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CONCEPTUAL DESIGN STUDY OF 1985 COMMERCIAL TILT ROTOR TRANSPORTS

Volume I - VTOL Design Summary

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Prepared by
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16. Abstract

This report presents results of a conceptual design study of 1985 commercial tilt rotor V/STOL transports for a NASA 200 n. mi. (370 km) VTOL mission. One of the main tasks of the study was to reach a conclusion regarding the largest size tilt rotor commercial transport that would be technically feasible if fabrication began in 1980.

Aircraft were synthesized in the 21-, 45-, and 100- passenger categories. Technological factors were considered and the 45-passenger point design, designated the D312, was selected. Variants of the D312 having sideline noise levels in hover of ± 5 PNdB were also studied. All three 45-passenger aircraft were analyzed for performance, weights, economics, handling qualities, noise footprints, aeroelastic stability and ride comfort.

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FOREWORD

This report is one of two volumes prepared by the Bell Helicopter Company (BHC), Fort Worth, Texas covering the VTOL portion of a conceptual design study of 1985 commercial tilt rotor V/STOL transports. The study, which will include STOL variants of the tilt rotor, is being conducted for the National Aeronautics and Space Administration, AMES Research Center, Moffett Field, California, under Contract NAS2-8259. Mr. D. R. Brown is the NASA Contracting Officer and Mr. H. K. Edenborough is the Technical Monitor for NASA on the VTOL portion of the effort. Mr. K. W. Sambell is the BHC Project Engineer for the study.

The technical contributions of Mr. G. Churchill and Mr. D. J. Guilianetti of NASA-AMES are especially noted. The assistance and advice of the following members of the BHC technical staff are gratefully acknowledged.

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Mr. E. E. Scroggs, Jr. - Weights
Dr. J. G. Yen - Aeroelasticity and Ride Comfort

The BHC tilt rotor aircraft design synthesis methods, available for use on this project, were developed principally by Mr. E. L. Brown. The engine scaling methods were developed by Mr. F. V. Engle.

The volumes prepared are as follows:

Volume I - Conceptual Design Study of 1985 Commercial Tilt Rotor Transports - VTOL Design Summary (BHC Report No. D312-099-002).

Volume II - Conceptual Design Study of 1985 Commercial Tilt Rotor Transports - VTOL Substantiating Data (BHC Report No. D312-099-003). NASA CR137602

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SYMBOLS AND ABBREVIATIONS

AIA	Aerospace Industries Association
APU	auxiliary power unit
askm	available seat kilometer
assm	available seat statute mile
bh	block hour
BHC	Bell Helicopter Company
BITE	built in test equipment
c.g.	center of gravity
cm	centimeter
C_T	rotor thrust coefficient
cu	cubic
$^{\circ}$, deg	degree
$^{\circ}$C	degrees Celsius
$^{\circ}$F	degrees Fahrenheit
DGW	design gross weight
DOC	direct operating cost
$M	dollars (millions)
FAA	Federal Aviation Authority
FAR	Federal Air Regulation
ft	feet
fpm	feet per minute
fps	feet per second
F/A	fore and aft
FS	fuselage station
g	acceleration due to gravity

SYMBOLS AND ABBREVIATIONS

GW	gross weight
HELO	helicopter
hp	horsepower
hr	hour
IGE	in ground effect
in.	inch
IRP	intermediate rated power (30-min rating)
km	kilometer
kph	kilometers per hour
kt	knot
kw	kilowatt
lb, lbf	pound force
max.	maximum
MCP	maximum continuous power
min	minute
min.	minimum
N	newton
n	normal acceleration
n. mi.	nautical mile
NASA	National Aeronautics and Space Administration
OGE	out of ground effect
PAX	passengers
pct, %	percent
PNL	perceived noise level

SYMBOLS AND ABBREVIATIONS

PNdB	perceived noise level decibels
psf	pounds per square foot
rad	radian
rpm	revolutions per minute
SCAS	stability and control augmentation system
SL	sea level
SLS	sea level, standard day
std.	standard
s. mi.	statute mile
sq	square
STOL	short takeoff and landing
$T_{1/2}$	time to one-half amplitude
T_2	time to double amplitude
V	velocity
V_{CON}	Airspeed at which transition is complete and the aircraft enters the aerodynamic flight regime.
V_D	dive speed
V_t	total airspeed
V/STOL	vertical and short takeoff and landing
VTOL	vertical takeoff and landing
yr	year
Δ	incremental
δ_e	elevator deflection
ζ	damping coefficient ratio

SYMBOLS AND ABBREVIATIONS

θ_m	mast angle (airplane: zero^{o})
σ	rotor solidity ratio
σ'	atmospheric density ratio
$\ddot{\psi}$	yaw acceleration
ω_n	undamped natural frequency

1. SUMMARY

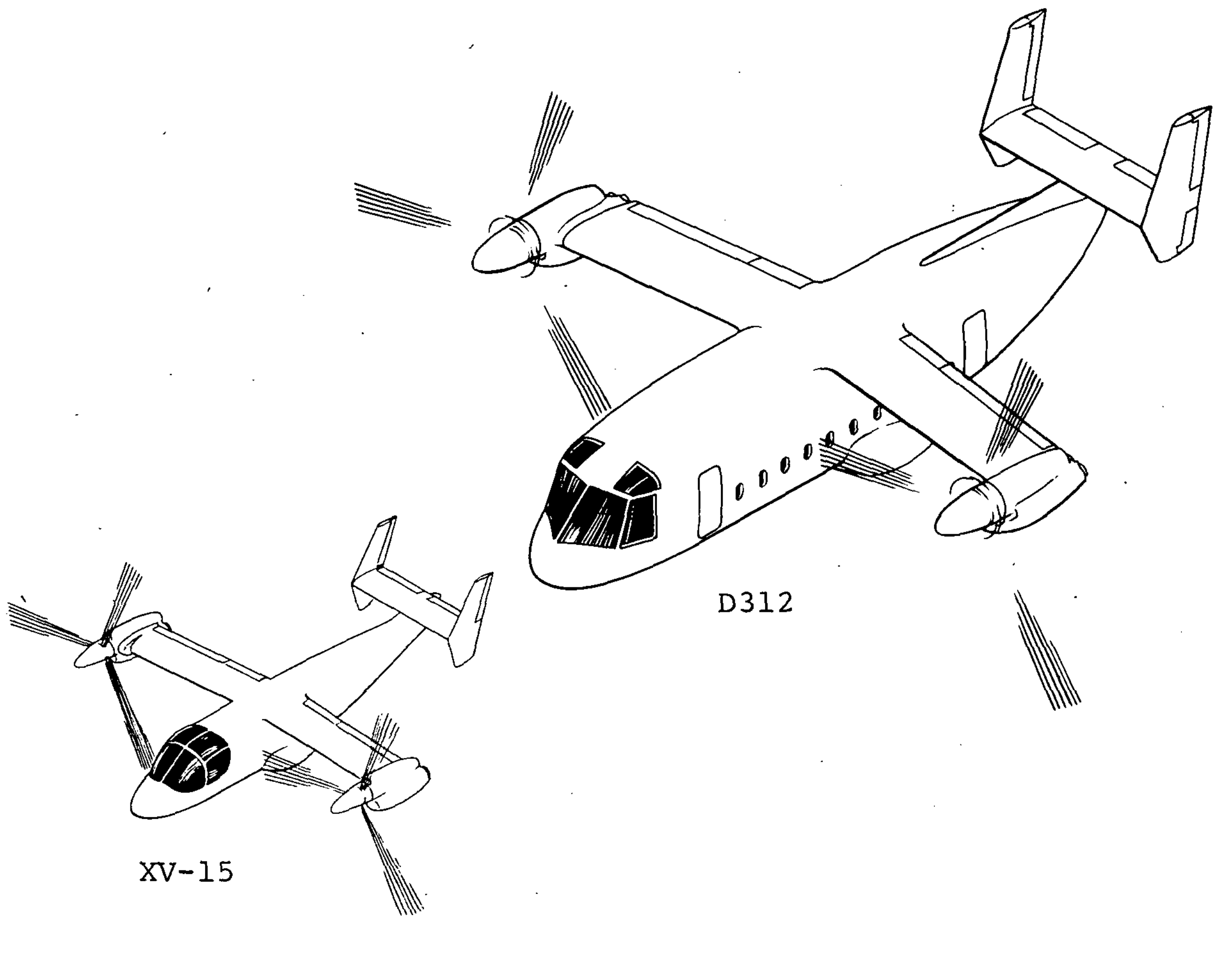
This report presents the results of a conceptual design study of 1985 commercial tilt rotor transports based on the NASA 200 n. mi. (370 km) VTOL mission. The purpose of the study is to generate transport designs to support V/STOL transportation system studies by NASA. One of the main tasks of the study was to reach a conclusion regarding the largest size tilt rotor commercial transport that would be feasible and practical if fabrication would begin in 1980.

To provide a data base for the recommendation, three size classes were investigated, each retaining the generic characteristics of the NASA-ARMY XV-15 Tilt Rotor Research Aircraft. Aircraft were synthesized in the 21-, 45-, and 100-passenger categories. Technological factors were considered and the 45-passenger point design, designated the D312, was selected. A comparison of the D312 and XV-15 is shown in Figure 1-1. A trade-off study was conducted to define versions of the aircraft having sideline noise levels in hover of -5 PNdB and +5 PNdB from the baseline. The prime design parameter varied was hover tipspeed. The values used were: 700 fps (213 m/sec) for the baseline, 550 fps (168 m/sec) for the -5 PNdB design and 850 fps (259 m/sec) for the +5 PNdB version.

All three 45-passenger aircraft were analyzed for performance, weights, economics, handling qualities, noise footprints, aeroelastic stability and ride comfort. The baseline aircraft was analyzed in greater depth for gusts, maneuvers, and the weight and cost increments to meet ride comfort criteria.

Significant results to be concluded from the study for the 45-passenger design are summarized in Table 1-1. In addition, it was concluded that important technology programs for the 1975 to 1979 period include tilt rotor flight simulations, XV-15 flight research, and advanced component technology programs. Important components to be considered for design with composite materials are the rotor and the wing. The advanced components need not be of the final size required by the transport to demonstrate that technology is in hand by 1979. These components, scaled to preserve the technological factors for the 1985 transport, should be planned for flight research on the XV-15.

Figure 1-1
COMPARISON OF XV-15 AND D312



(IDENTICAL SCALES)

TABLE 1-1
45-PASSENGER AIRCRAFT CHARACTERISTICS

NASA MISSION: 200 N.M. DESIGN RANGE
DESIGN HOVER: SEA LEVEL 90°F, ONE ENGINE INOPERATIVE

ITEM	UNIT		RELATIVE HOVER NOISE, PNdB -5		BASELINE		+5	
Noise at 500 ft. sideline, in hover	PNdB		92.2		97.2		102.2	
Direct Operating Cost, @ 200 n.m.	¢/assm	(¢/askm)	5.24	(3.26)	4.66	(2.90)	4.52	(2.81)
(per available seat statute mile) @ 400 n.m.	¢/assm	(¢/askm)	5.94	(3.69)	5.09	(3.16)	4.82	(3.00)
Area of 95 PNdB Contour, Take-off	acres	(km^2)	23.2	(.094)	49.0	(.198)	92.5	(.374)
Area of 95 PNdB Contour, Landing	acres	(km^2)	21.4	(.087)	46.5	(.188)	83.6	(.338)
Rotor Diameter	ft	(m)	45.8	(13.96)	43.6	(13.29)	42.6	(12.98)
Design Gross Weight	lbf	(N)	49388	(219687)	44848	(199493)	42797	(190369)
Weight Empty	lbf	(N)	37482	(166727)	33216	(147751)	31326	(139344)
Installed Horsepower, (Total, 30 Min. Rating, SLS)	hp	(kw)	9588	(7150)	9072	(6765)	9024	(6729)
Disc Loading	psf	(N/m^2)	15.0	(718)	15.0	(718)	15.0	(718)
Wing Loading	psf	(N/m^2)	80.0	(3830)	80.0	(3830)	80.0	(3830)
Hover Tip Speed	fps	(m/s)	550	(167.6)	700	(213.4)	850	(259.1)
Cruise Tip Speed	fps	(m/s)	450	(137.2)	600	(182.9)	600	(182.9)
Block Fuel	lbf	(N)	2220	(9875)	2015	(8963)	1912	(8505)
Block Time, Engines-On	hrs		.896		.858		.865	
Cruise Speed at 11000 feet, (3353 m) Std. Day	knots	(kph)	281	(520)	296	(548)	296	(548)

2. INTRODUCTION

The development of commercial transport aircraft has led to efficient and economic solutions for the high volume, long-range segments of the air transport network. However, the larger sizes, high powers, and frequent departures of these aircraft, when used to serve the entire network, have either strained the ability of the community to accept their presence or have caused airports to move farther from the centers they are intended to serve. The high-frequency, short-segment portion of the traffic can be served by V/STOL aircraft which will be quieter and capable of operating out of smaller airports close to the areas generating the traffic. The net result would be a higher level of community acceptance of the overall aircraft system and a lower expenditure of fuel to transport the short-haul traveler over his whole journey.

Several V/STOL concepts have emerged, and all are highly dependent on advanced technology to meet productive structural weight fractions. The various STOL concepts include the quiet turbofan, augmentor wing, externally blown flap, and internally blown wing configurations. The high disc loading direct lift and lift fan concepts also can be operated in the VTOL mode for increased operational flexibility.

V/STOL transport service is also within the operating realm of low disc loading aircraft, such as the helicopter, compound helicopter, and tilt rotor aircraft. Due to their low disc loading, these types have high efficiency for using energy to hover or climb steeply at very low speeds. Because low-speed lift is provided by rotors, fuselage attitudes can be essentially level in steep climbs and descents. These aircraft also have the capability of operating (more highly loaded) with running takeoffs to provide increased operational flexibility. The tilt rotor has the additional unique characteristics of efficient and quiet flight in the cruise mode. Modern research versions of the tilt rotor and the compound helicopter are being investigated by NASA in conjunction with the Army. The XV-15 Tilt Rotor Research Aircraft is being fabricated by Bell Helicopter Company (BHC), (Reference 2-1), and the Rotor System Research Aircraft (RSRA) is being fabricated by Sikorsky (Reference 2-2). Each of these will be important for carrying out flight research and technology programs in the 1975 to 1979 time frame to arrive at effective V/STOL aircraft.

The purpose of the study reported in this volume is to generate tilt rotor transport designs to support NASA V/STOL transportation system studies. The NASA studies will be aimed at identifying the most effective V/STOL technology programs for the overall transportation system. The STOL and lift-fan V/STOL concepts have been studied recently by Boeing, Lockheed, and McDonnell Douglas (e.g., References 2-3 through 2-6). The low disc loading concepts are being studied by Boeing-Vertol (Reference 2-7), Sikorsky (Reference 2-8), and Bell (Reference 2-9). The investigation reported in this volume covered the VTOL portion of the BHC study.

The ground rules for the BHC study are summarized in Table 2-1. They are based on the study contract Statement of Work and the Design Criteria and Study Guidelines of Reference 2-10. A key requirement was to recommend the largest size tilt rotor commercial transport that would be feasible and practical if fabrication begins in 1980. The technological viewpoint, rather than the economic, was to govern the recommendation. The separation of these factors is difficult. However, by making the assumption that passenger demand would be adequate to justify the largest aircraft considered, the technological factors can govern the selection. In this study, these factors were examined as a function of size. Three payload categories were identified, and related point designs were synthesized representing aircraft in each payload category having a baseline hover noise level in the 90 to 100 PNdB range and noise levels of -5 PNdB and +5 PNdB from the baseline. The resulting data provided technological trends as a function of payload and noise level which were considered in selecting the maximum payload category. No clear limit on size was identified, but a selection was made based on the assumption that applicable technology programs would be carried out in the 1975 to 1979 period.

The next section of this report (Section 3) presents the approach, procedures, and results for selecting the three candidate payload categories and the lift-propulsion parameters to obtain the baseline, -5 PNdB, and +5 PNdB point designs. (Extensive substantiating data for the nine point designs are included in Reference 2-11.) Section 4 includes the considerations used in selecting the final aircraft size, and Section 5 presents a description summary of the selected aircraft. Performance, handling qualities, aeroelastic stability and ride comfort, noise, and safety aspects are presented for the final selected aircraft and its -5 PNdB and +5 PNdB variants in Sections 6 through 10. The section on handling qualities is fairly detailed for a conceptual design study reflecting the emphasis placed on the subject by the Design Criteria and Study Guidelines. Additional handling characteristics for which the guidelines are not clearly applicable to the tilt rotor are included in the Appendix for comparison with other concepts. The conclusions presented in Section 11 are aimed at highlighting the significant inputs to the NASA V/STOL transportation system studies.

TABLE 2-1
STUDY CONSTRAINTS AND GUIDELINES

NASA 1985 COMMERCIAL TILT ROTOR TRANSPORT STUDY

NASA CONTRACT STATEMENT OF WORK ITEM:
"---REACH A CONCLUSION ON THE LARGEST SIZE COMMERCIAL TILT ROTOR AIRCRAFT THAT WOULD BE FEASIBLE AND PRACTICAL IF FABRICATION STARTED IN 1980."

CONSTRAINTS:
- o MAXIMUM PAYLOAD OF 100 PASSENGERS
- o HOVER NOISE, FOR BASELINE AIRCRAFT IN THE 90 TO 100 PNdB RANGE (500 FT. SIDELINE)
- o DEFINE AIRCRAFT HAVING +5 PNdB AND -5 PNdB NOISE LEVELS RELATIVE TO THE BASELINE AIRCRAFT
- o SELECT PAYLOAD SIZE FROM TECHNOLOGICAL RATHER THAN ECONOMIC FACTORS

DESIGN GUIDELINES:

- o MISSION
  - DESIGN HOVER SL 90^{O}F, ONE ENGINE OUT
  - 200 NM RANGE +50 NM ALTERNATE LEG + LOITER
- o PAYLOAD
  - 180 LB/PASSENGER, INC. BAGGAGE
  - 190 LB/CREWMAN, INC. GEAR
  - 140 LB/CABIN ATTENDANT, INC. GEAR
- o FUSELAGE
  - DOUBLE AISLE
- o EQUIPMENT
  - 2100 LB + SEATS
- o TECHNOLOGY LEVEL
  - 25% WEIGHT REDUCTION FROM PRESENT
    - \- BODY, EMPENNAGE, WING
    - \- ENGINE NACELLES
- o ENGINES
  - NASA-DEFINED CRITERIA
    - FUEL SFC = 0.42 LB/SHP.HR, TOP @ S.L. 90^{O}F.
    - SPECIFIC WEIGHT = 0.15 LB PER SHP
- o RIDE COMFORT
  - NASA-DEFINED CRITERIA
- o STABILITY & CONTROL
  - NASA-DEFINED CRITERIA
- o ECONOMICS
  - NASA-DEFINED UNIT COSTS FOR INITIAL COST
  - NASA-DEFINED AIA METHOD FOR D.O.C.

