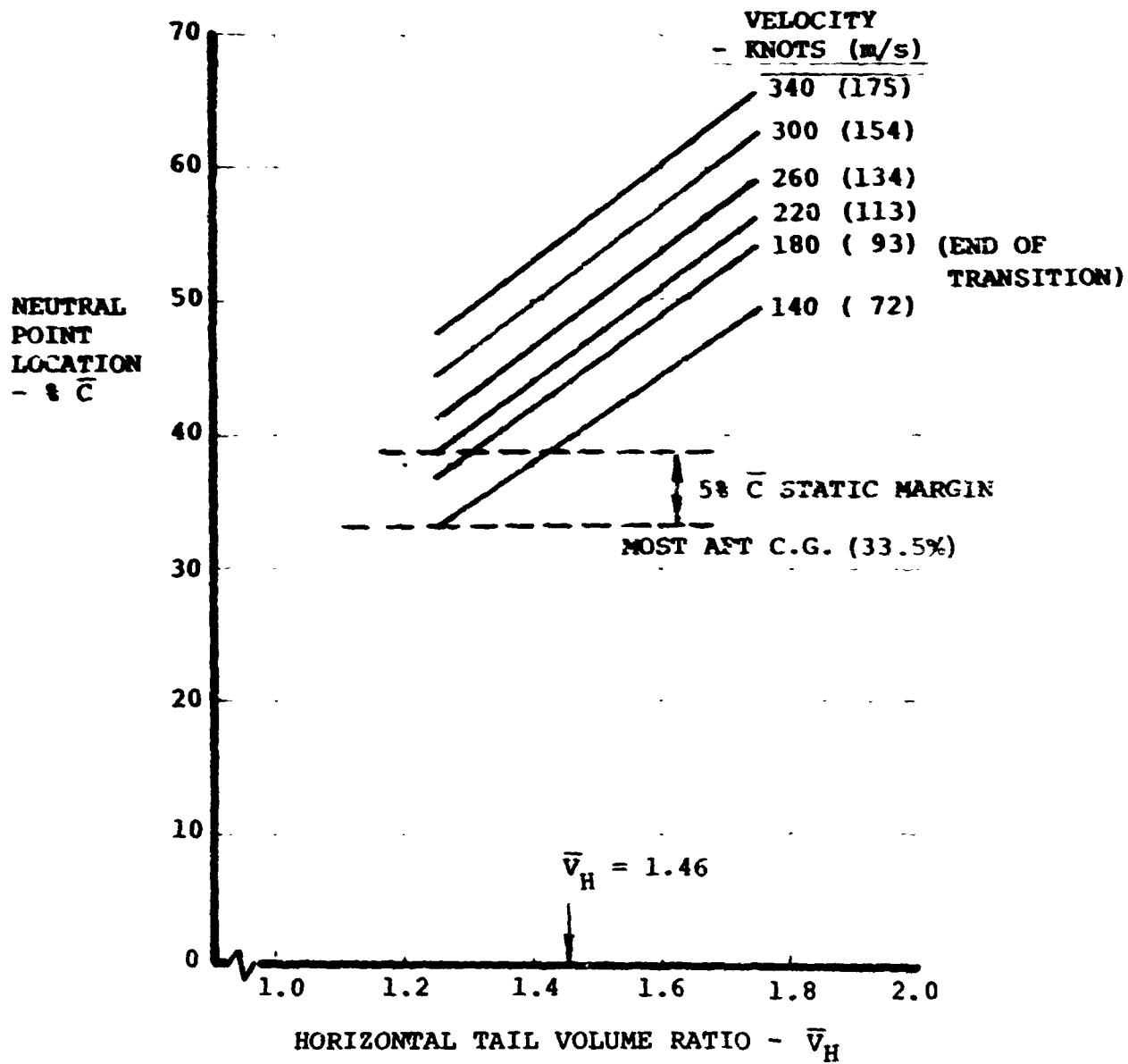
FIGURE 3.28. NEUTRAL POINT LOCATION AS A FUNCTION OF TAIL
VOLUME RATIO AND VELOCITY.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
GROSS WEIGHT = 68,493 LBS/31,068 Kg
CG AT 35% MAC

TRIMMED
FUSELAGE
ATTITUDE
(DEGREES)

— 14,000 FT/4,267 m
— 10,000 FT/3,048 m
— SEA LEVEL

NRP LIMIT

ELEVATOR
ANGLE TO
TRIM
(DEGREES)

NRP LIMIT

— 14,000 FT/4,267 m
— 10,000 FT/3,048 m
— SEA LEVEL

TRUE AIRSPEED - KNOTS
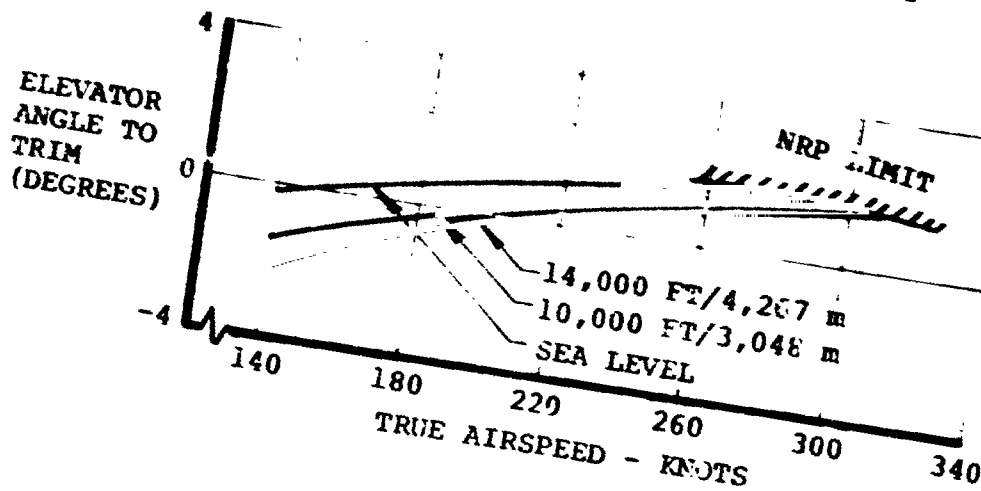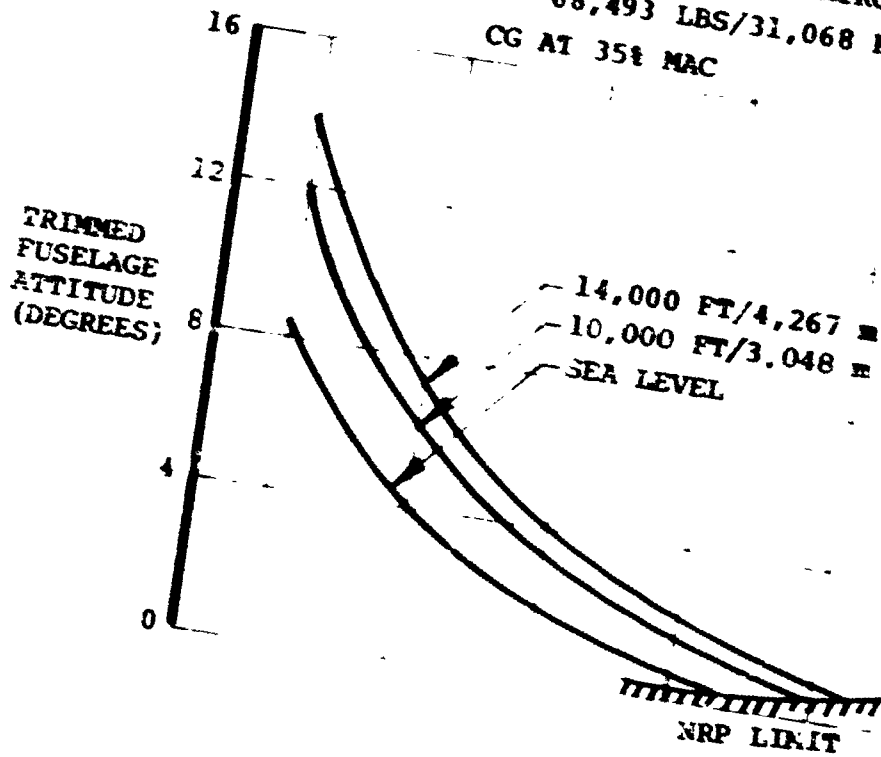
FIGURE 3.29. CRUISE TRIM CHARACTERISTICS - AFT CG.

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1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

GROSS WEIGHT = 47,500 LBS/21,546 Kg

CG AT 10% MAC

TRIMMED
FUSELAGE
ATTITUDE
(DEGREES)

14,000 FT/4,267 m
10,000 FT/3,048 m
SEA LEVEL

ELEVATOR
ANGLE TO
TRIM
(DEGREES)

14,000 FT/4,267 m
10,000 FT/3,048 m
SEA LEVEL

TRUE AIRSPEED - KNOTS

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
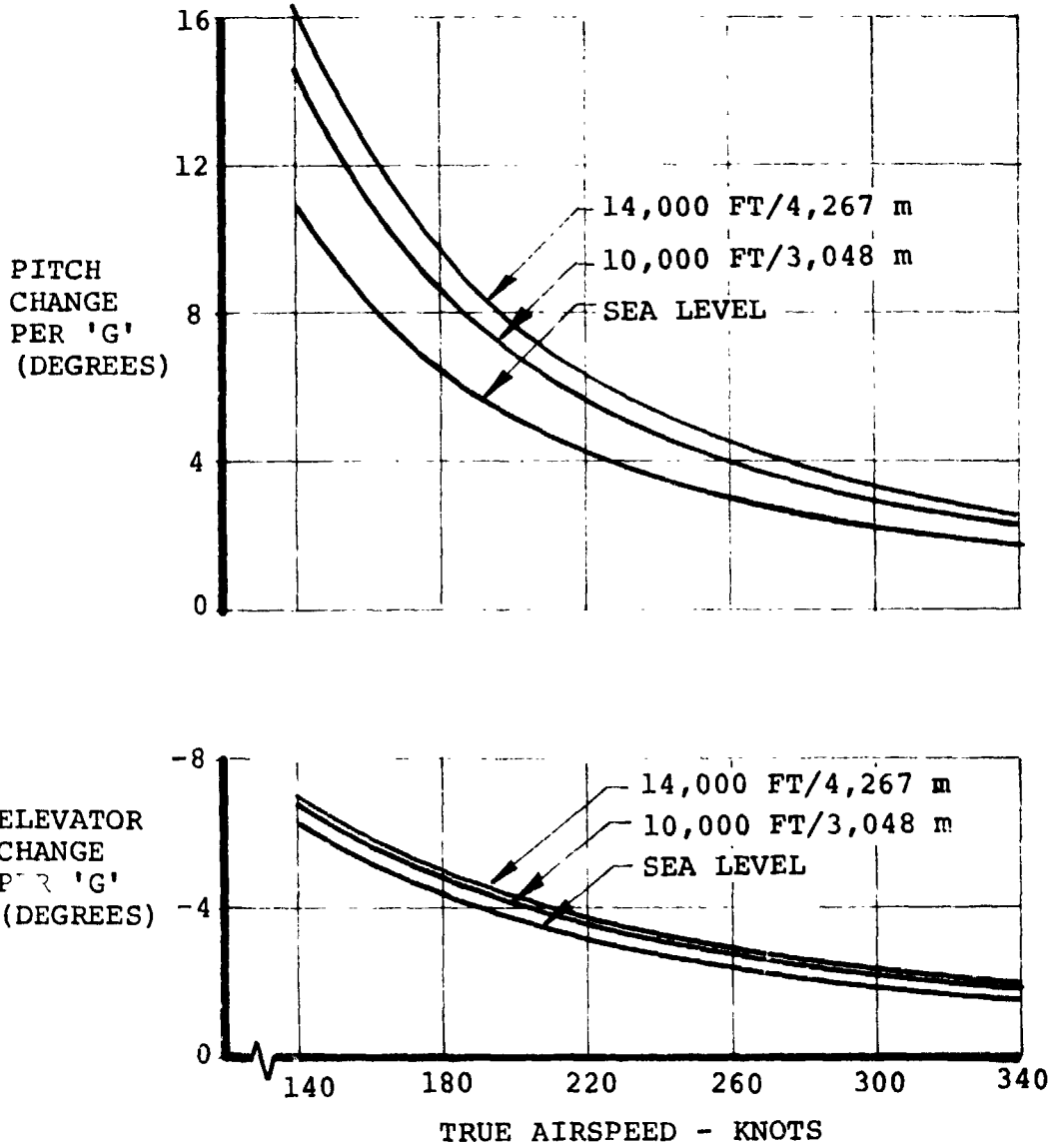
FIGURE 3.31. LONGITUDINAL CONTROL IN CRUISE - AFT CG.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

GROSS WEIGHT = 47,500 LBS/21,546 Kg

CG AT 10% MAC
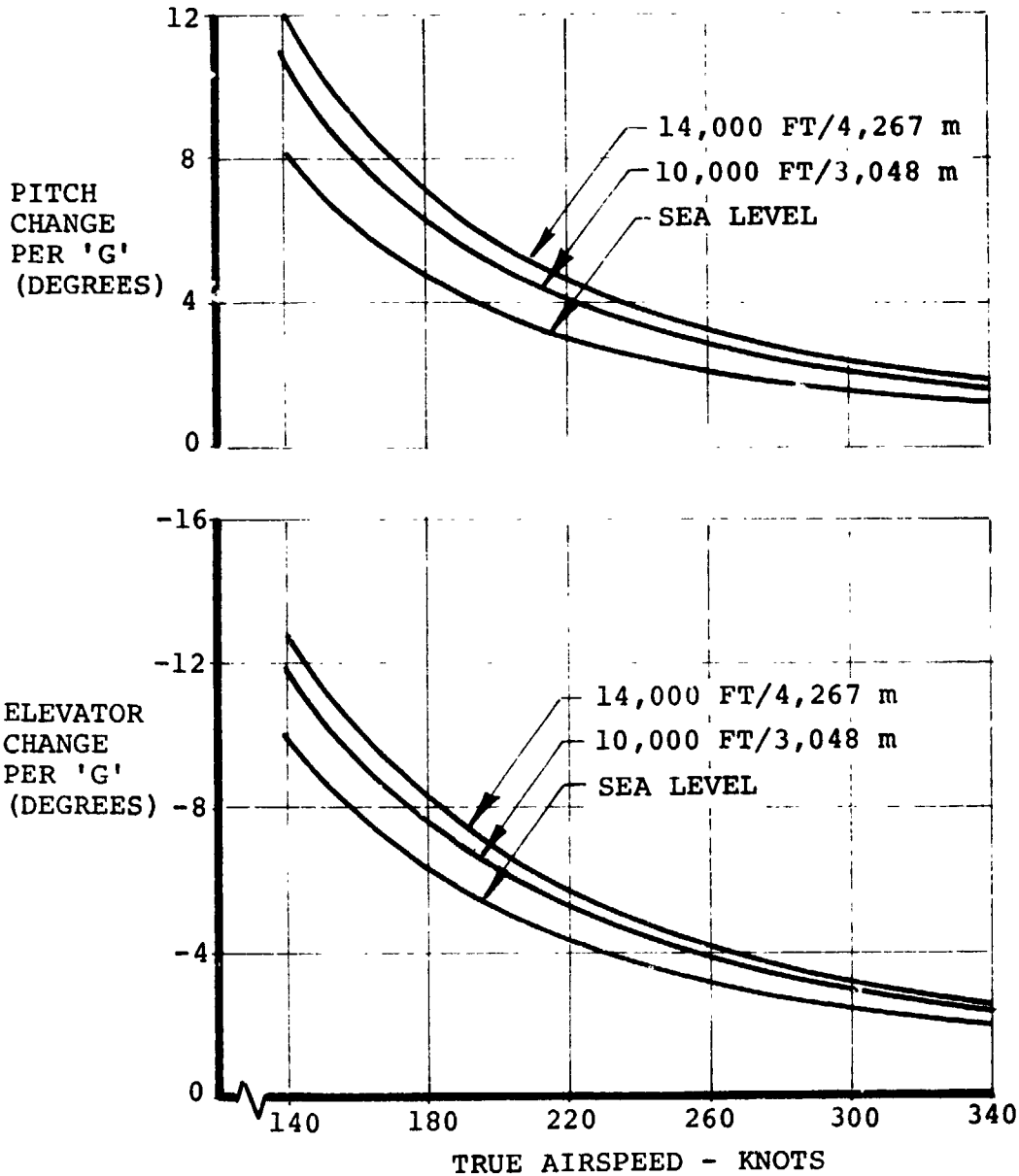
FIGURE 3.32. LONGITUDINAL CONTROL IN CRUISE - FORWARD CG.

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

SYMBOL	SPEED
▲	180 KNOTS
O	300 KNOTS
+	380 KNOTS

FWD CG (10% $\bar{C}$) ———
AFT CG (35% $\bar{C}$) — — —
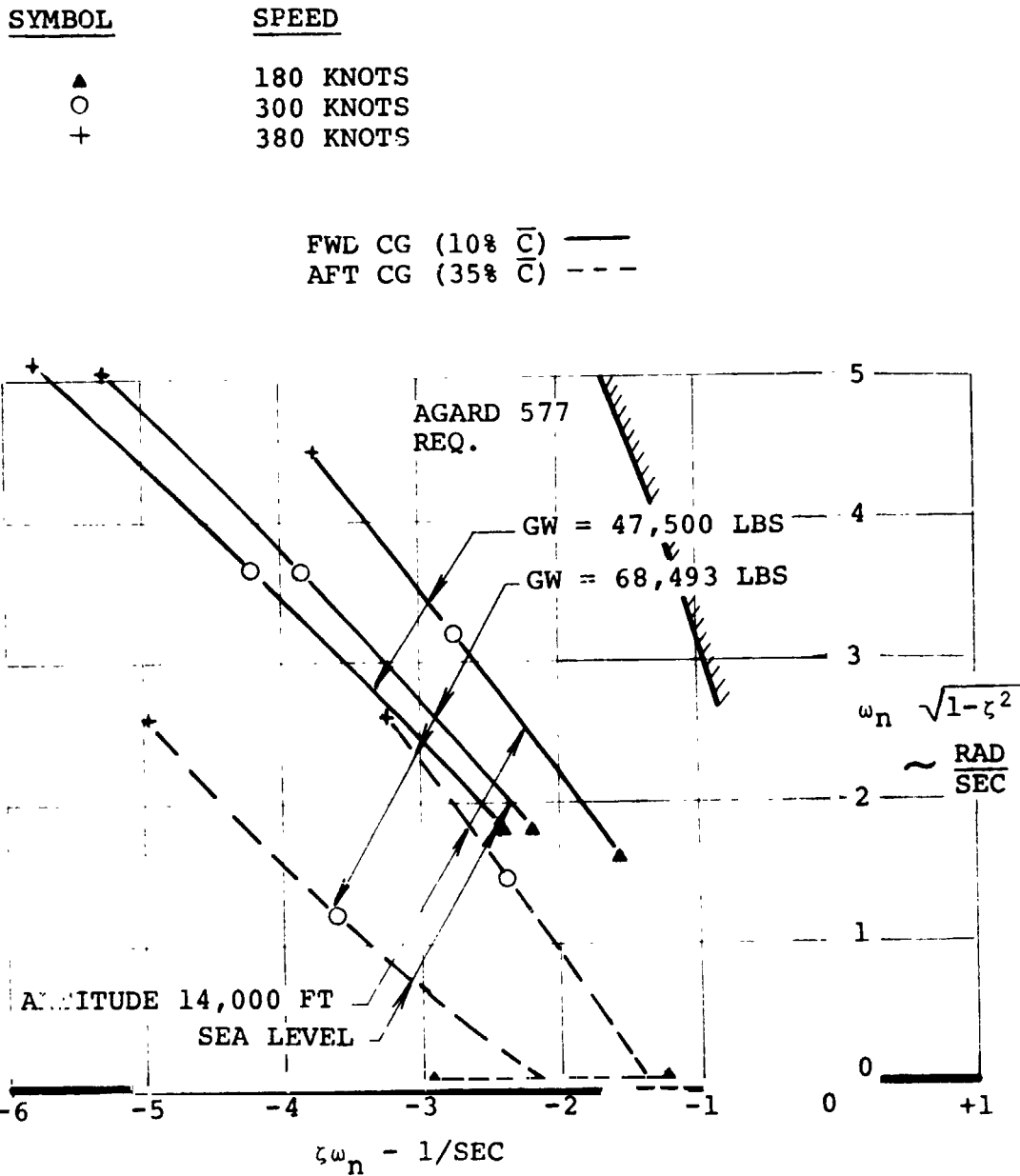
FIGURE 3.33. LONGITUDINAL DYNAMIC RESPONSE CHARACTERISTICS - SHORT PERIOD.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

GROSS WEIGHT = 68,493 LBS/31,068 Kg
C.G. AT 35% MAC
CRUISE AT SL/STD
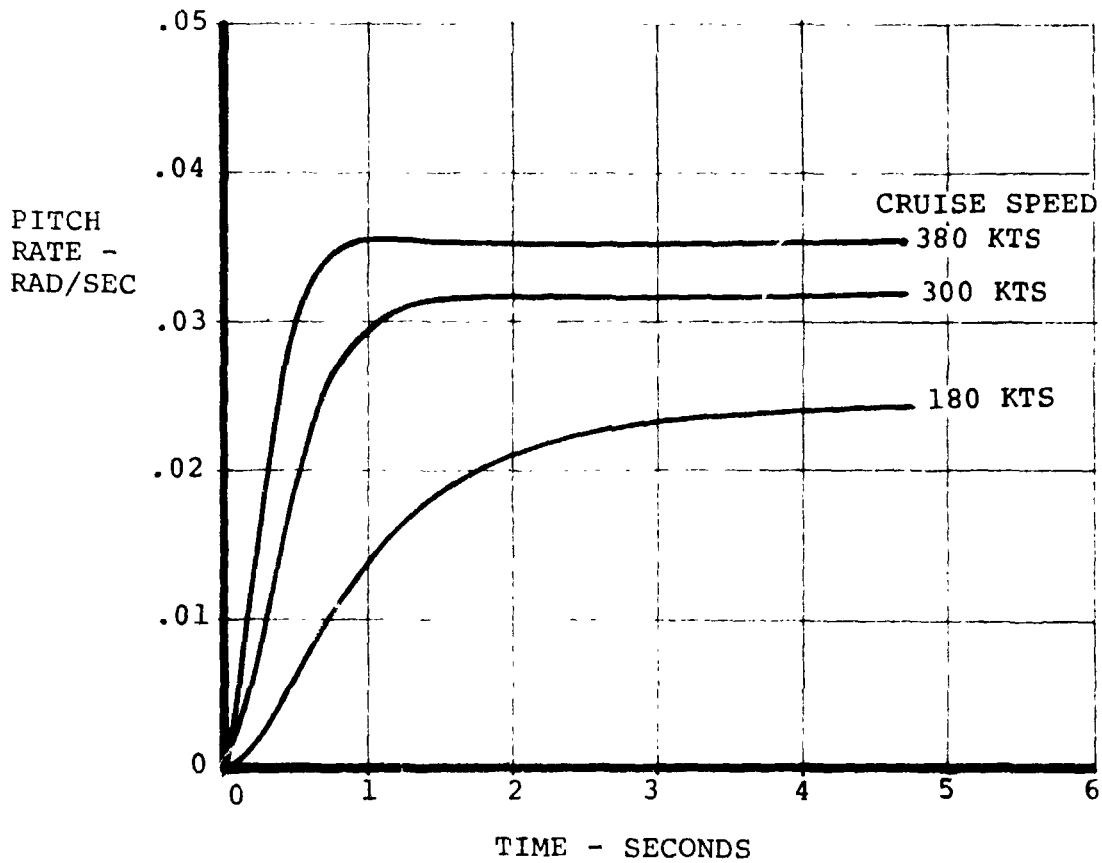
FIGURE 3.34. PITCH RATE RESPONSE TO UNIT STEP ELEVATOR INPUT.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

MIL-F-8785B(ASG)
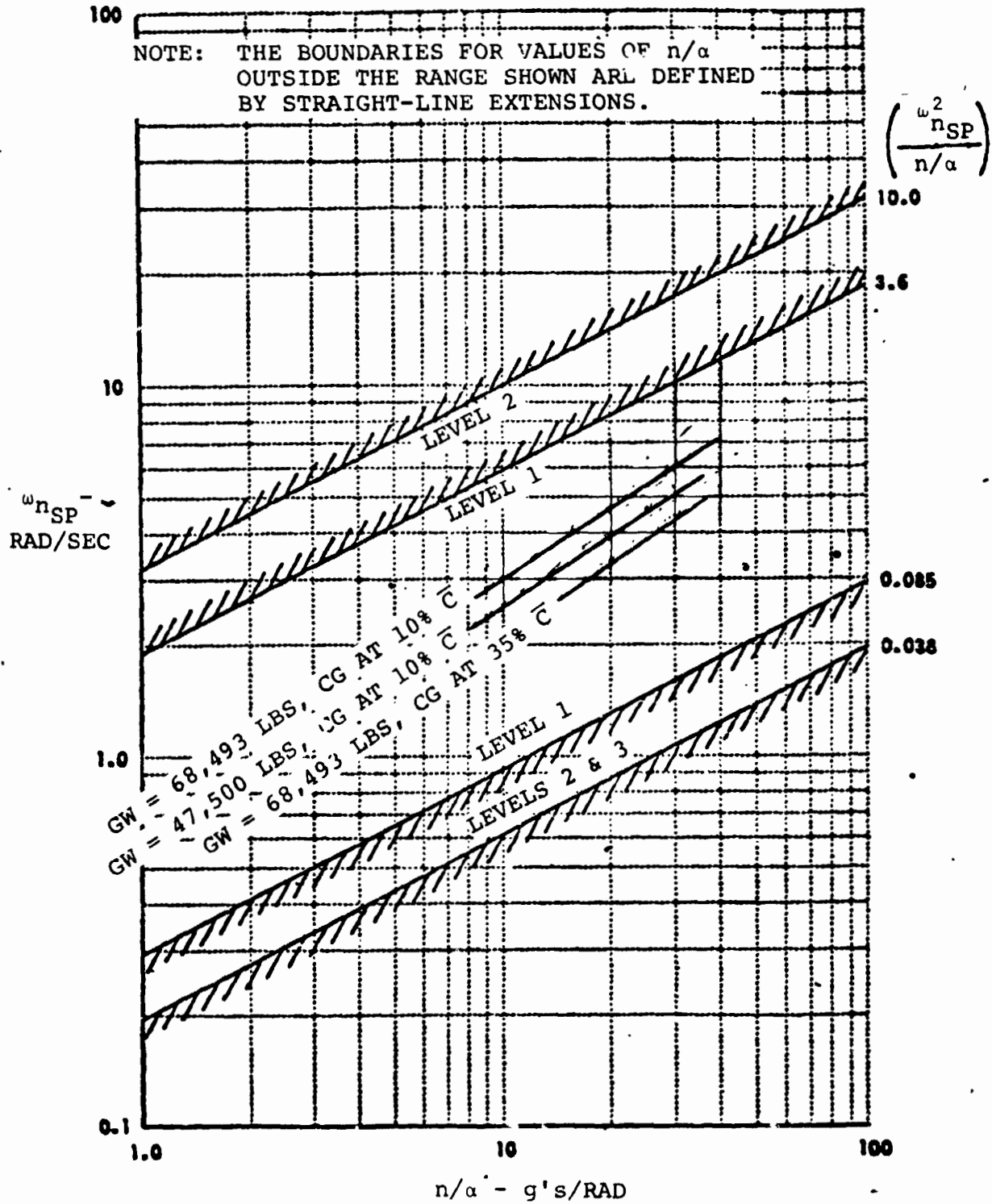
FIGURE 3.35.   SHORT-PERIOD FREQUENCY REQUIREMENTS.
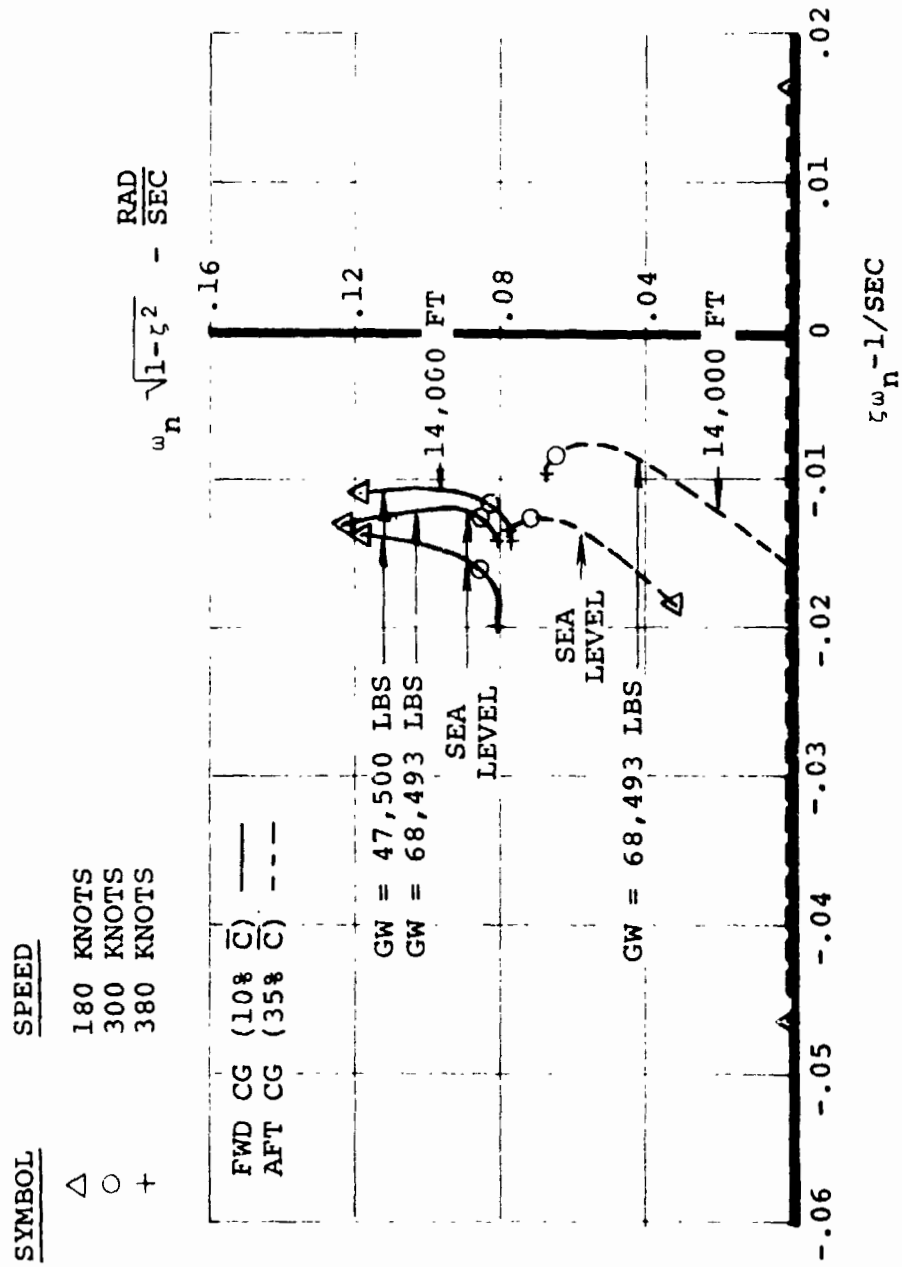
FIGURE 3.36.   LONGITUDINAL DYNAMIC RESPONSE CHARACTERISTICS -
PHUGOID.

The lateral static stability derivatives in cruise are
shown in Figure 3.37. The aircraft is statically direction-
ally stable (positive C_{n_β}) and the level increases with air-
speed. 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 indi-
cate a high C_{n_r} or yaw damping.

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
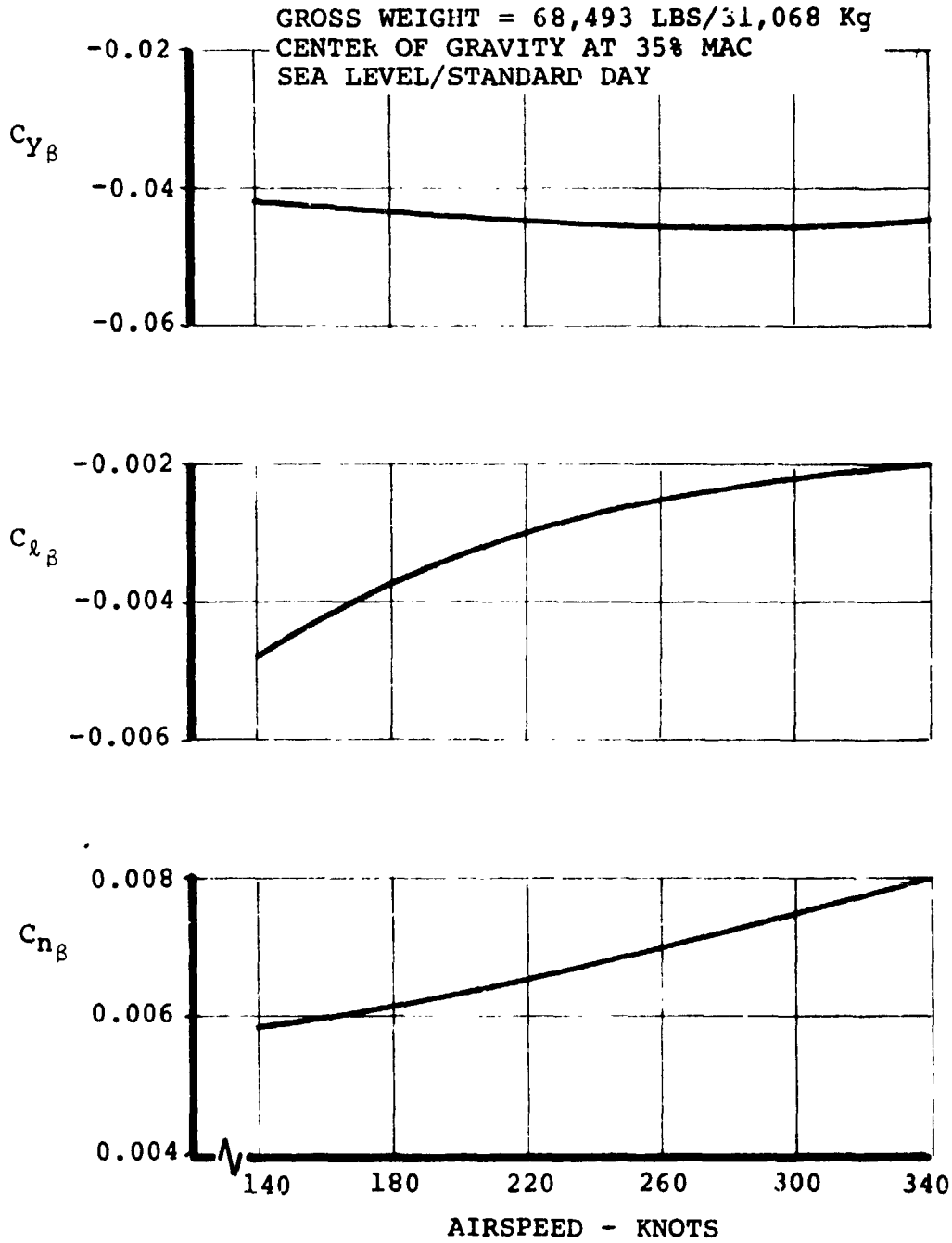
FIGURE 3.37.   STATIC LATERAL-DIRECTIONAL DERIVATIVES IN CRUISE.