3. DESCRIPTION OF CANDIDATE POINT DESIGNS

The tilt rotor point designs investigated in this study encompassed a wide range of payload capacity with 100 passengers as the upper limit. Three payload categories were used to span this range in order to provide technical data on which to base a recommendation of the largest size feasible if the fabrication phase started in 1980. In order to establish a candidate baseline configuration in each size class, it was necessary to identify possible -5 PNdB and +5 PNdB variants. A total of nine point designs was investigated as a basis for recommending a final size.

3.1 PAYLOAD CATEGORIES INVESTIGATED

Three payload categories were selected by starting with the 100 passenger, maximum-specified capacity and successively multiplying by 0.45. This yielded, after review of seating arrangements, 100-, 45-, and 21-passenger configurations. The next step downward in the progression (not studied) is 9 passengers which is the seating capacity, with NASA specified seats, of a fuselage the size of the XV-15 NASA-Army Tilt Rotor Research Aircraft. A comparison of the resulting fuselage sizes is shown in Figure 3.1-1.

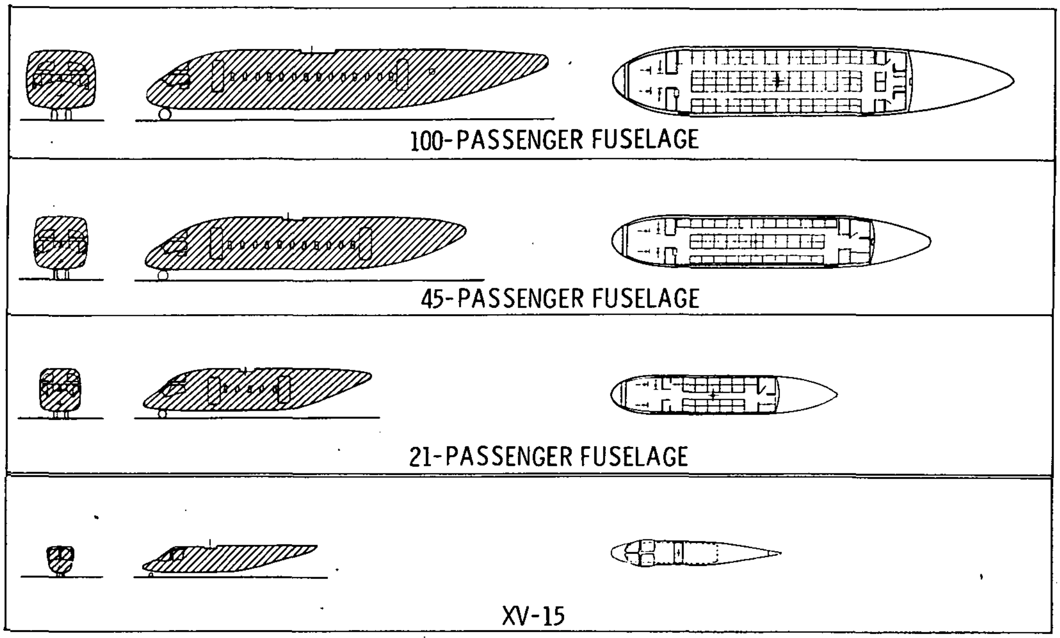
The fuselage dimensions and other structural cirteria were a portion of the input data for the Tilt Rotor Aircraft DEsign Synthesis (TRADES) method (BHC computer program OMSW02) used in this study. The fuselage size in any payload category was kept the same for the candidate baseline, -5 PNdB and +5 PNdB configurations. Only the lift propulsion system parameters were varied to arrive at solutions for the 200 n. mi. (370 km) mission in each payload category for the three noise level versions.

3.2 LIFT-PROPULSION PARAMETERS

In order to establish the baseline configuration in any payload category it was necessary that a -5 PNdB and +5 PNdB version be identifiable. The hover noise estimating method used was calibrated with test data obtained with the BHC Model 300 Tilt Rotor tested on the Wright Field whirl tower in March 1973. This method is sensitive to rotor thrust (design gross weight), tipspeed, disc loading and rotor blade loading coefficient.

3.2.1 HOVER TIPSPEEDS - Tipspeed was found to be the strongest design parameter affecting noise levels and so, early in the study, a sweep of hover tipspeeds was made in the 100-passenger category to define hover noise levels and direct

FIGURE 3.1-1
FUSELAGE SIZE COMPARISONS



operating cost as a function of tipspeed. It was found that at tipspeeds above 850 fps (259 m/sec), direct operating costs as well as noise levels increased. This tipspeed became the maximum considered and was used for the +5 PNdB designs in all payload categories. A tipspeed of 700 fps (213 m/sec) yielded 5 PNdB lower hover noise levels and was used for the baseline configurations in all sizes. To reach -5 PNdB levels from the baseline points a hover tipspeed of 550 fps (168 m/sec) was required. For the 100-passenger payload it was found that the baseline configuration would just fall within the range of 90 to 100 PNdB as required in the study ground rules for the candidate baseline aircraft. Therefore, it was assured that the smaller candidate baseline aircraft would also fall within this range. Tipspeeds could have been increased, as aircraft size was reduced, to keep noise levels constant. Instead it was elected to determine noise reductions at constant tipspeed.

3.2.2 DESIGN DISC LOADING - A disc loading of 15 psf (718 N/sq m) was used for each payload category. A higher disc loading in the 100-passenger class would have caused noise levels to exceed the study ground rules.

3.2.3 DESIGN C_T/σ - The values of design hover C_T/σ used were: 0.125, 0.124, 0.119 for tipspeeds of 550, 700, 850 respectively. These provide vertical and roll control margins without exceeding 95% of maximum rotor thrust in hover. Compressibility effects were considered in determining maximum hovering rotor thrust.

3.2.4 DESIGN WING LOADING - An investigation was made of the effect of design wing loading on direct operating costs. Based on the analysis and considerations of wing stall speeds at the low end of the conversion speed range, a wing loading of 80 psf (3830 N/sq m) was used for the 21- and 45-passenger point designs. A wing loading of 85 psf (4069 N/sq m) was used for the 100-passenger aircraft.

3.3 CHARACTERISTICS SUMMARY OF CANDIDATE BASELINE CONFIGURATIONS

As indicated in paragraph 3.2, several point designs were synthesized to arrive at basic lift-propulsion design parameters for all payload categories. Although nine designs were synthesized with the final lift-propulsion parameters, just the candidate baseline configurations are summarized here to show the variation of characteristics with size. The characteristics are presented in Table 3.3-1. The direct operating cost per available seat statute mile is seen to be a minimum and noise levels a maximum for the largest (100-passenger) aircraft, as expected. The three-views discussed in the following paragraphs were based on the dimensional data generated by the synthesis program.

TABLE 3.3-1
CANDIDATE BASELINE AIRCRAFT CHARACTERISTICS

NASA MISSION: 200 N.M. RANGE
DESIGN HOVER: SEA LEVEL 90^{o}F, ONE-ENGINE INOPERATIVE

ITEM	UNITS	PASSENGER CAPACITY		
		21	45	100
Noise at 500 ft. sideline, in hover	PNdB	95.6	97.2	99.0
Initial Cost, Including Spares	$M,1974	2.707	3.981	6.701
Direct Operating Cost (per available seat statute mile)	¢/assm	7.97	4.66	3.01
Design Gross Weight	lbs	28288	44848	81577
Weight Empty	lbs	22013	33216	57703
Useful Load	lbs	6274	11632	23873
Crew	lbs	520	520	660
Passengers	lbs	3780	8100	18000
Mission Fuel Including Reserves	lbs	1830	2857	5037
Trapped Fluids	lbs	144	155	176
Disc Loading	psf	15.0	15.0	15.0
Wing Loading	psf	80.0	80.0	85.0
Hover Tip Speed	fps	700	700	700
Cruise Tip Speed	fps	600	600	600
Rotor Diameter	ft	34.7	43.6	58.8
Blade Chord (Three Blades Per Rotor)	in	25.8	32.5	43.6
Wing Span	ft	49.1	62.0	82.8
Wing Chord	ft	7.2	9.0	11.6
Installed Horsepower, (Total, 30 Min. Rating, SLS.)	hp	5721	9072	16395
Number of Engines	-	4	4	4
Rated Power Per Engine, Required	hp	1430	2268	4099
Closest Engine Model Type	-	T700	PLT27	LTC4V-1
Block Fuel	lbs	1296	2015	3545
Block Time, Engines-On	hrs	.879	.858	.827
Cruise Speed at 11,000 feet,Std.Day	knots	287	296	311

3.4 AIRCRAFT FEATURES AND THREE-VIEWS

All tilt rotor aircraft in this study retain the same generic characteristics as the XV-15. A three-view of the baseline 21-passenger aircraft is shown in Figure 3.4-1. The 45-passenger and 100-passenger aircraft are shown in Figures 3.4-2 and 3.4-3. Significant features are the stiff-in-plane three bladed tilt rotor with a design disc loading of 15 psf. Gimbal hubs provide relief for one-per-rev flapping airloads and virtually eliminate Coriolis forces induced by flapping which reduces inplane bending moments. A moderate amount of hub restraint is used to increase control power and damping in helicopter mode without generating high blade loads.

The swept-forward wing of the XV-15 is retained to minimize mast length and save weight. The wing has a constant 23% thickness chord ratio and is swept forward 6.5°. Clearance is provided for 12° of blade flapping.

The four turboshaft engines are mounted in pairs on the rotor pylons. High transmission efficiency is possible since the normal rotor drive is via herringbone and planetary gears. The rotors are mechanically interconnected so that any engine can power either rotor.

The H-configuration empennage is sized by the same tail volume coefficients as used for the XV-15, and provides desirable flying qualities with SCAS off in the airplane cruise mode. The body is sized by the NASA Study Guidelines and Design Criteria and provides airline passenger accommodations with a double aisle. Passenger checked baggage volume, 2.5 cu ft (0.071 cu m) per passenger, is provided in the fuselage belly. These guidelines led to the non-circular fuselage cross sections shown. Additional overall system studies should investigate fuselage belly requirements to carry mail/freight and, if so, a circular cross section could be justified.

The cockpit is designed to provide adequate pilot visibility for V/STOL operations. Downward visibility of 25° is provided so that the touchdown point is in sight during final approach on a 25° glideslope. Typical fuselage attitudes are approximately 2° nosedown during the landing approach leg in the helicopter mode.

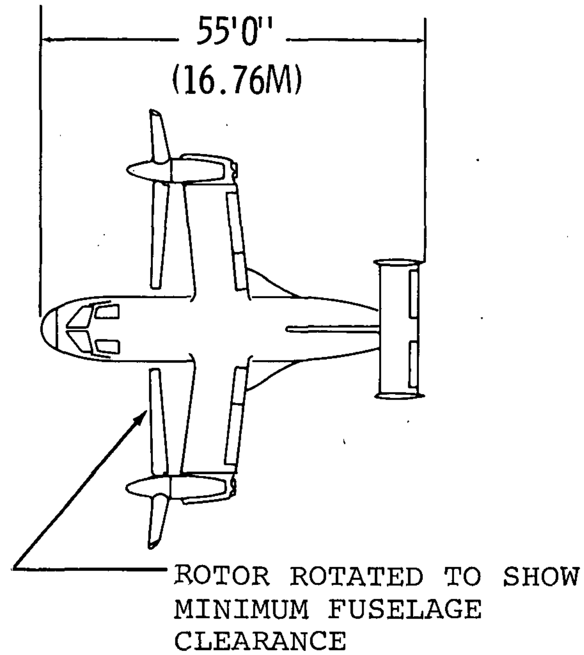
The landing gear is designed for rolling take-off and landing at speeds up to 80 knots (148 kph). Tip-over angle in any direction is a minimum of 27°.

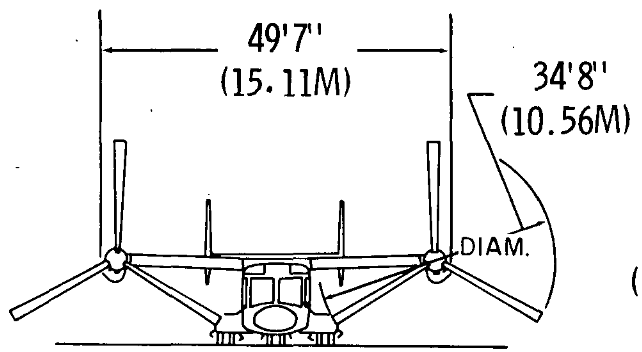
Economic analyses were conducted, based on NASA guidelines and the Aerospace Industries Association 1968 direct-operating-

FIGURE 3.4-1
TILT ROTOR TRANSPORT, 21-PASSENGER

BASELINE CONFIGURATION

DESIGN GROSS WEIGHT:	28288 lbf (125830 N)
INSTALLED POWER:	5721 shp (4266 KW)
DESIGN STAGE LENGTH:	200 n.mi.(370 KM) (WITH RESERVES)





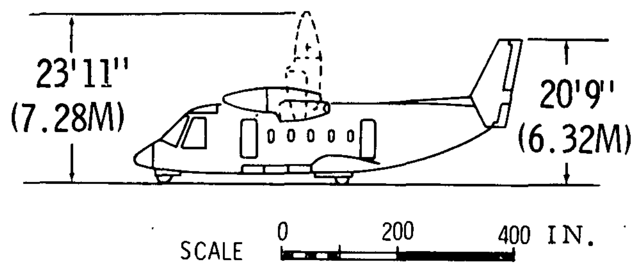


FIGURE 3.4-2
TILT ROTOR TRANSPORT, 45-PASSENGER

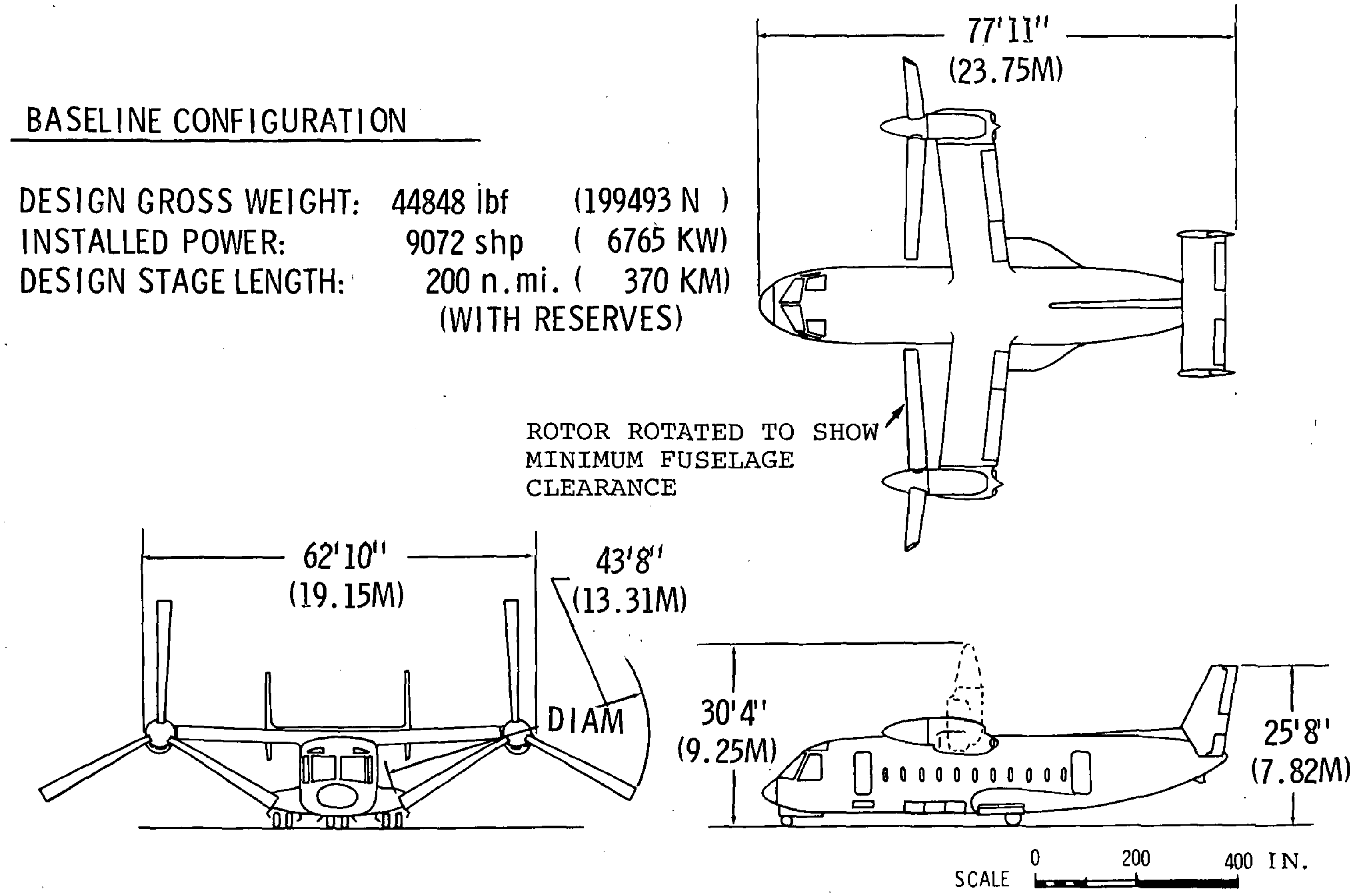
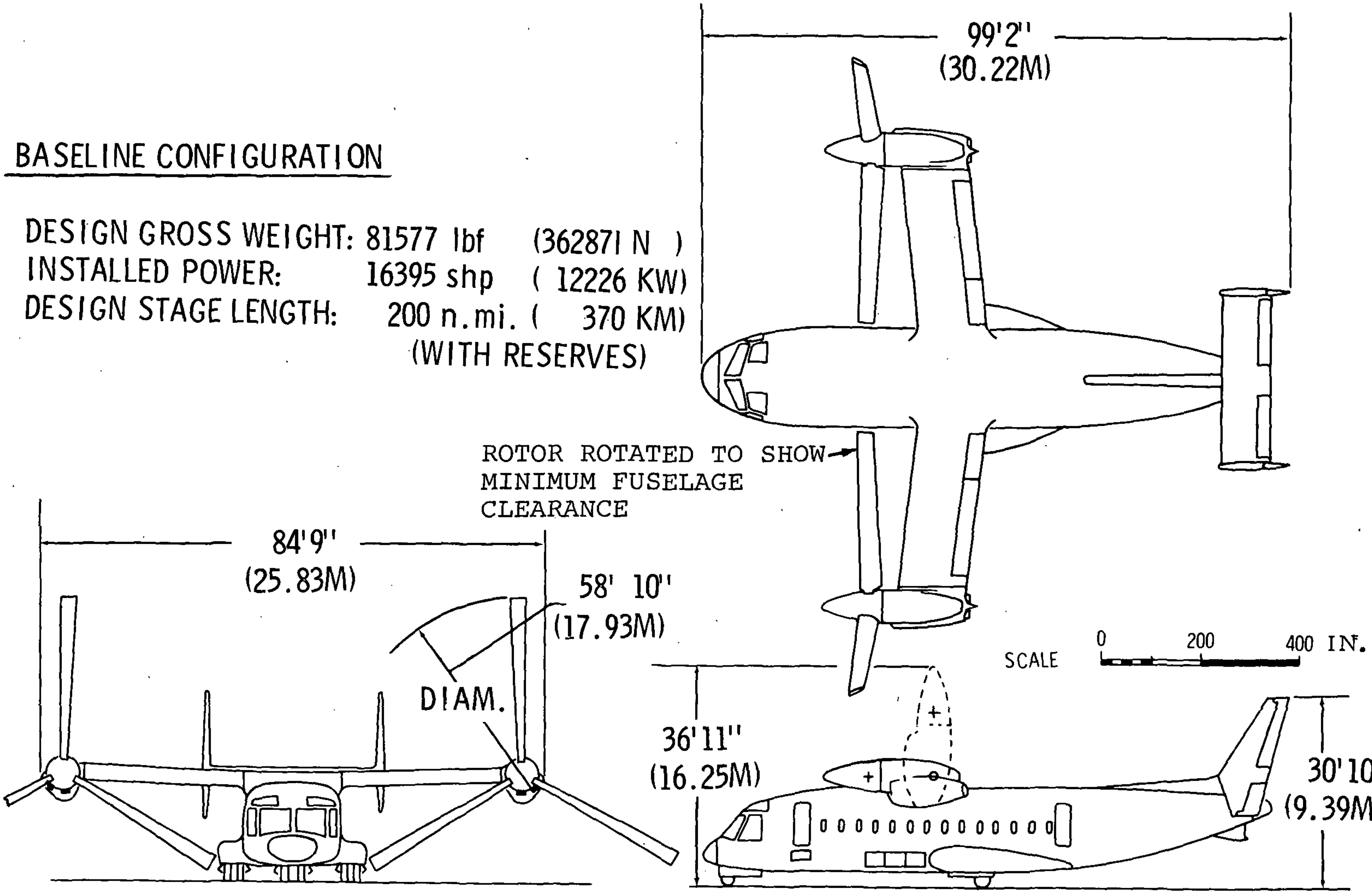