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
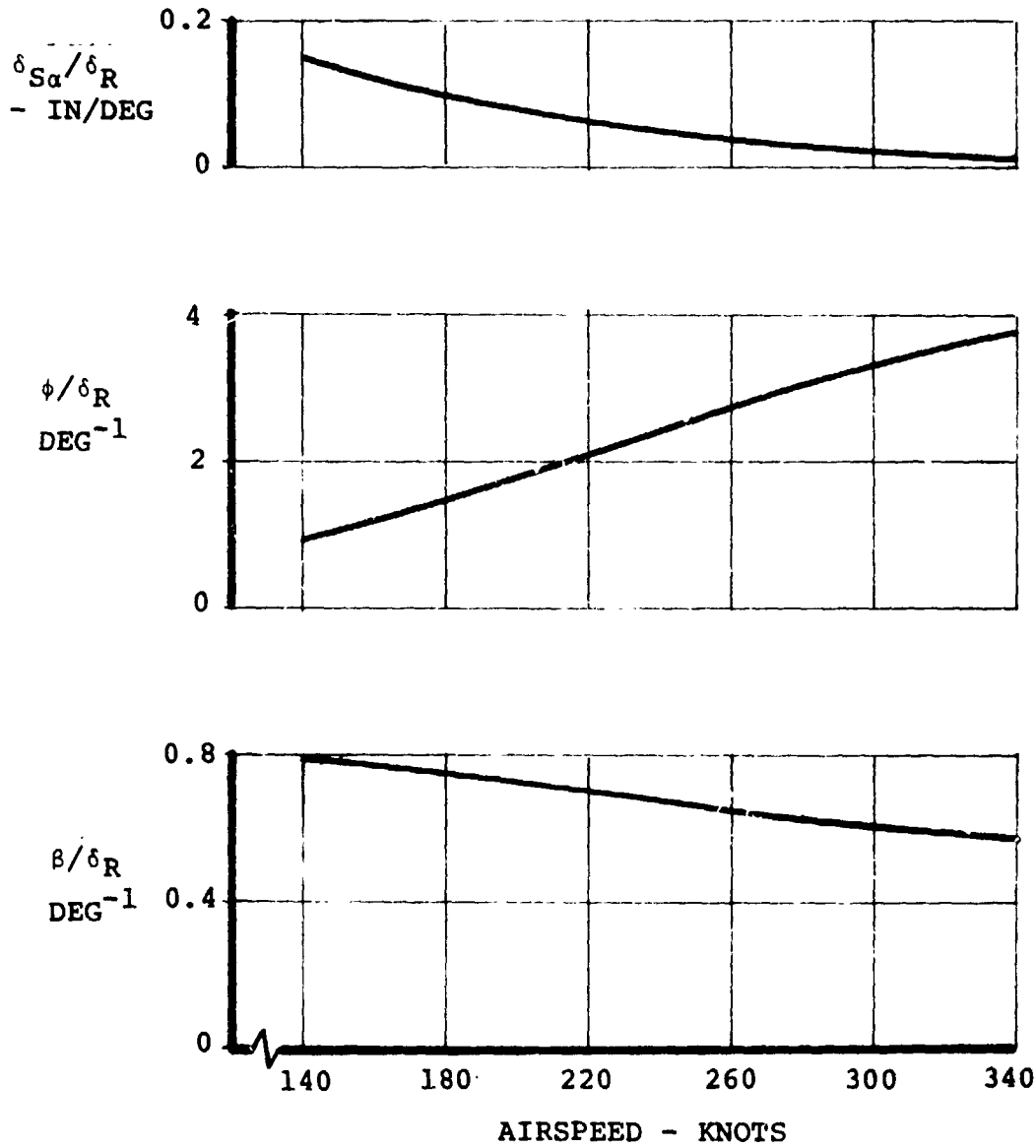
FIGURE 3.38. SIDE SLIP CHARACTERISTICS IN CRUISE.
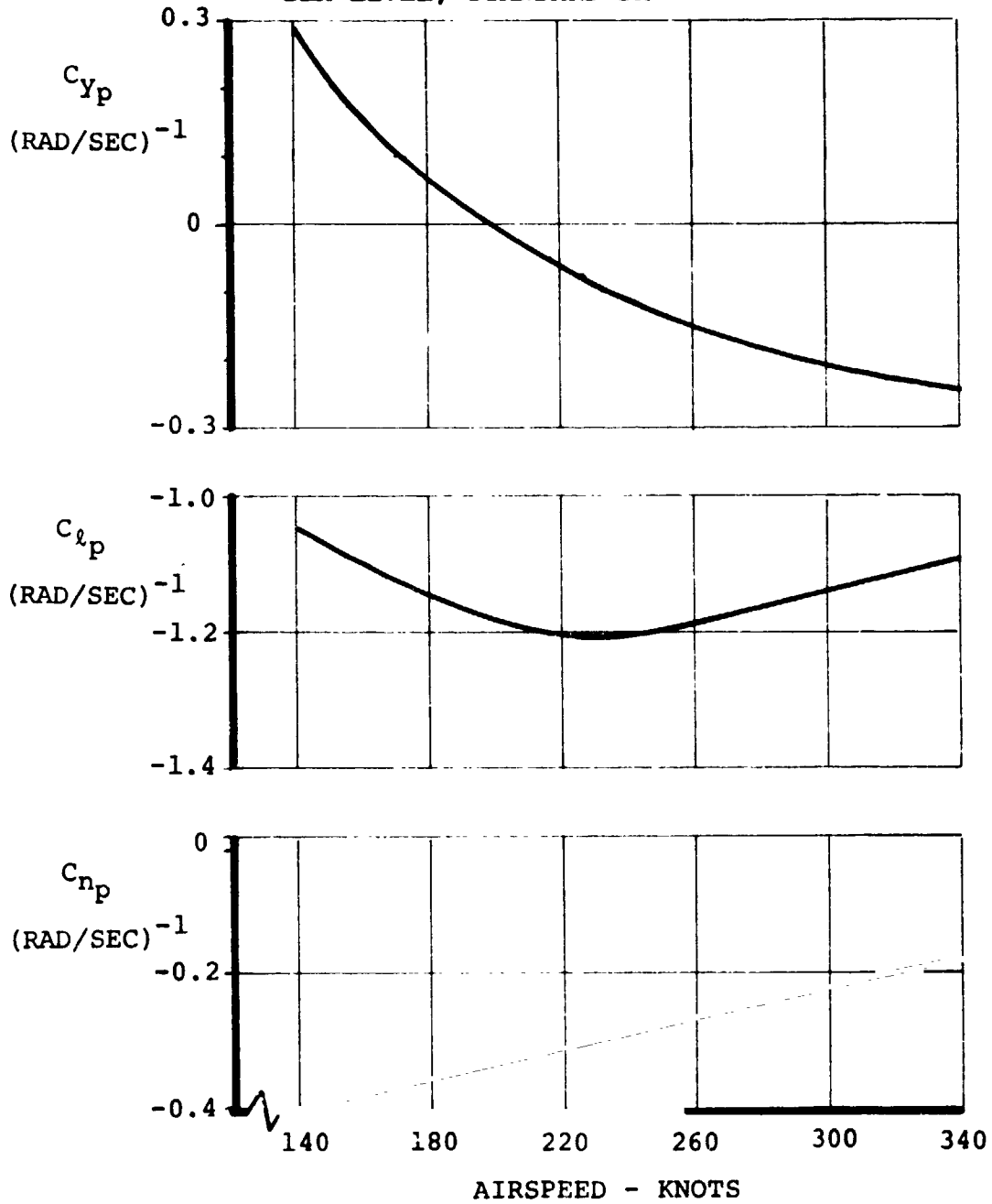
FIGURE 3.39. LATERAL-DIRECTIONAL ROLL RATE DERIVATIVES IN CRUISE.

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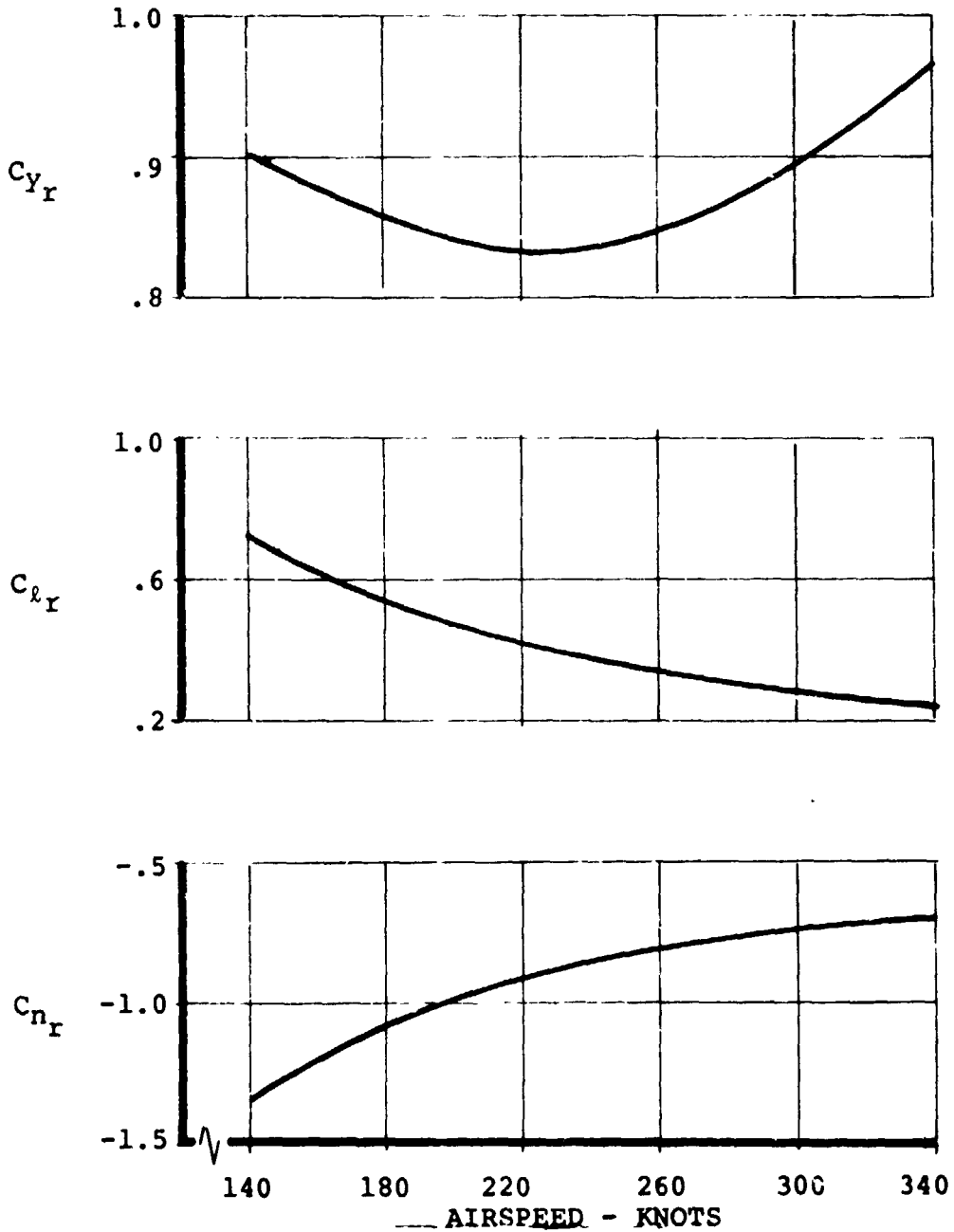
FIGURE 3.40. LATERAL-DIRECTIONAL YAW RATE DERIVATIVES IN CRUISE.

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

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<u>1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT</u>

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	-.045^8

C_{ℓ_β} - Rolling Moment Due to Sideslip

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

140	.00595	.00686	.00686	.00588	.00681
180	.00622	.00718	.00718	.00619	.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

SYMBOL	SPEED
▲	180 KNOTS
O	300 KNOTS
+	380 KNOTS

$$\omega_n \sqrt{1-\zeta^2} \frac{RAD}{SEC}$$

FWD CG (10% $\overline{C}$) ——
AFT CG (35% $\overline{C}$) ---

SEA
LEVEL

14,000 FT

GW = 68,493 LBS
GW = 47,500 LBS

$\zeta\omega_n$ - 1/SEC
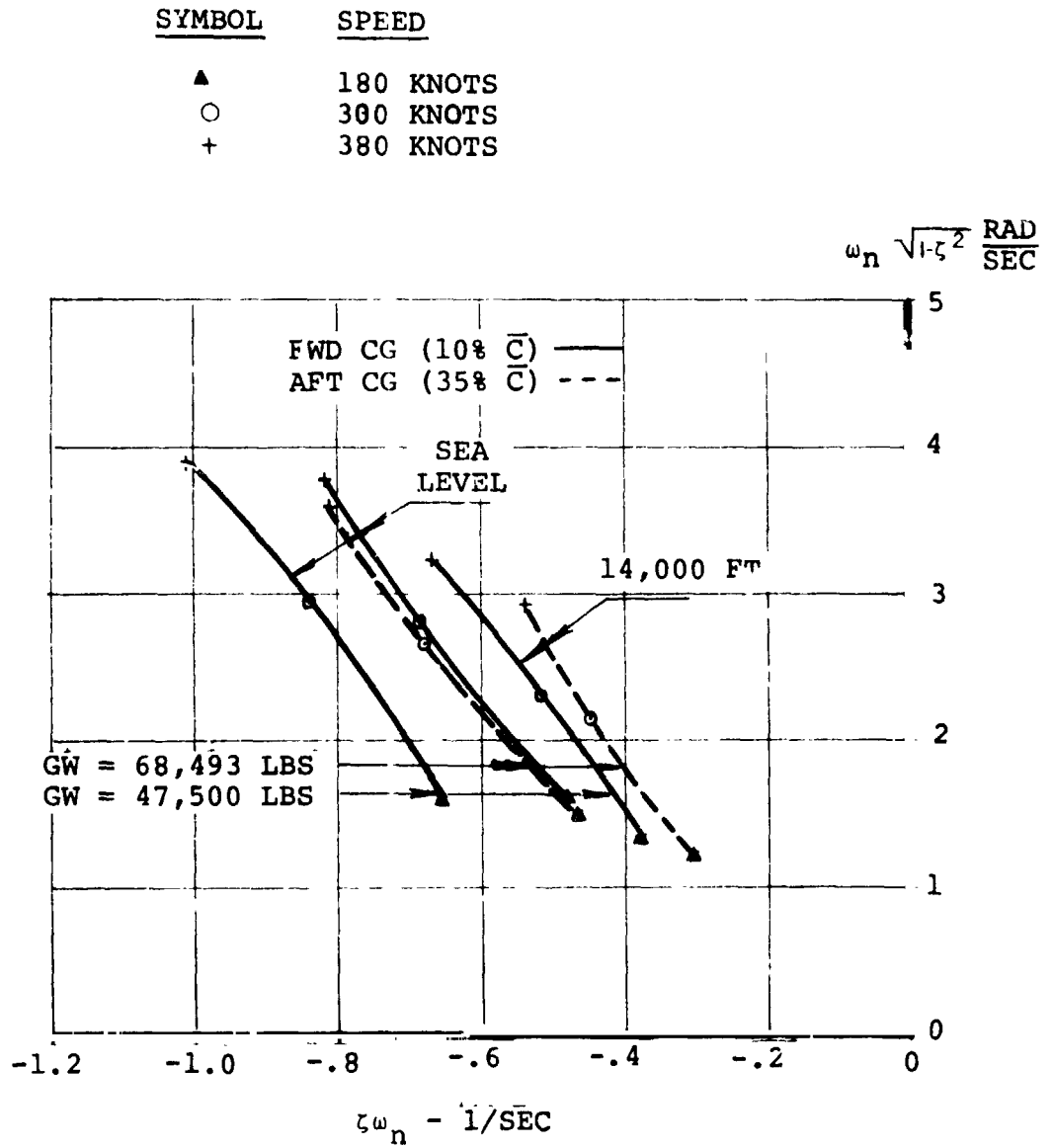
FIGURE 3.41. LATERAL-DIRECTIONAL DYNAMIC RESPONSE CHARACTERISTICS
- DUTCH ROLL MODE.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
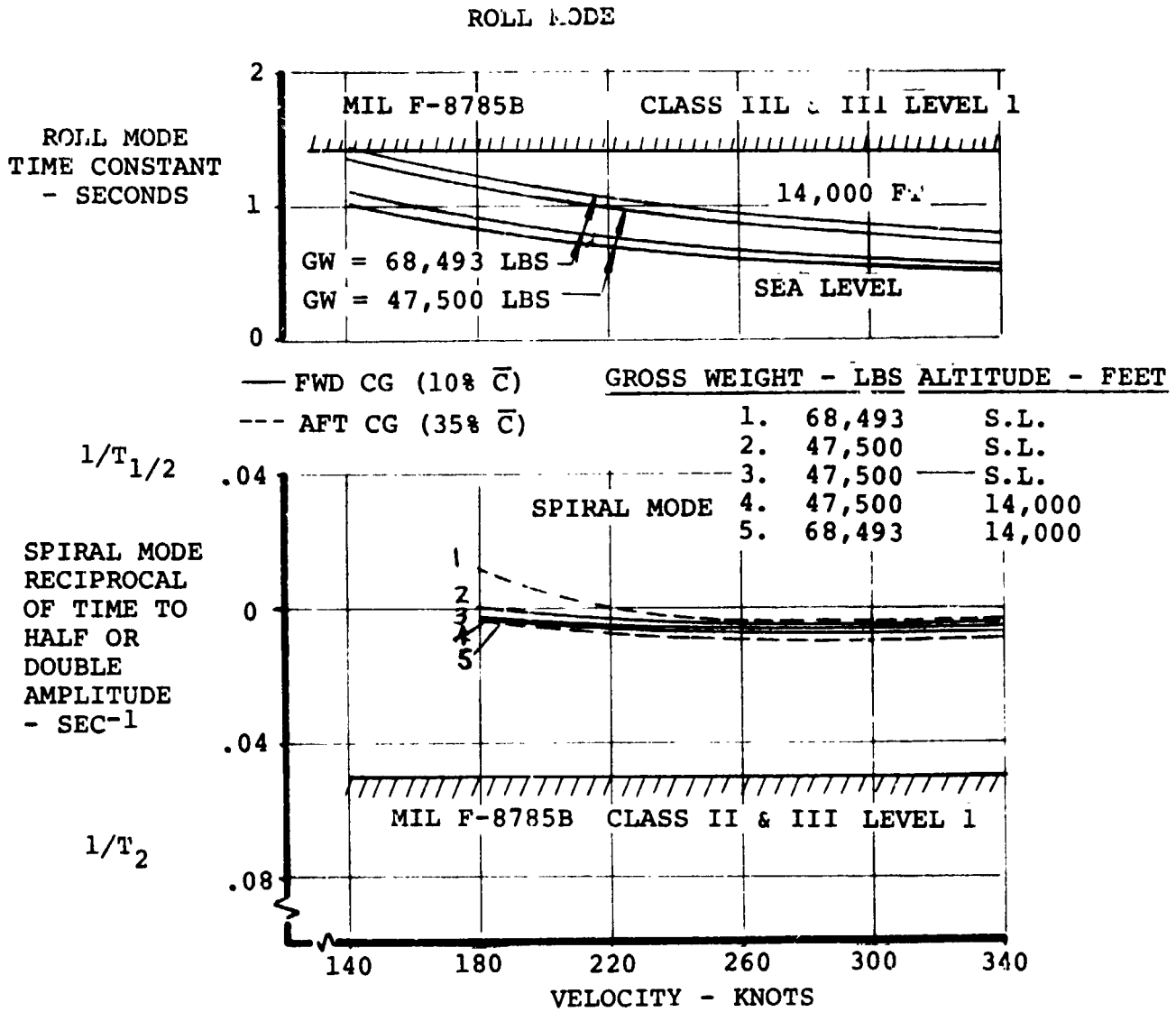
ROLL MODE

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

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
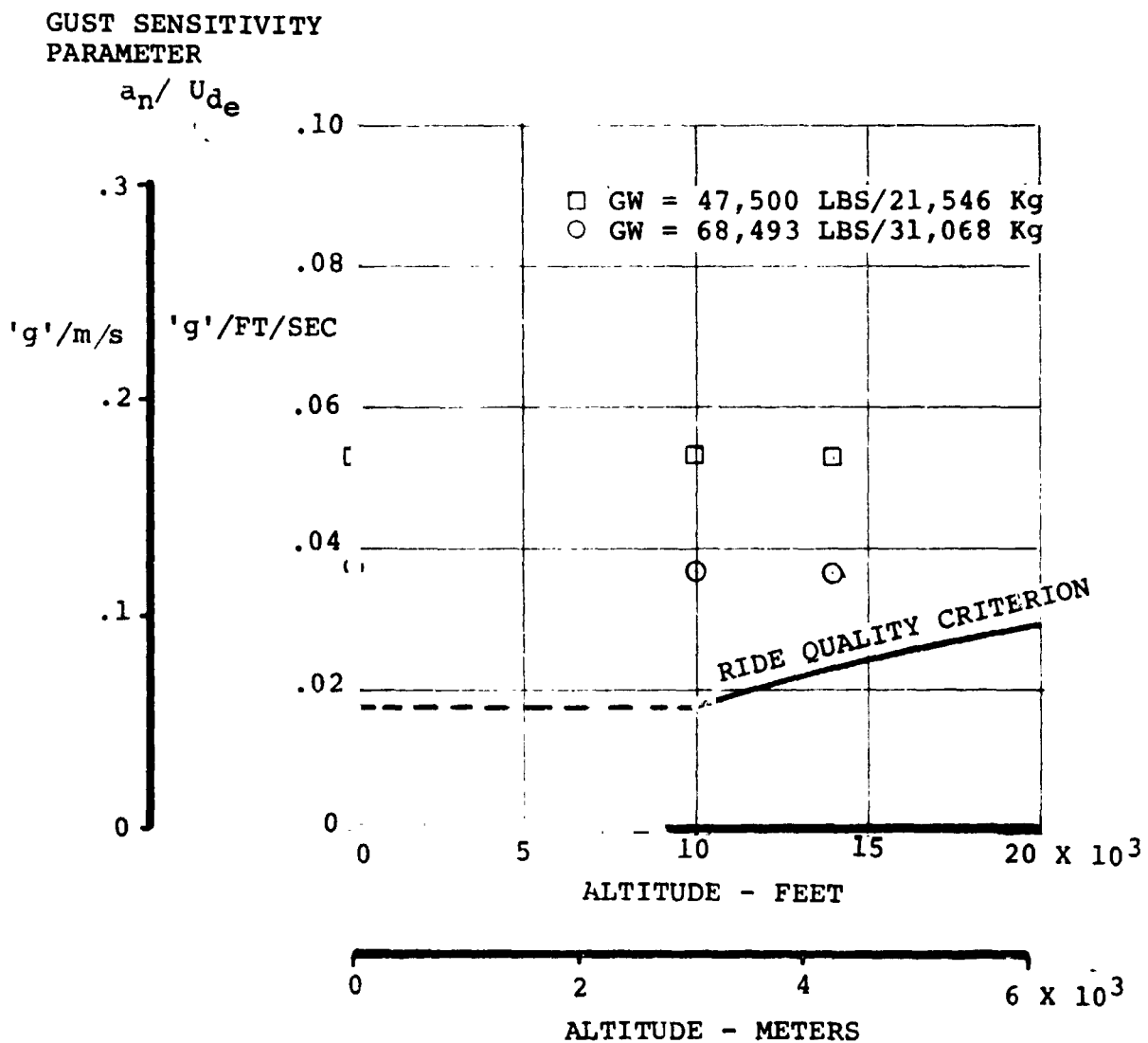
FIGURE 3.43.   GUST SENSITIVITY AT MAXIMUM CRUISE SPEED.

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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 investi-
gated 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

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COMPONENT	INPUT RPM	REDUCTION RATIO	CRITICAL TORQUE FOOT-POUNDS	SIZING CONDITION
ENGINE TRANSFER CASE	14430	4:1	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.
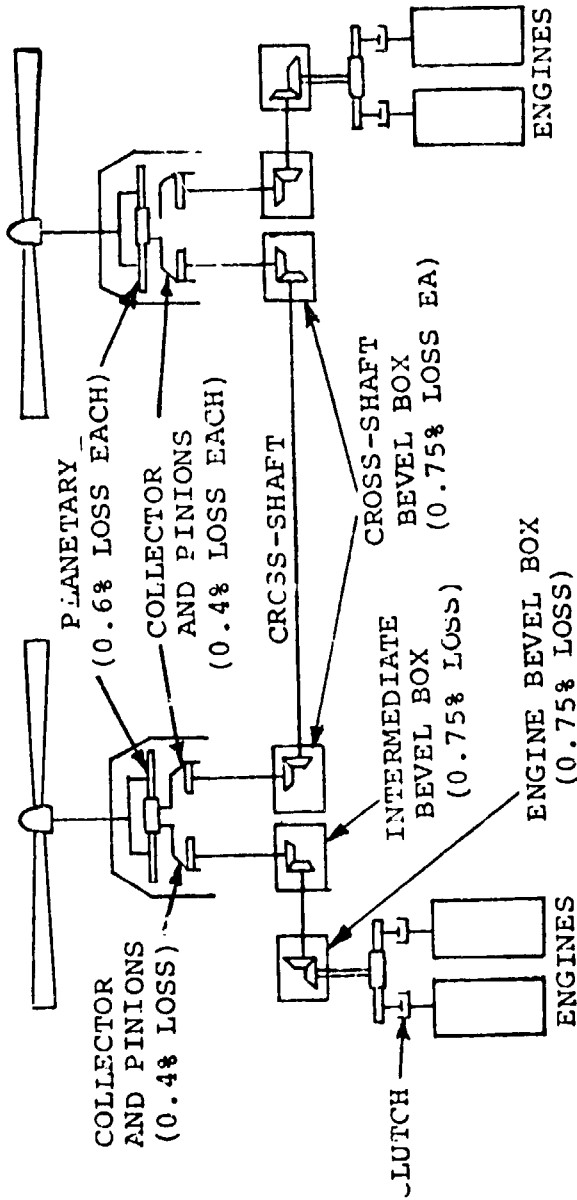
FIGURE 3.44. TRANSMISSION SCHEMATIC.

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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 trans-ducers 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

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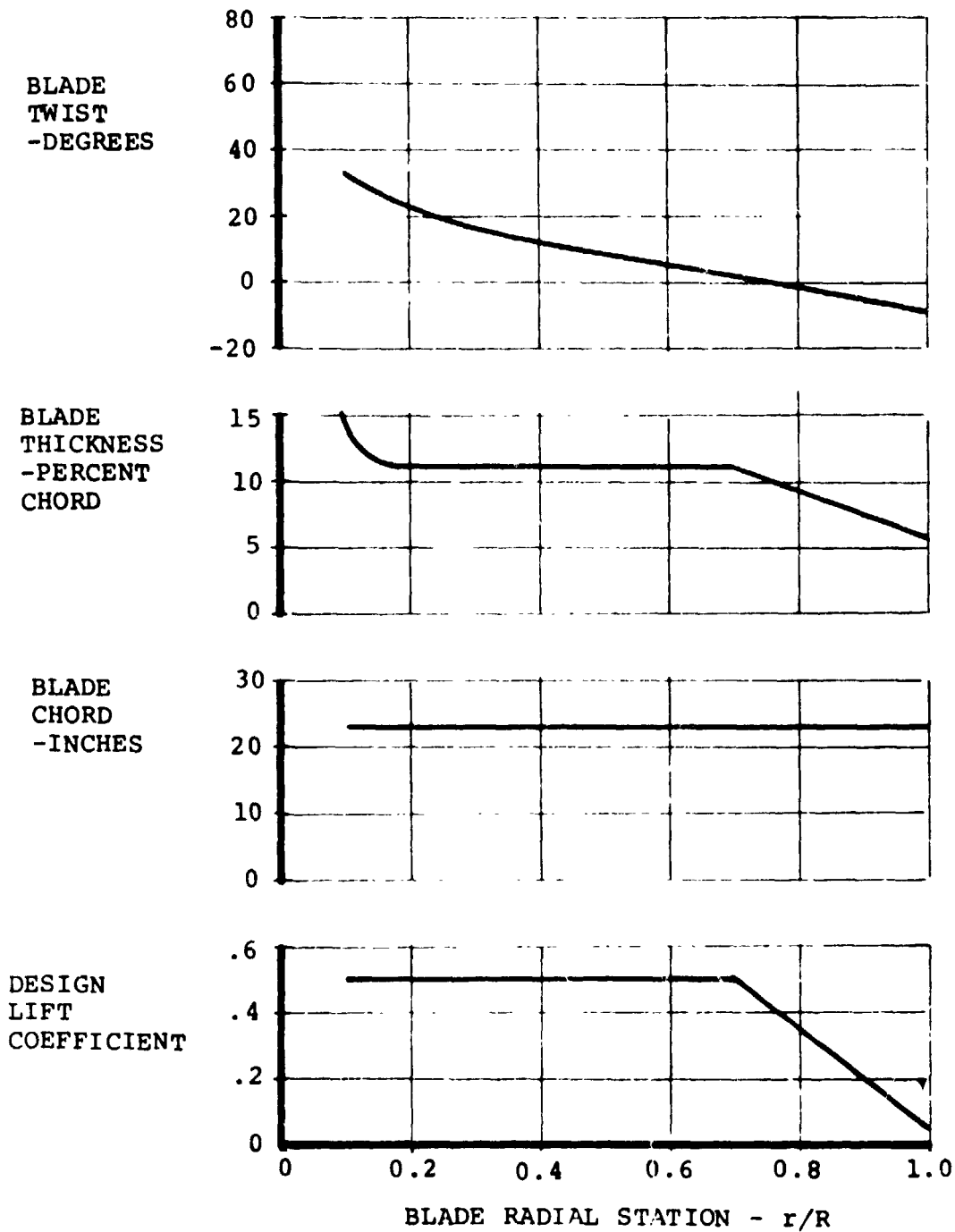
FIGURE 3.45. ROTOR BLADE GEOMETRY.
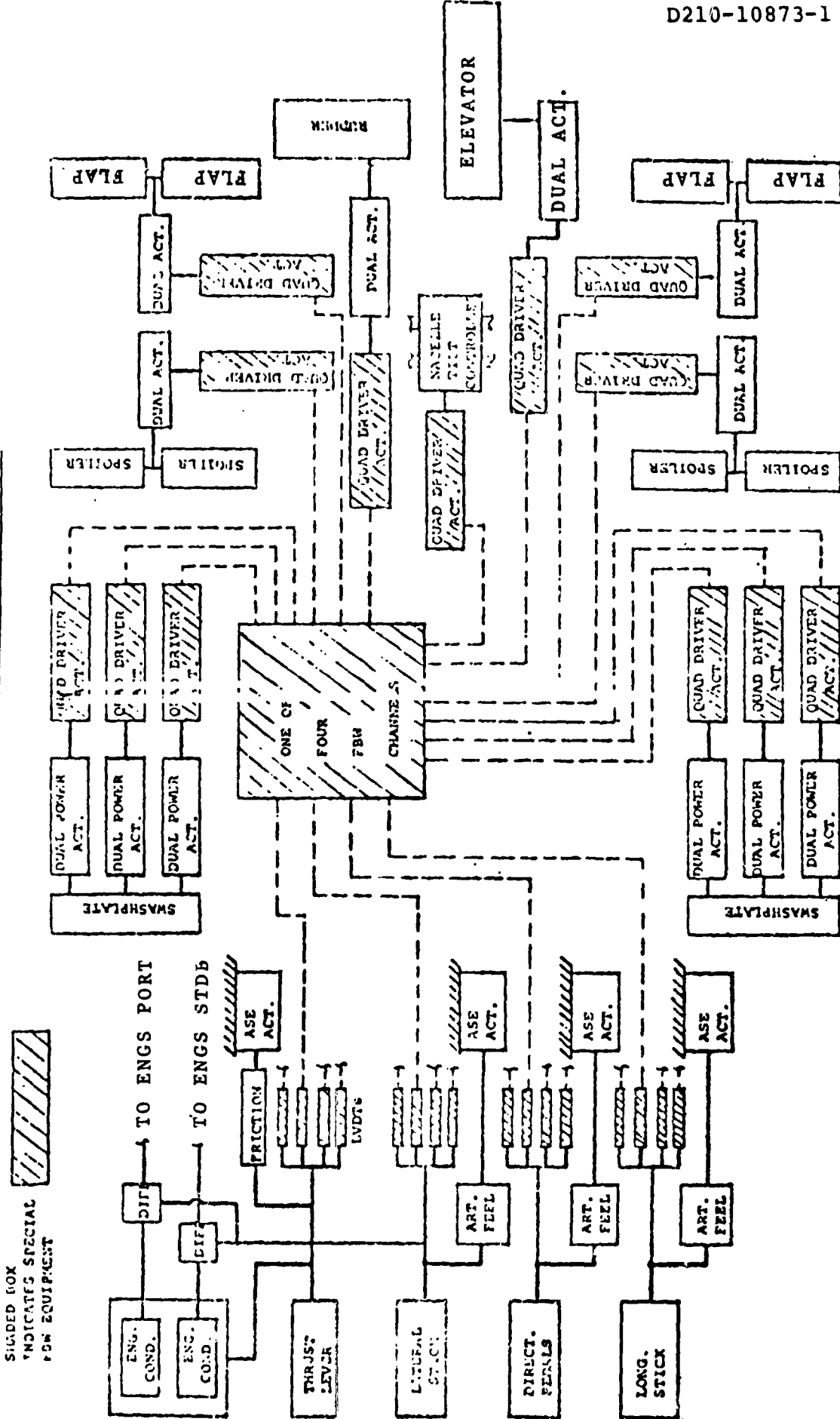
FIGURE 3.46. TILT ROTOR FLY-BY-WIRE CONTROL SYSTEM.

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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.

Tilt Rotor Fuel System

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

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	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

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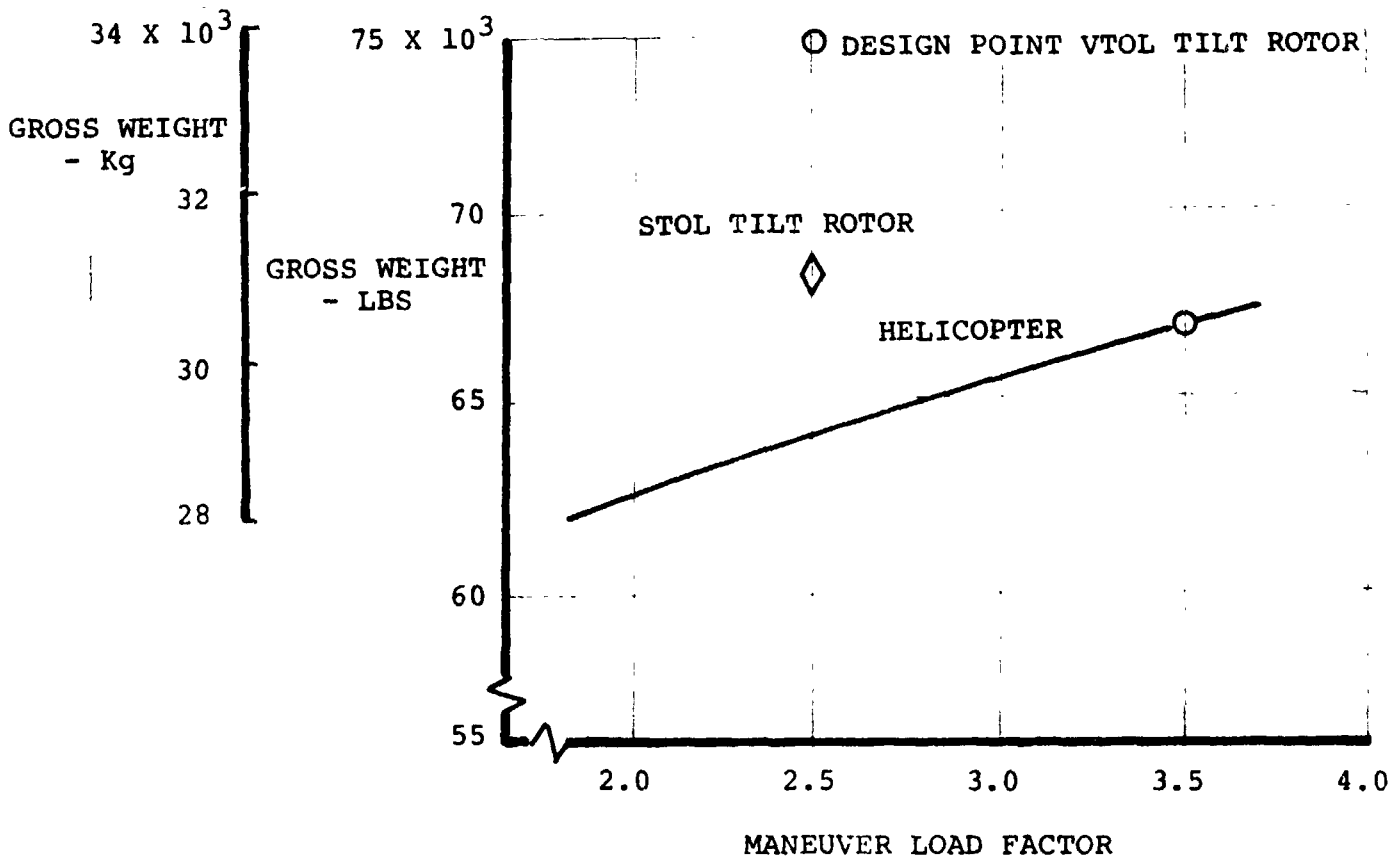
FIGURE 4.1.   EFFECT OF DESIGN MANEUVER LOAD FACTOR ON
AIRCRAFT SIZE.

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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 heli-

coper 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".

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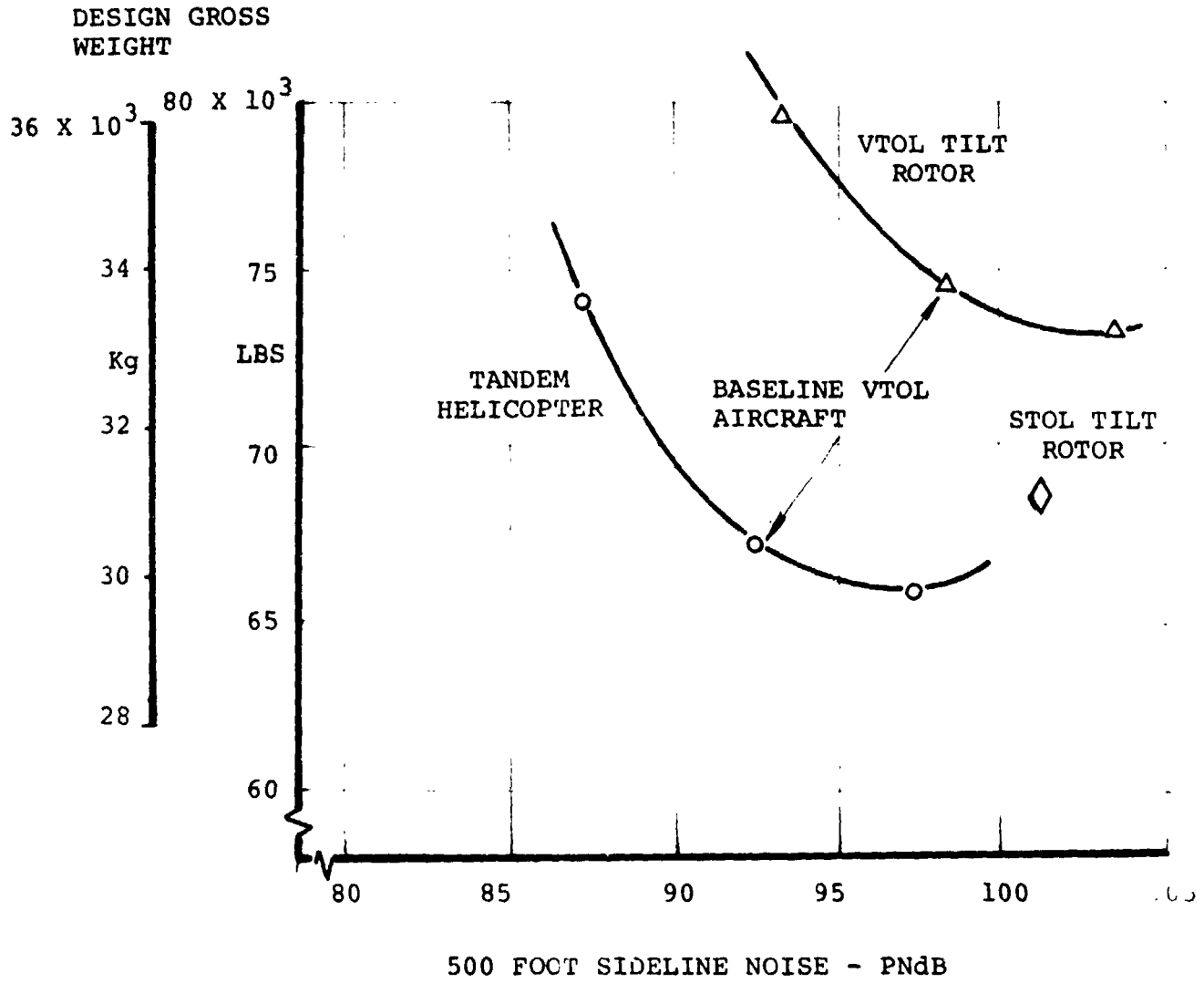
FIGURE 4.2. THE EFFECT OF EXTERNAL NOISE DESIGN CRITERIA ON AIRCRAFT DESIGN GROSS WEIGHT.

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1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
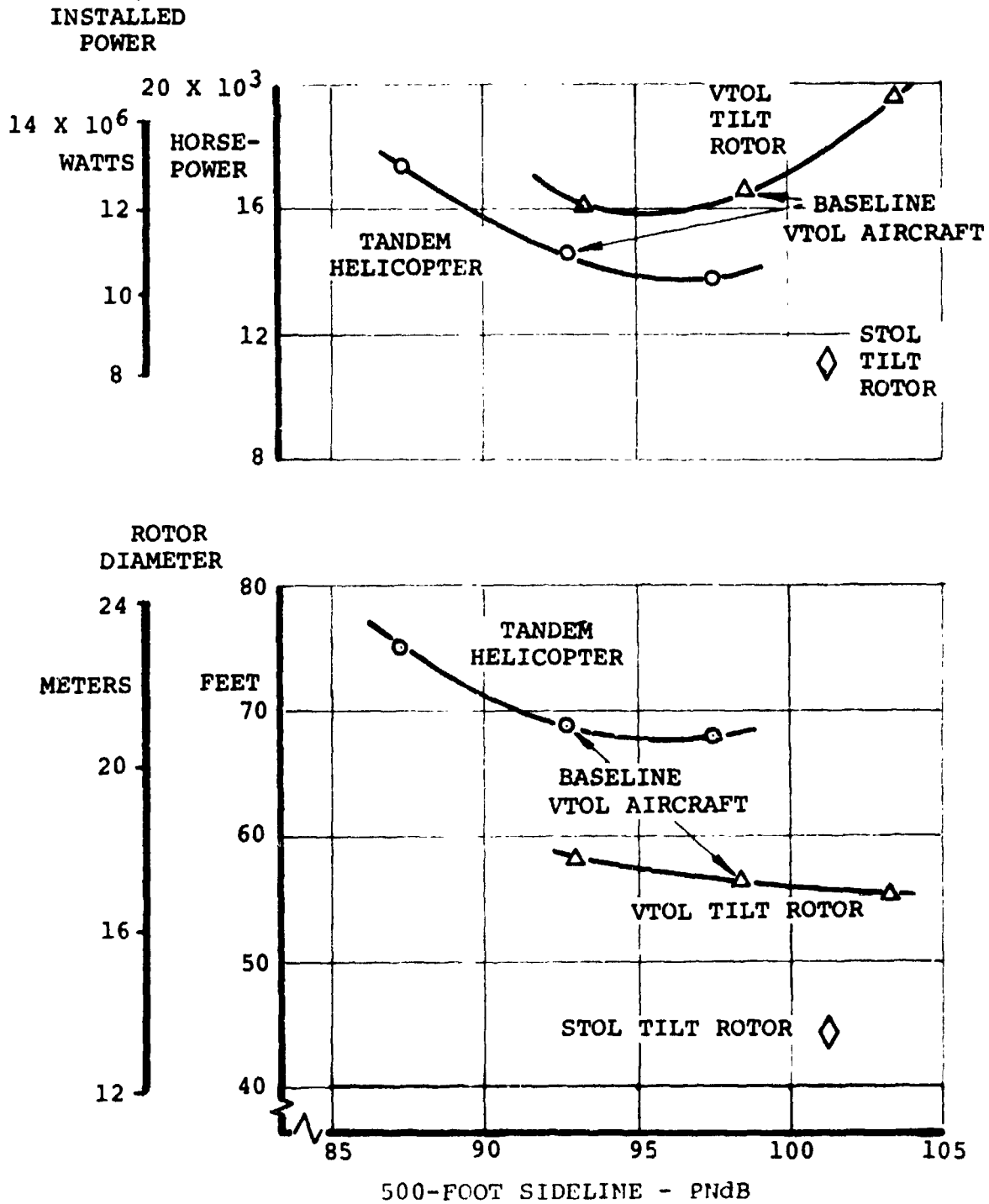
FIGURE 4.3. EFFECT OF EXTERNAL NOISE DESIGN CRITERIA ON ROTOR DIAMETER AND INSTALLED POWER

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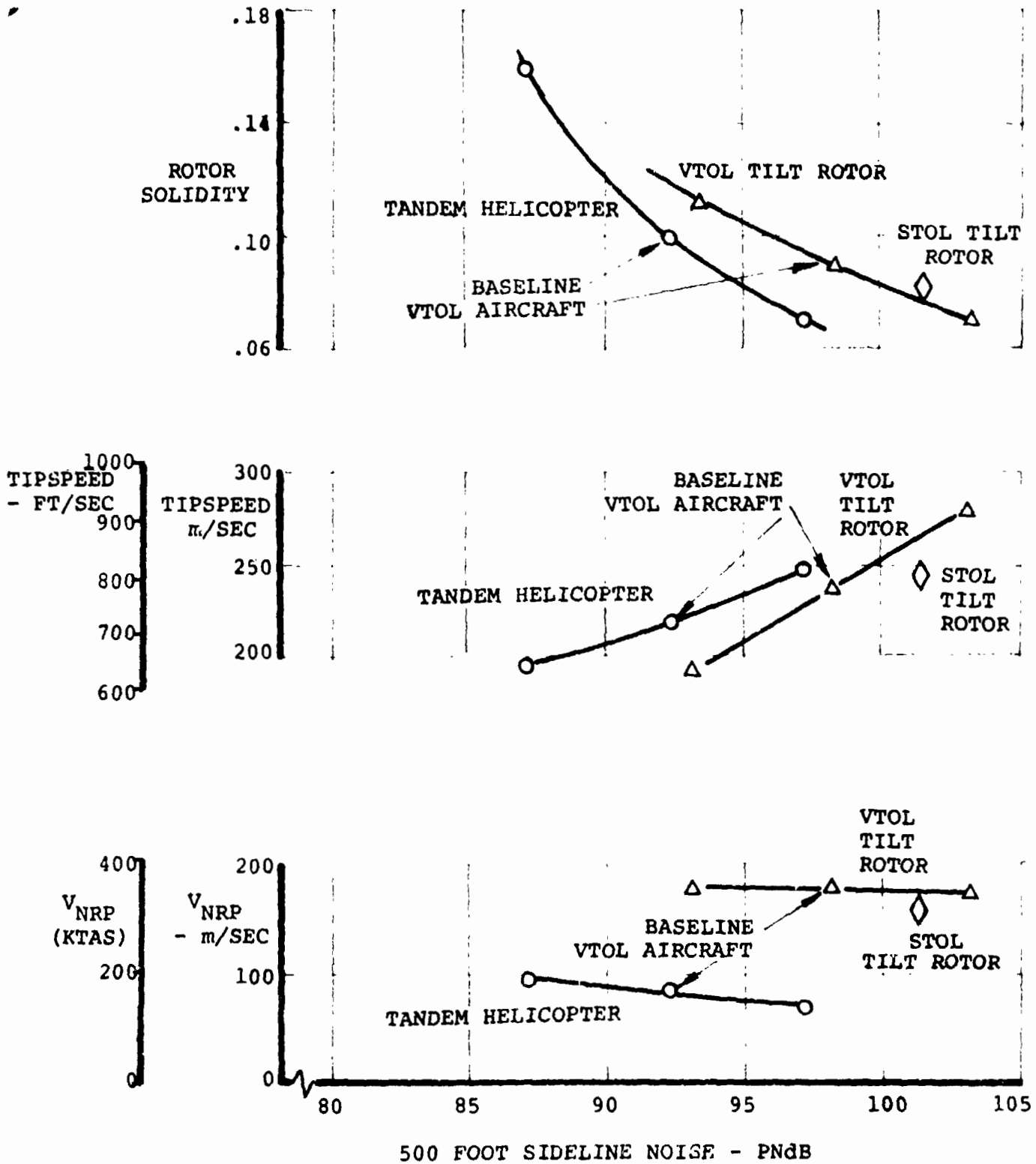
FIGURE 4.4. EFFECT OF EXTERNAL NOISE CRITERIA ON
DESIGN PARAMETERS.

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

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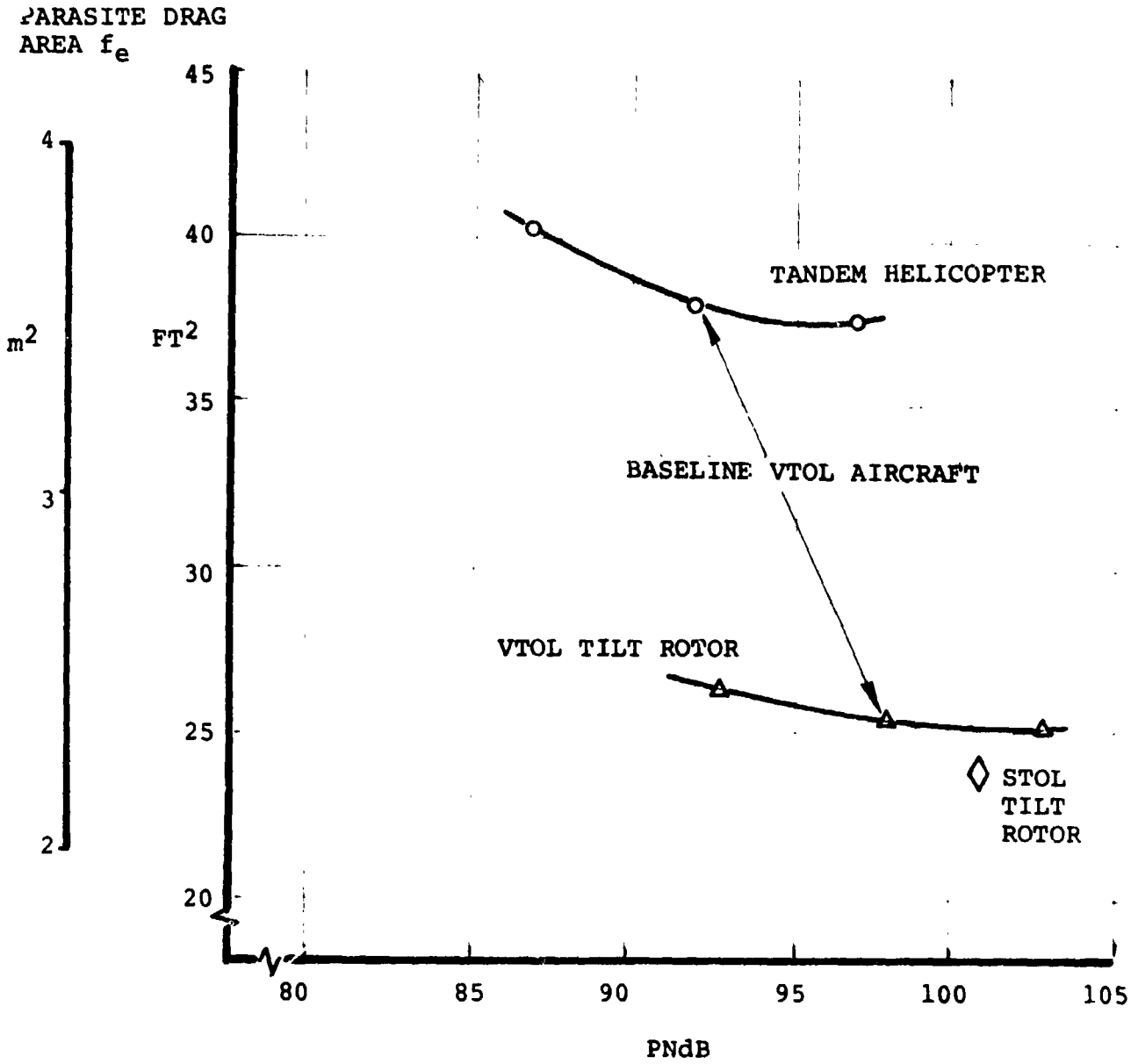
FIGURE 4.5. EFFECT OF EXTERNAL NOISE CRITERIA ON PARASITE
DRAG AREA f_e.

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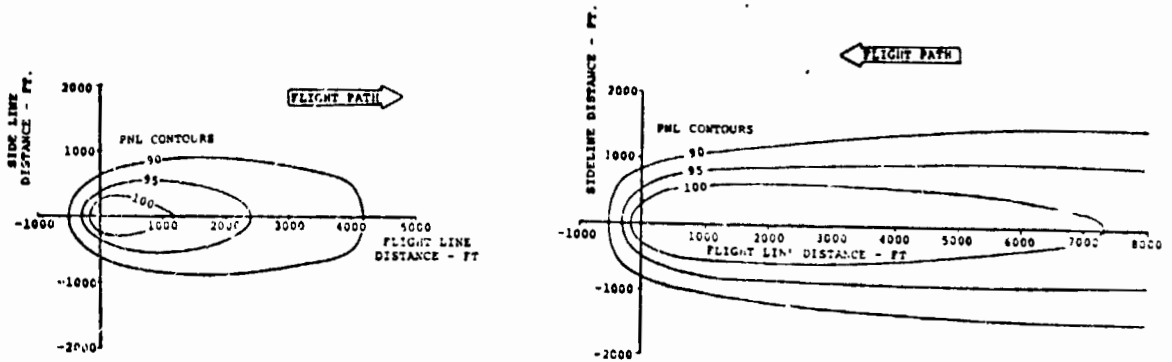
HELICOPTER TAKEOFF

HELICOPTER LANDING
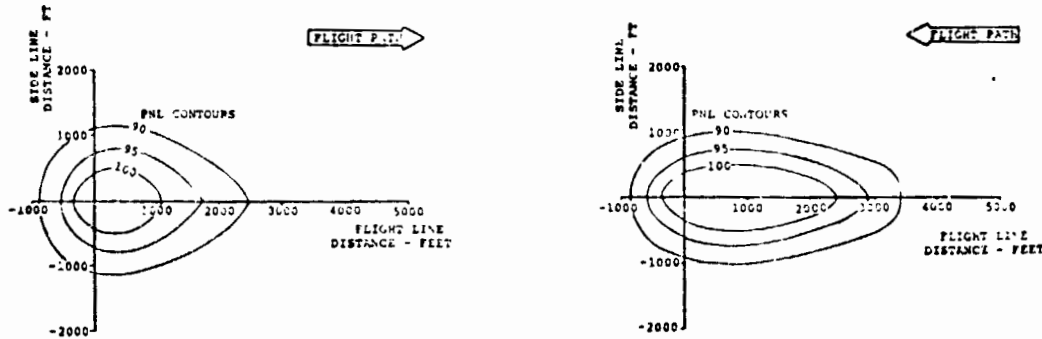
VTOL TILT ROTOR TAKEOFF

VTOL TILT ROTOR LANDING
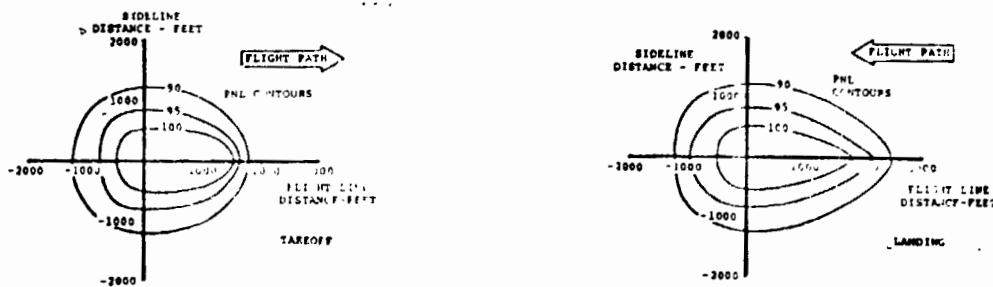
STOL TILT ROTOR TAKEOFF

STOL TILT ROTOR LANDING

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

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1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT
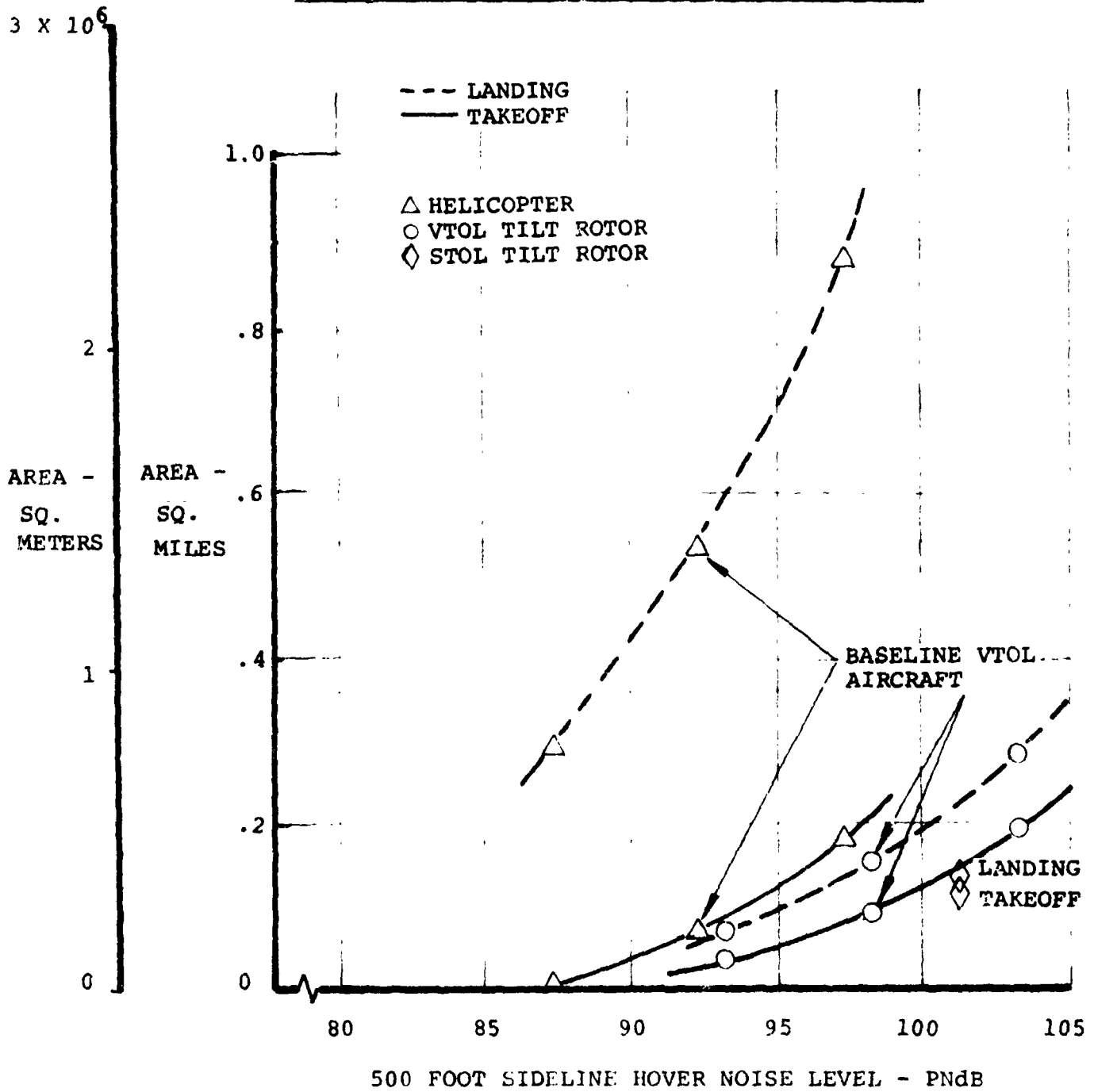
FIGURE 4.7. COMPARISON OF AREA IMPACTED BY 95 PNdB NOISE
LEVEL OR GREATER DURING TAKEOFF AND LANDING.

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 tan-
dem 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

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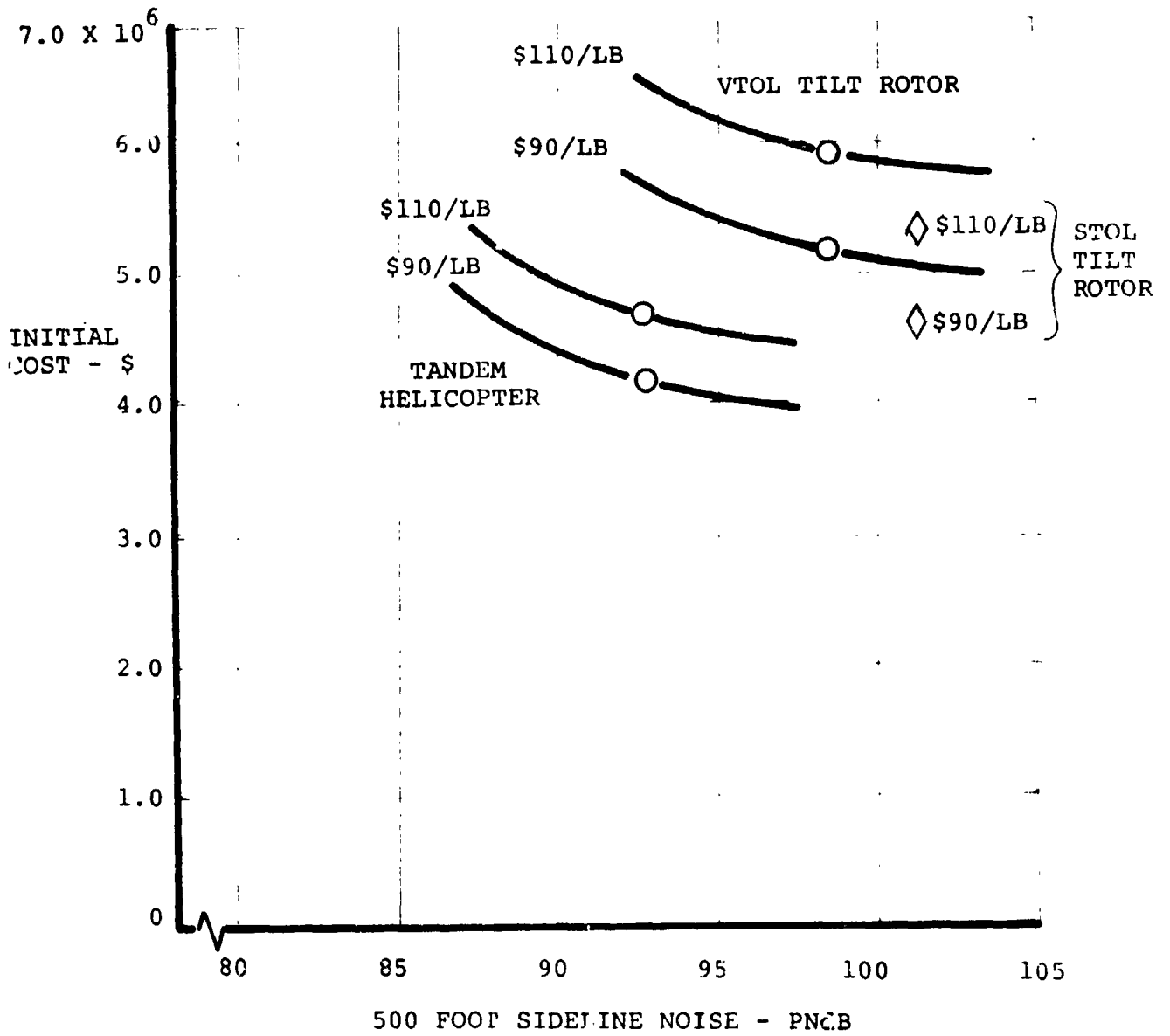
FIGURE 4.8.   EFFECT OF EXTERNAL NOISE DESIGN CRITERIA ON INITIAL COST.

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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.

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AIRFRAME COST $90/LB
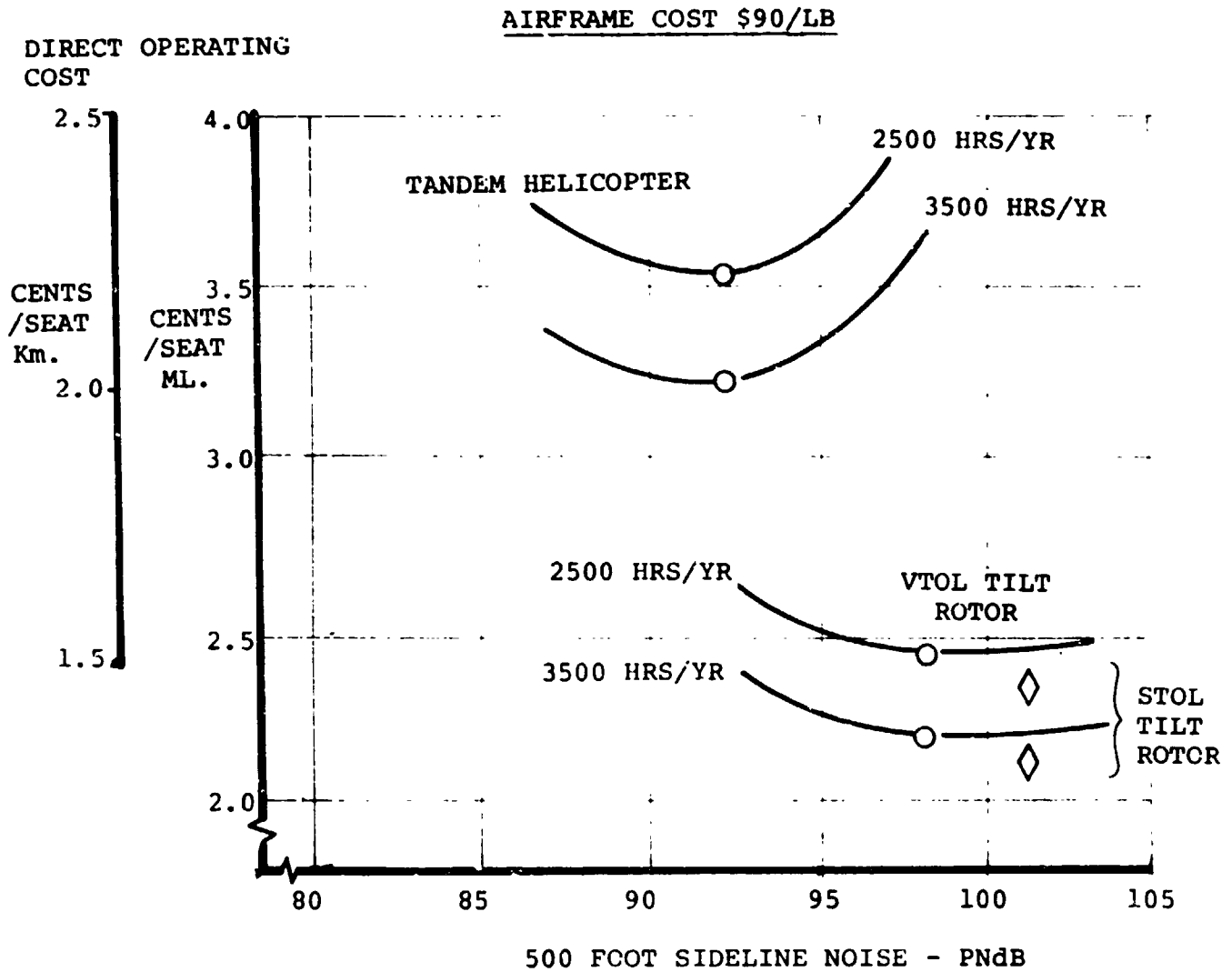
FIGURE 4.9. EFFECT OF EXTERNAL NOISE CRITERIA ON DOC AT 230 STATUTE MILES RANGE.

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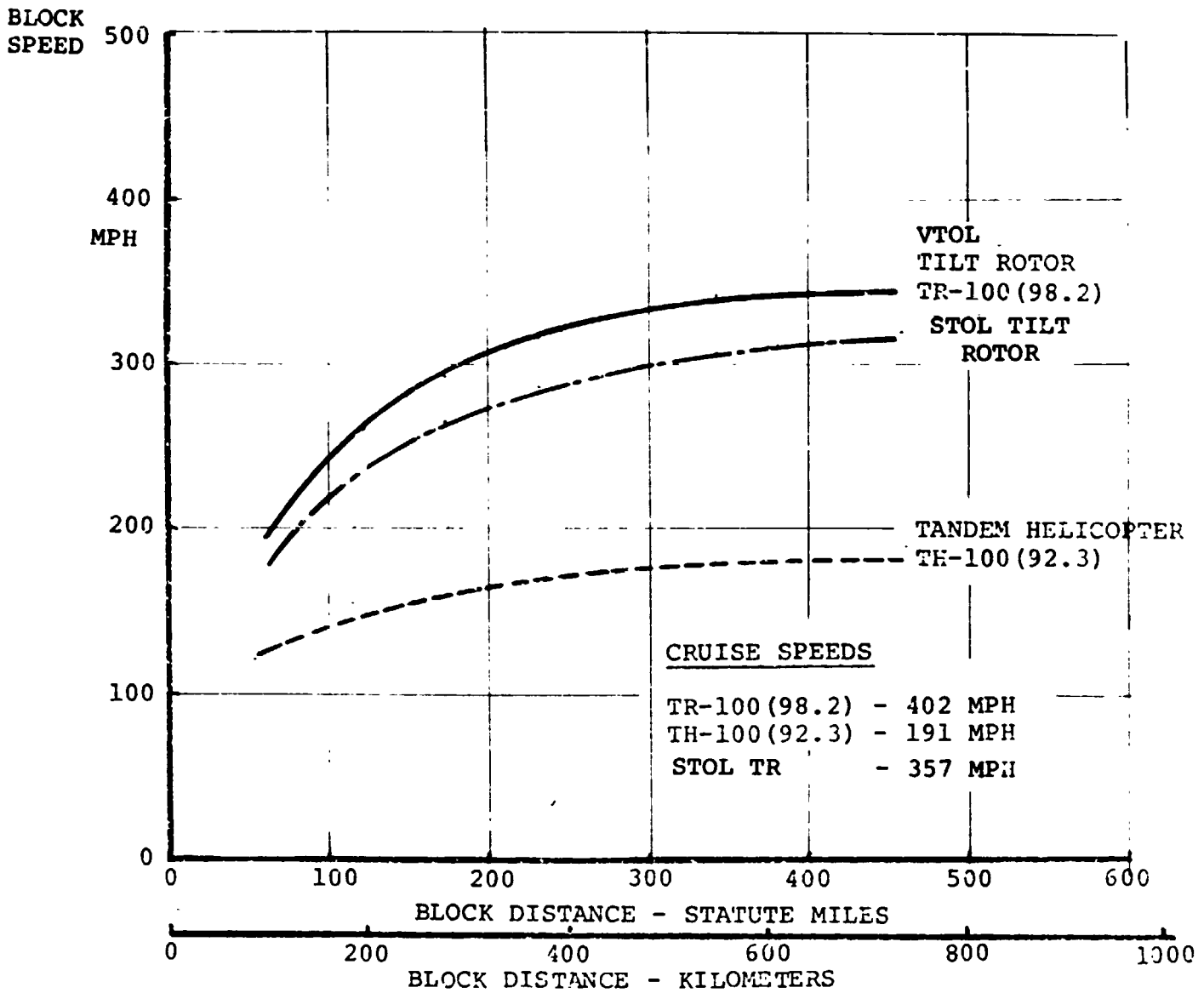
FIGURE 4.10. COMPARISON OF BASELINE DESIGN AIRCRAFT –
BLOCK SPEED AND DISTANCE.