FIGURE 3.4-3
TILT ROTOR TRANSPORT, 100-PASSENGER

BASELINE CONFIGURATION

DESIGN GROSS WEIGHT:	81577 lbf	(362871 N)
INSTALLED POWER:	16395 shp	(12226 KW)
DESIGN STAGE LENGTH:	200 n.mi.	(370 KM)
	(WITH RESERVES)	



cost (DOC) estimating method (Reference 3-1). Initial costs, including spares and avionics are shown in Table 3.4-1 and DOC versus hover noise for the 200 n. mi. mission are shown in Figure 3.4-4 for all nine point designs.

3.5 COMMENTS ON ALTERNATE MISSIONS

As with helicopters and airplanes, any size tilt rotor aircraft would have applications in several roles for the military services. Typical applications, based on independent tilt rotor aircraft design studies for each service, closely match the three sizes defined in this study. A comparison of the three tilt rotor aircraft is shown in Figure 3.4-5. For the Navy, the 21-passenger size is a close match to recent HX requirements (Reference 3-2); specifically, the HMX 17- to 23-troop Marine Assault aircraft. For the Army, the 21-passenger size is approximately that of the Army-BHC Model 266 Composite Tilt Rotor Aircraft, and the 45-passenger size is approximately that of the LTTAS (Reference 3-3 and 3-4). For the Air Force, the 100-passenger size is slightly larger than the LIT transports and rescue aircraft studied in 1967 through 1969 (Reference 3-5), and slightly smaller than the AMST (Advanced Medium STOL Transport) aircraft currently being fabricated. There are more possibilities for tri-service applications of the smaller size aircraft based on existing mission definitions.

TABLE 3.4-1
INITIAL COST, INCLUDING SPARES

1974 DOLLARS (MILLIONS)

SIZE-CLASS	-5 PNdB	BASELINE	+5 PNdB
21-PASSENGER	$2.904	$2.707	$2.552
45-PASSENGER	$4.430	$3.981	$3.790
100-PASSENGER	$7.692	$6.701	$6.301

FIGURE 3.4-4
DIRECT OPERATING COST VERSUS SIZE AND NOISE

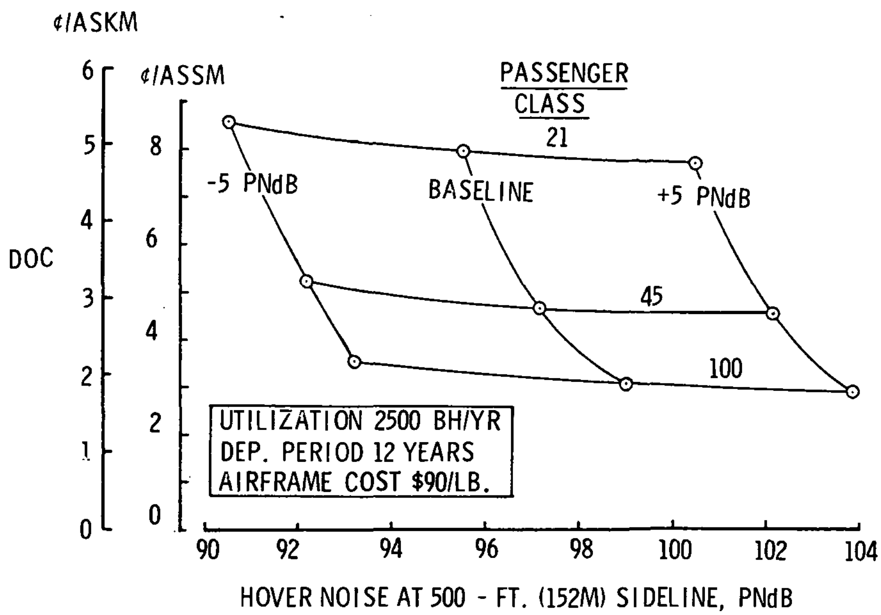
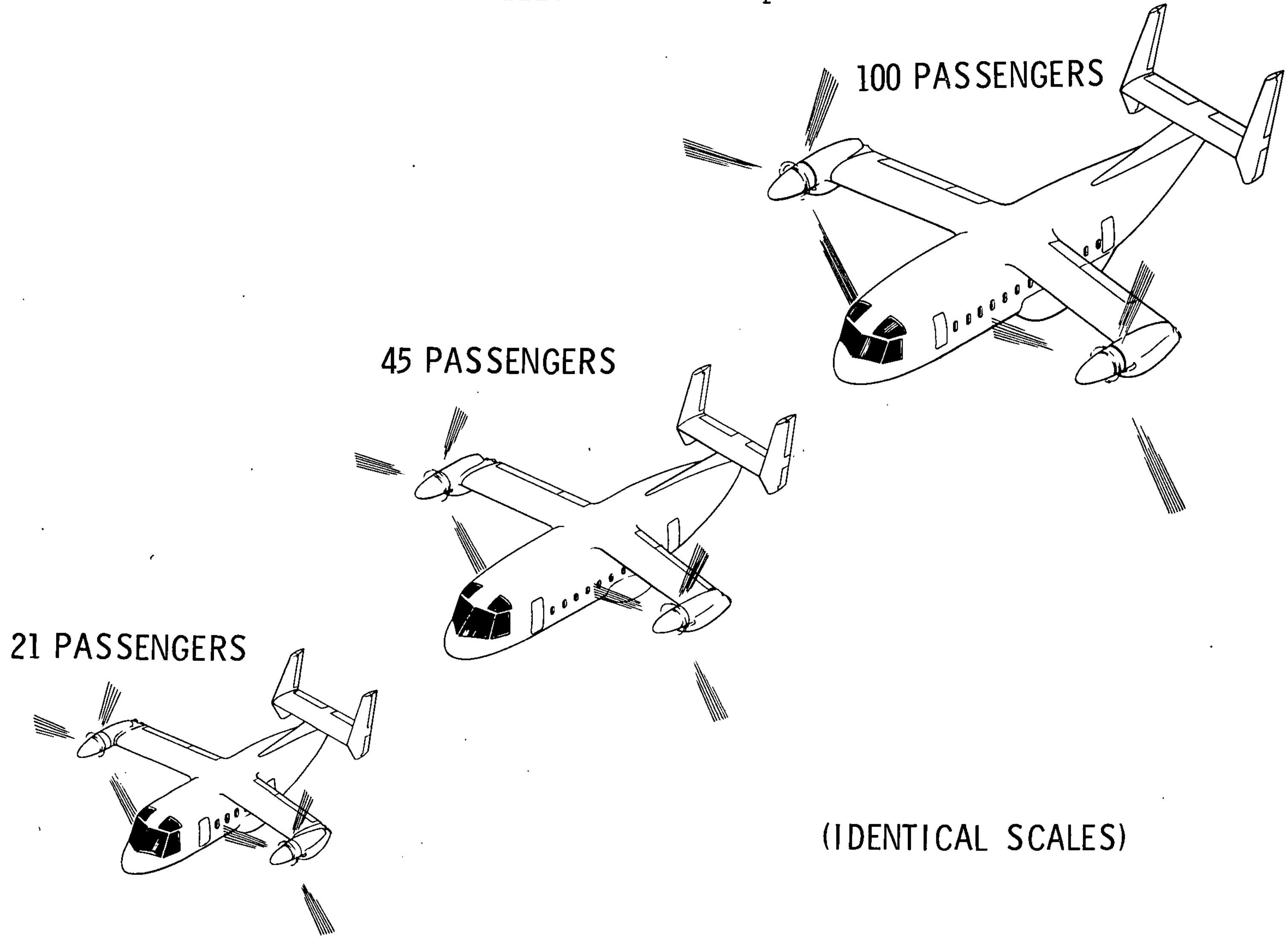


Figure 3.4-5
Tilt Rotor Transports



4. SELECTION OF AIRCRAFT SIZE

As discussed in the previous section, the largest payload aircraft studied (100 passengers) generated the lowest direct operating cost per available seat statute mile. There are at least two conditions which must be satisfied for selecting that size aircraft for the fabrication phase in 1980 in view of its potential operational economy. First, that sufficient passenger-trip demand can be projected to exist during the operational time period (e.g., in the year 1990, assuming a 1985 I.O.C.) to justify the fleet size on which the estimated costs are based; and second, that efforts can be completed to ensure that the technology is in hand by the fabrication phase start date (1980) so that predicted aircraft characteristics (size, economy, environmental compatibility, performance, etc.) can be achieved.

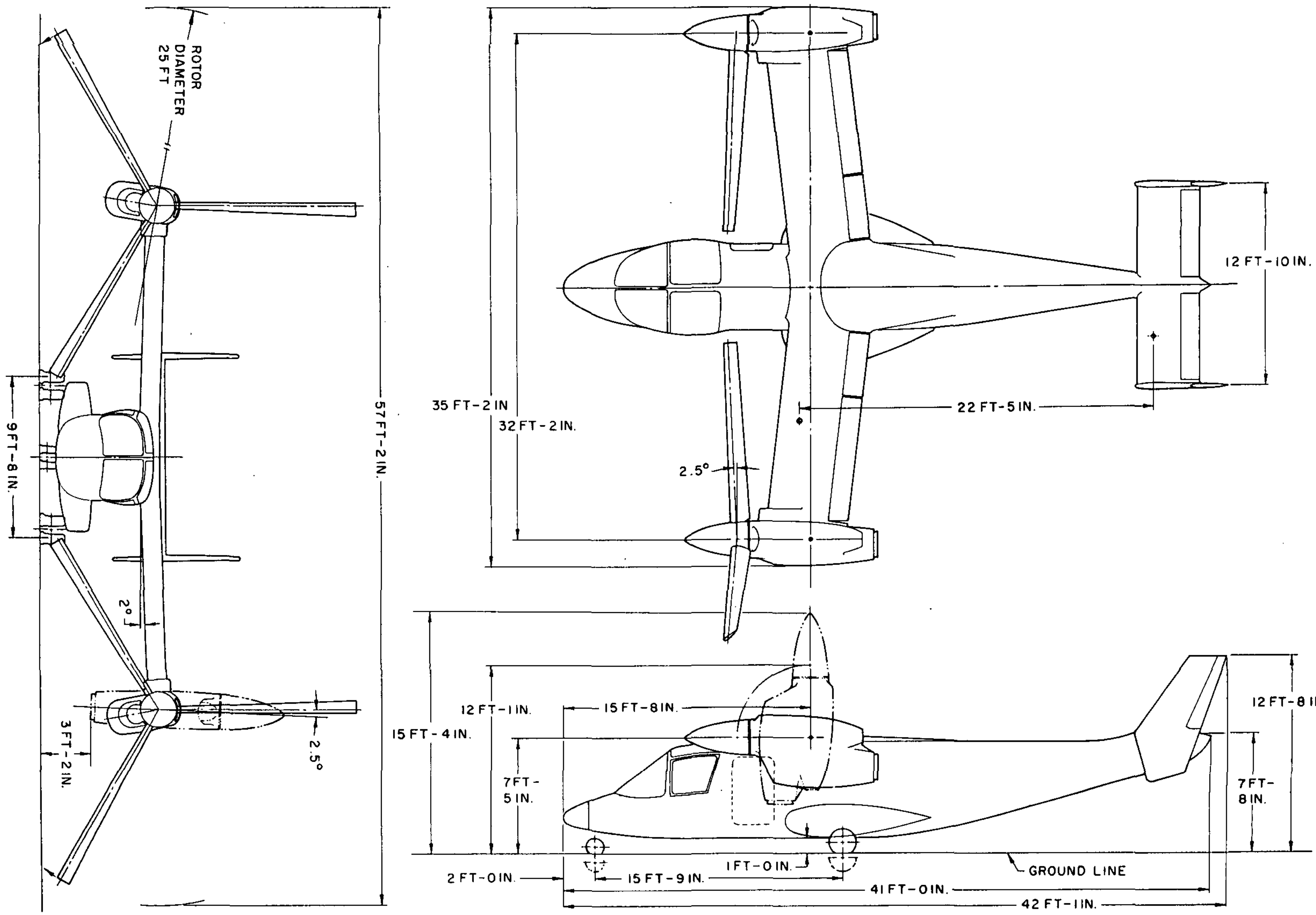
4.1 ECONOMIC FACTORS

The projection of demand for scheduled V/STOL service by 1980 was not within the scope of the present study. It is clear, however, that for a given demand and fleet size, a smaller aircraft will experience higher load factors and, therefore, lower direct operating costs. If demand is larger than allowable by the maximum practical load factors for a given size aircraft, then fleet size can be increased which results in reduced unit costs and, thereby, lower direct operating costs. Some point of increased demand, however, would favor a fleet of larger aircraft. In such a healthy situation, the technical, ecological, and economic data bases would have been established with the smaller aircraft to justify the larger one. Scheduled STOL operations generating the demand data base for V/STOL service in the 80- to 200-n. mi. (148- to 370-km) stage length range have only recently begun, and a realistic growth rate for this service needs to be determined. However, if the assumption is made that sufficient demand will be available by 1990 for the largest size aircraft considered, then the second condition, that technology is in hand by 1980, is the controlling one. This aspect is within the scope of this study.

4.2 TECHNOLOGICAL FACTORS

While helicopter components in this country have undergone development for gross weight classes over 50,000 lbf (222 410 N), only one tilt rotor aircraft, the XV-15 (Figure 4.2-1),at a design gross weight of 13,000 lbf (57 827 N),is currently undergoing development with modern technology. Helicopter and fixed wing technologies provide a sound foundation for tilt rotor aircraft, but there are characteristics unique to the

FIGURE 4.2-1. NASA-ARMY XV-15 TILT ROTOR RESEARCH AIRCRAFT



tilt rotor which depend on possible independent trends. These trends will not be clarified until new components for larger, operationally oriented tilt rotor aircraft are designed and verified through full-scale, or carefully scaled, component tests. Between 1975 and 1980, there could be time for an operational tilt rotor aircraft, slightly larger than the XV-15, to be designed, fabricated, and enter its flight test program. However, the technology levels would have to be those in hand for production tilt rotor aircraft design by, say, the end of 1975. This date precedes the first flight of the XV-15 scheduled for 1976; therefore, technology changes would be modest. Technology programs for a subsequent generation of aircraft would just be getting underway - those which would lay a sound basis for starting the fabrication phase for an advanced aircraft production design in 1980. The point designs synthesized in this study were based on technology levels which include those that could result from technology programs existing during the 1975 to 1979 time period.

Several areas could be explored with the XV-15 tilt rotor research aircraft, such as environmental compatibility, advanced control systems, and advanced technology components. Other areas, explored in helicopter and airplane programs, will be applicable to tilt rotor aircraft, such as advanced composite structures for fuselage and empennage assemblies. As a result of these programs in the 1975 to 1979 period, reductions in structural weight fractions for the fuselage, wing, tail surfaces and flight controls can be expected. A factor of 25% was used for these weight groups in this study (per NASA guidelines) for aircraft beginning fabrication in 1980.

The technology level of the fuselage and subsystems of the tilt rotor will be typical of any advanced aircraft of that time period. Risk levels for these assemblies will be lower than for V/STOL concepts which require control-propulsion elements buried in the fuselage.