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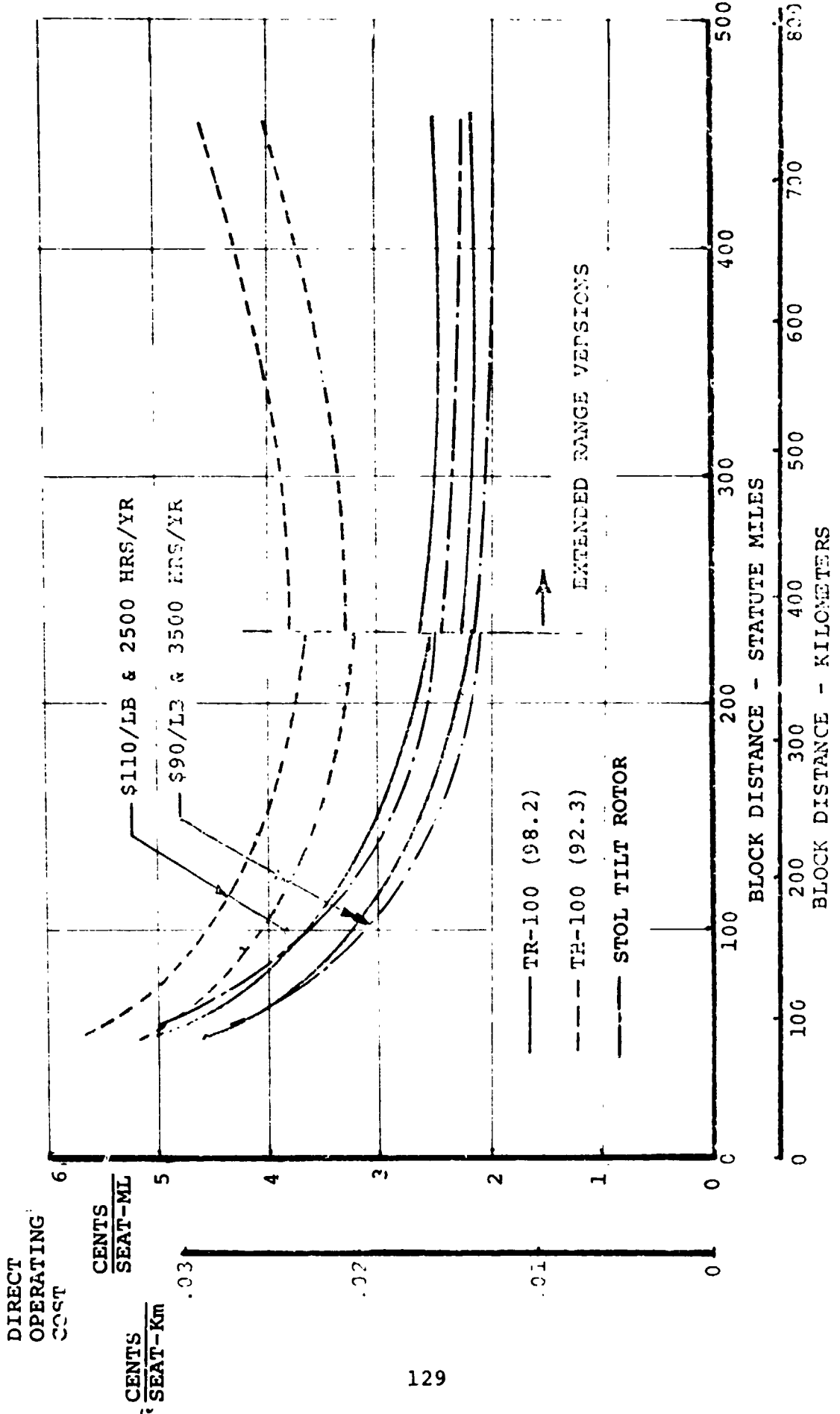
FIGURE 4.11. DIRECT OPERATING COST COMPARISON WITH TANDEM HELICOPTER AND VTOL TILT ROTOR.

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 operatir j 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 con-
figurations 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.

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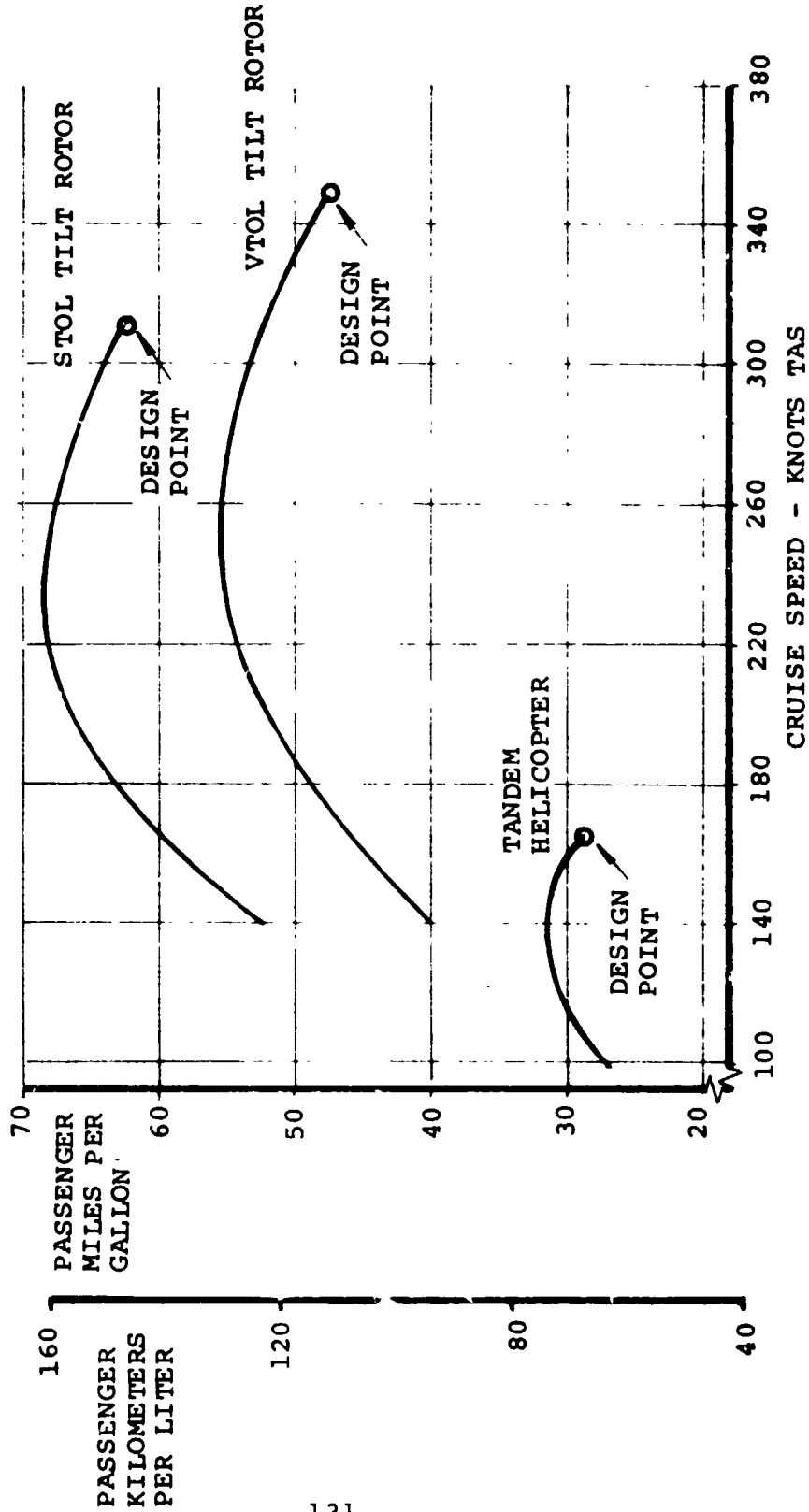
FIGURE 4.12. VARIATION OF FUEL CONSUMPTION WITH CRUISE SPEED.

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.

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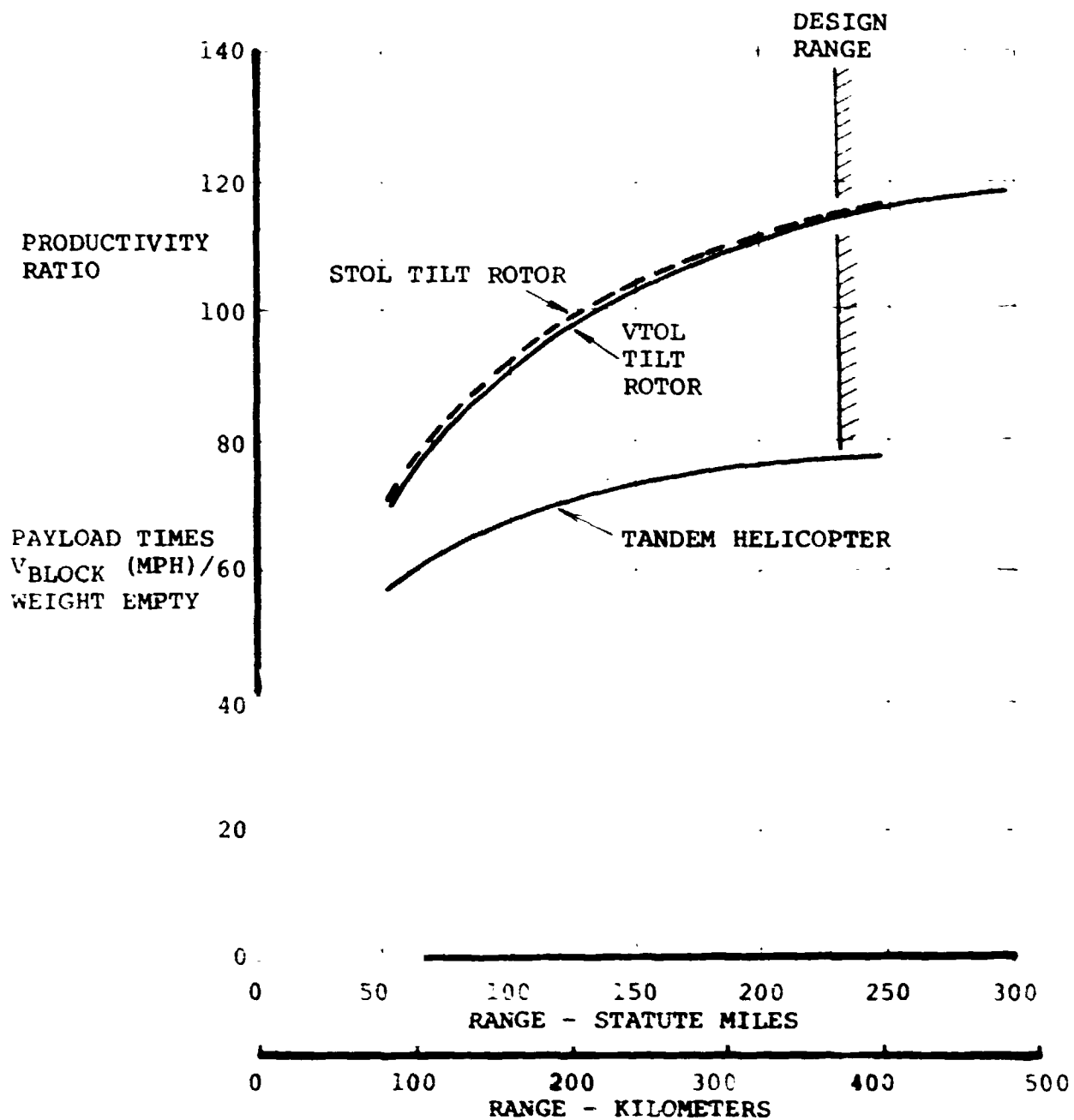
FIGURE 4.13. PRODUCTIVITY RATIO COMPARISON: TANDEM ROTOR
HELICOPTER AND TILT ROTORS.
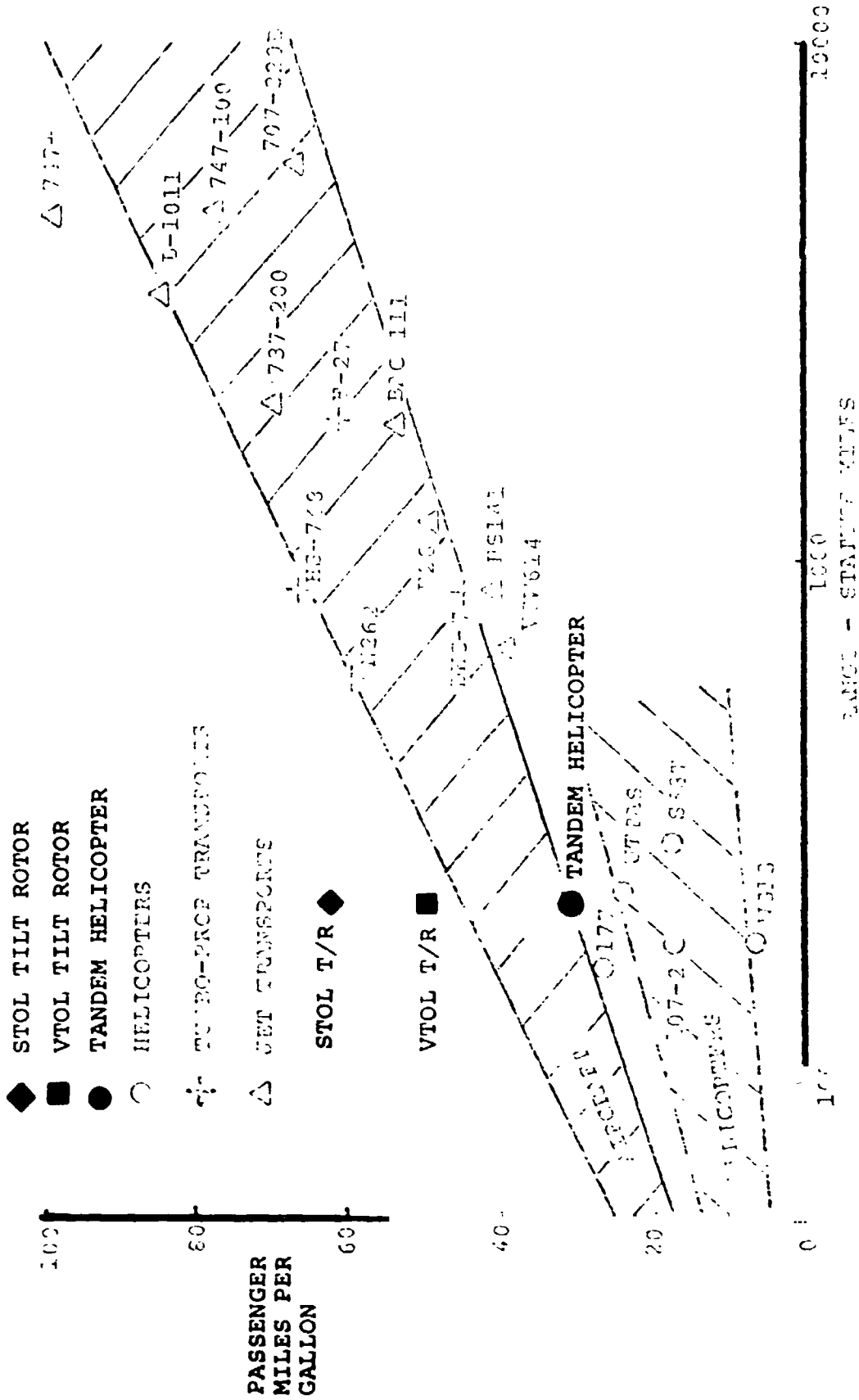
FIGURE 4.14. FUEL CONSUMPTION COMPARISON.

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 cir-
cumstances, 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 per-
formance, 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 configur-

ation 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 diffi-

culties 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.

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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.

140
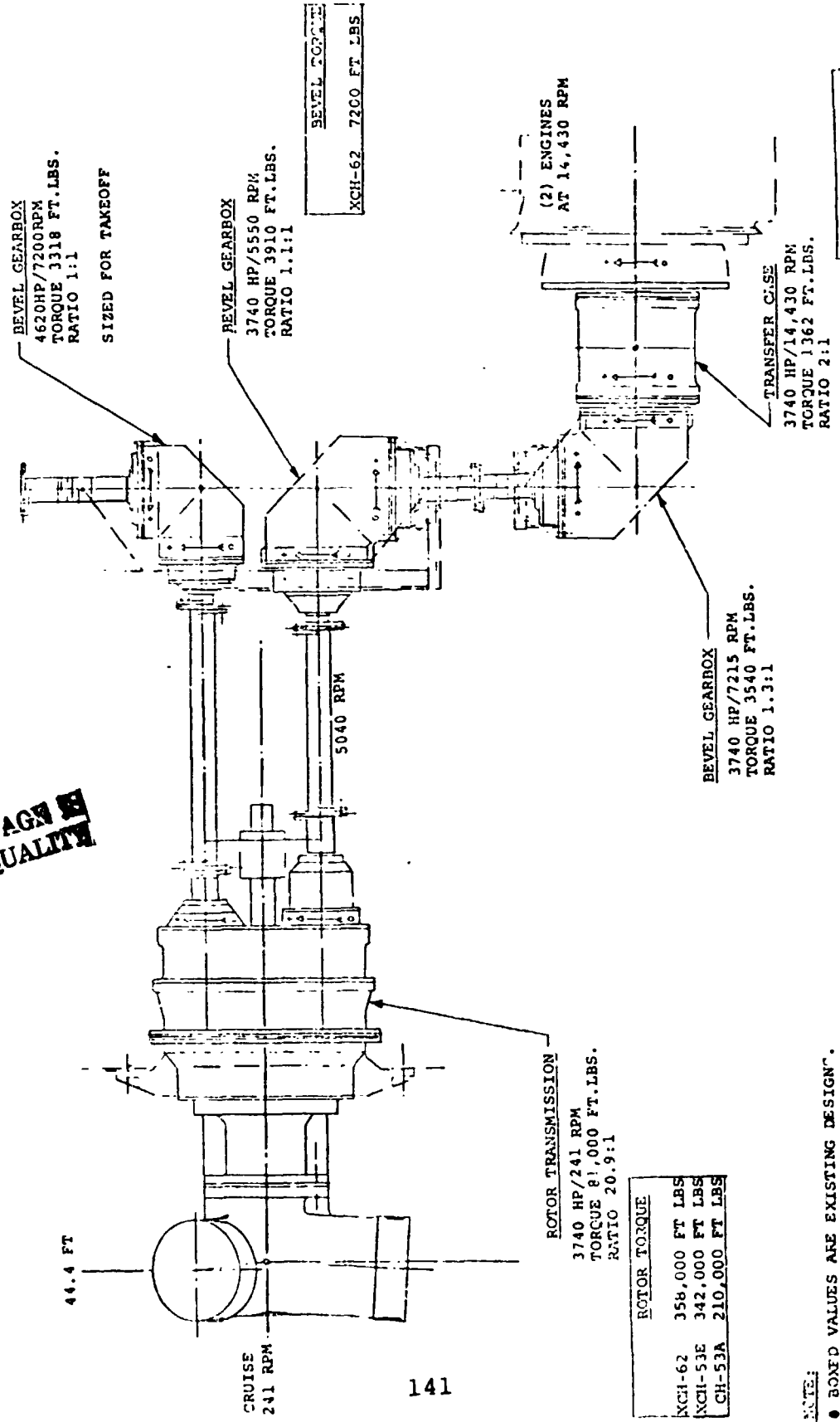
FIGURE 5.1.   COMPARISON OF TRANSMISSION DESIGN TECHNOLOGY
FOR STOL TILT ROTOR.

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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
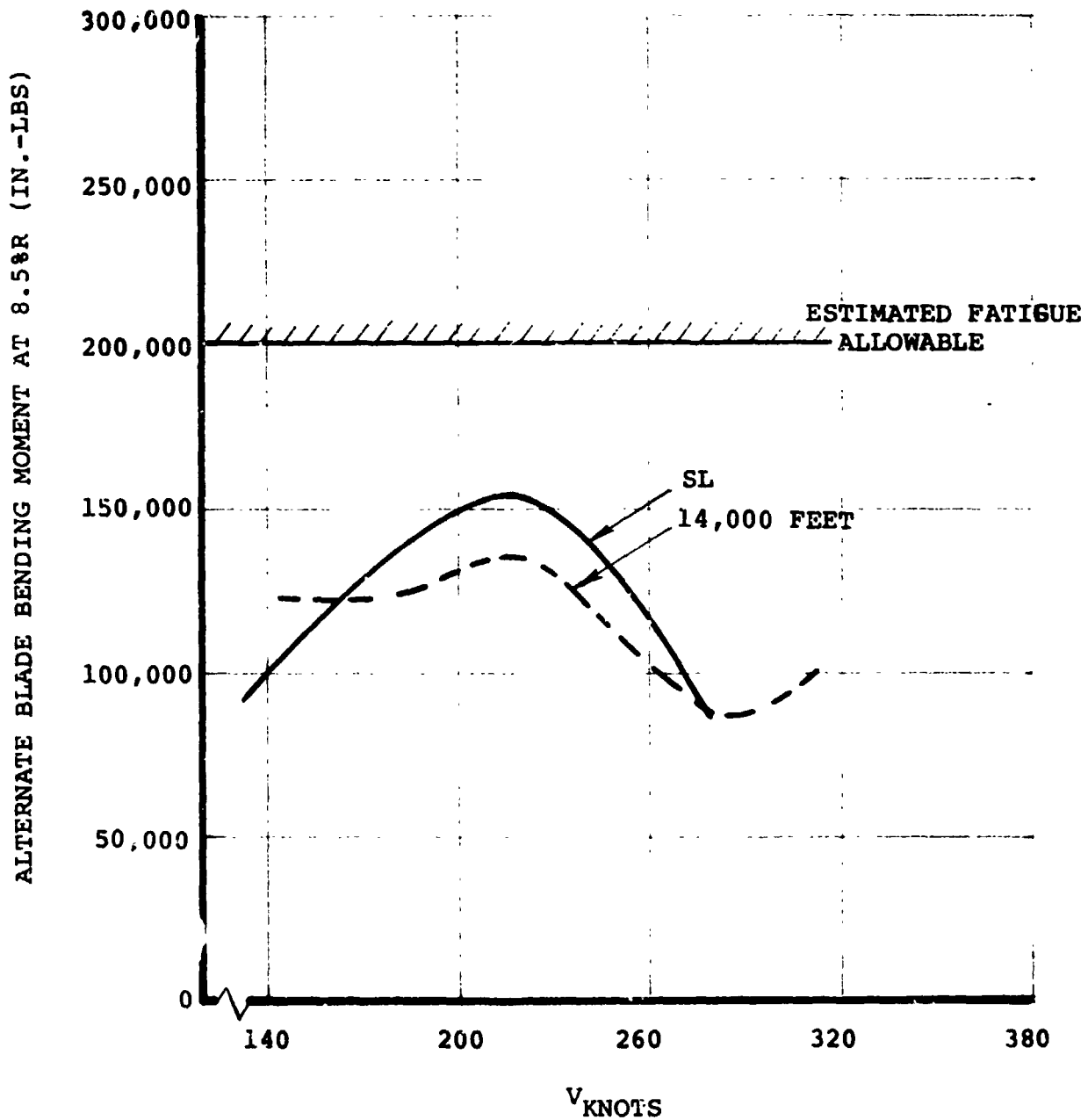
FIGURE 5.2.   ALTERNATE BLADE BENDING LOADS IN CRUISE - 1g
LEVEL FLIGHT.
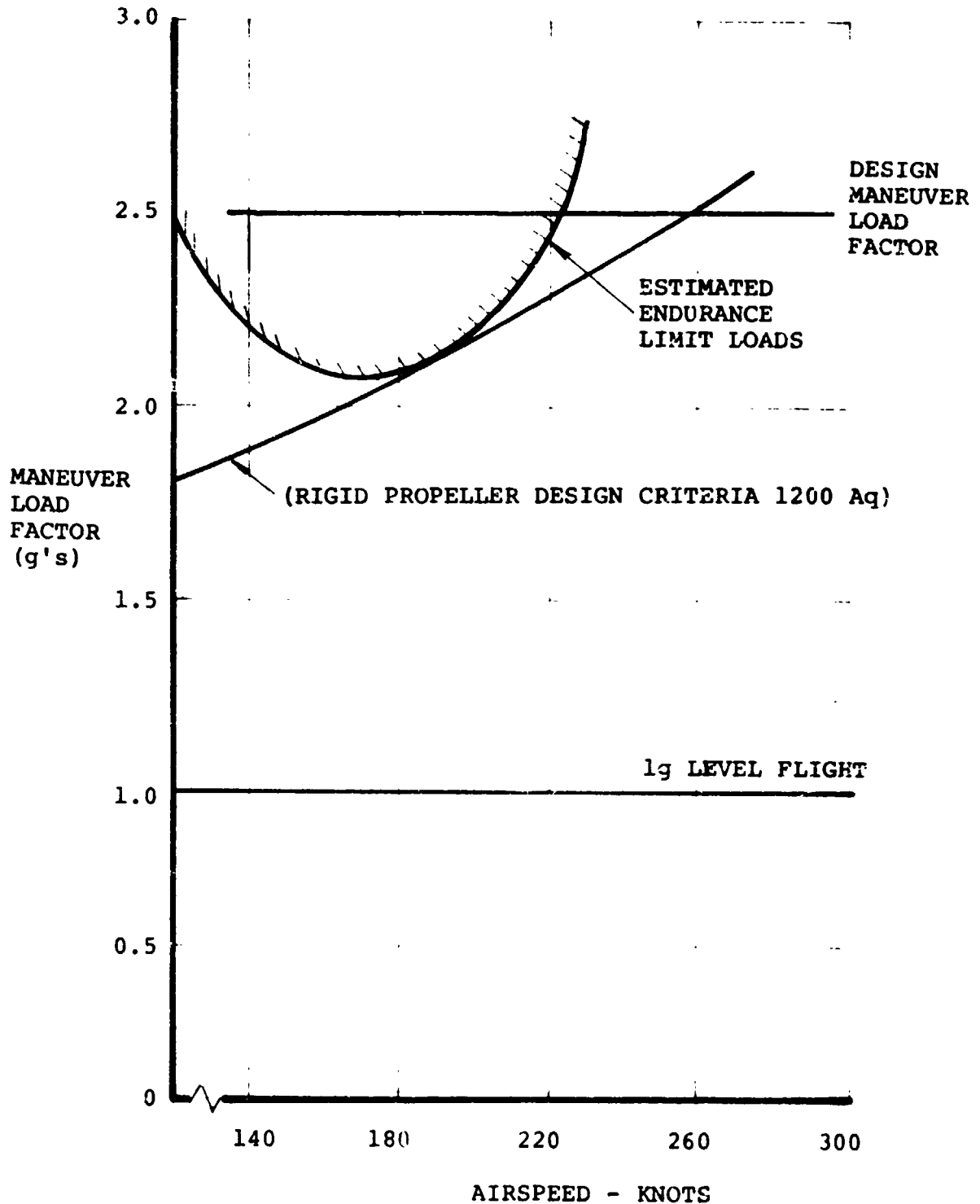
FIGURE 5.3. BLADE FATIGUE LIMITS ON MANEUVER ENVELOPE.

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

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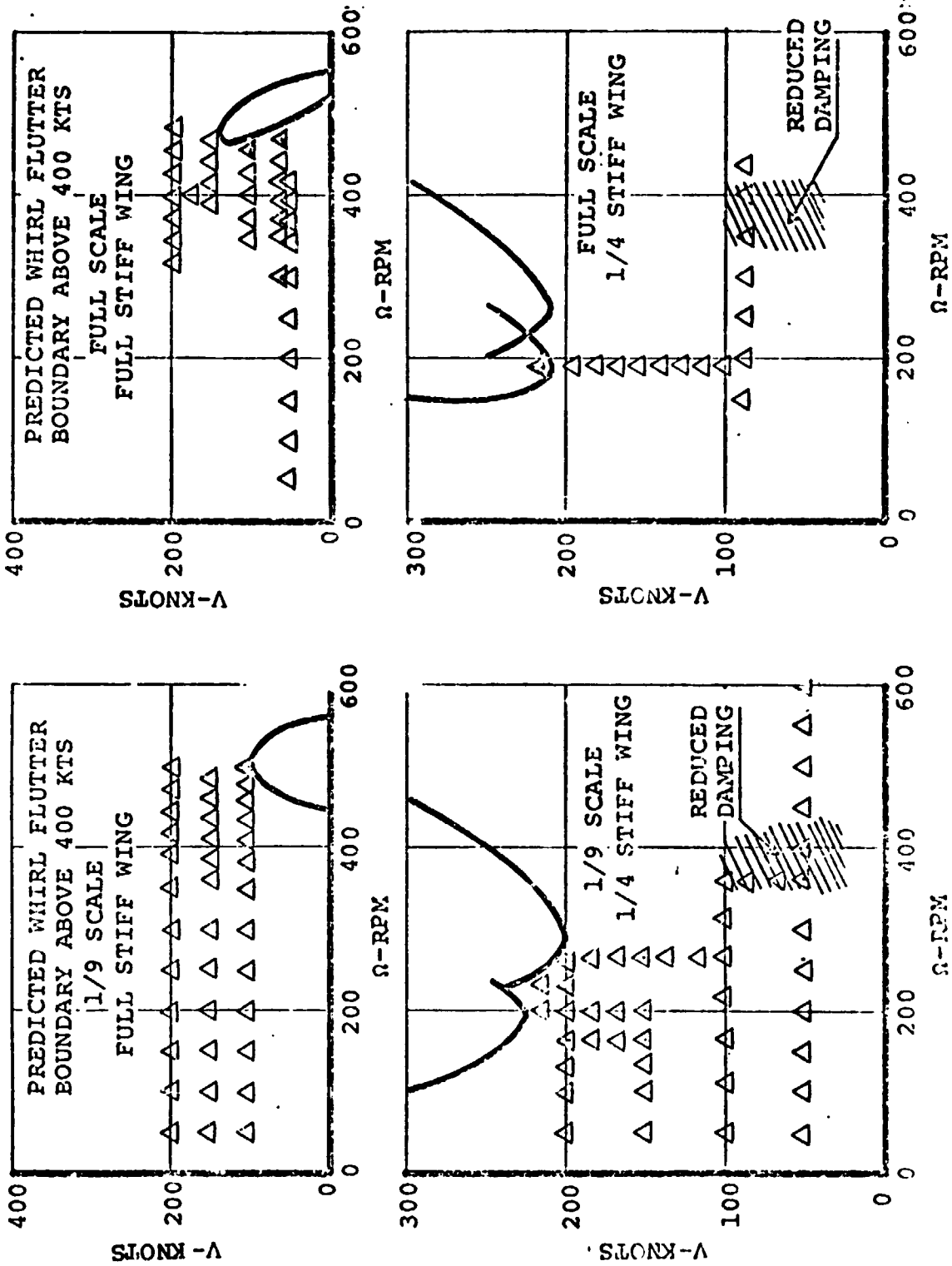
FIGURE 5.4. SCALING SUMMARY.

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 para-
meter 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

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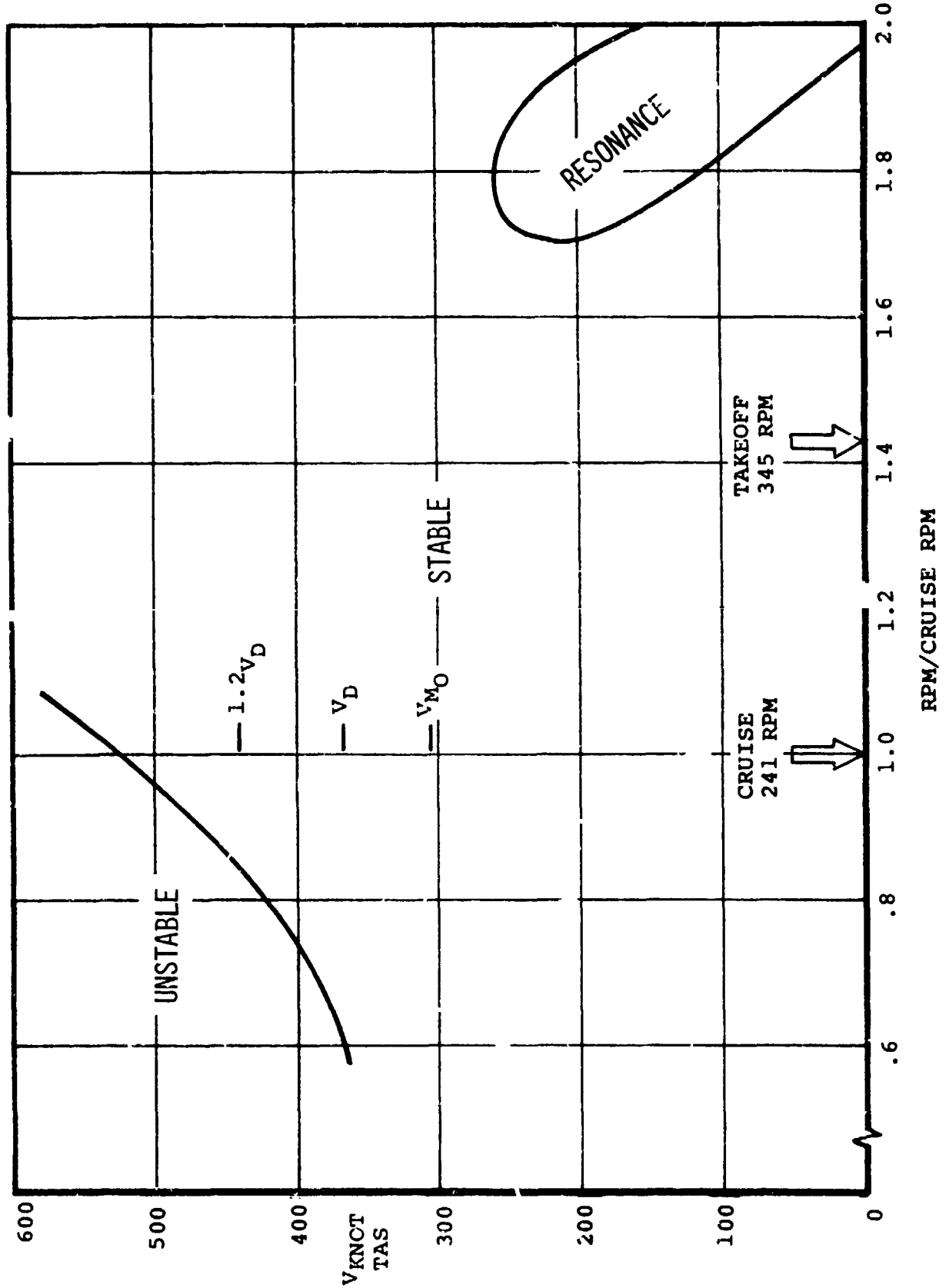
FIGURE 5.5.   AEROELASTIC STABILITY BOUNDARIES FOR STOL TILT ROTOR AT
SEA LEVEL.