The lift-propulsion assemblies of the tilt rotor are unique, however, and a data base for projections consists of one actual aircraft (of modern design), the XV-15, and design data for the Bell Model 266 composite tilt rotor aircraft (Figure 4.2-2) designed under Army contract (Reference 4-1). The lift-propulsion system consists of the rotors, engine-drive system, and the wing with its unique loading and stiffness requirements. The technology levels for each of these components are assessed, as they affect size selection, in the following paragraphs.

4.2.1 ROTOR SYSTEM - The prediction of rotor group weights for this study is based on a family of statistical weight equations which correlate well with actual helicopter rotor weights over a wide range of diameters, design gross weights, blade chords, number of blades, load factors and hub types. The resulting correlation chart is shown in Figure 4.2-3. One equation in this family is calibrated to the weight of the XV-15 rotor system, using its design parameters. The same equation was then applied to predict the rotor weights for the tilt rotor parameters of this study.

While the calibration constant used to represent the tilt rotor configuration is based on an existing technology rotor system, it cannot be assumed that all weights predicted by the equation are of current technology level. This is because the helicopter rotor data base covers a time span in which different size rotors came into being. The equations reflect a historical trend of improvements in technology as size is varied from small to large. Consequently, variations (with size) of ratios such as modulus/density and strength/density are built-in. "Current technology" is representative only of the latest, largest actual reference point in the data base. For tilt rotors, this is the XV-15.

The predicted rotor weight fractions for the candidate baseline, -5 PNdB, and +5 PNdB point designs in each of three payload classes are plotted in Figure 4.2-4. The technology trends versus size built into the prediction method make these target weights. One measure of technology level required to meet these target weights can be defined as the strength/density ratio of any given point design relative to the ratio used for the XV-15. Another measure of technology level is the effective modulus/density ratio which governs the ability to control frequency placement of the rotor structural modes to permit operation at helicopter and reduced airplane rpms. Many other measures exist to make up a total definition of technology level (airfoil selection, environmental protective measures, etc.), but these will be illustrative of aircraft size selection factors in this study.

<u>Effective Strength/Density Ratio</u> - Rotors having similar spanwise distributions of airfoil thickness, blade twist, mass, and stiffness, will have aerodynamic stresses which vary generally as tipspeed squared divided by solidity squared. Based on this loading, the required effective strength/density ratio of the rotor system to meet target rotor weight fractions is proportional to the rotor radius divided by the rotor blade loading coefficient (C_T/σ), solidity, and the target rotor group weight fraction. This relationship indicates that if technology improvements are not made as size increases, the rotor group weight fraction for geometrically similar rotors will increase as the rotor radius (---the square-cube law in action). Statistically, this has not happened.

FIGURE 4.2-2. MODEL 266 COMPOSITE AIRCRAFT

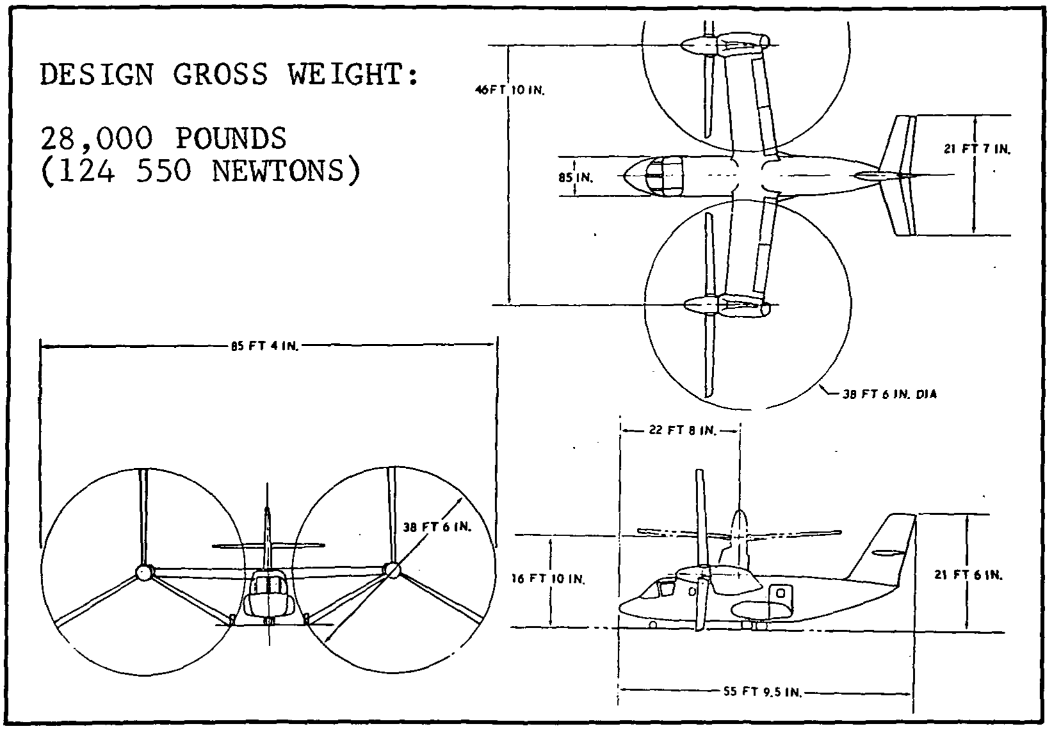
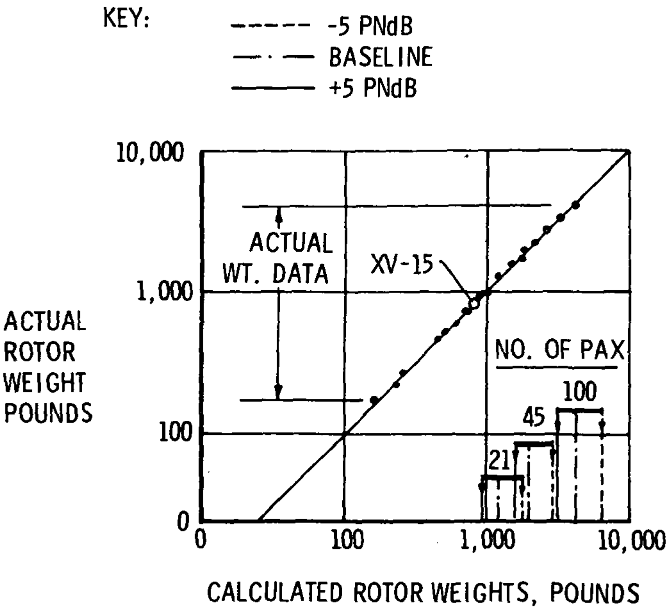


FIGURE 4.2-3. ROTOR WEIGHT CORRELATION



Improvements in effective strength/density ratio have been made, either by material or rotor structural configuration changes. Where such improvements may have been offset by additional operational requirements, disc loadings have been increased to minimize the rotor weight fraction.

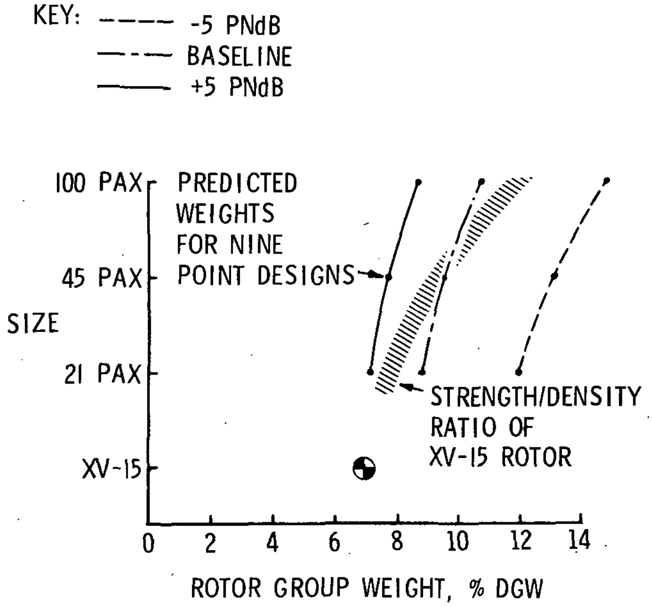
The reference value for rotor strength/density ratio of the XV-15 is taken at an index of 1.0. Since the design C_T/σ values for the nine points are essentially equal to the value used for the XV-15, the strength/density indexes vary as radius divided by the solidity and the rotor group weight fractions. For reference, a contour of constant strength/density index of 1.0 is superimposed on Figure 4.2-4. This indicates that the rotor for the candidate baseline 21-passenger point design is possible within the rotor technology (steel blades, titanium hub) of the XV-15. (This index for the Model 266 composite tilt rotor aircraft, which also was designed with steel blades and titanium hub, is .995.)

The rotor for the candidate baseline 45-passenger point design can be met with essentially the same strength/density ratio, and the candidate baseline 100-passenger point design requires an increase in strength without violating stringent component service life goals for commercial operation. Of the point designs studied, the rotor for the 45-passenger aircraft is the largest which matches the levels of strength/density ratio typical of current technology rotors as required by NASA guidelines.

<u>Effective Stiffness/Density Ratios</u> - For similar spanwise distributions of airfoil thickness, blade twist, mass and stiffness, ratios of structural natural frequencies to rotor rotational speed are placed similarly if the effective modulus/density ratio varies as tipspeed squared divided by solidity squared. This relationship indicates that the rotors for the three candidate baseline aircraft, designed to meet the noise guidelines with 700 fps (213.4 m/sec.) tipspeeds, all require an effective modulus/density ratio at least 49% of that for the XV-15 rotor. This ratio is approximately that which has been experienced on helicopter fiberglass blades relative to steel blades. For the -5 PNdB designs, the ratio may reduce to 12% due to their low tipspeed and high solidity. For the +5 PNdB designs, the ratio must increase to at least 163%, which would mean heavy emphasis on increased use of high modulus filaments. Some mix of high modulus filaments with fiberglass for the main blade structure appears promising for design tipspeeds over 700 fps.

<u>Other Factors</u> - Gravity (static droop) deflections and stresses of the tilt rotor on the XV-15 are negligible. However, as size increases, these factors become more important.

FIGURE 4.2-4
EFFECT OF STRENGTH/DENSITY RATIO
ON ROTOR WEIGHT FRACTION



The rotor for the -5 PNdB point designs, with relatively limber blades, would have increased droop. In the 100-passenger class it would be an important design consideration, especially when considering environmental effects such as static icing loads.

4.2.2 ENGINES AND DRIVE SYSTEM - Two areas of interest are whether engines are available in the power class required and an identification of any unusual drive system requirements.

Engines - The installed powers required by the nine point designs are plotted in Figure 4.2-5. The variation in power requirements from the -5 PNdB design to the +5 PNdB design is small in any one of the three size classes. Further, it is seen that although "rubber" engines were used in the study, the powers required for each of the three size classes coincide with advanced technology engines which have undergone some degree of development or flight testing. For the 21-, 45- and 100-passenger aircraft, typical engines are the T-700, PLT-27, and LTC4V-1, respectively. No constraint on size selection of the tilt rotor aircraft exists due to engine availability for the time period of the study.

Drive System - The rotor on each pod is driven by a main transmission mounted between the rotor and the pod tilt axis. The transmissions in both wing tip pods are interconnected by a cross-shaft in the wing so that the rotors turn together as with the XV-15. Unlike the XV-15, two engines (rather than one) are mounted in, and tilt with, each pod. They drive through freewheeling units into a combining gear stage which drives the main transmission and interconnect shaft. The combining of two engines to drive a main rotor transmission is common on many production helicopters.

The overall gear ratio between the engine and the rotor shaft varies over a wide range for the nine point designs synthesized. These ratios are shown in Figure 4.2-6. At gear reductions of approximately 70:1 it is likely that one more reduction stage than used in the XV-15 will be required. While gear reductions greater than this are required in some helicopters, the size and shape constraints on the transmission envelope are somewhat different for the tilt rotor configuration. Optimization of transmission configuration, where an additional reduction stage is required, represents an extra task over simply scaling up the XV-15 transmission. The 45-passenger aircraft (baseline version) would require an overall reduction ratio of 58:1, whereas the 100-passenger aircraft would require 75:1. The 45-passenger aircraft would have almost the identical reduction that the XV-15 transmission has without its adapter gearbox for the T53 engine.

FIGURE 4.2-5
POINT DESIGN POWER REQUIREMENTS

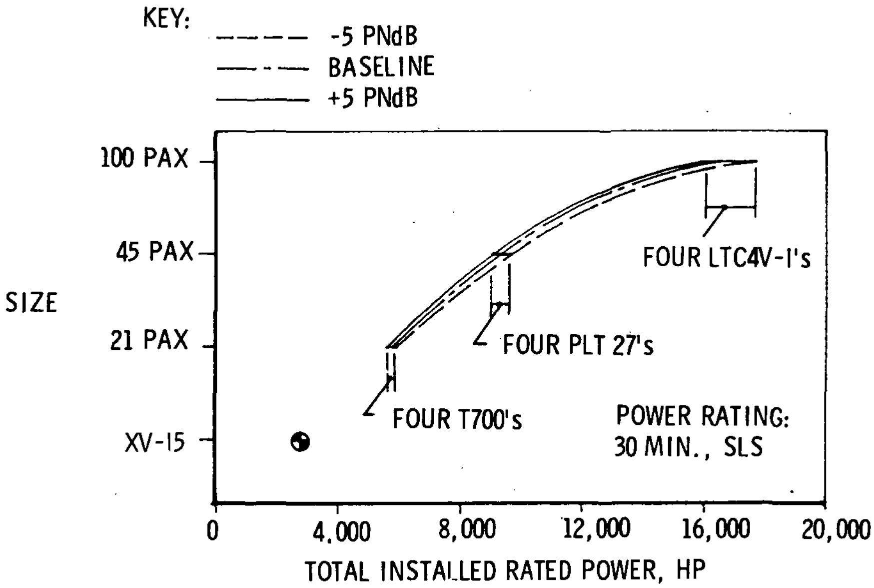
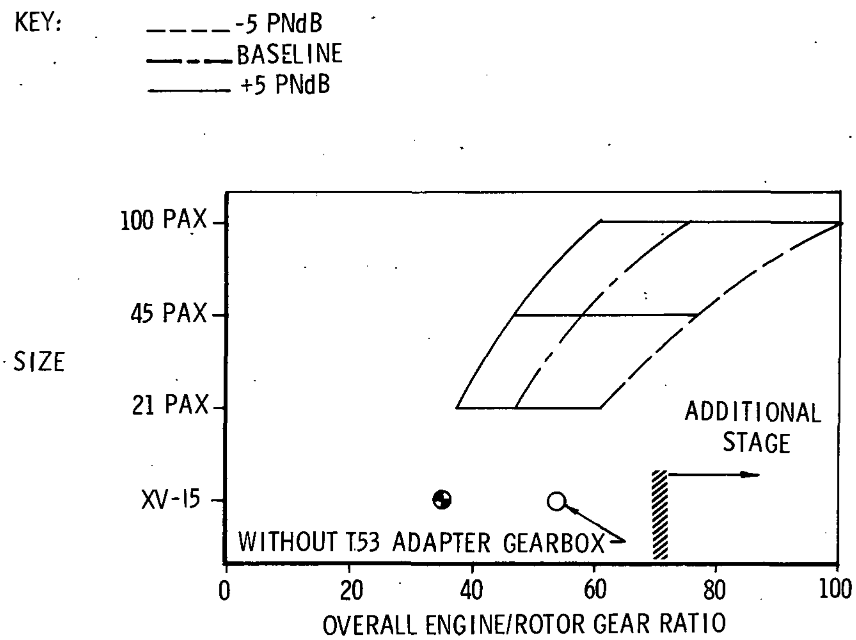


FIGURE 4.2-6
POINT DESIGN ENGINE/ROTOR GEAR RATIOS



4.2.3 WING - The variation of wing span and chord for the three size classes indicated that aspect ratios increase slightly with size (span is dictated by design disc loading and chord by design wing loading). The aspect ratios are similar to that for the XV-15, and are slightly lower for the -5 PNdB designs than for the +5 PNdB designs. The wing dimensions for the 45-passenger aircraft are less than twice those of the XV-15. The wing dimensions for the 21-passenger aircraft are approximately those of the Model 266, and for the 45-passenger aircraft only slightly larger.

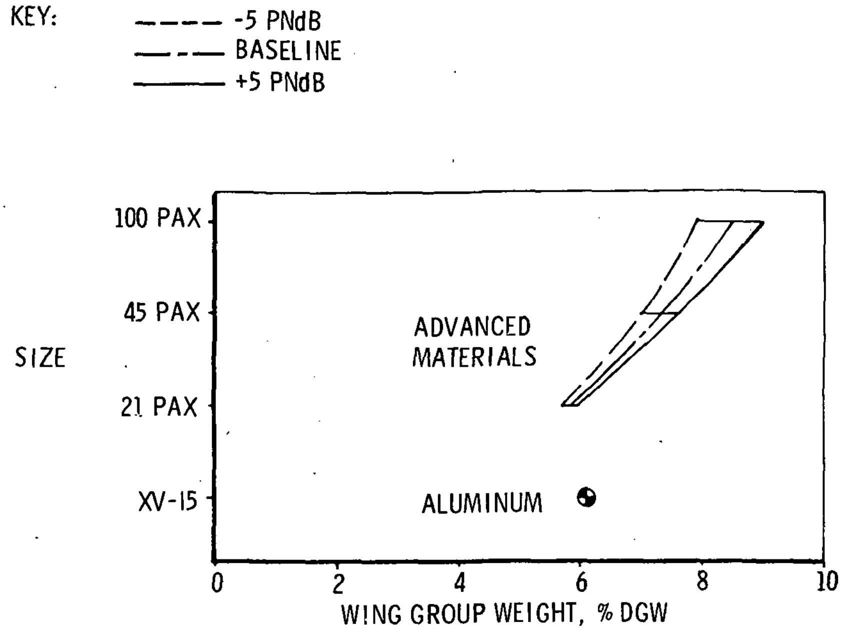
In order to predict wing structural weight for the unique constraints on tilt rotor aircraft wing design, jump takeoff loads and wing torsional stiffness requirements are checked in the design synthesis program to define basic wing structural configuration. Allowances are then made for nonprimary structure based on the XV-15 wing design. Effective properties of advanced composite material are assumed. The estimated wing panel skin thicknesses (assumed the same for inner and outer skins) and overall panel thickness were determined. A relatively small departure in size, compared to the XV-15, was found for the 21- and 45-passenger point designs.

The resulting wing weight fractions are shown in Figure 4.2-7 compared to the XV-15 design which uses aluminum and aluminum honeycomb panel construction. The weight fractions are shown to increase with size. The higher aspect ratio designs show a higher weight fraction to maintain stiffness and/or strength requirements which follows expected trends. The effective properties and potential weight savings of a composite tilt rotor aircraft wing, using the XV-15 requirements as base (detail loads and criteria are available), are being determined under a BHC IR&D initial design and analysis task. Results to date indicate that the weight fraction improvements shown are reasonable to expect (25%). Such improvements are required to maintain productive weight fractions in larger size aircraft.

4.3 CONCLUSION ON SIZE

A review of technological factors indicates that there is no distinct limit on size for the tilt rotor aircraft. Rather, there are various sizes which can be selected, depending on the degrees of certainty one wishes to use for meeting target characteristics in a certain time frame. Given the assumption that V/STOL service demand is adequate for aircraft up to 100-passenger capacity (in the time frame of the study), technological factors become the controlling ones. Based on the assumption of a 1980 go-ahead for the fabrication phase, time will be available in the 1975 to 1979 period to improve the technology levels for several of the key components and systems

FIGURE 4.2-7
POINT DESIGN WING WEIGHT FRACTIONS



which would be used in the 1985 transport. Assuming such technology programs occur, the largest size aircraft which meets the predicted characteristics in this study is the 45-passenger design.

Of the various components considered, unique with the tilt rotor concept, an advanced design rotor system is identified as the most important area for demonstrating that the required technology is in hand for meeting performance, weight, environmental compatibility, and service life goals. The strength/density ratio of the XV-15 steel and titanium rotor meets the requirements of the 45-passenger baseline point design; however, the strength, stiffness, and potential life characteristics of an advanced design composite rotor will provide an opportunity to optimize rotor technology for the 1985 transport. Such characteristics can be demonstrated on a rotor size other than the final size. However, care must be taken to rigorously address the technological factors necessary for meeting the goals established for the final size. Another component, unique with the tilt rotor concept, identified as important for the 1985 transport, is an advanced design wing in which composite materials are employed to meet strength and stiffness requirements for the aircraft without incurring the projected weight growth for an aluminum wing. Opportunities would exist to incorporate advanced airfoils, assuming adequate data become available for the section thickness required. No new engine development program is required, and no unusual characteristics have been identified for the drive system.

5. DESCRIPTION OF SELECTED AIRCRAFT

5.1 GENERAL

The 45-passenger payload size class, as discussed in the preceding section, is considered to be the largest size which would be feasible and practical from a technological viewpoint, if fabrication began in 1980. The 45-passenger baseline aircraft is designated the D312 and is shown in Figure 3.4-2.

The 45-passenger fuselage concept, shown in Figure 5.1-1, has four-abreast seating. As required by the NASA guidelines, the following are provided: two doors, two aisles, space for one cabin attendant, one lavatory, beverage service, coat rack, ticket center, and built-in air stair. In the fuselage belly, baggage compartments are provided to allow 2.5 cu ft per passenger. These accommodation requirements were adequately met by a noncircular cross section. The fuselage external width and height is 150 in. (3.81 m), and overall length is 915 in. (23.24 m).

It is recommended that detailed trade studies be conducted to determine if the second aisle affects loading and unloading times sufficiently to justify its extra weight penalty in this size class.

Fuselage pressurization is provided to hold cabin pressure at the equivalent of 3000 ft (914 m) pressure altitude. This was sized by the NASA requirement that pressure rate-of-change not exceed the equivalent of a descent rate of 300 fpm (91.4 m/min).

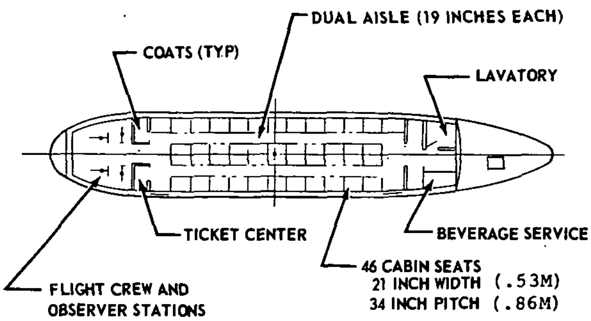
The fuselage layout is common to all three point design 45-passenger aircraft. However, the body group weight varies slightly between point designs because the synthesis method is sensitive to design gross weight and cruise speed.

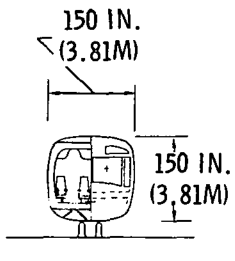
5.2 DESCRIPTION OF THREE FINAL POINT DESIGNS

The baseline aircraft, the +5 PNdB and the -5 PNdB aircraft were synthesized to meet the 200 n. mi. (370 km) NASA mission. The prime design parameter varied to meet the noise criteria was hover tipspeed, which varied from 850 to 550 fps (259 to 167 m/sec). Rotor disc loading and wing loading were analyzed from a DOC viewpoint, and values of 15 and 80 psf (718 and 3830 N/sq m), respectively, were found to be close to the minimum DOC solution. These values also preserve closely the generic values of the XV-15 and were, therefore, selected for all three final point designs. General characteristics are shown in Table 5.2-1 which shows noise, costs, weights, dimensional

FIGURE 5.1-1
FUSELAGE LAYOUT, 45-PASSENGER CLASS

- SOUNDPROOFED
- PRESSURIZED, MCA = 3000 FT. (914M)
- SAME FUSELAGE IS USED FOR BASELINE, +5 PNdB and -5 PNdB CONFIGURATIONS
- MAIN LANDING GEAR LOCATION DEPENDS ON FINAL SIZE OF LIFT-PROPULSION SYSTEM





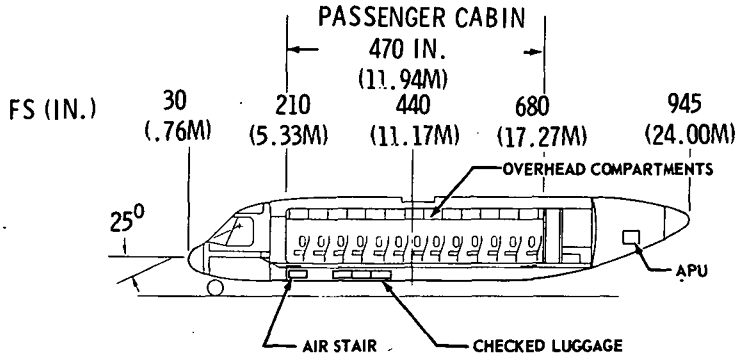


TABLE 5.2-1
CHARACTERISTICS OF THREE 45-PASSENGER POINT-DESIGN AIRCRAFT

NASA MISSION: 200 N.M. RANGE
DESIGN HOVER: SEA LEVEL 90°F, ONE-ENGINE INOPERATIVE

ITEM	UNITS	RELATIVE HOVER NOISE, PNdB		
		-5	0	+5
Noise at 500 ft. sideline, in hover	PNdB	92.2	97.2	102.2
Initial Cost, Including Spares	$M,1974	4.430	3.981	3.790
Direct Operating Cost (per available seat statute mile)	¢/assm	5.24	4.66	4.52
Design Gross Weight	lbs	49388	44848	42797
Weight Empty	lbs	37483	33216	31326
Useful Load	lbs	11905	11633	11471
Crew	lbs	520	520	520
Passengers	lbs	8100	8100	8100
Mission Fuel Including Reserves	lbs	3128	2858	2698
Trapped Fluids	lbs	157	155	153
Disc Loading	psf	15.0	15.0	15.0
Wing Loading	psf	80.0	80.0	80.0
Hover Tip Speed	fps	550	700	850
Cruise Tip Speed	fps	450	600	600
Rotor Diameter	ft	45.8	43.6	42.6
Blade Chord (Three Blades Per Rotor)	in	54.8	32.5	22.4
Wing Span	ft	64.4	62.0	60.9
Wing Chord	ft	9.6	9.0	8.8
Installed Horsepower, (Total, 30 Min. Rating, SLS)	hp	9588	9072	9024
Number of Engines	-	4	4	4
Rated Power Per Engine, Required	hp	2397	2268	2256
Closest Engine Model Type	-	PLT27	PLT27	PLT27
Block Fuel	lbs	2220	2015	1912
Block Time, Engines-On	hrs	.896	.858	.865
Cruise Speed at 11,000 feet, Std. Day	knots	281	296	296
Hover Ceiling, All Engines, 30-Min. Rating, Std. Day, OGE	ft	7300	7100	6000

and performance data. A planview comparison of the -5 PNdB and the +5 PNdB aircraft is shown in Figure 5.2-1. The penalties for the quieter version include a larger wing, tail, and rotor and, therefore, higher design gross weight and direct operating costs.

5.3 MISSION ANALYSIS

The NASA mission was represented by 21 segments which allowed for the basic 200 n. mi leg, the 50 n. mi. alternate leg, and the 20-minute hold. A mission schematic is shown in Figure 5.3-1. Engine fuel flow estimation was based on matching the NASA reference point at sea level 90°F (32.2°C) with typical engine technology (Reference 5-1) of the 1980-85 time frame.

5.3.1 MISSION PROFILE DEFINITION - The BHC-defined items of the mission profile: climb speed, cruise speed, and cruise altitude were determined from minimum DOC considerations. The fuel cost ($.02/lbf) defined by NASA for this study, favored a mission profile which minimized mission time at the expense of mission fuel.

Climb Speed - Point design solutions were synthesized for the NASA 200 n. mi. mission with climb speeds of 1.2 x stall speed (close to speed for maximum rate-of-climb) and 1.8 x stall speed. The higher climb speed required more fuel and required a higher design gross weight (DGW) aircraft; however, the mission time reduced, and this produced a lower DOC solution. Table 5.3-1 shows, for the mission at 11,000 ft (3353 m) cruise altitude, that there is a 1.1 percent penalty on DOC if the slower climb speed is used.

Cruise Speed - Point design solutions were synthesized for cruise speeds corresponding to 99% of maximum range and a higher speed requiring 90% of maximum continuous power. The higher speed required more fuel and a higher DGW solution, but the mission time reduced, and this produced a lower DOC solution. Table 5.3-1 shows, for the mission at 11,000 ft cruise altitude, that there is an 8.7% penalty on DOC if the slower cruise speed is used.

Cruise Altitude - Point design solutions were synthesized for cruise altitudes of 11,000, 15,000, and 20,000 ft (3353, 4572 and 6096 m). The lower altitude required more fuel and a higher DGW solution, but the reduced mission time produced a lower DOC solution. Table 5.3-1 shows that there is a 2.1% penalty on DOC if the 20,000 ft altitude is used compared to the 11,000 ft altitude. Thus, from DOC considerations the lowest possible altitude should be selected. To avoid being limited by the 250 knot (463 kph), 10,000 ft (3048 m) restriction of the Design Criteria, paragraph 2.4, a cruise altitude

FIGURE 5.2-1
EFFECT OF NOISE GUIDELINES ON
LIFT PROPULSION SYSTEM SIZE

45-PASSENGER AIRCRAFT:
----- -5 PNdB VERSION
——— +5 PNdB VERSION

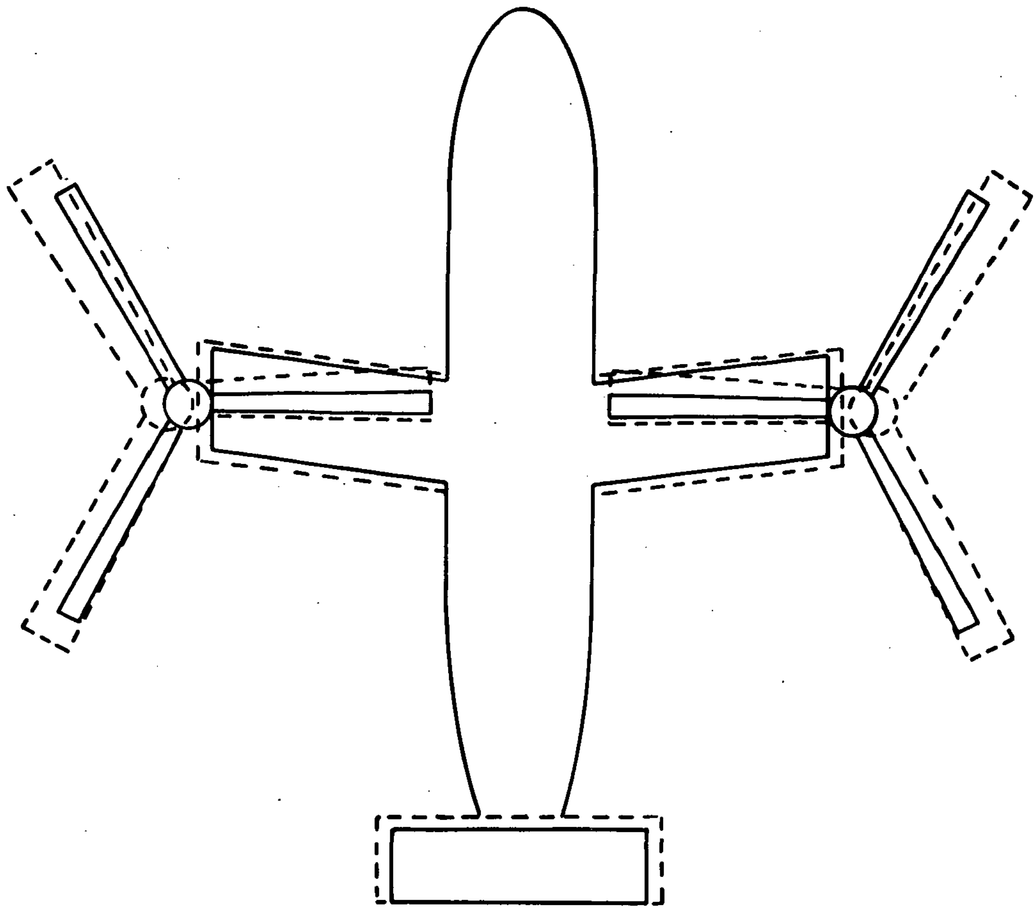
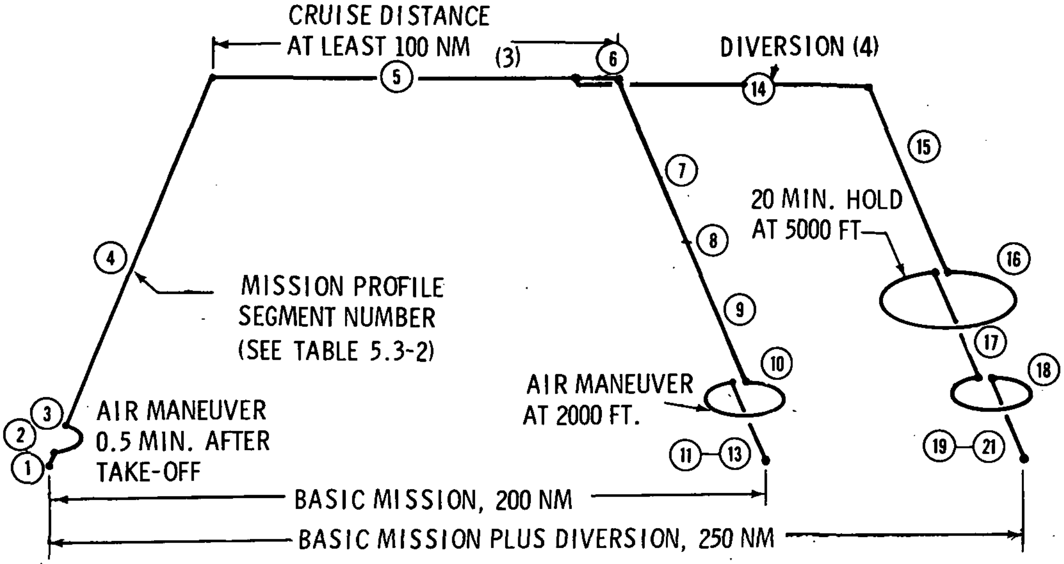


FIGURE 5.3-1
NASA 200 NM V/STOL MISSION PROFILE



NOTE 1. CRITICAL SIZING FOR TAKE-OFFS AND LANDING AT SL 90°F.
2. MISSION FUEL ANALYSIS FOR STANDARD DAY.
3. CRUISE ALTITUDE AND SPEED SELECTED USING MINIMUM DOC AS A GUIDE.
4. DIVERSION AT SPEED FOR BEST RANGE AT CRUISE ALTITUDE.

TABLE 5.3-1
MISSION PROFILE FACTORS AND RELATIVE DOC

CLIMB SPEED, CRUISE SPEED, CRUISE ALTITUDE

	BASELINE	ALTERNATE
CLIMB SPEED	1.8 V_{stall}	1.2 V_{stall}
RELATIVE DOC	100%	101.1%
CRUISE SPEED	90% MCP	99% Max. Range
RELATIVE DOC	100%	108.7%
CRUISE ALTITUDE	11,000 Ft.	20,000 Ft.
RELATIVE DOC	100%	102.1%

of 11,000 ft was selected.

5.3.2 MISSION SEGMENT ANALYSIS - Results of the 200 n. mi. mission analysis for each of the three 45-passenger point design aircraft are shown in Table 5.3-2. Calculations are shown for time required, distance covered, and fuel required for 21 mission segments. Cumulative values are shown for the mission status at the end of each segment. Significant results are that the baseline aircraft used only 2015 lbf (8963 N)of fuel (4.5% DGW), and the mission time (.858 hr) was within 10% of that of current turbofan airliners. Reserve fuel for the baseline aircraft was 843 lbf (3750 N), or 42% of fuel consumed.

The BHC synthesis program compares the fuel required, as determined above, at a trial design gross weight with the fuel available following a weight estimate. New trial design gross weights are automatically tested until a fuel balanced design gross weight is achieved within specified limits. When this is achieved, a group weight statement for the design point solution may be defined as described next.

5.4 GROUP WEIGHT STATEMENTS

The NASA guidelines allowed a 25% weight reduction from present technology for the following components: body, empennage, wing, engine nacelles and nonrotating flight controls. The BHC weight estimating method was based on the following:

Rotor Group:

> Actual weights for the XV-15 rotor group, detailed design study of the Bell Model 266 tilt rotor (DGW = 28,000 lbf) and general helicopter experience.