151
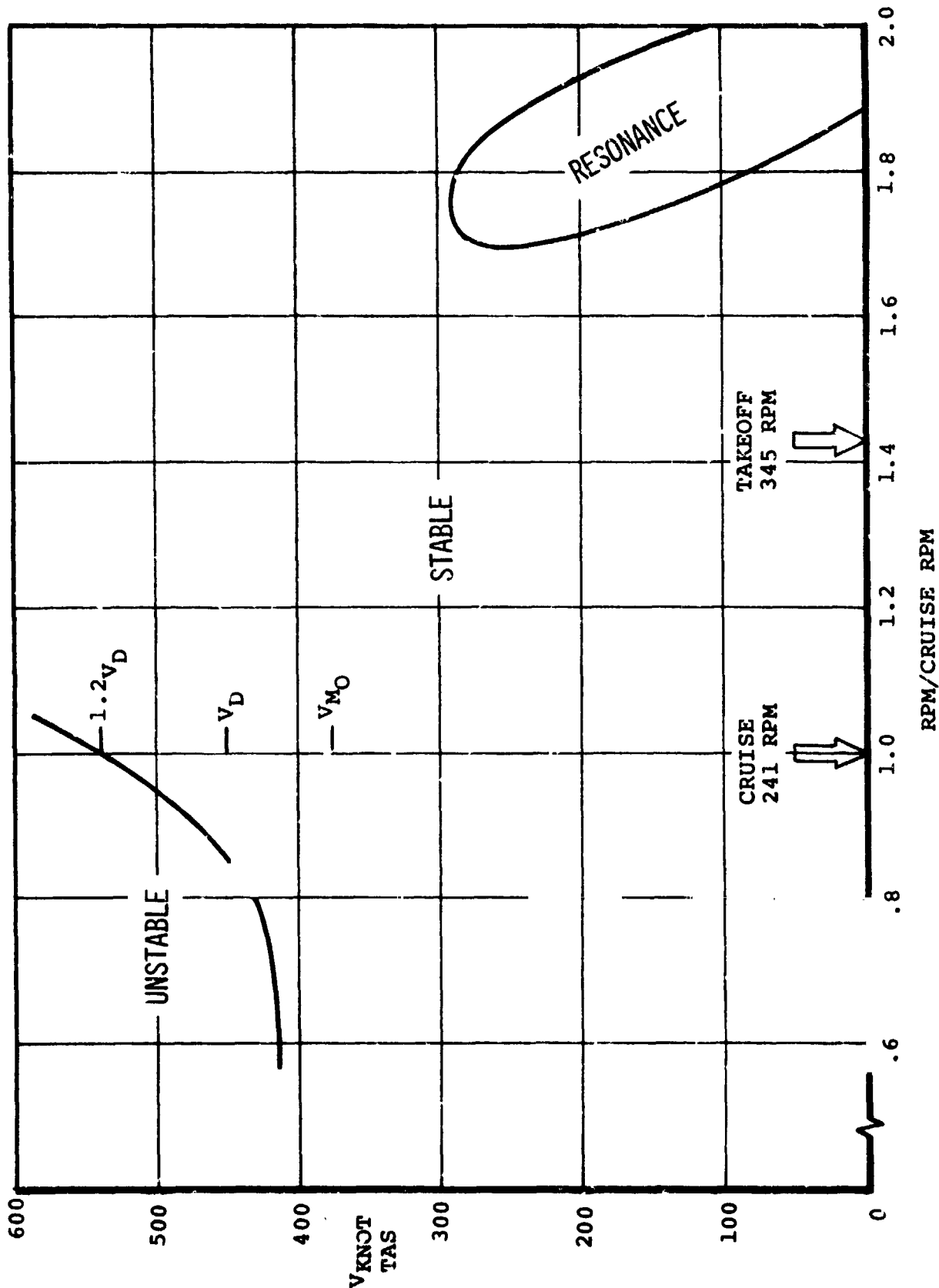
FIGURE 5.6. AEROELASTIC STABILITY BOUNDARIES FOR STOL TILT ROTOR AT 13,300 FEET ALTITUDE.

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.

SELECTED
OPTIMUM AIRCRAFT
FOR
COMMERCIAL APPLICATION

COMPONENT DEVELOPMENT AIRCRAFT (APPROXIMATELY)

100 PASSENGER TANDEM ROTOR HELICOPTER

PASSENGERS

ROTOR DISC LOADING - PSF

50 75 100
15
12.5
10

DIRECT OPERATING COST

CENTS / SEAT MILE

4.0 3.6 3.2 2.8 2.4 2.0
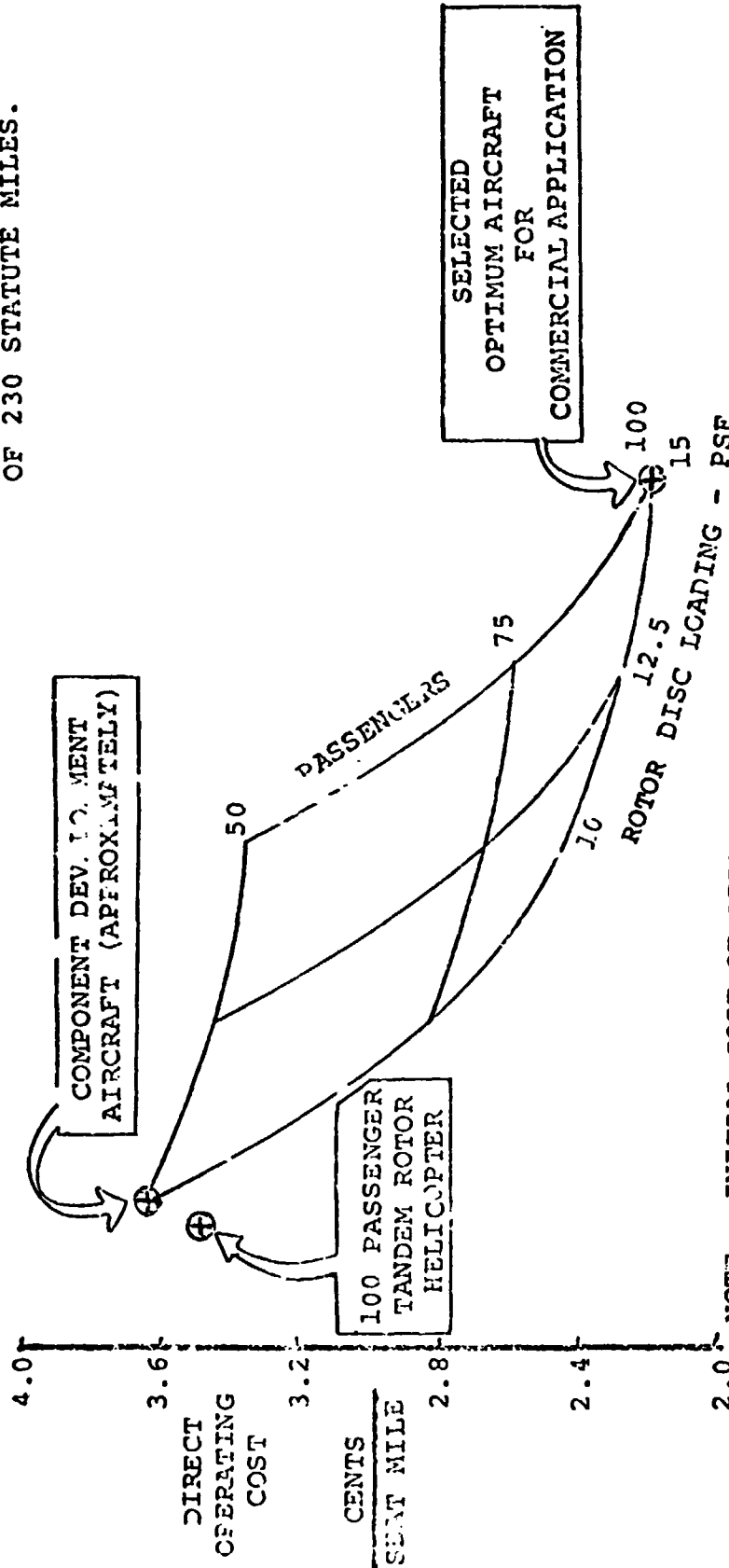
NOTE: INITIAL COST OF AIRFRAME BASED ON $90 PER LB AND UTILIZATION OF 2500 HOURS PER YEAR AND A RANGE OF 230 STATUTE MILES.

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

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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 con-
struction 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

155
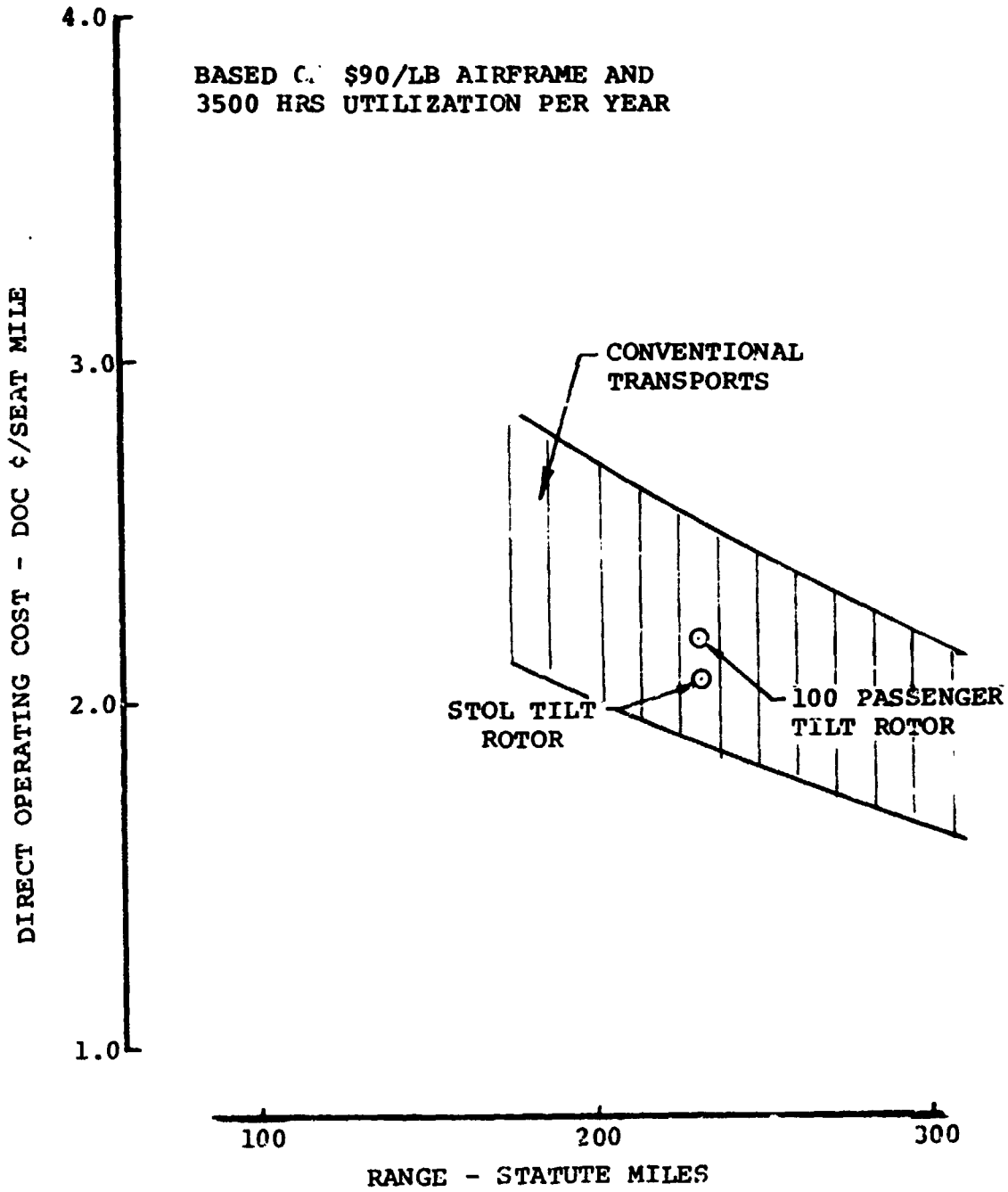
FIGURE 5.8.   COMPARISON OF STOL AND VTOL TILT ROTOR AIRCRAFT
DOC WITH CONVENTIONAL TRANSPORTS.

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:

o Initial cost is reduced from $5.15M to $4.62M.

o Fuel consumption is improved 32.1% from 47.3 to 62.5 passenger miles per gallon.

c Direct operating cost is reduced from 2.19 to 2.09 cents per seat mile.

o 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:

o 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.

o 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.

o 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.

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

A1

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

A2

1985 1C0 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

V_{NRP} @ 14,000 FT/4,267 m

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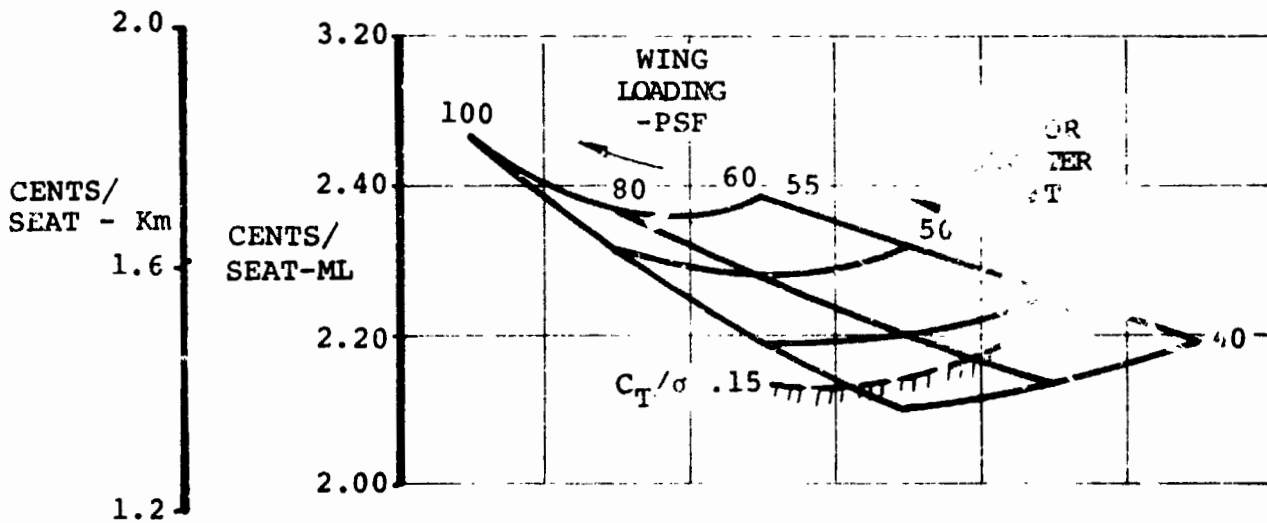
FIGURE A-3. VARIATION OF MAXIMUM CRUISE SPEED AND DIRECT
OPERATING COST WITH WING LOADING AND ROTOR
DIAMETER (T/W = 0.8).

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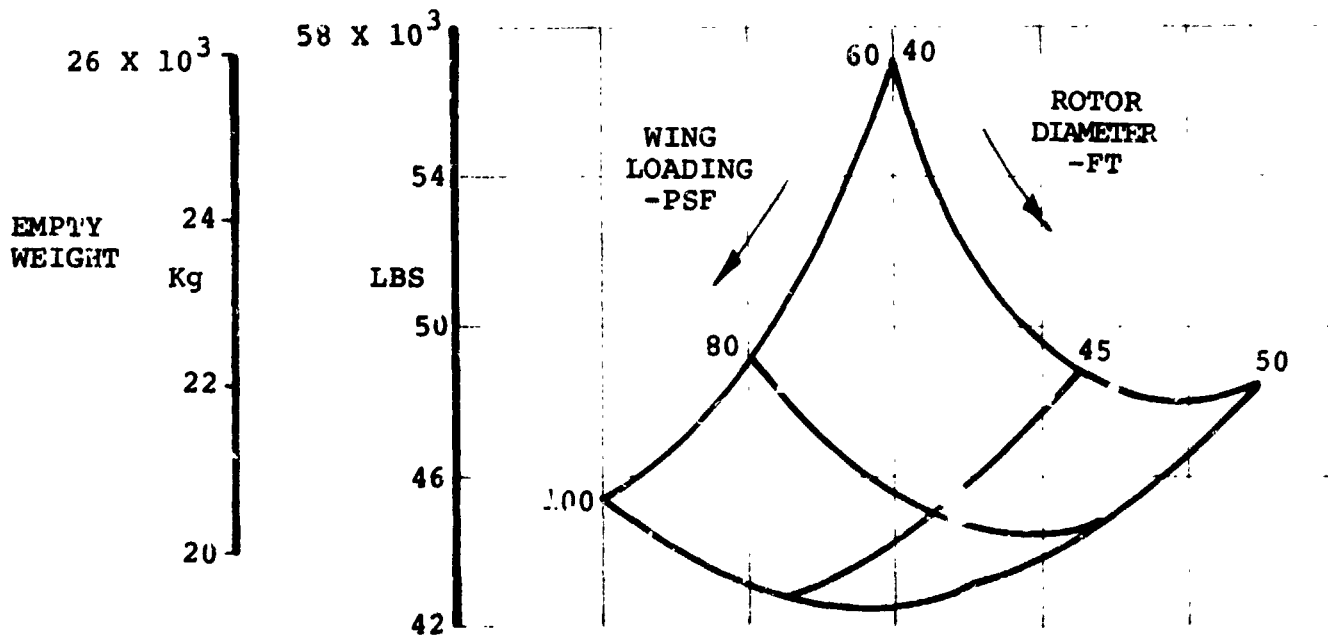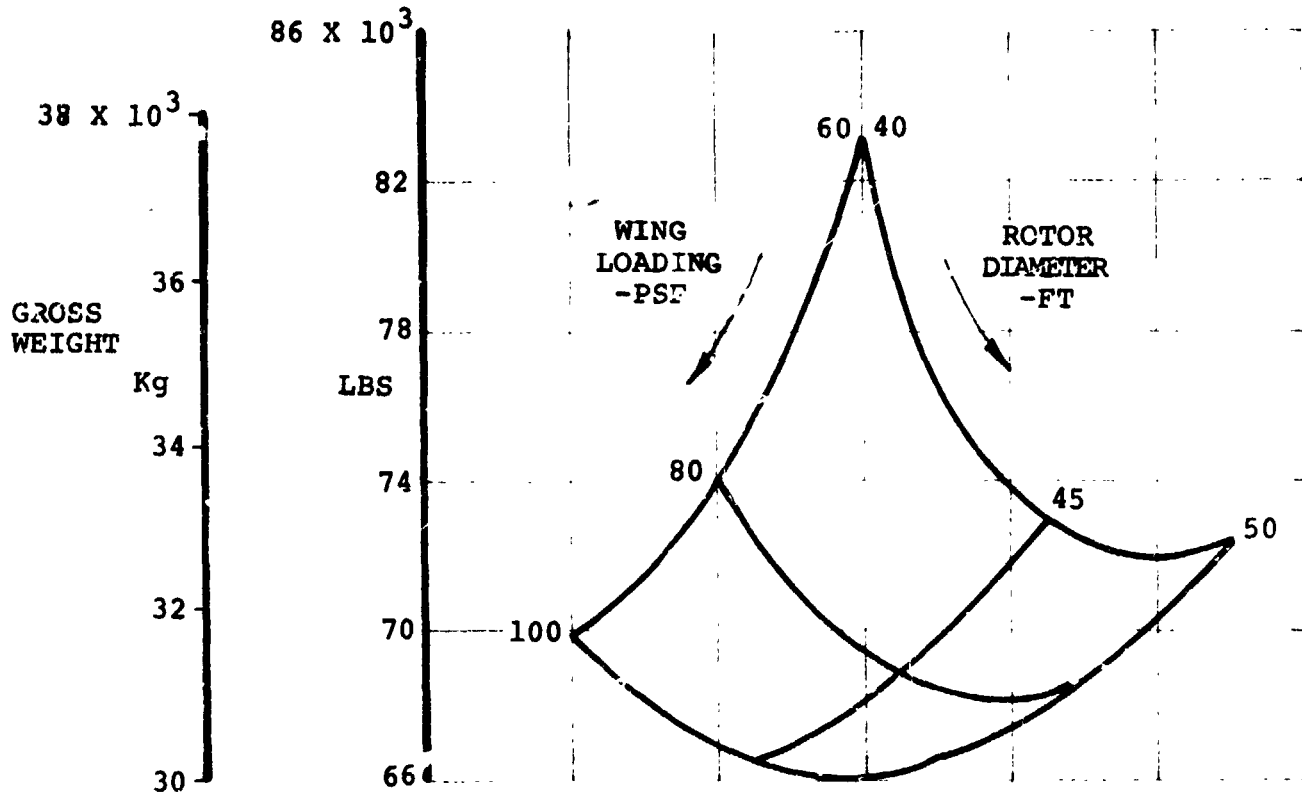
FIGURE A-4. VARIATION OF GROSS WEIGHT AND EMPTY WEIGHT WITH
            WING LOADING AND ROTOR DIAMETER (T/W = 0.95).

A6

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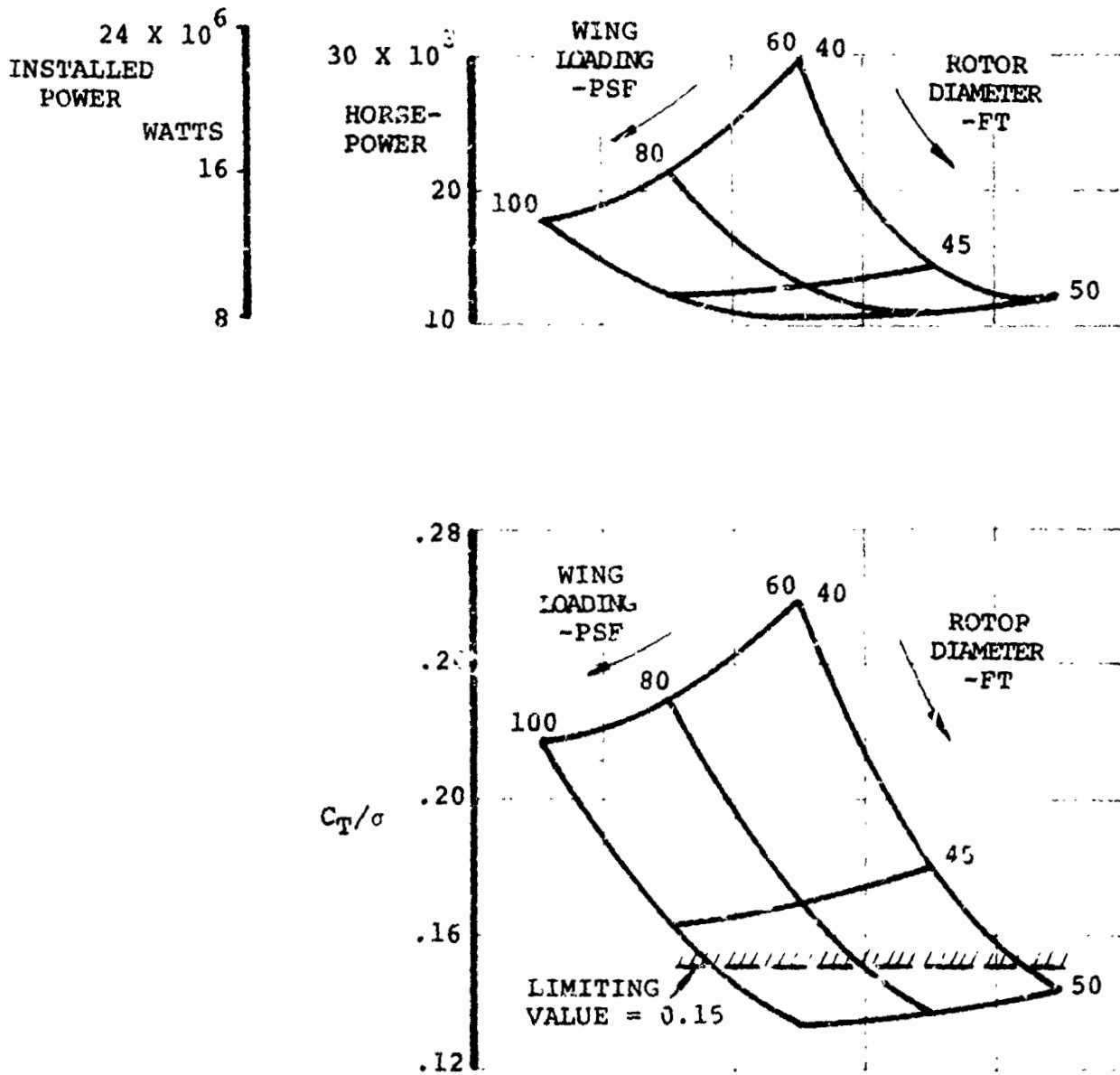
FIGURE A-5. VARIATION OF INSTALLED POWER AND ROTOR LOADING
WITH WING LOADING AND ROTOR DIAMETER (T/W = 0.95).

A7

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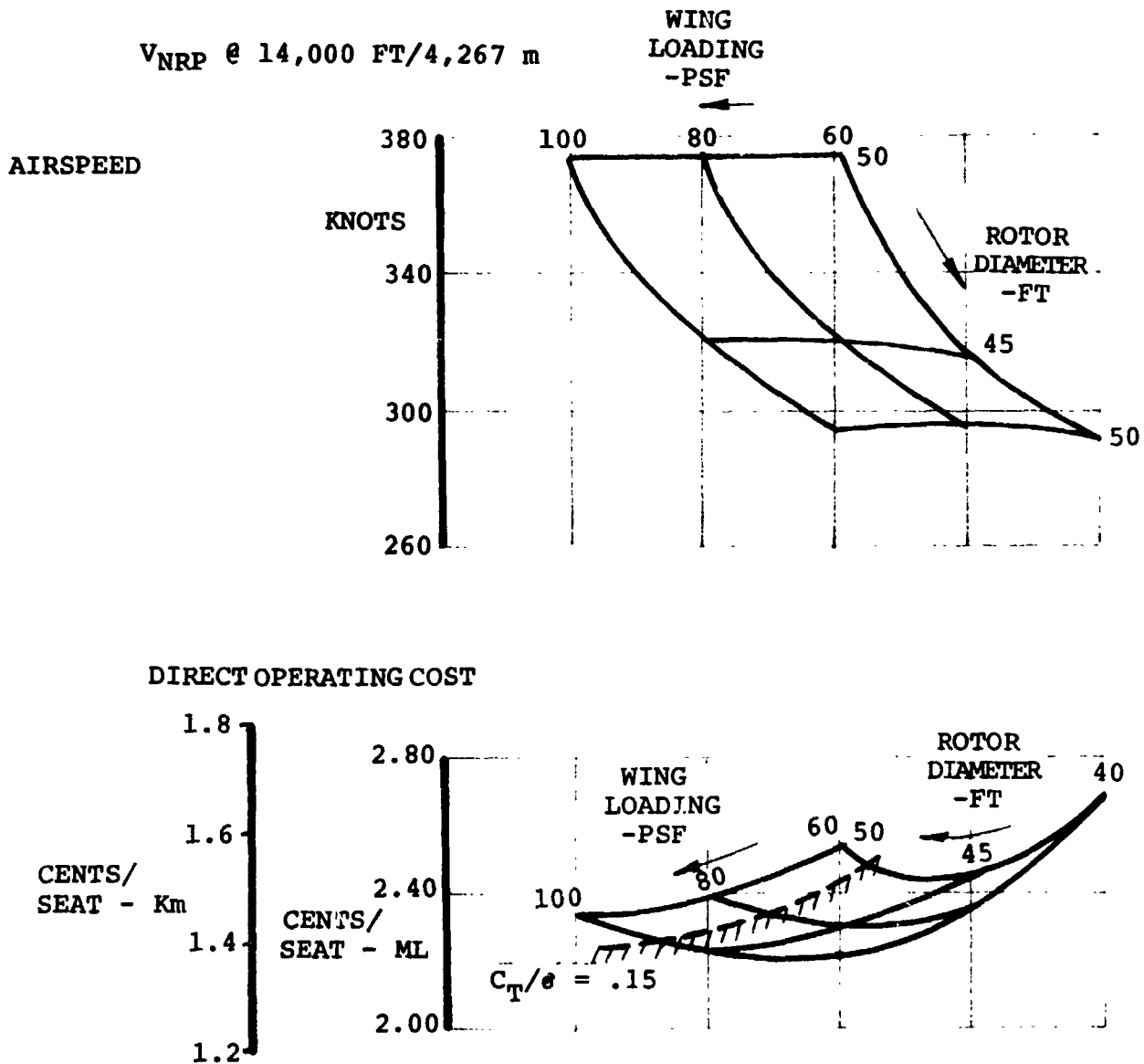
FIGURE A-6. VARIATION OF MAXIMUM CRUISE SPEED AND DIRECT OPERATING COST WITH WING LOADING AND ROTOR DIAMETER (T/W = 0.95).

A8

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ASPECT RATIO   = 7.14
ROTOR SOLIDITY   = 0.09
ROTOR TIPSPEED   = 775 FT/SEC @ T.0.
STATIC THRUST/WEIGHT = 1.05
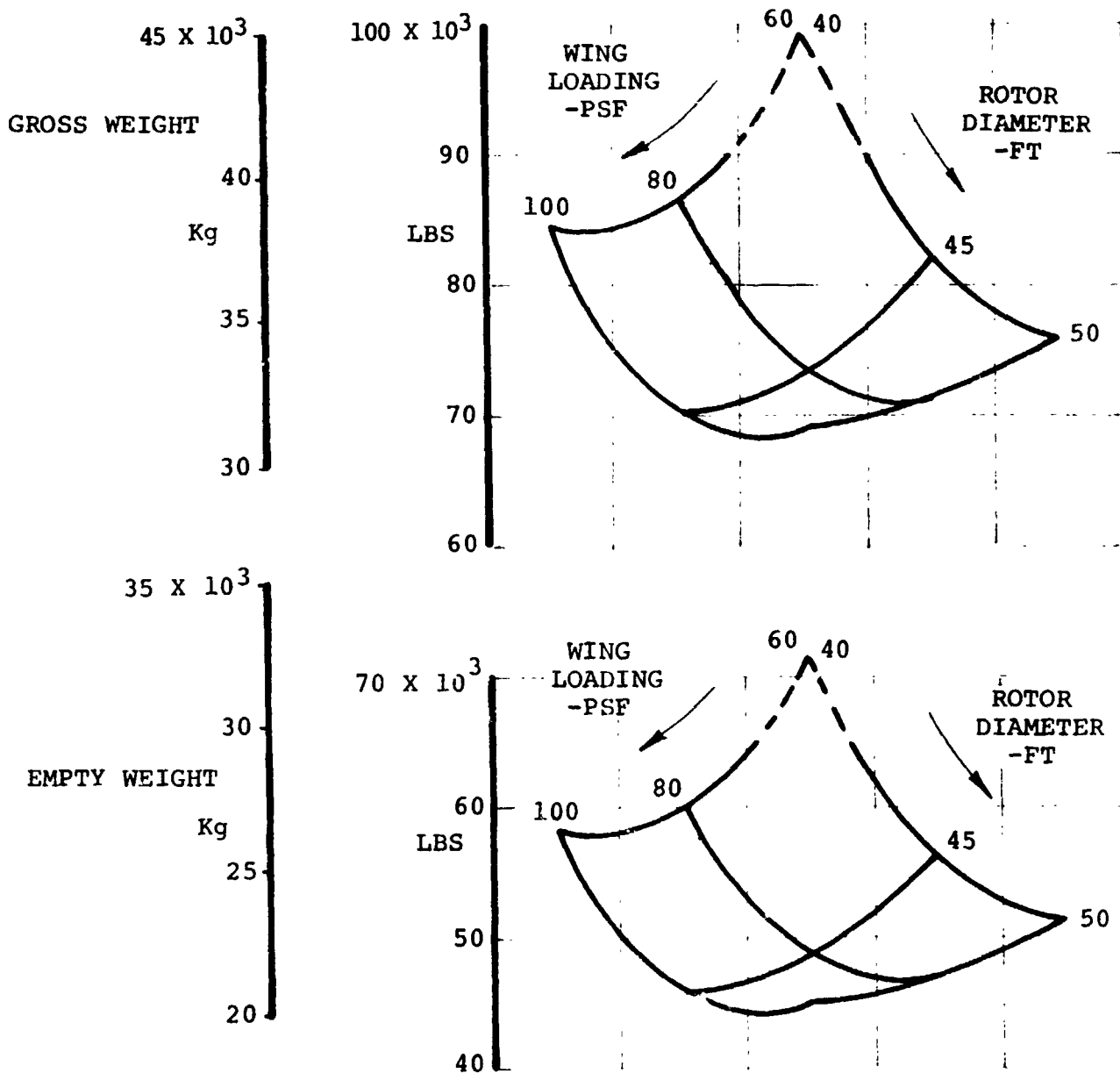
FIGURE A-7. VARIATION OF GROSS WEIGHT AND EMPTY WEIGHT WITH
WING LOADING AND ROTOR DIAMETER (T/W = 1.05).

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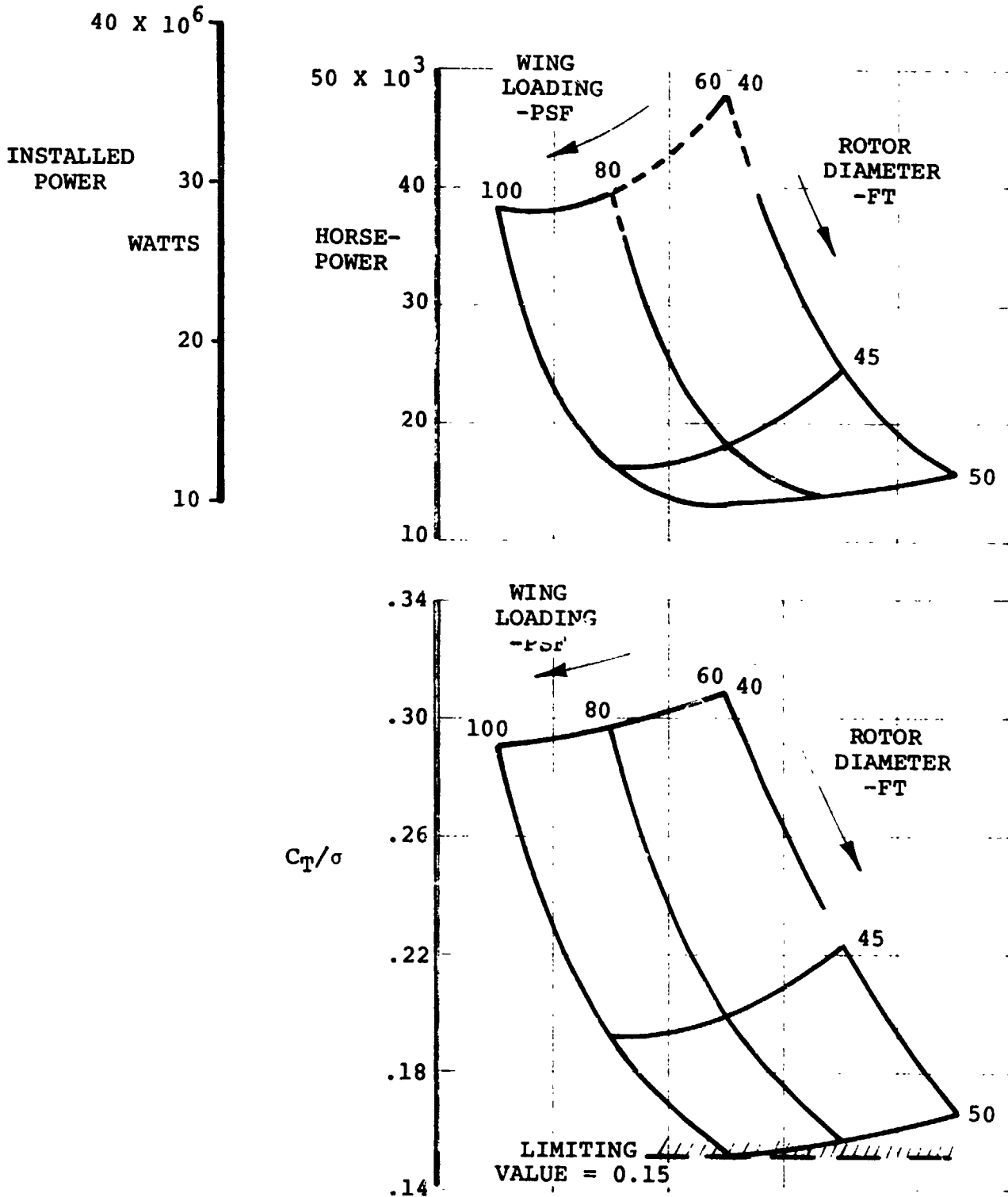
FIGURE A-8. VARIATION OF INSTALLED POWER AND ROTOR LOADING
WITH WING LOADING AND ROTOR DIAMETER (T/W = 1.05).