Drive System:

> General helicopter experience at BHC.

Wing Group:

> Analytical method based on calculated design conditions. No statistics were found to be applicable to wings for tilt rotor aircraft.

Engine Group:

> Basic engine specific weight was defined by the NASA Study Guidelines and is considered to be representative of 1980 technology.

TABLE 5.3-2
MISSION SEGMENT ANALYSIS,
45-PASSENGER AIRCRAFT

SEGMENT			-5 PNdB			BASELINE			+5 PNdB		
MODE NO.	/TYPE	ALT. FT.	TIME HRS.	DIST. N. MI.	FUEL LBS.	TIME HRS.	DIST. N. MI.	FUEL LBS	TIME HRS.	DIST. N. MI.	FUEL LBS.
1	WUP	0	0.017	0	37	0.017	0	35	0.017	0	35
2	HOV	0	0.050	0	140	0.050	0	133	0.050	0	133
3	ALO	0	0.067	0	167	0.067	0	158	0.067	0	156
4	ACL	11000	0.245	42	703	0.195	30	536	0.228	38	569
5	ACR	11000	0.752	184	2062	0.714	184	1867	0.721	184	1773
6	ACR	11000	0.756	185	2070	0.718	185	1874	0.725	185	1780
7	DSC	11000	0.756	185	2070	0.718	185	1874	0.725	185	1780
8	ACR	11000	0.761	186	2078	0.722	186	1882	0.729	186	1787
9	DSC	2000	0.821	200	2113	0.782	200	1914	0.789	200	1817
10	ALO	2000	0.846	200	2151	0.807	200	1950	0.814	200	1850
11	DSC	1000	0.862	200	2166	0.824	200	1965	0.831	200	1864
12	DSC	0	0.879	200	2182	0.841	200	1980	0.848	200	1878
13	GND	0	0.896	200	2219	0.858	200	2015	0.865	200	1912
14	ACR	11000	0.983	238	2501	0.937	237	2268	0.944	237	2144
15	DSC	5000	1.023	247	2524	0.977	247	2290	0.984	247	2164
16	ALO	5000	1.356	247	3016	1.310	247	2750	1.317	247	2594
17	DSC	2000	1.376	250	3025	1.330	250	2758	1.337	250	2602
18	ALO	2000	1.401	250	3063	1.355	250	2793	1.362	250	2635
19	DSC	1000	1.418	250	3078	1.372	250	2808	1.379	250	2648
20	DSC	0	1.435	250	3094	1.389	250	2823	1.396	250	2662
21	GND	0	1.452	250	3131 (13927 N)	1.405	250	2858 (12713 N)	1.412	250	2697 (11997 N)

MODES: ACL - AIRPLANE CLIMB
ACR - AIRPLANE CRUISE
ALO - AIRPLANE LOITER
DSC - DESCENT
GND - GROUND OPERATION
HOV - HOVER
WUP-WARM UP

- MISSION FLOWN ON STANDARD DAY

- FOR RESERVE LEG:
INITIAL CONDITIONS OF
SEGMENT 14 EQUAL
FINAL CONDITIONS SEGMENT 5.

Body Group:

Commercial airliner statistical data.

All other components and systems were based on statistical weight data available to BHC.

The group weight statement for each 45-passenger point design aircraft is shown in Table 5.4-1. The empty weight ratio for the baseline aircraft is 0.741, for the +5 PNdB aircraft it is 0.732, and for the -5 PNdB aircraft it is 0.759.

5.5 MISSION WEIGHT SUMMARY

Mission weight summaries for the three 45-passenger aircraft are shown in Table 5.5-1. Crew and passenger weights are per NASA guidelines:

Pilot Crew (2) .... 190 lbf (845 N), each, including gear

Cabin Attendant (1) .. 140 lbf (623 N), including gear

Passengers (45) ...... 180 lbf (801 N), each, including baggage

The design gross weight for each aircraft shows that to achieve a 5 PNdB reduction the gross weight has to increase approximately 10%, and if a 5 PNdB increase is allowed, the DGW can reduce approximately 5%. These changes also impact on economics as discussed next.

5.6 ECONOMICS

The economic analysis was based on NASA guidelines and the 1968 Aerospace Industries Association method to estimate direct operating costs. This approach to economics is considered by BHC to be adequate at the conceptual design stage. The AIA method estimates the DOC of V/STOL aircraft by allowing for the initial cost and weight of the dynamic systems and then adding this to the airframe and engine costs. BHC compared the AIA method to BHC methods used in Reference 3-4 and found good correlation. It should be noted that if the AIA method was used on an alternative V/STOL concept with a large number of small components, but which had the same total weight and initial cost as the tilt rotor aircraft, then the maintenance cost predicted would be the same and, therefore, would probably be optimistic for the alternative concept.

The three 45-passenger point designs were analyzed for initial cost and direct operating cost for the NASA design mission with climb rates, cruise speeds and altitude selected to

TABLE 5.4-1
GROUP WEIGHT STATEMENTS, 45-PASSENGER AIRCRAFT

WEIGHT ITEM	AIRCRAFT CONFIGURATION -5 PNdB			BASELINE			+5 PNdB		
ROTOR GROUP			6509 LBS			4298 LBS			3321 LBS
WING GROUP			3130			3001			3028
TAIL GROUP			524			438			402
HORIZONTAL		264			216			196	
VERTICAL		260			222			206	
BODY GROUP			5562			5350			5251
LANDING GEAR			1884			1738			1672
NOSE		440			413			401	
MAIN		1414			1298			1245	
AUXILIARY		30			27			26	
FLIGHT CONTROLS GROUP			3243			3103			3066
NONROTATING		2243			2172			2139	
ROTATING		516			480			478	
CONVERSION SYSTEM		484			451			449	
ENGINE SECTION			676			617			594
PROPULSION GROUP			9102			7734			7119
ENGINE INSTALLATION		1709			1653			1647	
EXHAUST SYSTEM		99			96			96	
LUBRICATION SYSTEM		355			336			334	
FUEL SYSTEM		198			192			189	
ENGINE CONTROLS		232			229			227	
STARTING SYSTEM		122			119			118	
DRIVE SYSTEM		6387			5109			4508	
GEARBOXES	5497			4396			3865		
SHAFTING	890			713			643		
INSTRUMENT GROUP			293			293			293
HYDRAULIC GROUP			361			355			352
ELECTRICAL GROUP			495			495			495
AVIONICS GROUP			458			458			458
FURNISHINGS AND EQUIPMENT GROUP			3608			3699			3638
ENVIRONMENTAL CONTROL GROUP			1299			1299			1299
AUXILIARY POWER UNIT			338			338			338
OTHER			0			0			0
LOAD HANDLING GROUP			0			0			0
WEIGHT EMPTY			37482 LBS (166727 N)			33216 LBS (147751 N)			31326 LBS (139344 N)

TABLE 5.5-1
MISSION WEIGHT SUMMARY 45-PASSENGER AIRCRAFT

	-5 PNdB	BASELINE	+5 PNdB
WEIGHT EMPTY	37482 (166727)*	33216 (147751)	31326 (139344)
CREW	520 (2313)	520 (2313)	520 (2313)
PAYLOAD	8100 (36030)	8100 (36030)	8100 (36030)
AUXILIARY TANK	0 (0)	0 (0)	0 (0)
TRAPPED FLUIDS	157 (698)	155 (689)	153 (680)
FUEL AVAILABLE	3129 (13918)	2857 (12708)	2698 (12001)
DGW	49388 (219687)	44848 (199493)	42797 (190369)

*UNITS. . .LBF (N)

minimize DOC.

The baseline cost data used were:

- airframe cost, $90 per pound
- dynamic system cost, $80 per pound
- utilization, 2500 block hours per year
- depreciation period, 12 years

The avionics group cost ($0.25M) has been included in the initial cost and in the depreciation cost, but it has not been included in the airframe maintenance cost equations. All other costs were computed per NASA guidelines and the AIA cost method.

5.6.1 INITIAL COSTS - Table 3.2-1 shows an initial cost of $M 3.981 (1974 dollars) for the baseline 45-passenger aircraft. This includes $M 0.25 for avionics and $M 0.563 for spares. The -5 PNdB aircraft costs 11.3% more, and the +5 PNdB aircraft costs 4.8% less.

5.6.2 DIRECT OPERATING COSTS - Direct operating costs were analyzed over ranges from 50 to 500 s. mi. The design range was 200 n. mi. For longer ranges, extra fuel capacity was installed. Payload was reduced to keep take-off weight at design gross weight. Results are shown in Figure 5.6-1 and Table 5.6-1. For the baseline aircraft, minimum DOC of 4.66 ¢/assm occurs at the design range of 200 n. mi. At 50 s. mi. the DOC is 10.21 ¢/assm, and at 500 s. mi. the DOC is 5.28 ¢/assm. Similar trends occur for the other two 45-passenger aircraft.

5.6.3 DIRECT OPERATING COSTS VERSUS NOISE AND UTILIZATION

Airframe Cost: $90 per Pound - All three 45-passenger point design aircraft were analyzed for DOC for the design mission of 200 n. mi. Airframe cost was $90 per pound, per NASA guidelines. Utilization was 2500 and 3500 block hours per year, per NASA guidelines. Results are shown in Figure 5.6-2, and DOC varies from 4.15 to 5.24 ¢/assm.

Airframe Cost: $110 per Pound - The above analysis was repeated with an airframe cost of $110 per pound, per NASA guidelines. Results are shown in Figure 5.6-3, and DOC varies from 4.29 to 5.43 ¢/assm.

FIGURE 5.6-1
DIRECT OPERATING COST VERSUS RANGE,
45-PASSENGER CLASS AIRCRAFT

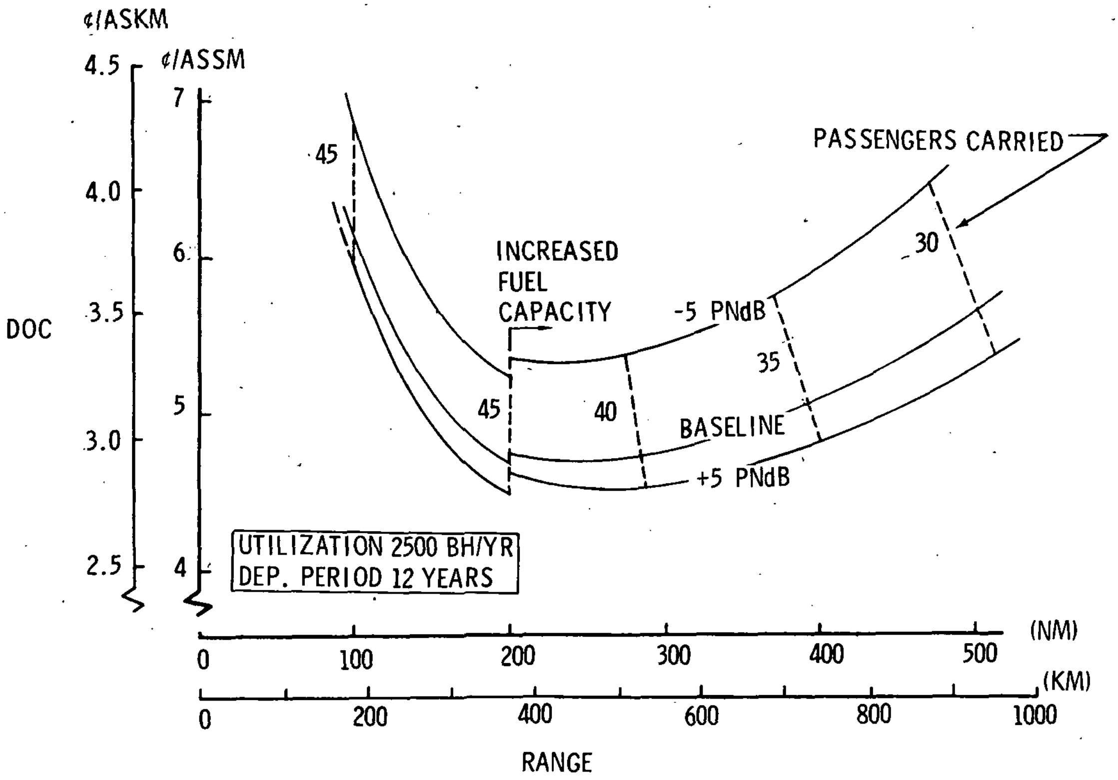


TABLE 5.6-1
DOC VERSUS RANGE, 45-PASSENGER CLASS

RANGE, STAT. MILES (KM)	DIRECT OPERATING COST, ¢/ASSM, (¢/ASKM) -5 PNdB DGW = 49388 LB (219687 N)	BASELINE DGW = 44848 LB (199493 N)	+5 PNdB DGW = 42797 LB (190369 N)
50 (80)	10.80 (6.71)	10.21 (6.35)	9.72 (6.04)
100 (161)	7.30 (4.54)	6.55 (4.07)	6.40 (3.98)
200 (322)	5.45 (3.39)	4.87 (3.03)	4.70 (2.92)
300[1] (483)	5.40 (3.36)	4.72 (2.93)	4.58 (2.85)
400[1] (644)	5.72 (3.55)	4.88 (3.03)	4.66 (2.90)
500[1] (805)	6.15 (3.82)	5.28 (3.28)	5.06 (3.14)

NOTE 1. ADDITIONAL FUEL CAPACITY INSTALLED

FIGURE 5.6-2

DIRECT OPERATING COST VERSUS NOISE AND UTILIZATION, AIRFRAME COST $90 PER POUND

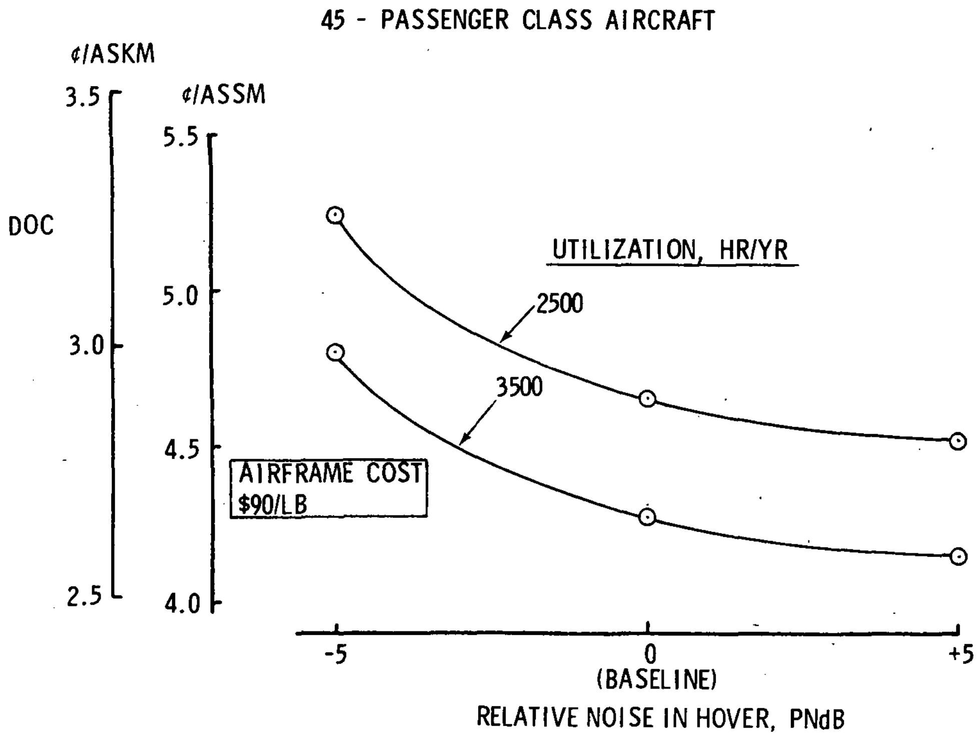
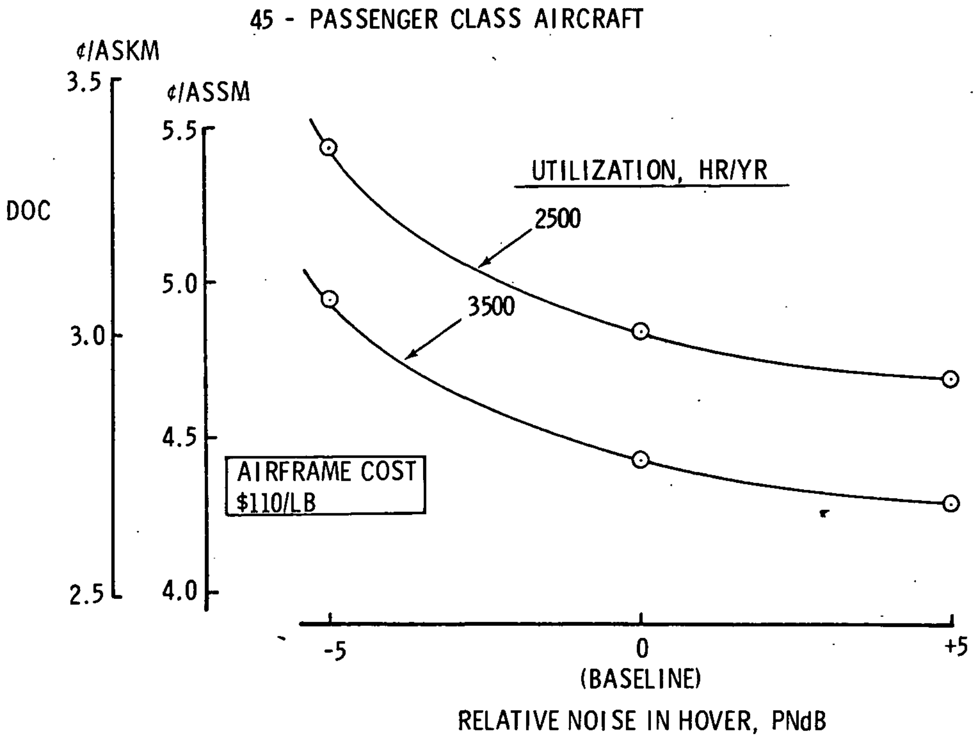


FIGURE 5.6-3
DIRECT OPERATING COST VERSUS NOISE AND UTILIZATION, AIRFRAME COST $110 PER POUND



6. PERFORMANCE

The three 45-passenger point design aircraft were analyzed in the major performance areas of hover capability, cruise envelope, and climb and descent in the helicopter mode.