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ASPECT RATIO   = 7.14
ROTOR SOLIDITY   = 0.09
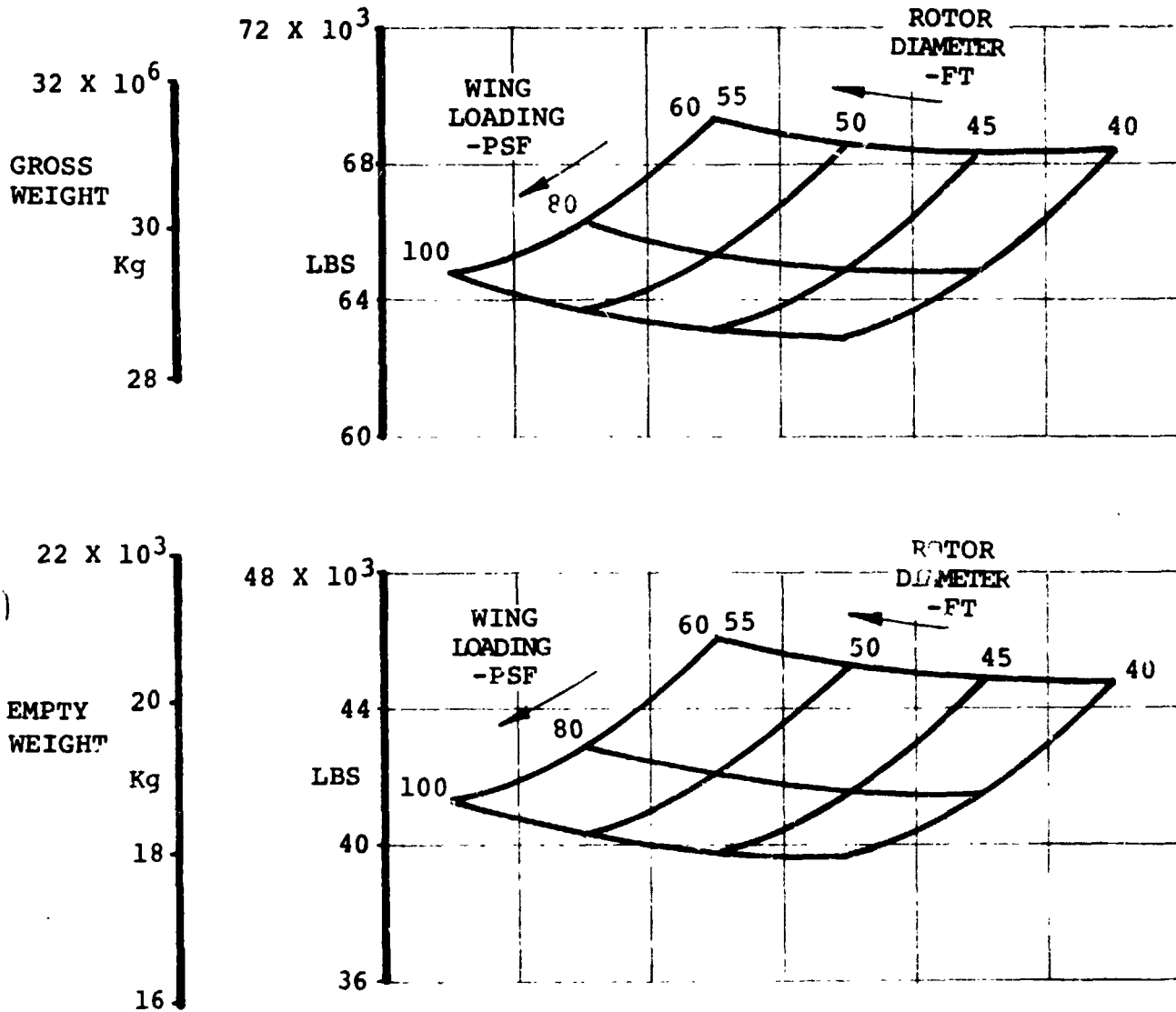ROTOR TIPSPEED   = 775 FT/SEC @ T.O.
STATIC THRUST/WEIGHT = 1.05
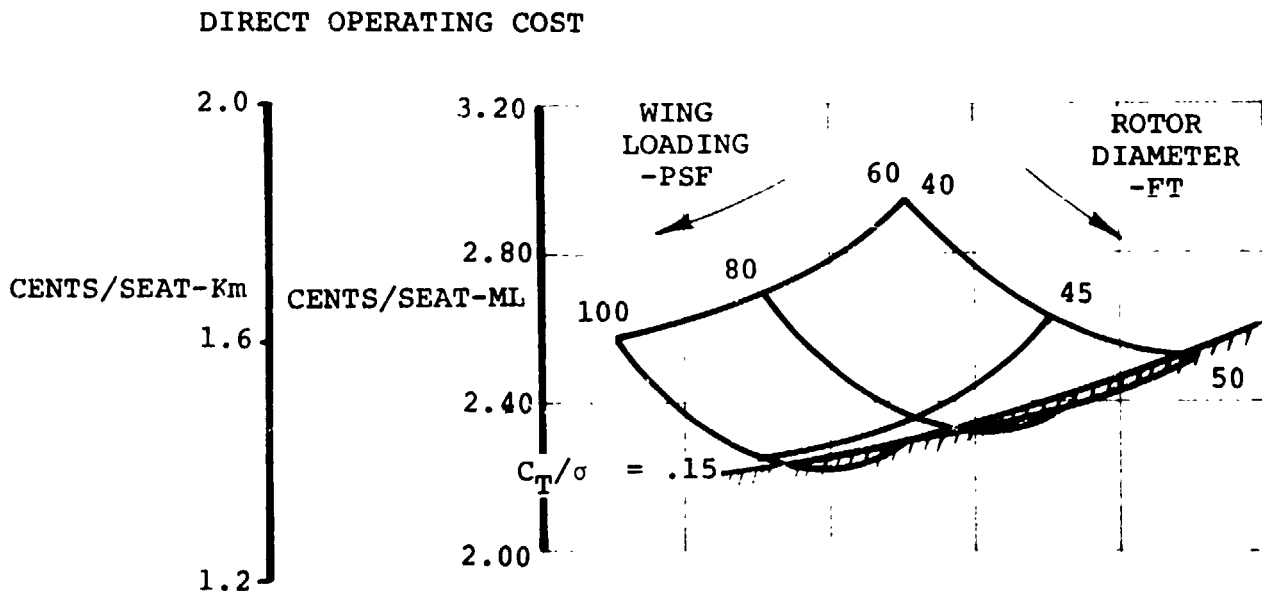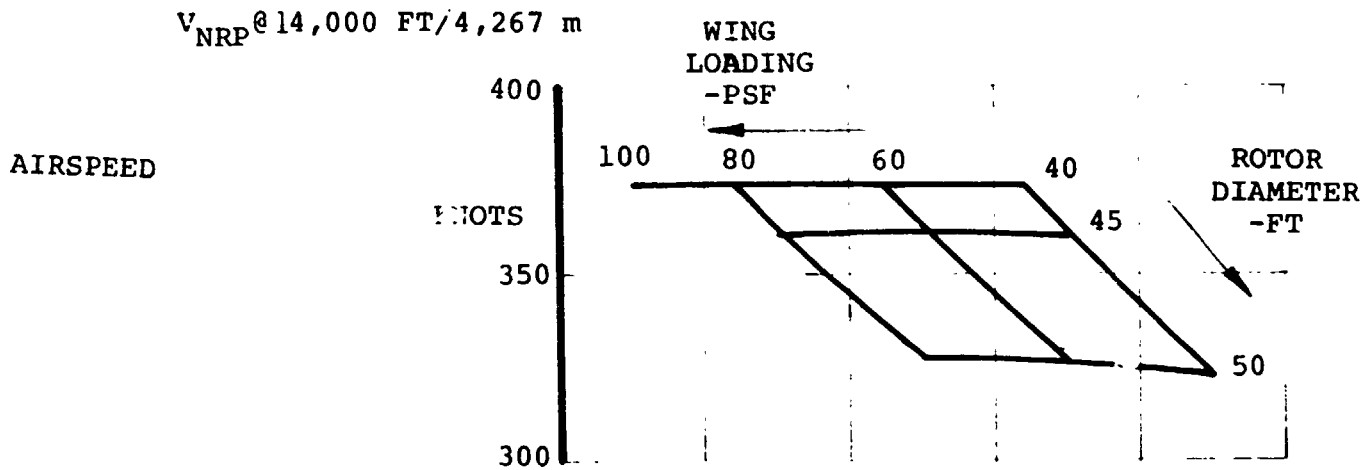
FIGURE A-9. VARIATION OF MAXIMUM CRUISE SPEED AND DIRECT OPERATING COST WITH WING LOADING AND ROTOR DIAMETER (T/W = 1.05).

A11

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 operat-

ing 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 para-

meters selected. The results indicated that the takeoff

performance was marginal in that field lengths of slightly

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ASPECT RATIO   = 7.14
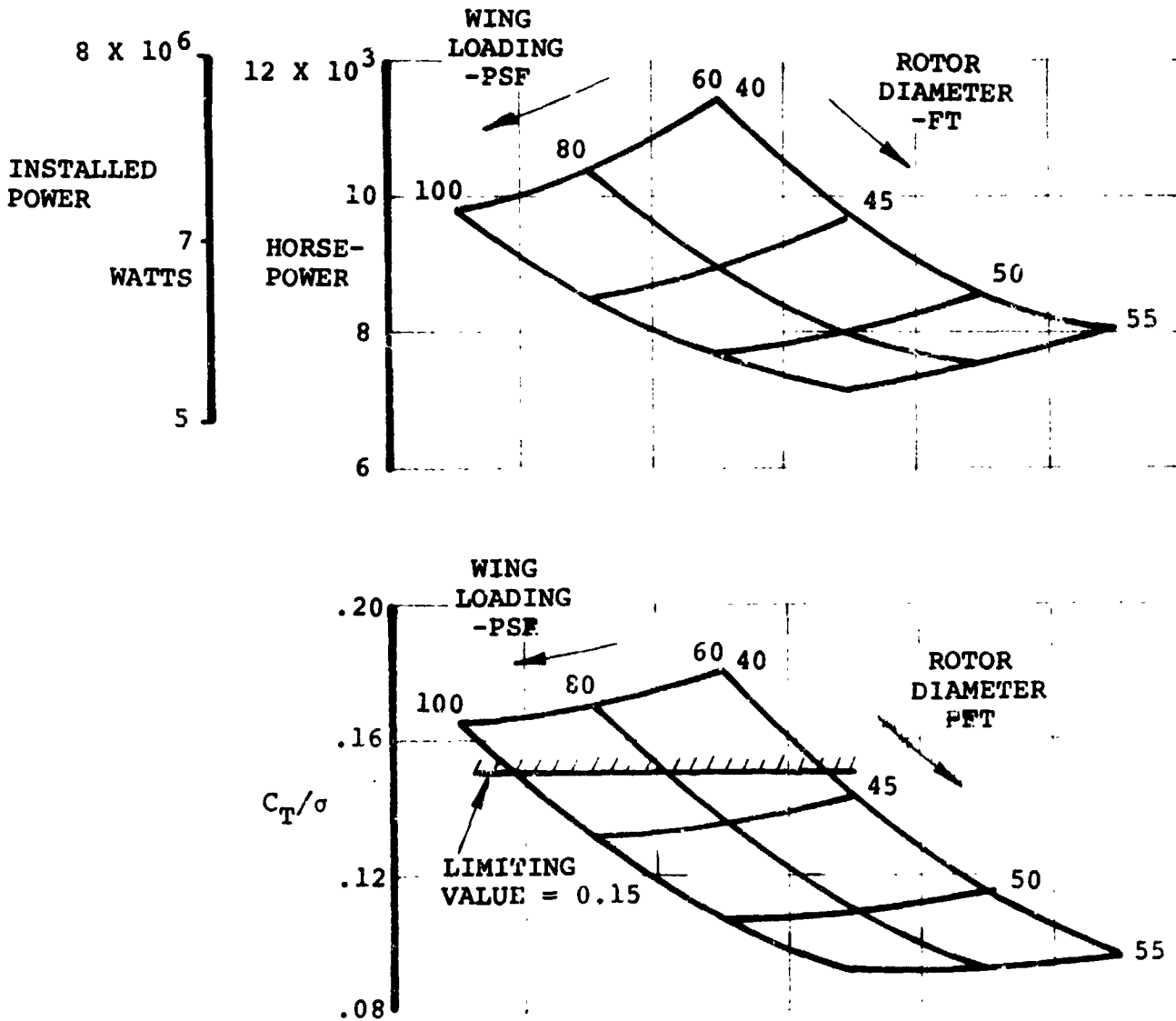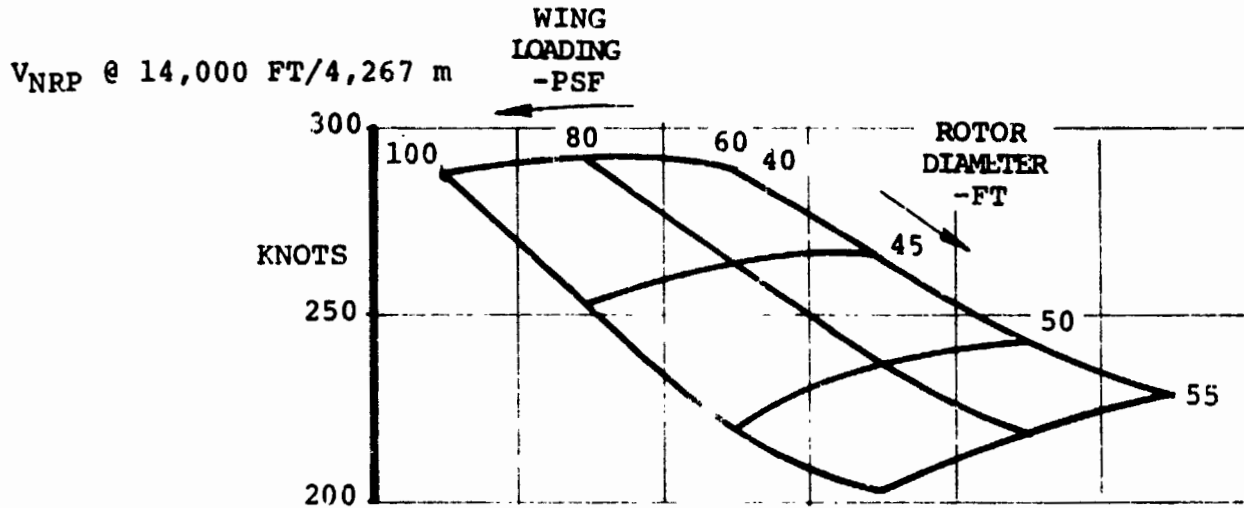ROTOR SOLIDITY = 0.09
ROTOR TIPSPEED = 775 FPS AT TAKEOFF

C_T/σ = 0.15
W/S   = 100 PSF   } SIZING CONSTRAINTS


ROTOR DIAMETER FOR MINIMUM DOC IS 44.4 FEET (13.53 m)

T/W FOR MINIMUM DOC IS 0.88
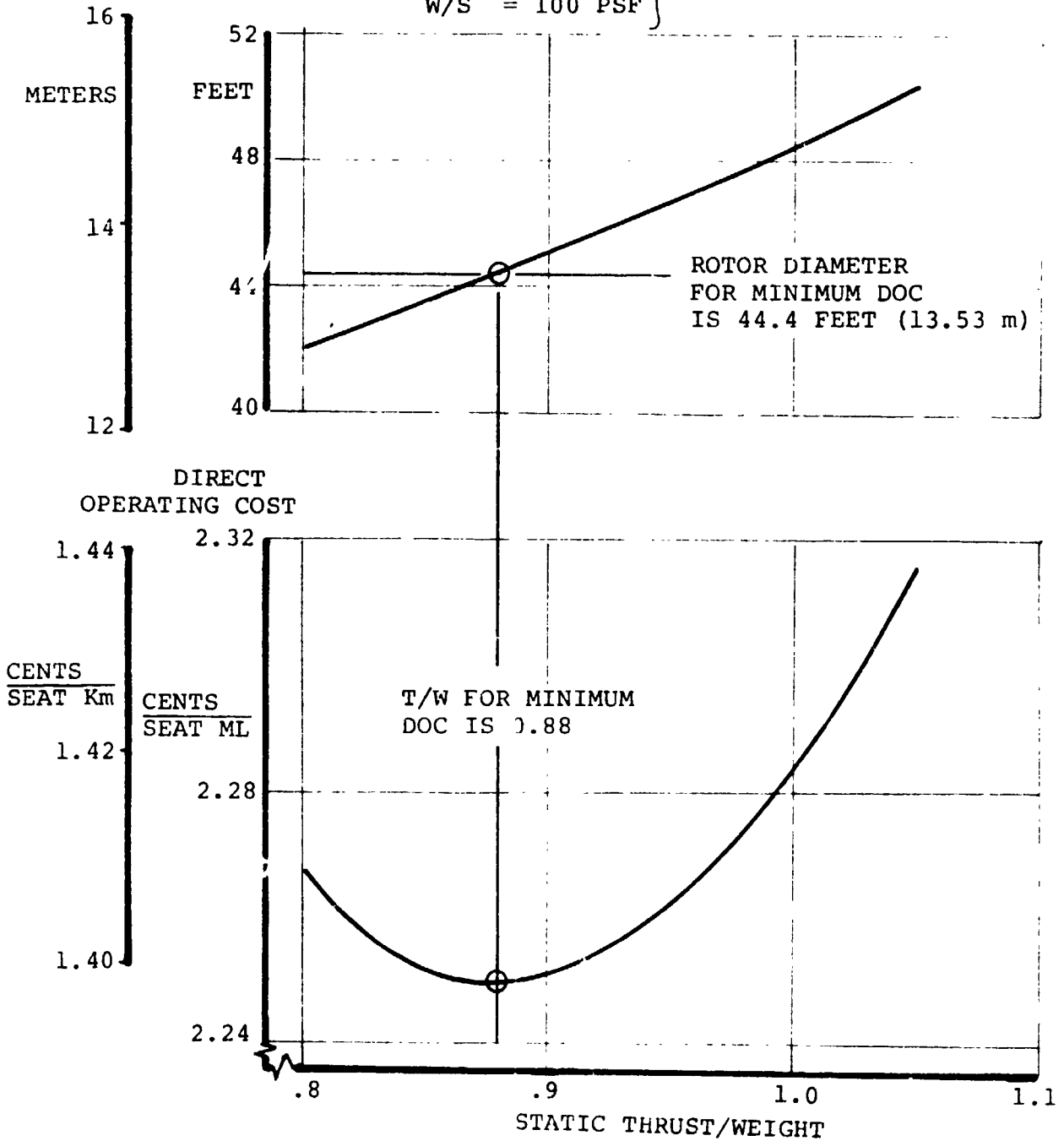
FIGURE A-10.   VARIATION OF ROTOR DIAMETER AND MINIMUM DIRECT OPERATING COST WITH STATIC THRUST/WEIGHT RATIO.

A13

greater than the required 2,000 feet were required. How-
ever, the process of optimizing the aircraft parameters had
not been completed and detail geometry related to the take-
off configuration had not been defined, so a considerable
improvement in takeoff performance could be expected. Con-
sequently, 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, take-
off 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 calcu-
lations 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

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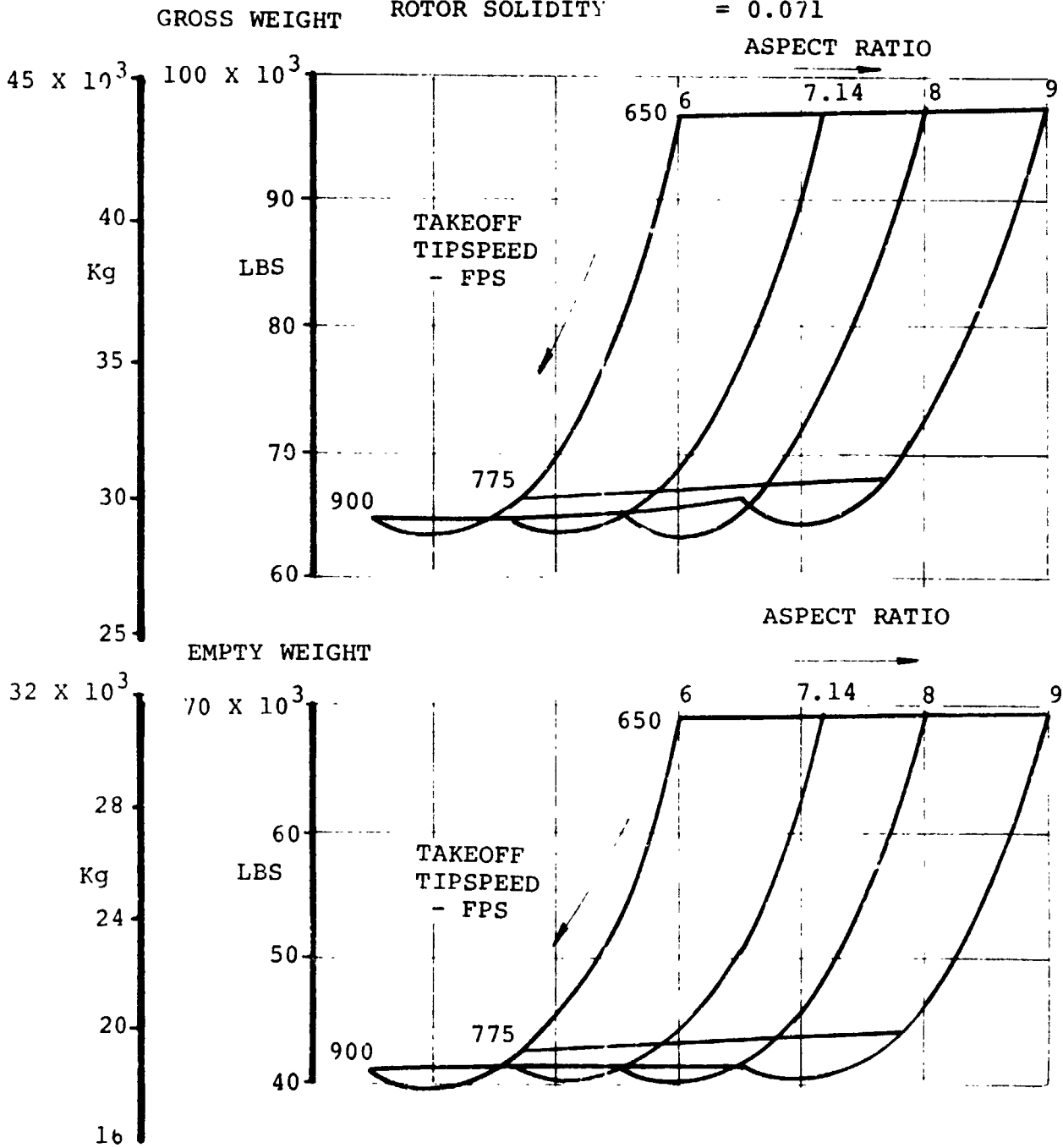
FIGURE A-11.   VARIATION OF GROSS WEIGHT AND EMPTY WEIGHT WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.071).

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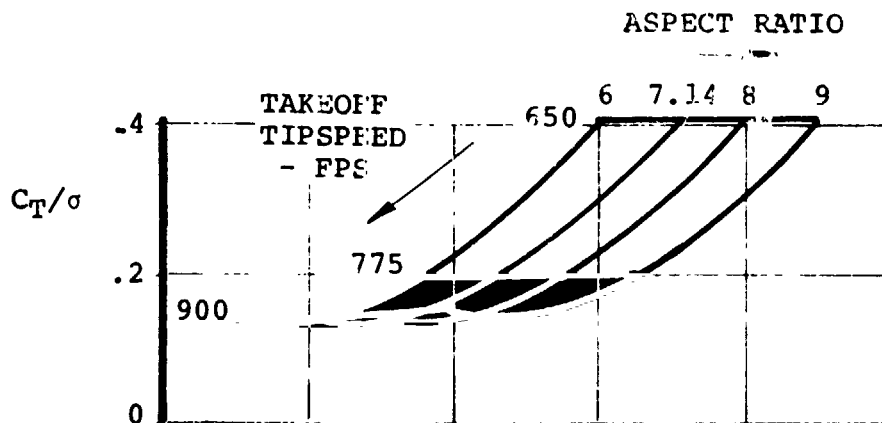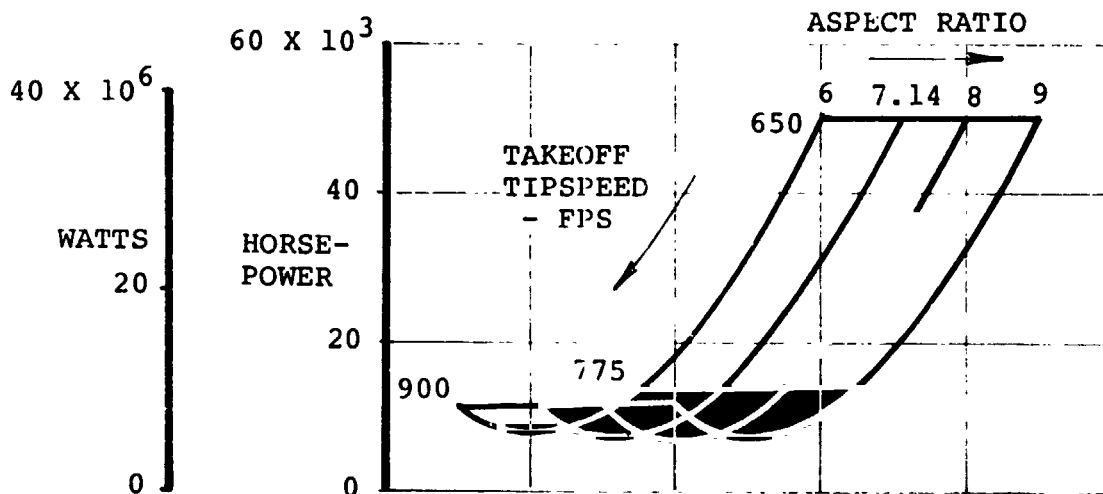
FIGURE A-12. VARIATION OF INSTALLED POWER AND ROTOR LOADING WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.071).

A16

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

V_{NRP} AT 14,000 FT/4,267 m

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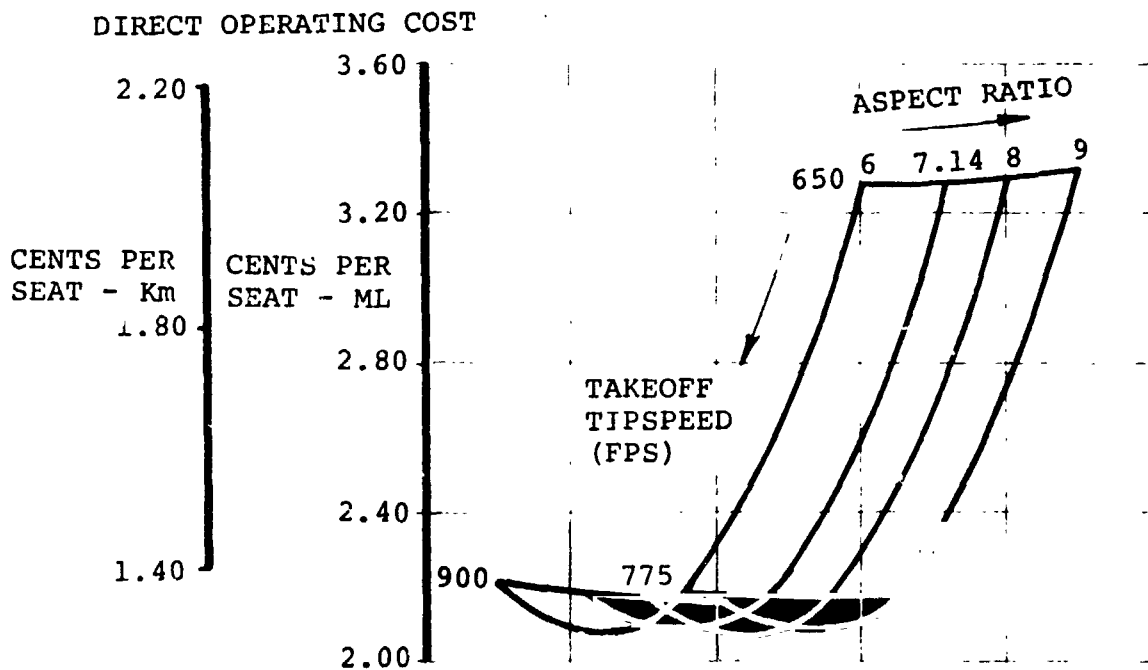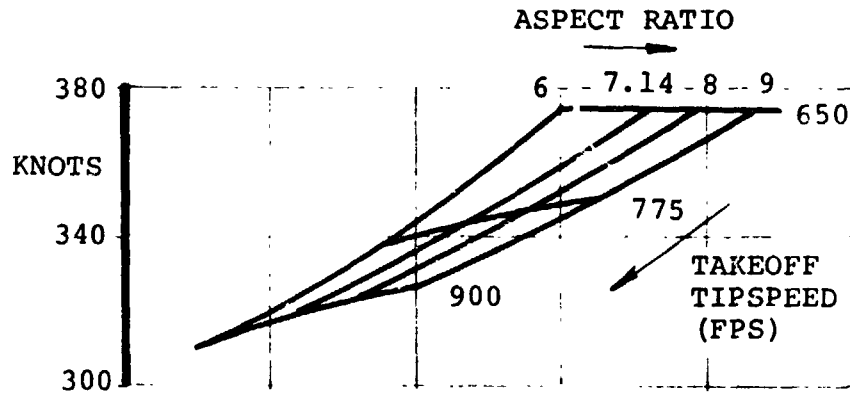
FIGURE A-13.   VARIATION OF MAXIMUM CRUISE SPEED AND DIRECT
OPERATING COST WITH ASPECT RATIO AND TAKEOFF
TIPSPEED (σ = .071).