6.1 HOVER CEILINGS

The hover criterion for each point design was sea level 90°F (32.2°C) with one engine inoperative and the remaining three engines at contingency power (1.09 x ten minute rating). Hover ceilings were determined for all engines operating at maximum continuous power (MCP) and the 30-minute intermediate rated power (IRP), on a standard day. The results are shown in Figures 6.1-1, 6.1-2 and 6.1-3 for the baseline, -5 PNdB, and +5 PNdB configurations, respectively. The hover ceilings for all three aircraft with four engines at IRP on a standard day are from 6000 to 7300 ft (1829 to 2225 m).

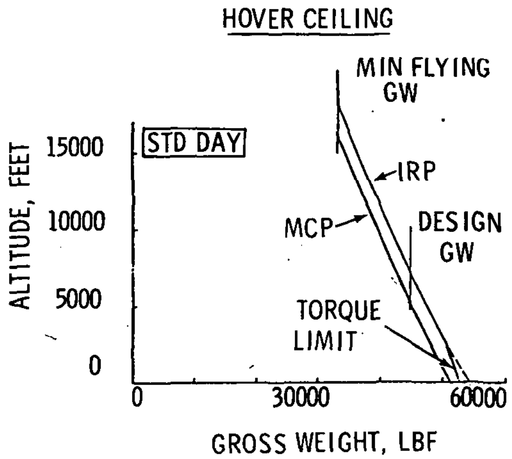
6.2 AIRPLANE CRUISE ENVELOPE

The airplane cruise envelopes for the three point designs are also shown in Figures 6.1-1, 6.1-2 and 6.1-3. The lower limit is at 1.2 x wing stall speed (based on a maximum wing lift coefficient of 1.65, flaps retracted). The upper boundary is limited by maximum continuous power or by the torque limit of the drive system. All three aircraft have maximum speed capability between 295 and 310 knots (546 and 574 kph).

6.3 CLIMB AND DESCENT PERFORMANCE

Rate of climb capability in helicopter mode is shown in Figure 6.3-1 for the baseline 45-passenger point design, at sea level 90°F. Maximum rate of climb is shown to be 2000 fpm (610 m/min) at 70 knots (130 kph) with all engines operating at IRP. A typical minimum noise approach condition of 1000 fpm (305 m/min) descent rate at 40 knots (74 kph) is shown to require a 45% power setting on all four engines.

FIGURE 6.1-1
HOVER AND CRUISE PERFORMANCE, BASELINE AIRCRAFT



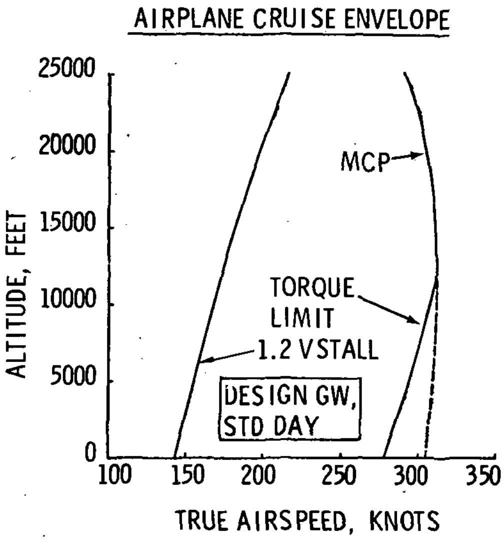


FIGURE 6.1-2
HOVER AND CRUISE PERFORMANCE, -5 PNdB AIRCRAFT

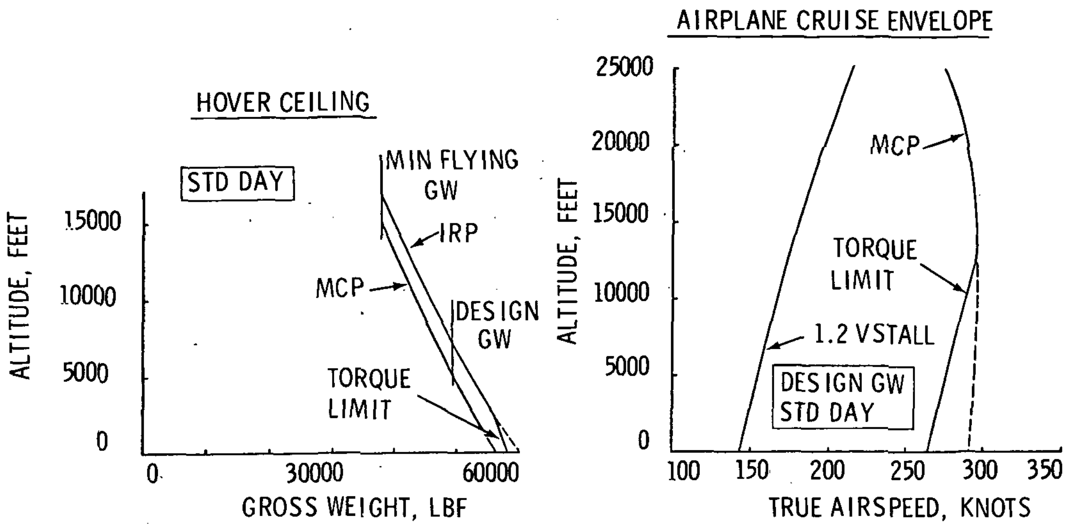


FIGURE 6.1-3
HOVER AND CRUISE PERFORMANCE, +5 PNdB AIRCRAFT

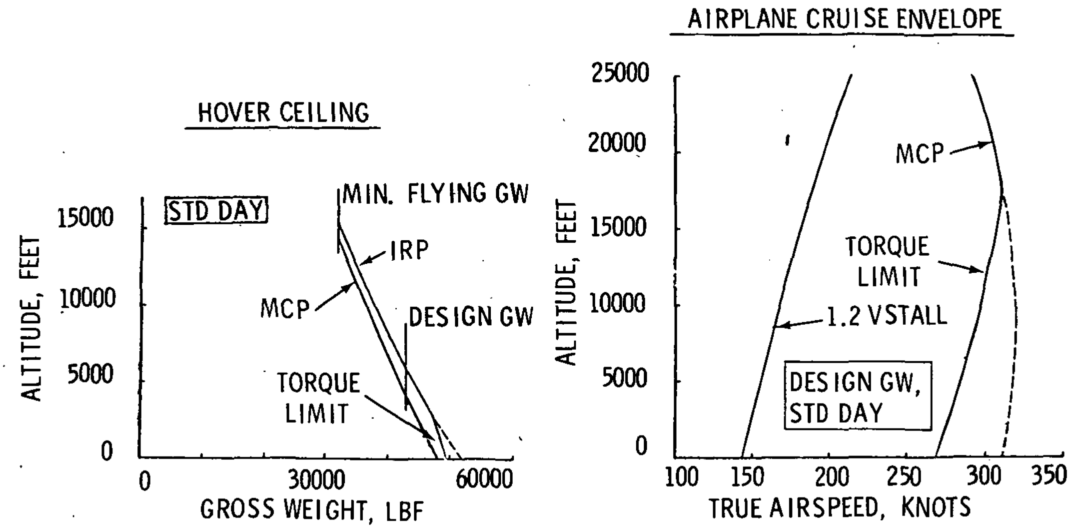
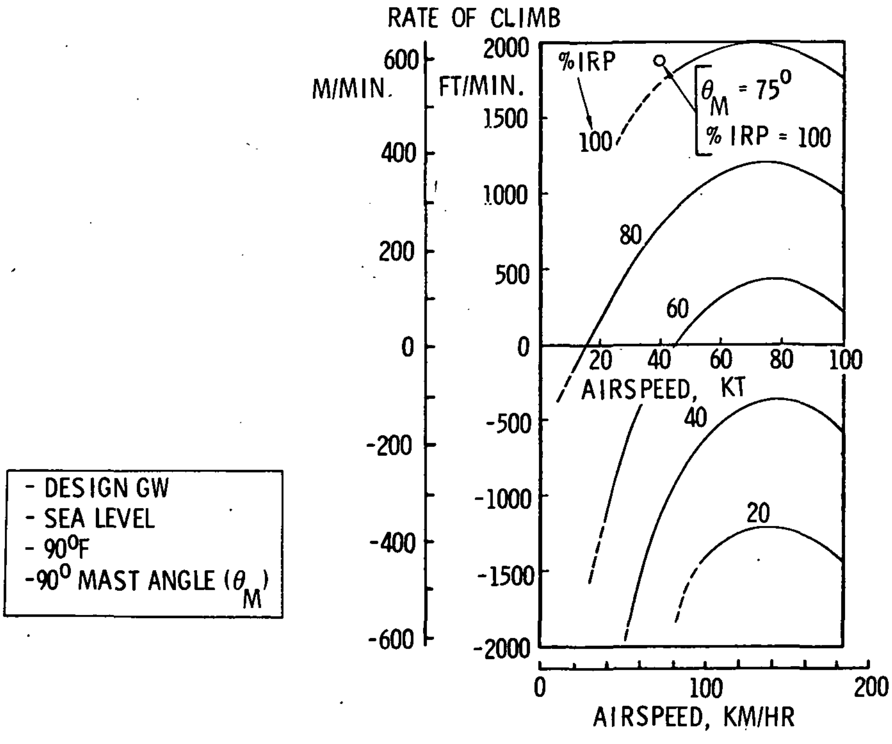


FIGURE 6.3-1
CLIMB PERFORMANCE, ALL ENGINES OPERATING, BASELINE AIRCRAFT



7. HANDLING QUALITIES

The stability, control and handling qualities analyses of the three 45-passenger point designs are based on the results obtained from a digital version of the NASA tilt rotor flight simulation computer program. This program is described in paragraph 7.1. Definition of the configurations studied, inputs for the program and the relationship to the XV-15 are described in paragraph 7.2. The following handling qualities topics are discussed in the subsequent paragraphs: Static trim stability, dynamic stability, control response and maneuver capability, and cruise flight maneuver stability. The low speed gust response is described in paragraph 10.3 under "Safety Aspects". Yaw control power in conversion and airplane modes, which are not the normal modes for final approach and landing, are discussed in the Appendix.

7.1 BASIS FOR ANALYSIS

The stability, control and handling qualities analysis is based on the results obtained from a digital version of the NASA tilt rotor flight simulation program designated BHC Program IFHB74. This particular math model has a six degree-of-freedom trim iteration routine which provides the capability to analyze lateral/directional characteristics, including the effects of a steady-state crosswind condition throughout the flight envelope. Gust and control response predictions are included in the dynamic phase of the model; however, inputs are currently limited to step functions for both cases. The math model limitations currently preclude evaluation with the Stability and Control Augmentation System (SCAS) on. Several additional modifications to the original BHC digital program, IFHB04, (being utilized for XV-15 evaluation) include those of the engine and fuselage aerodynamic portions to accommodate advanced tilt rotor configurations.

7.2 CONFIGURATION DEFINITIONS

Each of the three 45-passenger configurations analyzed for the study (baseline, -5 PNdB, +5 PNdB) possess certain identical characteristics to those of the XV-15 tilt rotor aircraft as follows: blade section properties (i.e., twist, lift and drag coefficients, precone angle and tip loss factor); wing/flap/flaperon and empennage aerodynamic coefficients; rotor-on-wing and rotor-on-empennage induced flow characteristics; cockpit control travels, rotor cyclic rigging (with the exception of differential cyclic/pedal position) aerodynamic surface riggings; and rotor and engine governor characteristics.

Parameters which are scaled from the XV-15 include the following: rotor blade dynamic characteristics (including the flapping hub restraint), fuselage dimensional aerodynamic derivations (ratioed by wing area), ground-effect roll moment values, engine rated power, total aircraft inertias, and maximum rotor thrust coefficient. The scaling factors were determined from the output of the synthesis program.

Design gross weight is used for each configuration for the entire analysis. Center-of-gravity range was that defined in paragraph 4.7 of the Design Criteria; i.e., payload shift of ±5 percent of the passenger cabin length. The basic geometric data (rotor, fuselage, wing/pylon, empennage and landing gear sizes and locations), weight, center-of-gravity, rotor rpm and the scaled parameters discussed above were varied for each of the three configurations as defined by the design synthesis method.

7.3 STATIC TRIM STABILITY

Longitudinal control position and aircraft pitch attitude for each 45-passenger point design (baseline, -5 PNdB and +5 PNdB) are shown in Figures 7.3-1, 7.3-2, and 7.3-3, respectively, for trimmed level flight throughout the speed and conversion angle ranges. The basic data shown are for aft center-of-gravity, helicopter mode rpm, and a flap/flaperon setting of 40/25°. Aft center-of-gravity, airplane mode rpm, zero-degree flap/flaperon data are shown for the baseline configuration which represents sea level tropical day flight conditions. Also shown in Figure 7.3-1 for the baseline configuration are data at forward center-of-gravity for low speeds at each conversion angle.

The fuselage pitch attitudes are all within the specified limits of paragraph 5.1 of the Design Criteria (+20° to -10°) with the exception of the -5 PNdB configuration in helicopter mode above 105 knots (194 kph). However, the 12° nose-down attitude at 120 knots (222 kph) could be decreased with nose-down fixed stabilizer incidence and not significantly influence the remaining stability. A typical conversion from helicopter to airplane mode would begin in the vicinity of 80 knots (148 kph) and therefore the fuselage attitude need not exceed 7° nose-down in this condition.

A stable stick gradient for each conversion angle exists throughout the speed range with the exception of helicopter mode in transition between hover and 20 knots (37 kph) forward speed for all configurations. While this instability is not within the requirements of paragraph 2.6.1 of AGARD-R-577-70 (positive gradient), the baseline and +5 PNdB configurations meet both paragraph 3.2.10 of MIL-H-8501A and Level 3

FIGURE 7.3-1
STATIC TRIM STABILITY, BASELINE AIRCRAFT

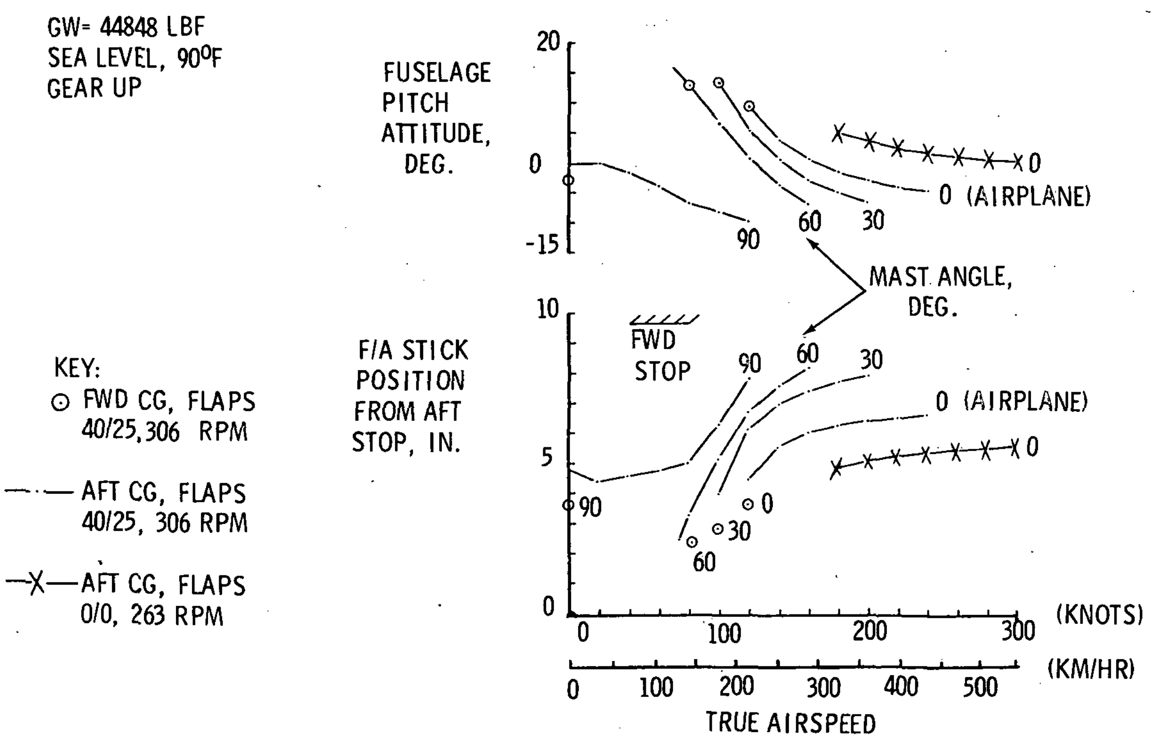
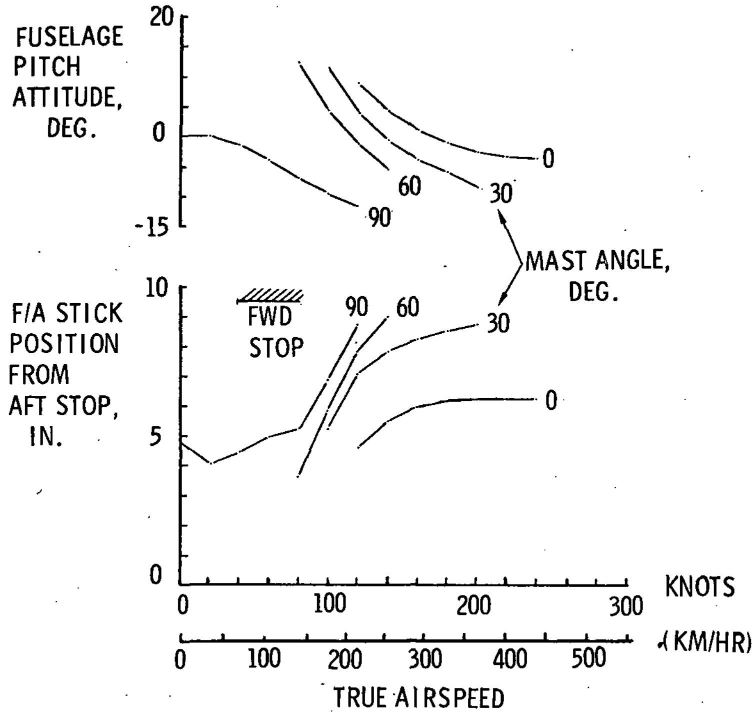
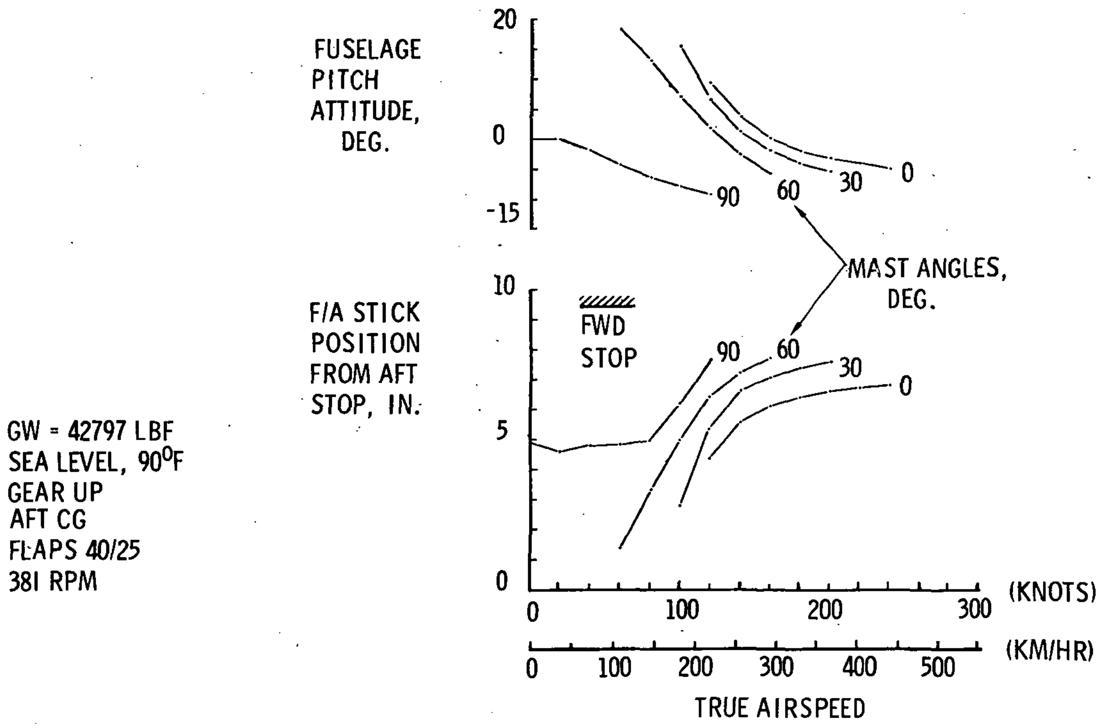