A17

1985 100 PASSENGER STOL TILT ROTOR ░ ░ ░FT

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

GROSS WEIGHT
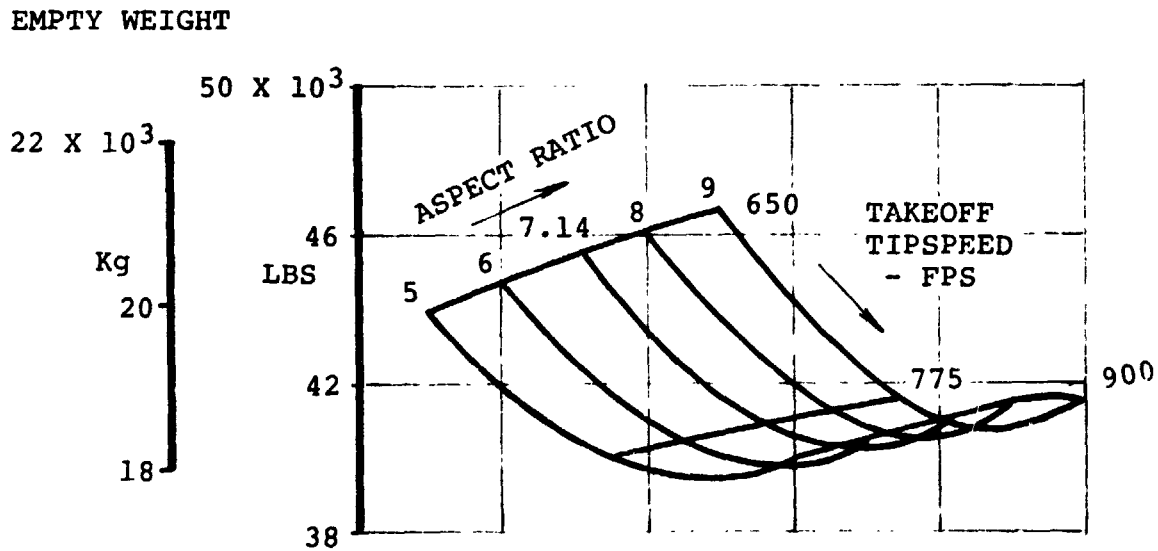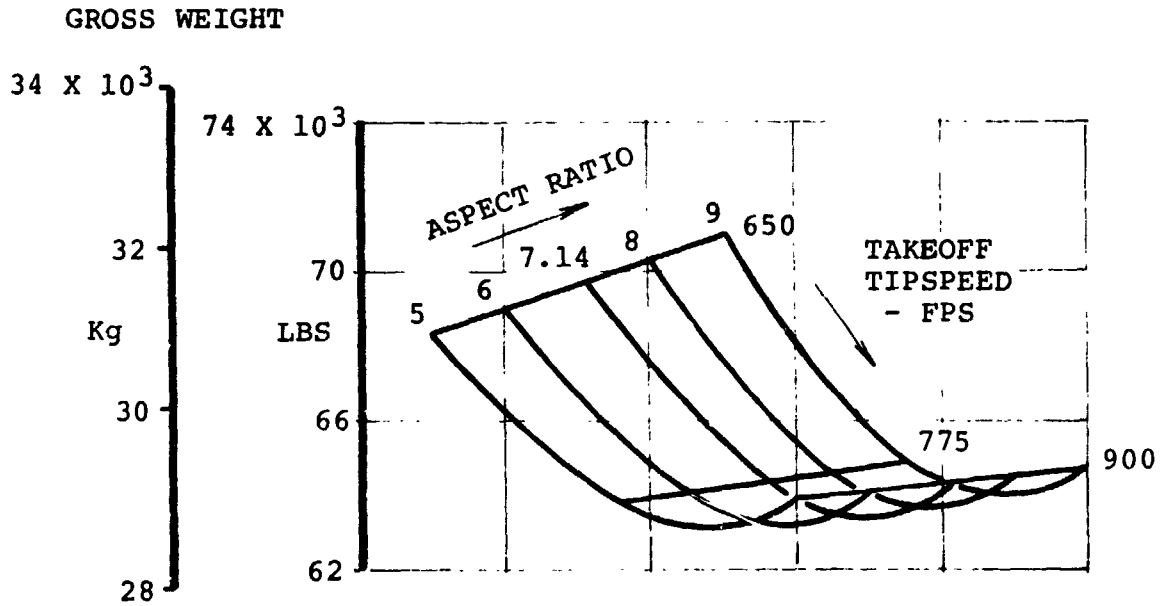
EMPTY WEIGHT

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

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

ROTOR DIAMETER = 44.4 FT/13.53m
WING LOADING = 100 PSF/488Kg/m^2
STATIC THRUST/WEIGHT = 0.88
ROTOR SOLIDITY = 0.09
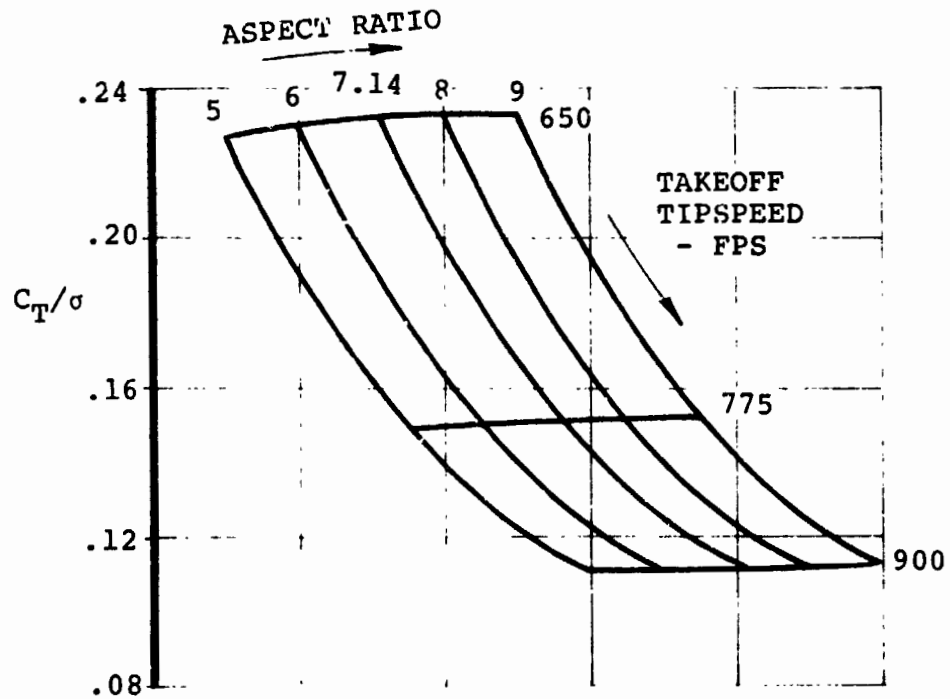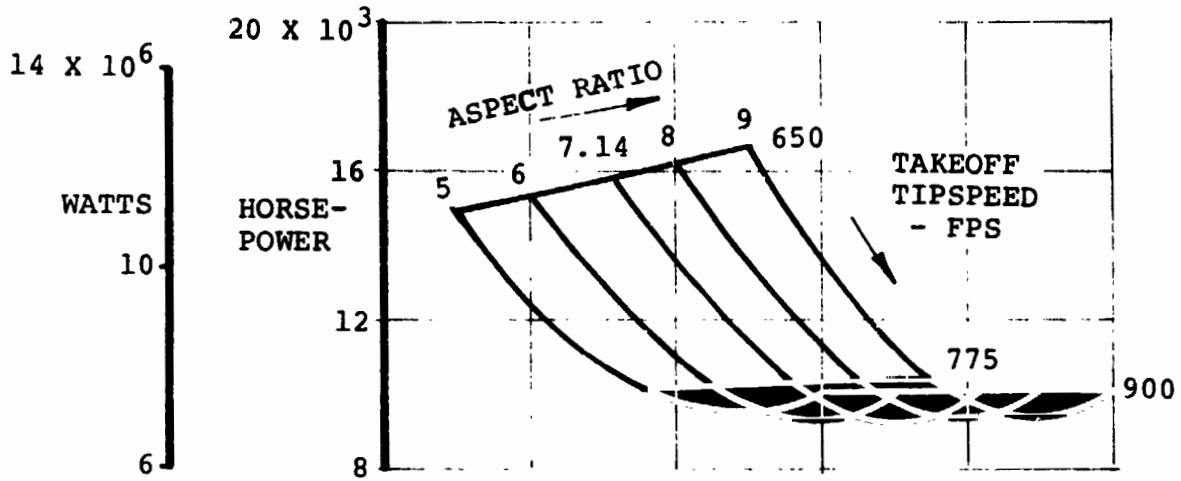
FIGURE A-15. VARIATION OF INSTALLED POWER AND ROTOR LOADING WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.09).

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	= 0.09

V_{NRP} AT 14,000 FT/4,267 m
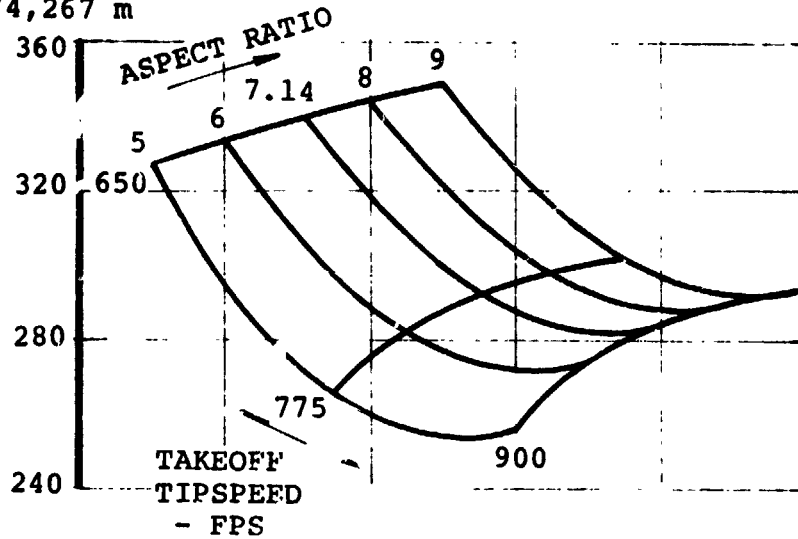
DIRECT OPERATING COST
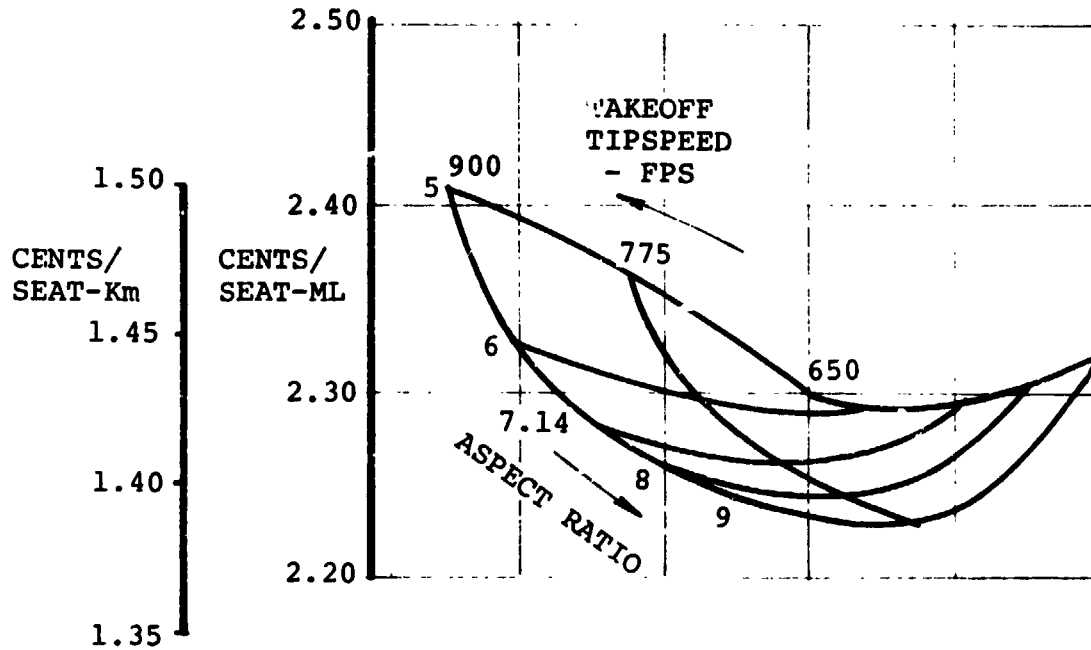
FIGURE A-16.   VARIATION OF MAXIMUM CRUISE SPEED AND DIRECT OPERATING COST WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.09).

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

GROSS WEIGHT
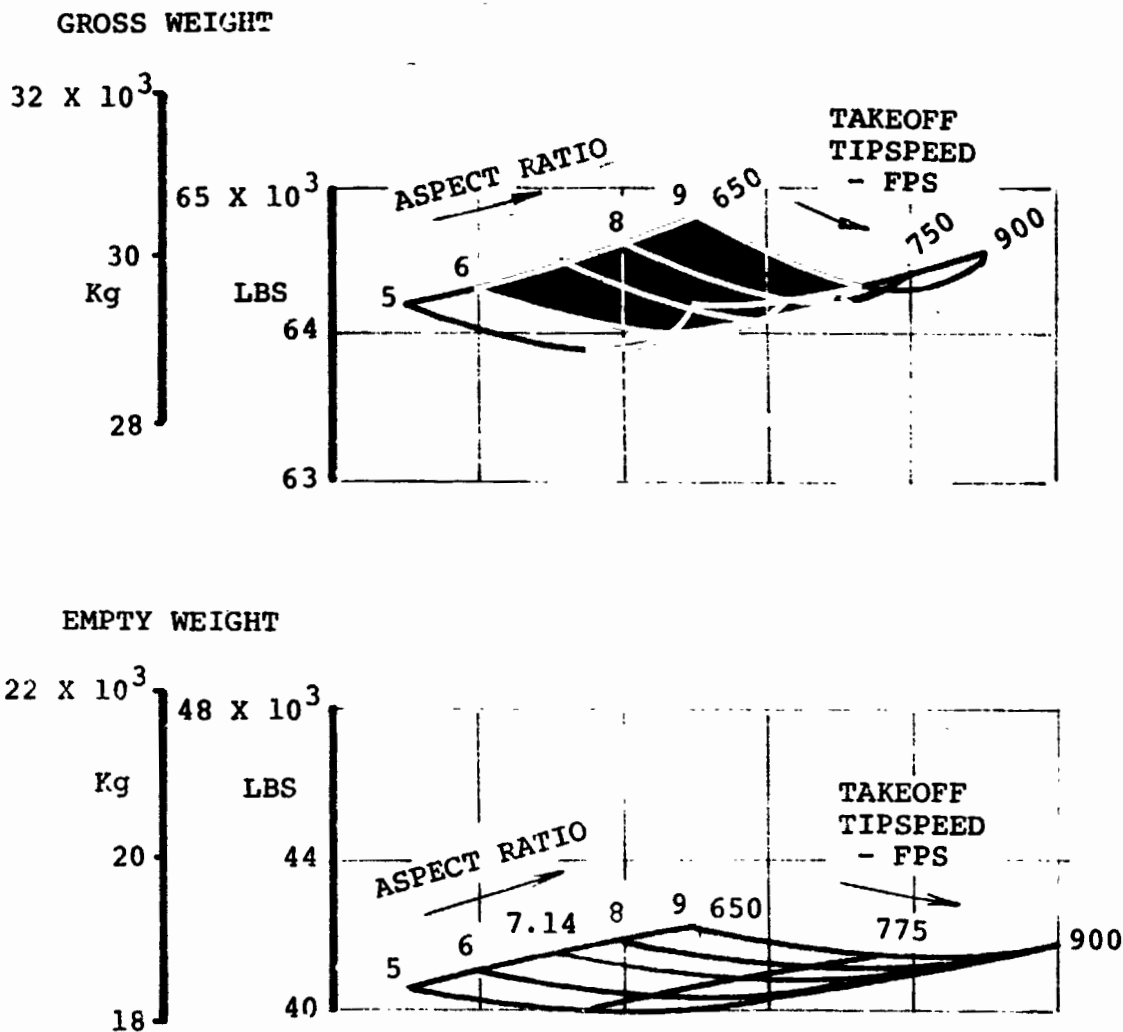
EMPTY WEIGHT

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

A21

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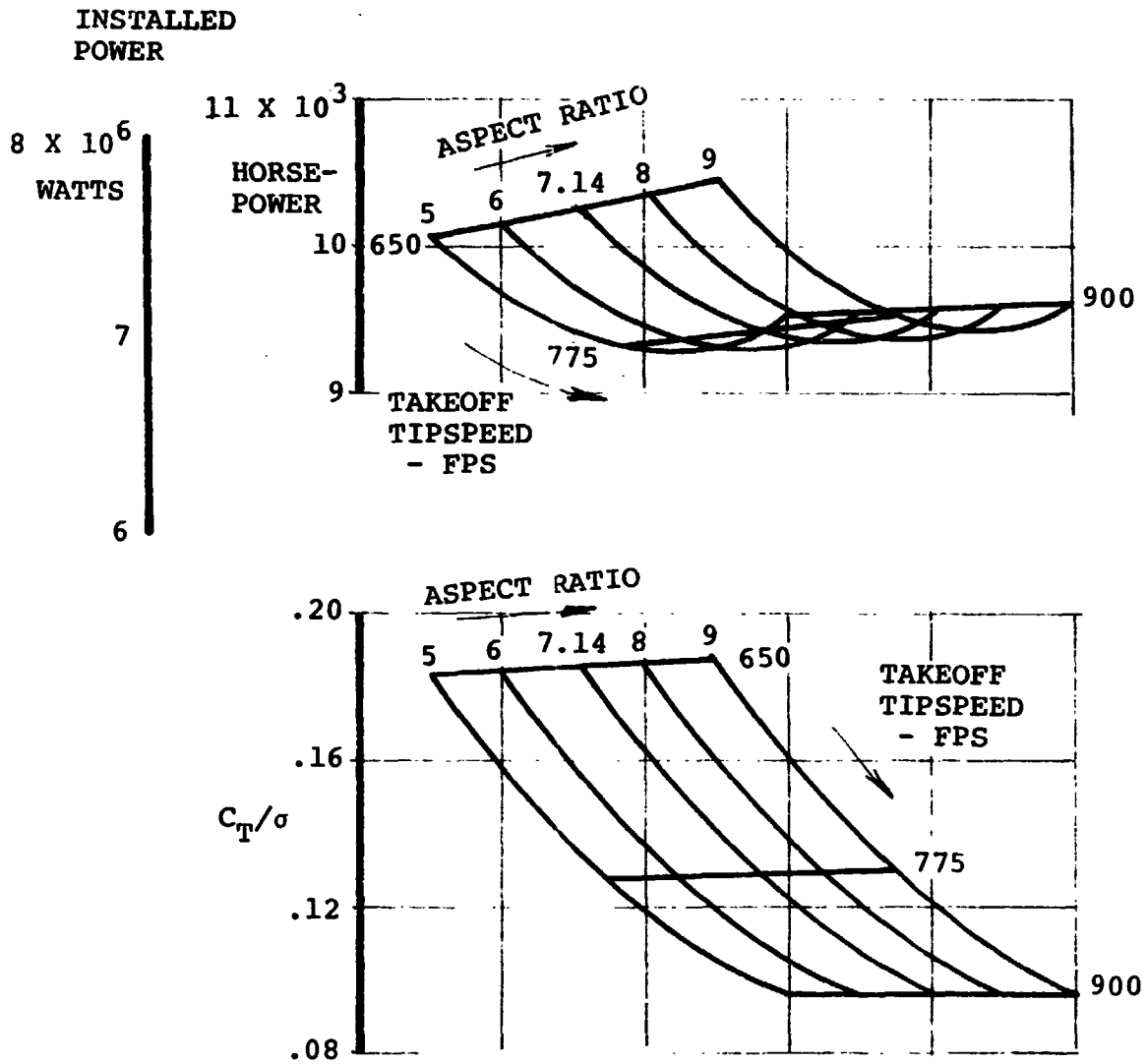
FIGURE A-18. VARIATION OF INSTALLED POWER AND ROTOR LOADING WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.105).

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

V_{NRP} AT 14,000 FT/4,267 m
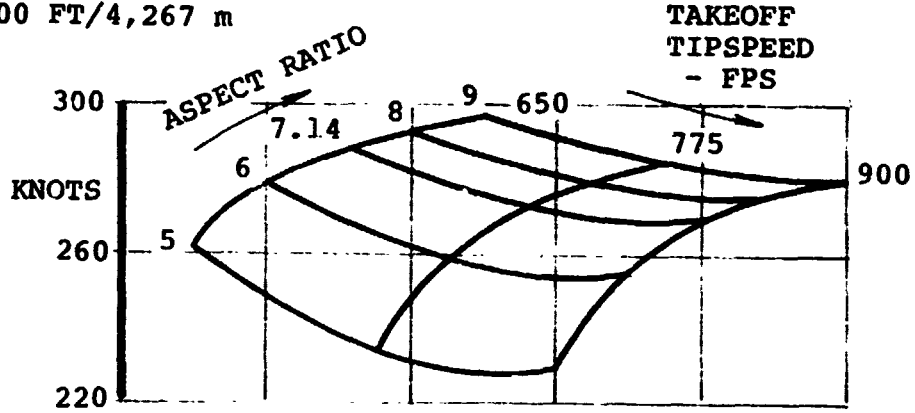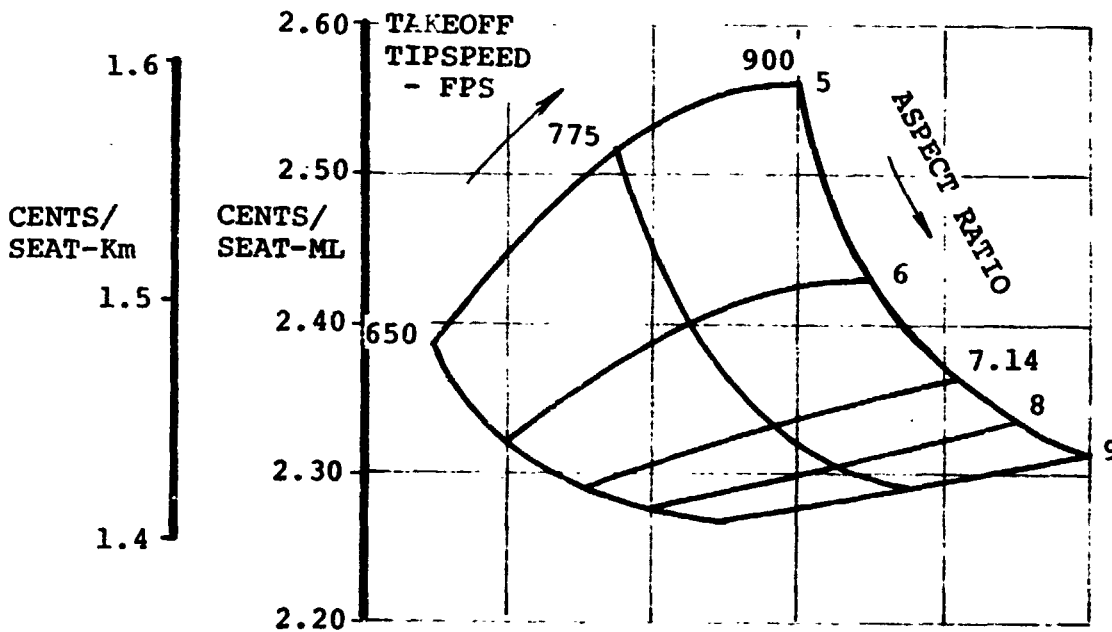
FIGURE A-19. VARIATION OF MAXIMUM CRUISE SPEED AND DIRECT OPERATING COST WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.105).

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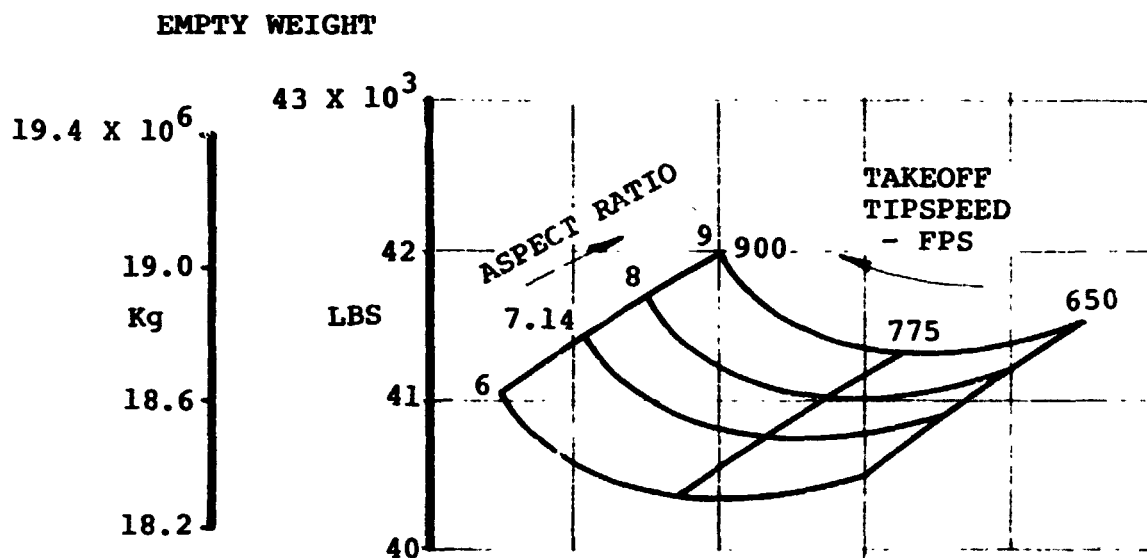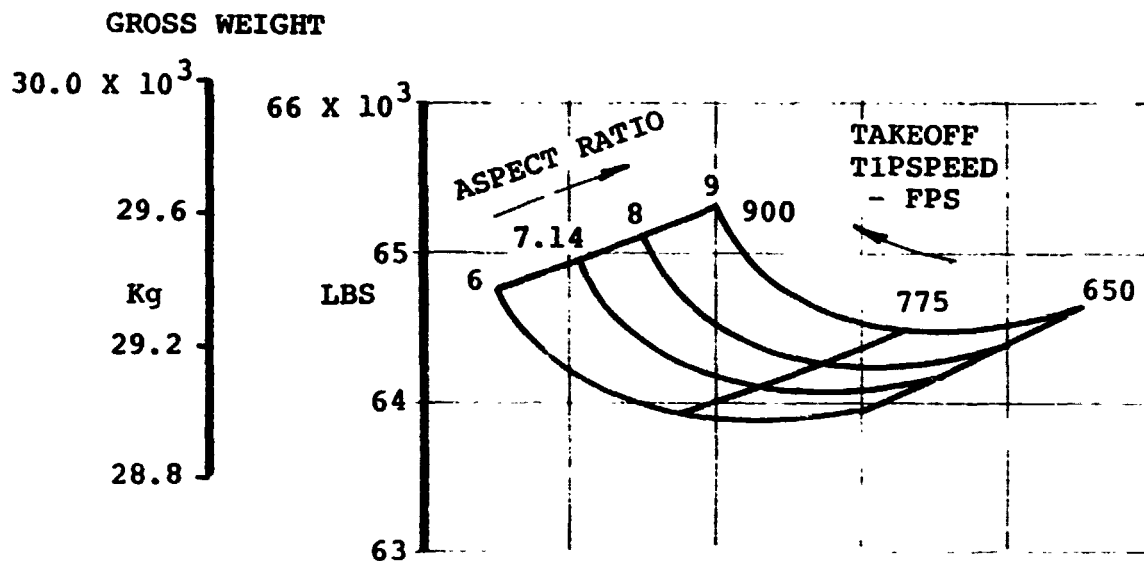
FIGURE A-20.   VARIATION OF GROSS WEIGHT AND EMPTY WEIGHT WITH ASPECT
RATIO AND TAKEOFF TIPSPEED (σ = 0.12).

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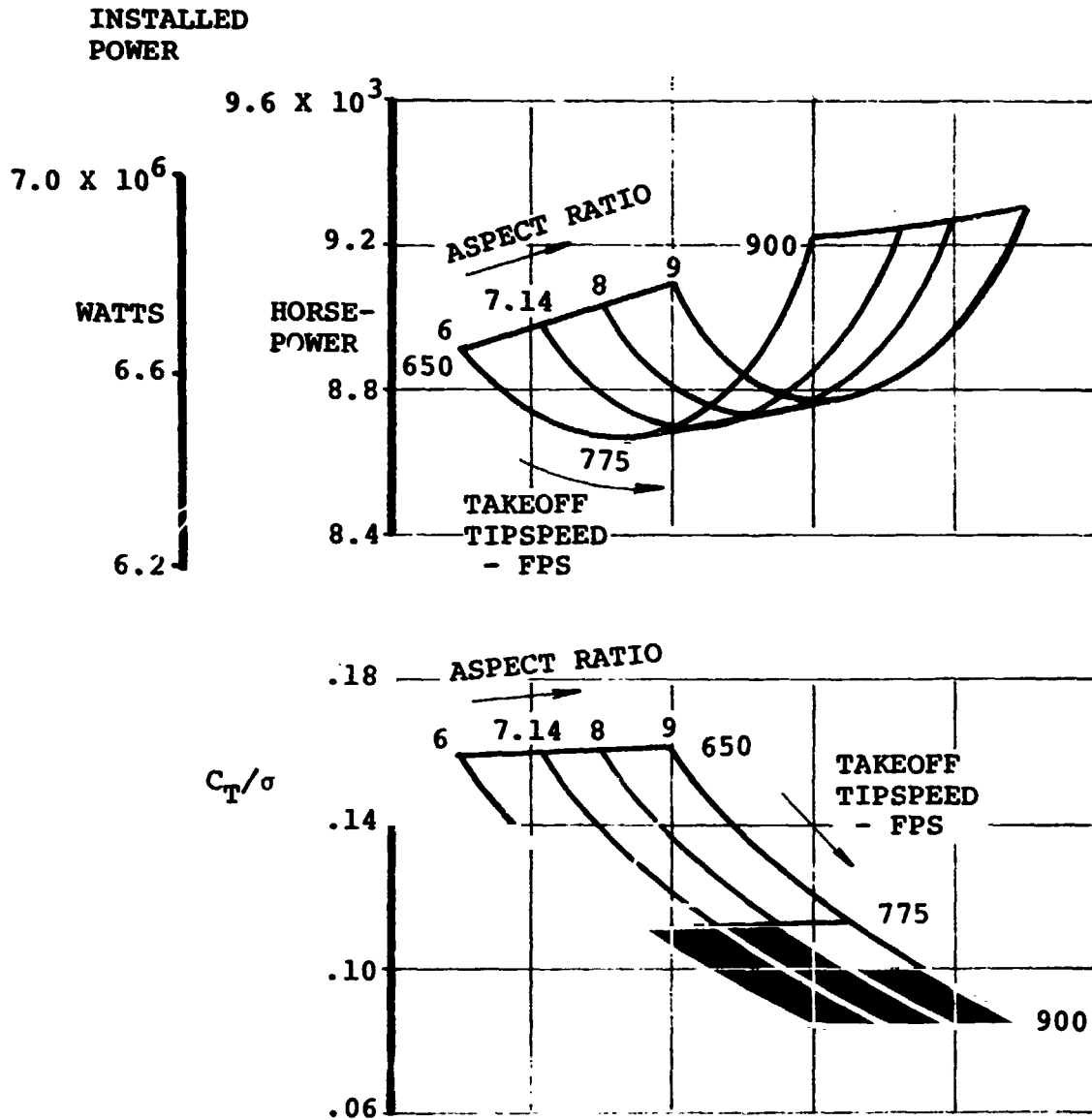
FIGURE A-21. VARIATION OF INSTALLED POWER AND ROTOR LOADING WITH ASPECT RATIO AND TAKEOFF TIPSPEED (σ = 0.12).

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

WING LOADING = 100 PSF
ROTOR DIAMETER = 44.4 FT
STATIC THRUST/WEIGHT = 0.88
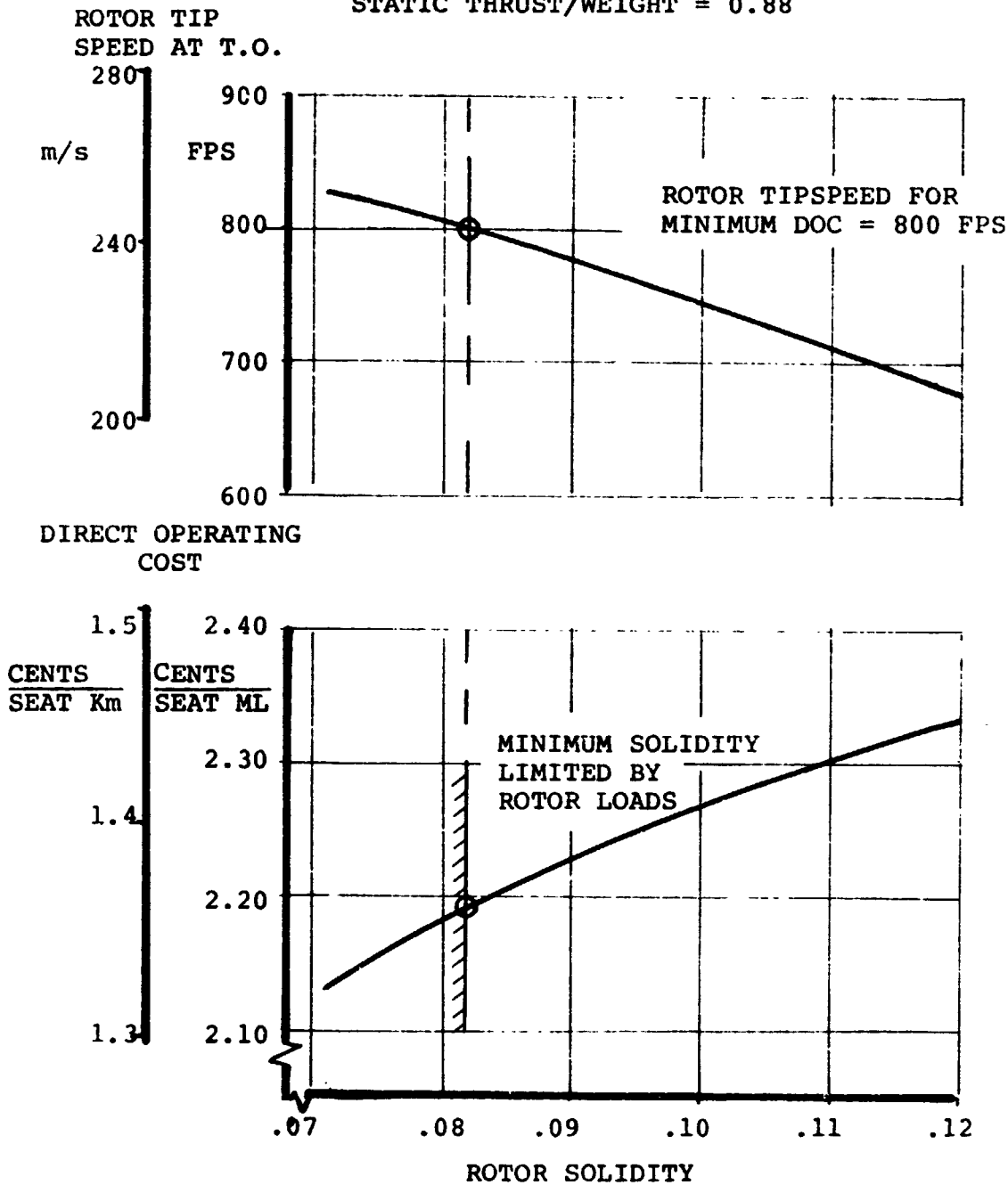
FIGURE A-23. DESIGN POINT SELECTION. VARIATION OF ROTOR
TIPSPEED AND MINIMUM DIRECT OPERATING COST
WITH ROTOR SOLIDITY.

A28

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

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

ENGINE MAINTENANCE COSTS

LABOR $\quad \$/FH = 0.65$ (AIA COSTS)

MATERIAL $\$/FH = 0.65$ (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 Cl.

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 Cl.

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	ORIGINAL PAGE IS OF POOR QUALITY		
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	68491			

FORM 26391 (2/73)

TABLE C-1. STOL TILT ROTOR WEIGHT SUMMARY. C2

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

```
              ↓
        ┌──────────────┐
        │ PROPULSION   │◁──────────────┐
        │ WEIGHTS      │    (W_f)_a     │
        └──────────────┘                │
              │ W_P                      │
        ┌──────────────┐                │
        │ STRUCTURES   │                │
        │ WEIGHTS      │                │
        └──────────────┘                │
              │ W_ST                     │
        ┌──────────────┐                │
        │ FLIGHT       │                │
        │ CONTROLS     │                │
        │ WEIGHTS      │                │
        └──────────────┘                │
              │ W_FC                     │
        ┌─────────────────────────────┐ │
```

$$(W_f)_A = W_G - (W_P + W_{ST} + W_{FC}) - W_{FE} - W_{FUL} - W_{PL}$$

$$WE = W_{ST} + W_P + W_{FC} + W_{FE}$$

$$OWE = WE + W_{FUL}$$

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
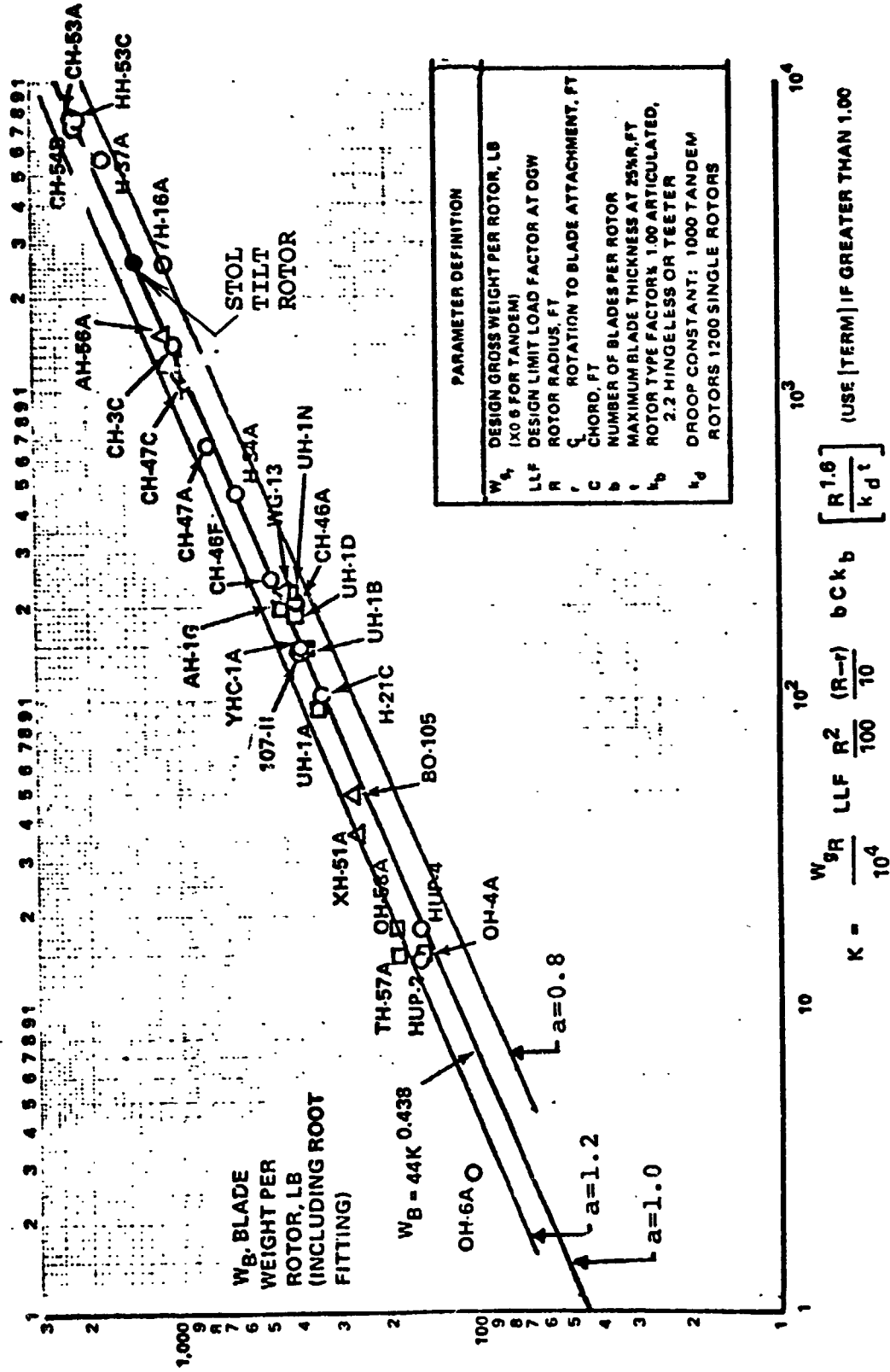
FIGURE C-2. ROTOR BLADE WEIGHT TREND.

$$K = \frac{W_{g_R}}{10^4} \; LLF \; \frac{R^2}{100} \; \frac{(R-r)}{10} \; bCk_b \; \left[\frac{R^{1.6}}{k_d^{\;t}} \right] \quad \text{(USE |TERM| IF GREATER THAN 1.00)}$$

PARAMETER DEFINITION	
W_g	DESIGN GROSS WEIGHT PER ROTOR, LB (X0 5 FOR TANDEM)
LLF	DESIGN LIMIT LOAD FACTOR AT DGW
R	ROTOR RADIUS, FT
r	ROTATION TO BLADE ATTACHMENT, FT
C	CHORD, FT
b	NUMBER OF BLADES PER ROTOR
t	MAXIMUM BLADE THICKNESS AT 25%R, FT
k_b	ROTOR TYPE FACTOR: 1.00 ARTICULATED, 2.2 HINGELESS OR TEETER
k_d	DROOP CONSTANT: 1000 TANDEM ROTORS 1200 SINGLE ROTORS

W_B, BLADE WEIGHT PER ROTOR, LB (INCLUDING ROOT FITTING)

$W_B = 44K^{0.438}$
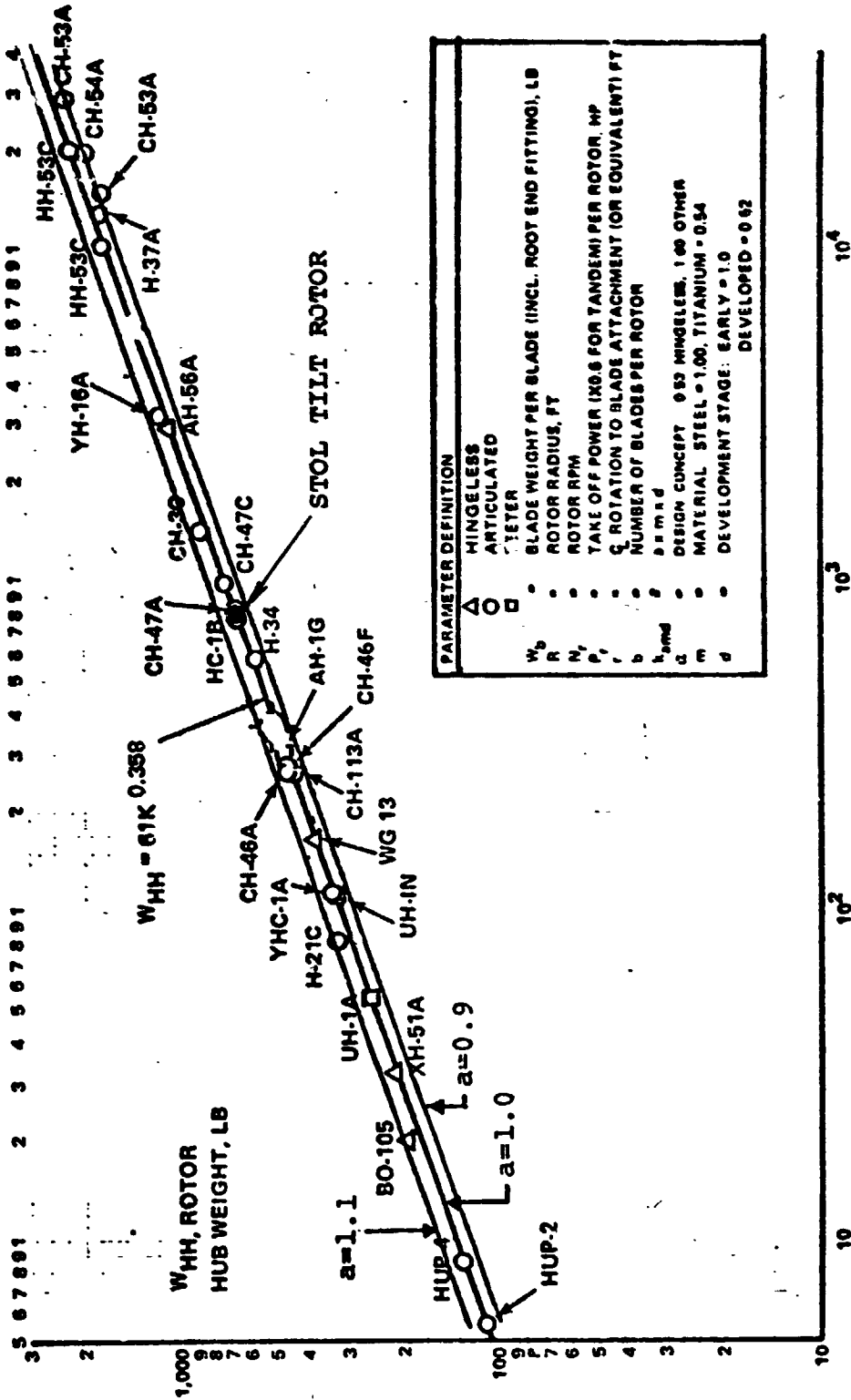
STOL TILT ROTOR

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.
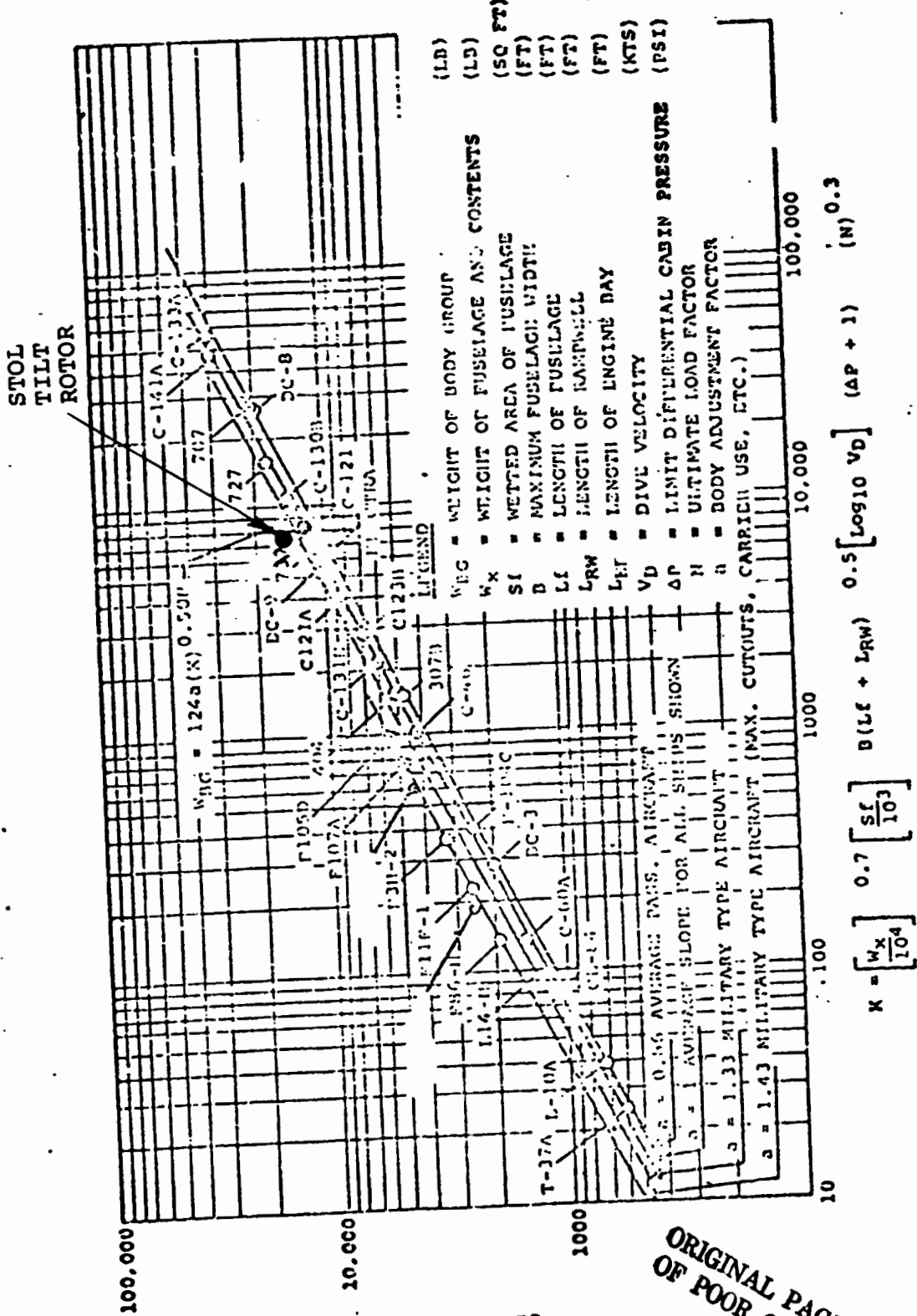
FIGURE C-4. AIRPLANE BODY GROUP WEIGHT TREND.

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

FIGURE C-6. WING WEIGHT TREND.
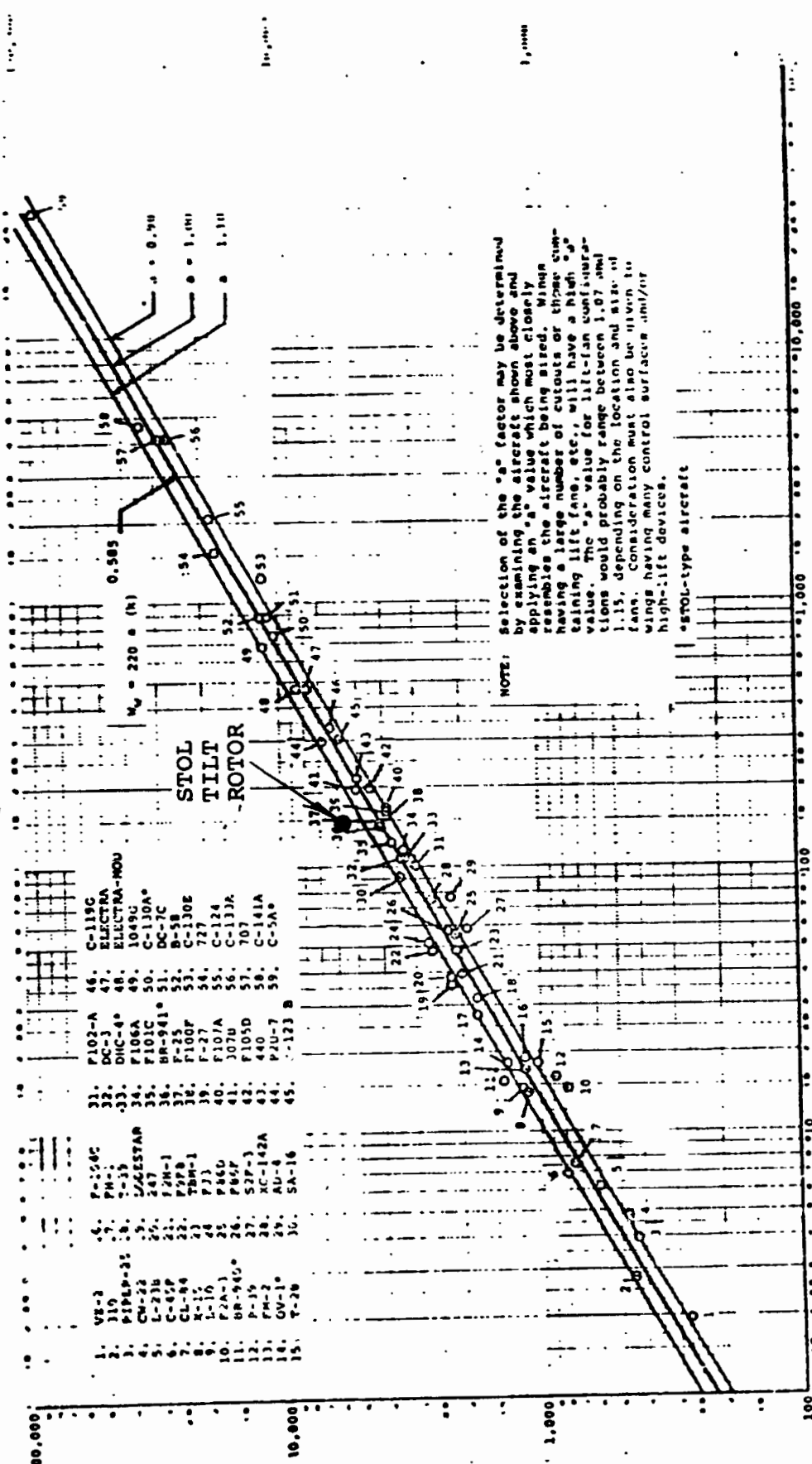
ORIGINAL PAGE IS
OF POOR QUALITY

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

$W = 380(K)0.54$

STOL TILT ROTOR

△ "T" TAIL CONFIGURATIONS

1. HELIO
2. APACHE
3. MODEL 76
4. AZTEC
5. T37A
6. AT-9
7. FH-1
8. F2H
9. T38A
10. T39A
11. S2F
12. XC-142A
13. F105
14. F101
15. F100
16. 440
17. DC-5
18. A3J
19. C-130A
20. C-130E
21. 707
22. C-133A
23. 727

WEIGHT OF VERTICAL TAIL (WVT)

(K)
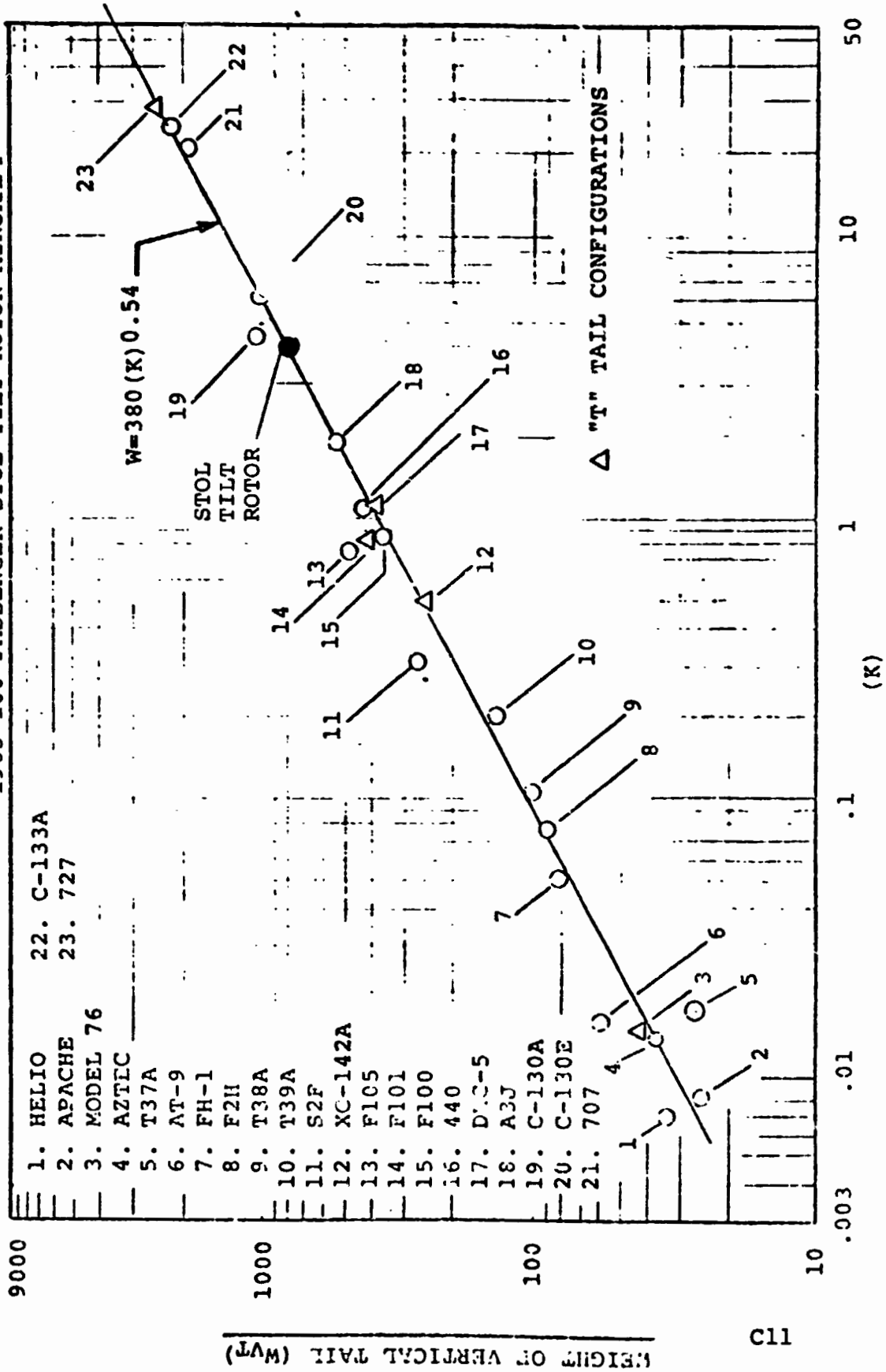
FIGURE C-7. VERTICAL TAIL WEIGHT TREND.

C11

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

$W_{HT} = 350K^{0.54}$

STOL TILT ROTOR

1. MODEL 76	21. 727
2. APACHE	22. C-130E
3. HELIO	23. C-133A
4. AZTEC	
5. AT-9	
6. T37A	
7. T38A	
8. FH-1	
9. T39A	
10. S2F	
11. F2H	
12. F3H	
13. F101	
14. F105	
15. XC-142A	
16. F100	
17. DHC-5	
18. F89D	
19. 440	
20. C-130A	

○ STANDARD TAIL WITH ELEVATOR
□ FULL FLYING TAIL
△ HORIZ. TAIL MOUNTED ON VERT. TAIL

WEIGHT OF HORIZONTAL TAIL (W_{HT})

(κ)
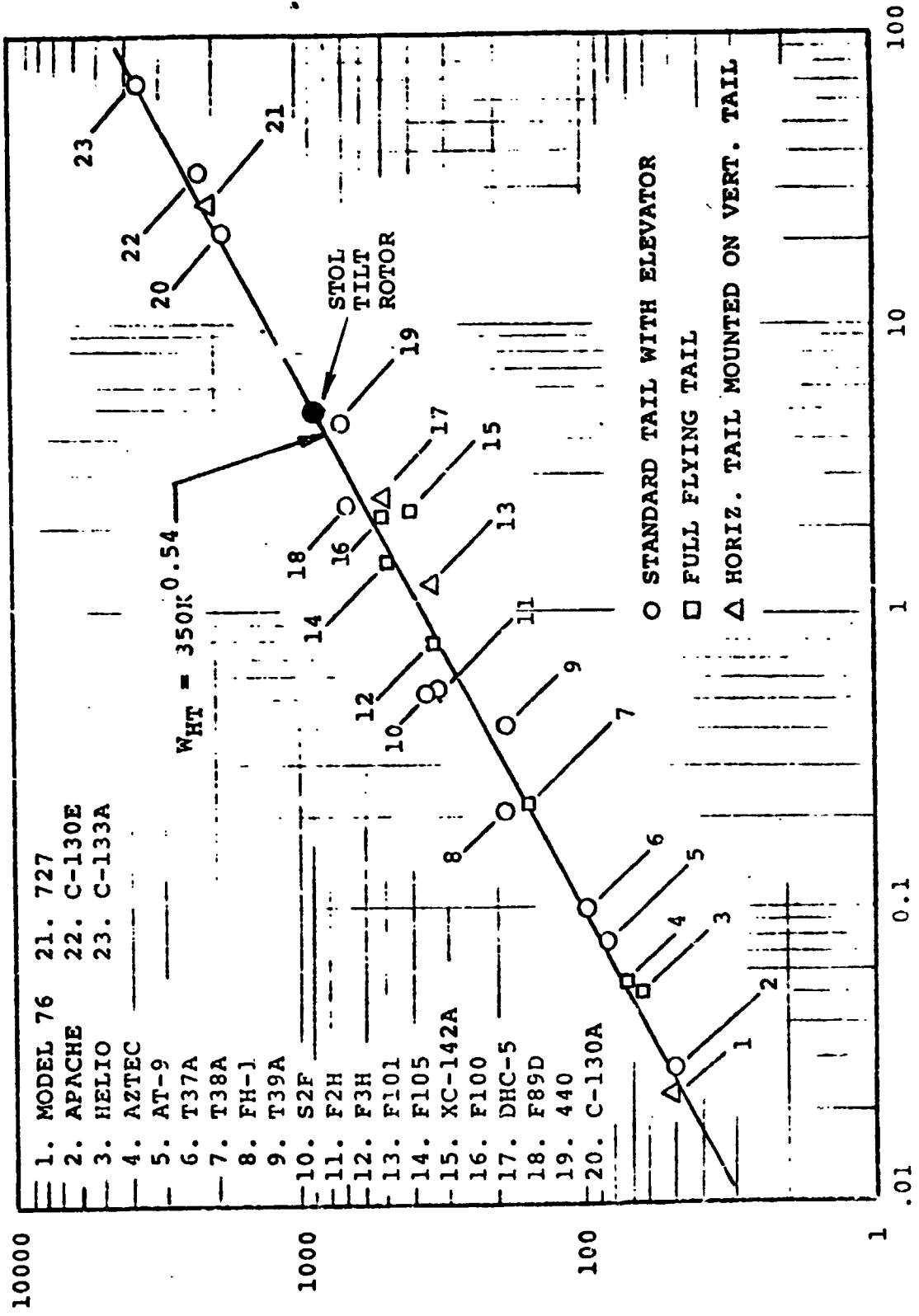
FIGURE C-8.   HORIZONTAL TAIL WEIGHT TREND

C12

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. Adjust-
ments 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 fiber-
glass 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.

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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.

C16

1985 100 PASSENGER STOL TILT ROTOR AIRCRAFT

	737-200 88 PASSENGERS		100 PASSENGERS	
	Kg	(Lbs)	Kg	(Lbs)
APU	139.7	308 ⎫		
Instruments	146.5	323 ⎬	544.2	1200*
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	257.5	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	55	23	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   = <u>0.50%</u> 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.

C20

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.

C21

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.

C22

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.

C23

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.

	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			

FORM 28391-2

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.

① CH-47A, CH-46A, 107II, XH-16A, etc.
② XC-142A, YH-54A, MODEL 76, 119, etc.
③ 707, DC-6B, DC-3, 404, etc.
④ B-26, B-29, B-47, B-52, etc.

① TANDEM HELICOPTERS
$k_y = 0.28L$

② VTOL AIRPLANES
$k_y = 0.19L$

STOL TILT ROTOR

③ TRANSPORTS
$k_y = 0.17L$

④ BOMBERS
$k_y = 0.16L$

LEGEND:
W = GROSS WEIGHT
g = MASS 32.17
L = LENGTH OF FUSELAGE

NOTES:
1. Select appropriate value for k_y from above. Include decimal only on weight input sheet.
2. Mass moments of inertia
$$\frac{W}{g} (k_y)^2$$

PITCH RADIUS OF GYRATION, k_y - FEET

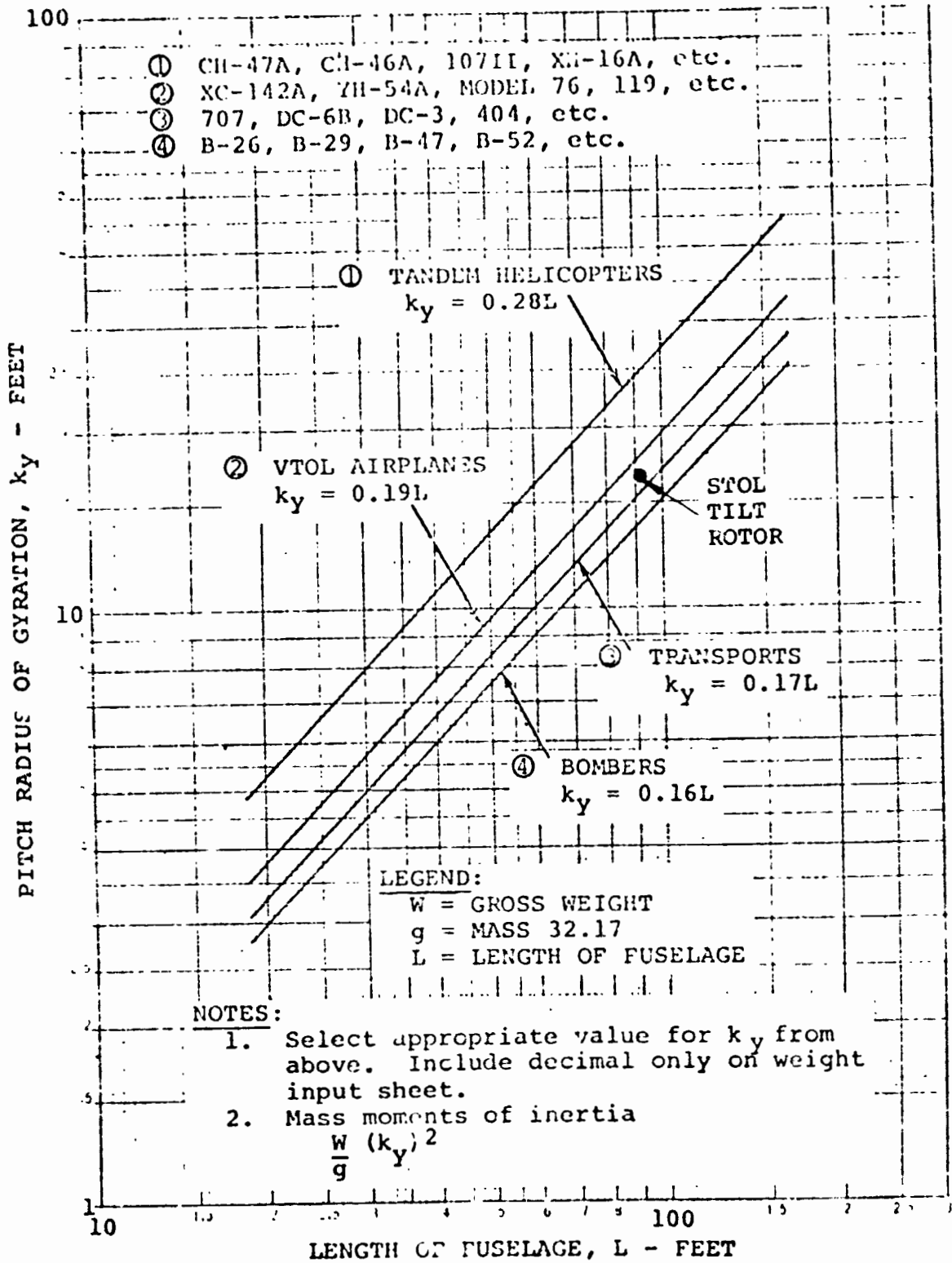LENGTH OF FUSELAGE, L - FEET
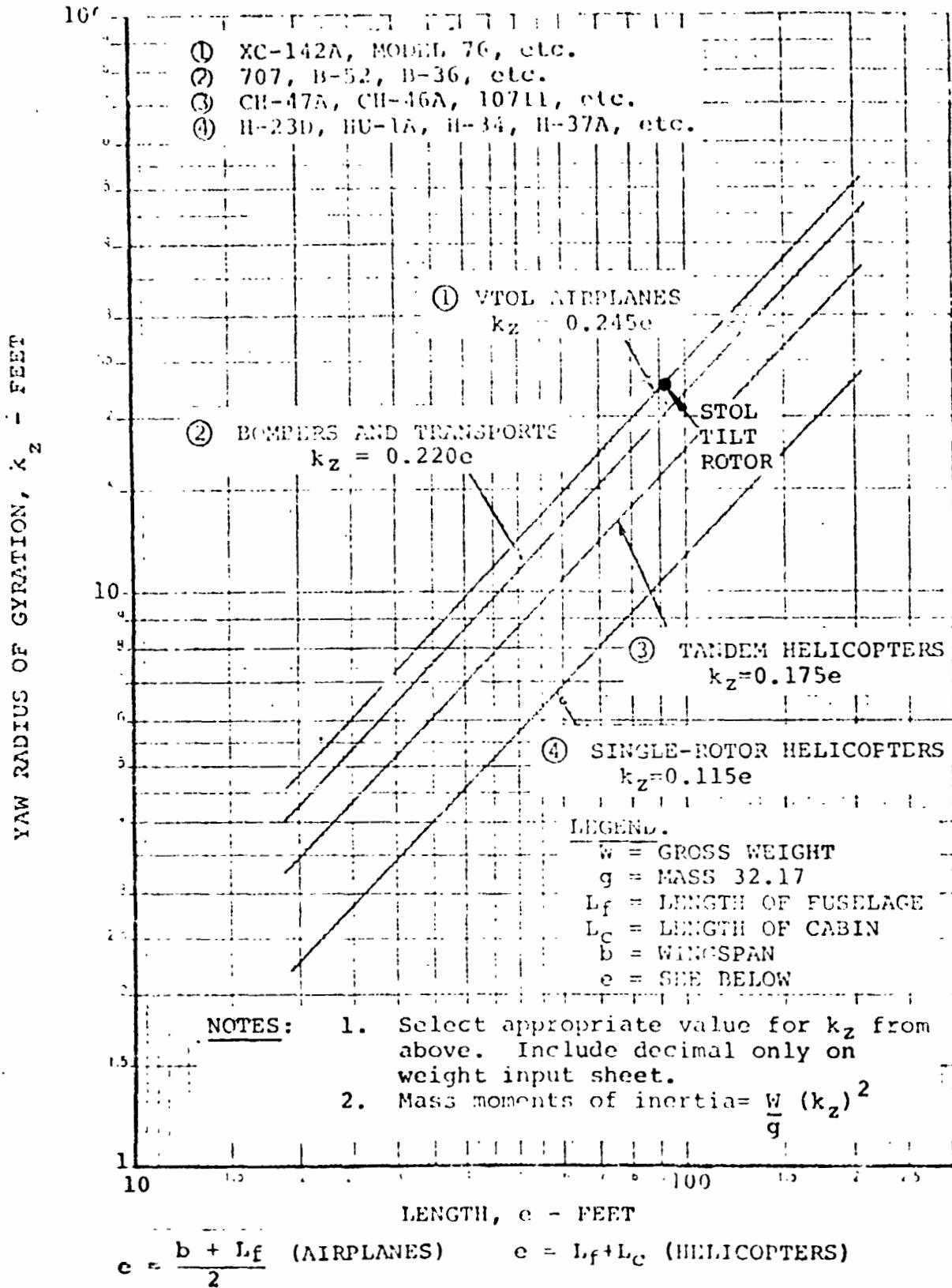
FIGURE C-9. RADIUS OF GYRATION TREND - PITCH.

C27

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

DESIGN POINT TILT ROTOR WEIGHT SUBSTANTIATION

WING (REFERENCE - WEIGHT TREND CURVE, FIGURE C6)

$$5,249 \text{ LBS}$$

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

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

$$\left(Log_{10} \, AR\right)$$

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 $\quad$ = 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

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

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

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

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

ADD: WING-POD ATTACHMENTS

.036 (POD WEIGHT)

.036 (11,720) $\quad = + \quad \underline{422}$

$\quad \quad \quad 7338$

REDUCE FOR COMPOSITES

0.302 (6916) $\quad = - \quad \underline{2089}$

$\quad \quad \quad 5249$

ROTORS

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

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

$$\left[\frac{(22.2)^{1.6}}{(1000)(.23)}\right]$$

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

$$= 2660$$

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

<u>HUB</u>   (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   = $\mathcal{C}$ 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 = $61(791)^{0.358} = 665 \text{ LBS PER ROTOR}$

TOTAL ROTOR WEIGHT

BLADES = 1392

HUB   = <u>665</u>

2057 X 2 = 4,114 LBS

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

618 LBS

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

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

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

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

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

$$= 46.89$$

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

$$= 5.61$$

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

REDUCTION FOR COMPOSITE

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

HORIZONTAL TAIL WEIGHT $\qquad = 618$

VERTICAL TAIL (REFERENCE WEIGHT TREND CURVE, FIGURE C7)

$$528 \ \text{LBS}$$

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

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

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

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

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

$$= \quad 31.78$$

$$K = \left[(31.78) + \left(\frac{29.7 \times 6.89}{2 \times 29.7}\right)\right]\left(\frac{185.0}{10^2}\right)\left(\frac{Log_{10}\ 375}{42.05 \times 1.579}\right)$$

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

$$= 3.96$$

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

REDUCTION FOR COMPOSITE

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

VERTICAL TAIL WEIGHT   = 528

BODY (REFERENCE - WEIGHT TREND CURVE, FIGURE C4)

8,497 LBS

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

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

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

WHERE:

W_B = WEIGHT OF BODY

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

GROSS WEIGHT = 68,493 LBS

LESS

C35

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

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

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

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

= 7351

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

REDUCE FOR COMPOSITE

.302 X 11,600 $\qquad = \underline{-3,503}$

$\qquad\qquad\qquad\qquad\qquad 8,097$

ADD AIR STAIRS (1) $\qquad \underline{+400}$

$\qquad\qquad\qquad\qquad\qquad \overline{8,497}$

<u>LANDING GEAR (4% DESIGN GROSS WEIGHT)</u> $\qquad$ 2,740 LBS

$\qquad$ 0.04 (68,493) = 2,740

<u>ENGINE SECTION (52% ENGINE WEIGHT)</u> $\qquad$ 637 LBS

$\qquad$ 0.52 (1755) $\qquad = \qquad$ 913

$\qquad$ REDUCE FOR COMPOSITE

$\qquad$ .302 (913) $\qquad = \qquad \underline{-276}$

$\qquad$ ENGINE SECTION WEIGHT $\qquad = \qquad$ 637 LBS

<u>ENGINES</u> $\qquad$ 1,755 LBS

<u>ENGINE INSTALLATION (31% ENGINE WEIGHT)</u> $\qquad$ 544 LBS

$\qquad$ (0.31) (1755) $\qquad = \qquad$ 544

<u>FUEL SYSTEM (4.9% FUEL)</u> $\qquad$ 168 LBS

$\qquad$ (0.049) (3425) $\qquad = \qquad$ 168

<u>DRIVE SYSTEM</u> (REFERENCE - WEIGHT TREND CURVE, FIGURE C5)

$\qquad\qquad\qquad\qquad\qquad\qquad\qquad\qquad\qquad$ 4,201 LBS

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

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

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)   = $\underline{-43}$

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

C39

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