FIGURE 7.3-2
STATIC TRIM STABILITY, -5 PNdB AIRCRAFT



GW = 49388 LBF
SEA LEVEL, 90°F
GEAR UP
AFT CG
FLAPS 40/25
229 RPM

FIGURE 7.3-3
STATIC TRIM STABILITY, +5 PNdB AIRCRAFT



of paragraph 3.2.1.3 of MIL-F-83300 (0.5 in. (1.27 cm) maximum allowable stick reversal). See References 7-1, 7-2, and 7-3. This characteristic is due to the upwash on the horizontal stabilizer during transition as shown by wind-tunnel model test results for the XV-15.

The table shown for Level 1 longitudinal control power in paragraph 1.1.1 of the Design Criteria indicates values of .33 and .30 rad/sec^2 for minimum available pitch acceleration below and above 40 knots (74 kph) respectively. Using this criterion to define a control margin, the baseline configuration exceeds the requirement at aft c.g. and the maximum speeds shown in Figure 7.3-1 for each conversion angle. This same criterion is also satisfied for the forward c.g. and minimum speeds shown for each conversion angle.

7.4 DYNAMIC STABILITY

Level flight dynamic stability from hover and low-speed flight through conversion to 160 knots (296 kph) in airplane mode, is summarized in Tables 7.4-1, 7.4-2 and 7.4-3 for the baseline, -5 PNdB and +5 PNdB configurations, respectively. The anslyses were made at design gross weight, aft c.g., sea level 90°F (32.2°C), helicopter rpm and flaps 40/25° with SCAS inoperative. This method allows an assessment of speed/conversion angle combinations which require SCAS to meet both stability levels as defined by paragraph 1.1.4 and Figure 1 of the Design Criteria.

7.4.1 Low-Speed Oscillatory Modes - The oscillatory modes in the tables are also shown in Figure 7.4-1, 7.4-2, and 7.4-3. The longitudinal short period modes are within the Level 1 optimum zone above 80 knots in helicopter mode and the other oscillatory modes, Dutch Roll and Phugoid, are stable above 80 knots as shown in the lower portion of the figures. The aperiodic Roll and Spiral modes are also stable. Therefore, each configuration satisfies the Level 1 criteria of the Guidelines above 80 knots without SCAS.

7.4.2 Low-Speed Aperiodic Modes - Aperiodic modes in both the longitudinal and lateral/directional (Spiral mode) axes possess time-to-double amplitude values of less than 12 seconds below 80 knots for each configuration. Therefore, SCAS is required in this flight regime to satisfy the requirements for both Levels 1 and 2 aperiodic modes as defined in paragraph 1.1.4. The Dutch Roll mode of the +5 PNdB point design meets Level 1, while this lateral/directional mode for the other point designs meets Level 2 (unstable with T_2 >12 sec. and ω_n < .84 rad/sec). The lateral/directional limits of AGARD-R-577-70 indicate that the SCAS-off Dutch Roll mode characteristics for the

TABLE 7.4-1
DYNAMIC STABILITY SUMMARY, BASELINE AIRCRAFT

	OSCILLATORY MODES				APERIODIC MODES				CRITERIA REFERENCES AND COMMENTS	SCAS REQ'D FOR LEVEL 1
	LONGITUDINAL		LATERAL/DIR.		LONGITUDINAL		LATERAL/DIR.			
V/θ_m kt/deg	$2\zeta\omega_n$ 1/sec	ω_n rd/sec	$2\zeta\omega_n$ 1/sec	ω_n rd/sec	$T_{1/2}$ sec	T_2 sec	$T_{1/2}$ sec	T_2 sec		
HOVER AND LOW SPEED FLIGHT[1]										
0/90	.492	.388	-.071 (T_2=19.5 SEC.)	.332	3.9	2.2	1.0	4.2	Re: Guideline Para. 1.1.4: "Unstable aperiodic modes shall have a time to double > 20 seconds for Level 1."	Yes
40/90	.242	.336	-.072 (T_2=19.3 SEC.)	.481	.42	1.6	.6	5.9		
80/90	2.21 .046	1.52 .276	.049	.52			.4 12.7		All modes meet Level 1 of Guideline Para. 1.1.4.	No
120/60	2.38 .042	2.09 .233	.35	1.07			.34 9.9			
140/30	2.54 .045	2.24 .203	.58	1.0			.4 6.8			
160/0	2.69 0.042	1.95 .186	1.13	1.33			.76 5.4			
	[1] Helicopter RPM. Flaps 40/25, SL, 90°F									
CRUISE FLIGHT[2]										
280/0	2.8 .021	3.21 .123	1.06	2.03			.48 18.56		All modes meet Level 1, Category B of MIL-F-8785B.	No
	[2] Airplane RPM, Flaps 0/0, 11000 ft, 20°F									
Common Conditions: Level flight, Aft CG, SCAS Off, Gross Weight = 44848 lbs										

TABLE 7.4-2
DYNAMIC STABILITY SUMMARY, -5 PNdB AIRCRAFT

	OSCILLATORY MODES				APERIODIC MODES				CRITERIA REFERENCES AND COMMENTS	SCAS REQ'D FOR LEVEL 1
	LONGITUDINAL		LATERAL/DIR.		LONGITUDINAL		LATERAL/DIR.			
$V/\theta m$ kt/deg	$2\zeta\omega n$ 1/sec	ωn rd/sec	$2\zeta\omega n$ 1/sec	ωn rd/sec	$T_{\frac{1}{2}}$ sec	T_2 sec	$T_{\frac{1}{2}}$ sec	T_2 sec		
HOVER AND LOW SPEED FLIGHT[1]										
0/90	.576	.508	-.076 (T_2 = 18.3 SEC.)	.362	3.5	1.52	.97	3.5	Re: Guideline Para. 1.1.4: "Unstable aperiodic modes shall have a time to double >20 seconds for Level 1."	Yes
40/90	.035	.267	-.043 (T_2 = 31.9 SEC.)	.537	.41	6.3	.57	5.96		
80/90	2.35 .048	1.69 .277	.039	.5			.36 9.2		All modes meet Level 1 of Guideline Para. 1.1.4.	No
120/60	2.45 .038	2.28 .244	.324	1.0			.29 8.0			
140/30	2.53 .037	2.32 .21	.54	.94			.31 5.8			
160/0	2.69 .041	1.82 .2	1.04	1.22			.52 5.12			
	[1] Helicopter RPM, Flaps 40/25, SL, 90°F									
CRUISE FLIGHT[2]										
280/0	3.69 .023	3.26 .123	.87	1.87			.35 16.1		All modes meet Level 1, Category B of MIL-F-8785B.	No
	[2] Airplane RPM, Flaps 0/0, 11000 ft, 20°F									
COMMON CONDITIONS: Level flight, Aft CG, SCAS Off, Gross Weight = 49388 lbs										

TABLE 7.4-3
DYNAMIC STABILITY SUMMARY, +5 PNdB AIRCRAFT

	OSCILLATORY MODES				APERIODIC MODES				CRITERIA REFERENCES AND COMMENTS	SCAS REQ'D FOR LEVEL1
	LONGITUDINAL		LATERAL/DIR.		LONGITUDINAL		LATERAL/DIR.			
V/ θ m kt/deg	$2\zeta\omega_n$ 1/sec	ω_n rd/sec	$2\zeta\omega_n$ 1/sec	ω_n rd/sec	$T_{\frac{1}{2}}$ sec	T_2 sec	$T_{\frac{1}{2}}$ sec	T_2 sec		
HOVER AND LOW SPEED FLIGHT[1]										
0/90	.425	.309	.08	.312	4.3	3.04	1.16	5.41	Re: Guideline Para. 1.1.4:"Unstable aperiodic modes shall have a time to double > 20 seconds for Level 1."	Yes
40/90	.222	.309	.091	.428	.42	1.52	.65	6.9		
80/90	2.1 .069	1.41 .27	.056	.53			.43 12.8		All modes meet Level 1 of Guideline Para. 1.1.4.	No
120/60	2.36 .064	1.99 .227	.38	1.09			.37 10.34			
140/30	2.52 .066	2.2 .2	.64	1.06			.47 7.2			
160/0	2.69 0.042	2.0 .18	1.13	1.37			.93 5.6			
	[1] Helicopter RPM, Flaps 40/25, SL, 90°F									
CRUISE FLIGHT[2]										
280/0	3.88 .021	3.23 .121	1.02	2.07			.55 19.7		All modes meet Level 1, Category B of MIL-F-8785B.	No
	[2] Airplane RPM, Flaps 0/0, 11000 ft, 20°F									
Common Conditions: Level flight, Aft CG, SCAS Off, Gross Weight = 42797 lbs										

FIGURE 7.4-1
DYNAMIC STABILITY, BASELINE AIRCRAFT

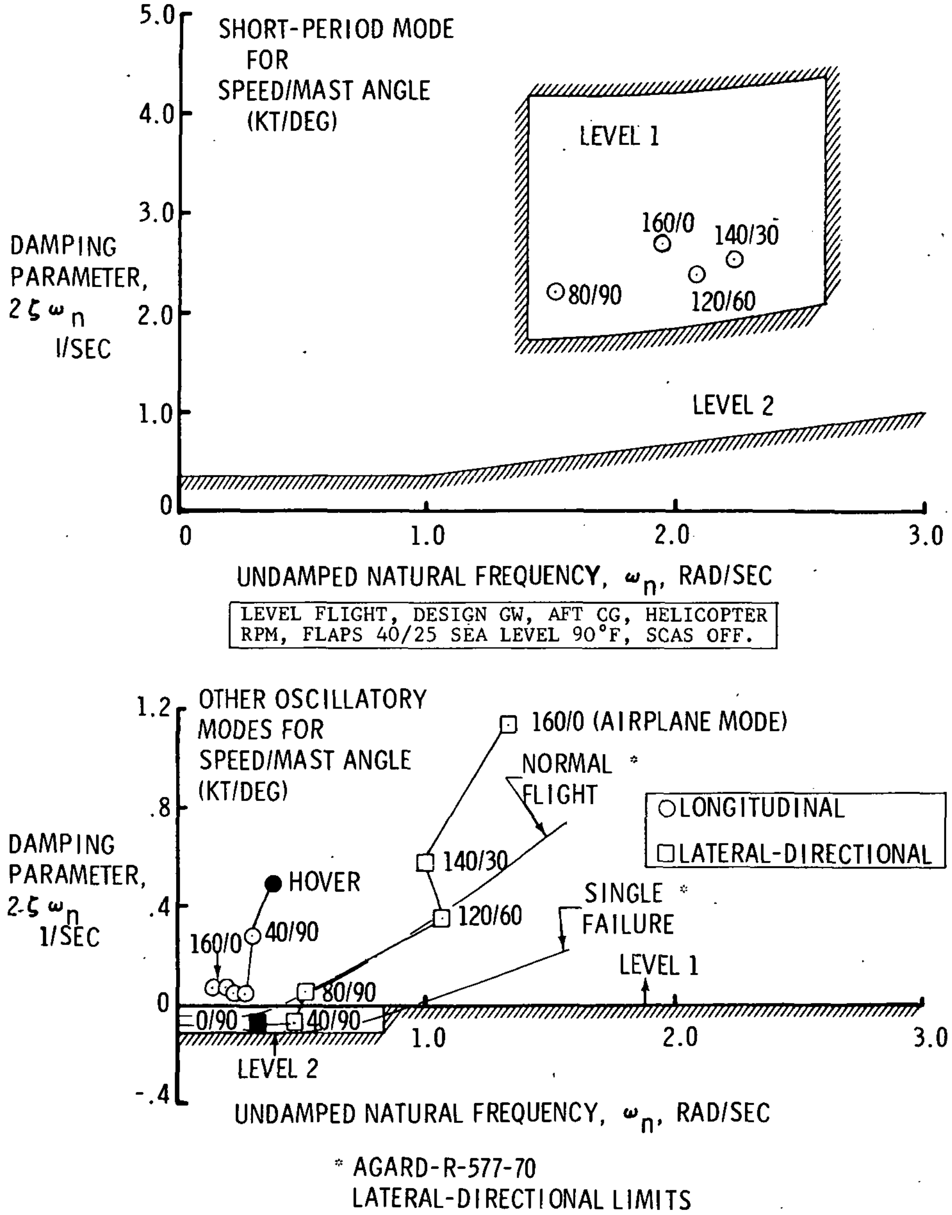


FIGURE 7.4-2
DYNAMIC STABILITY, -5 PNdB AIRCRAFT

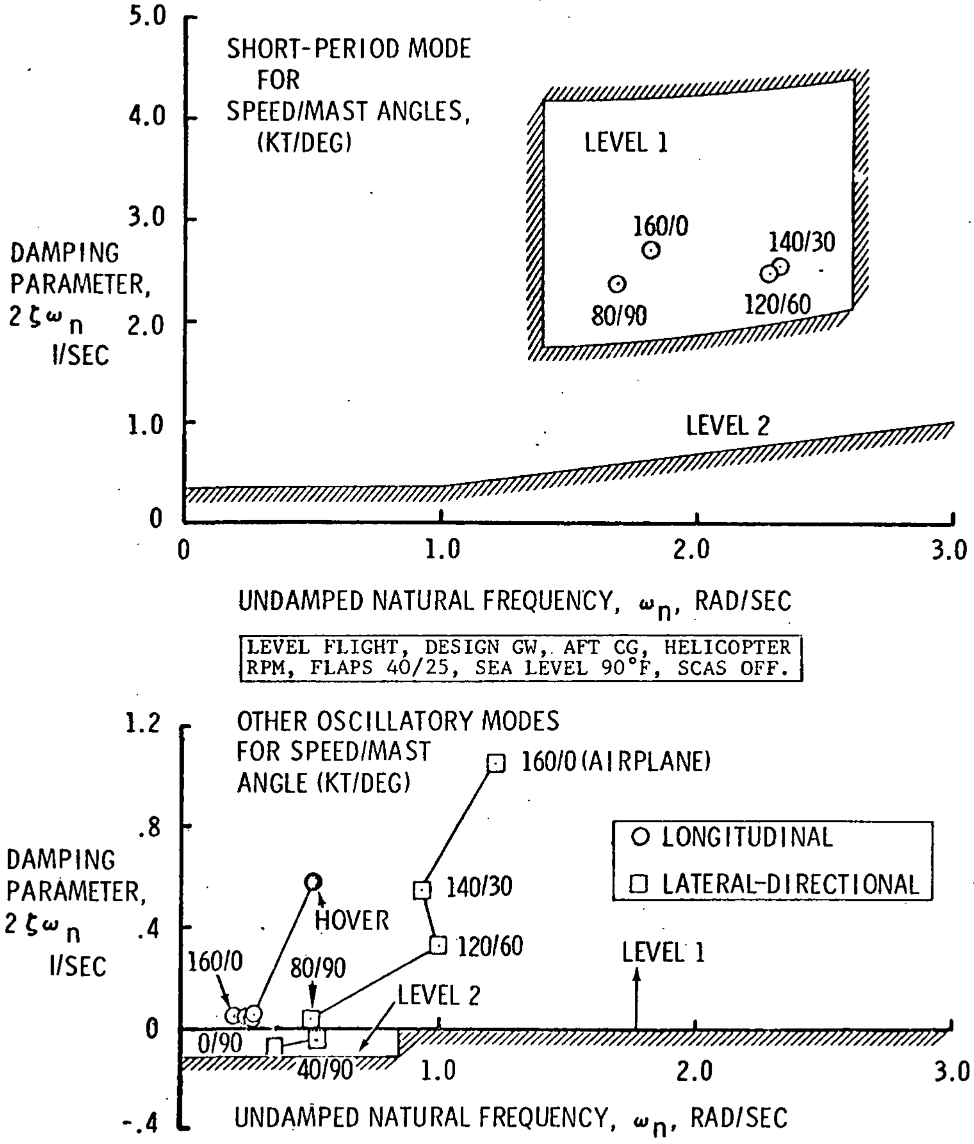
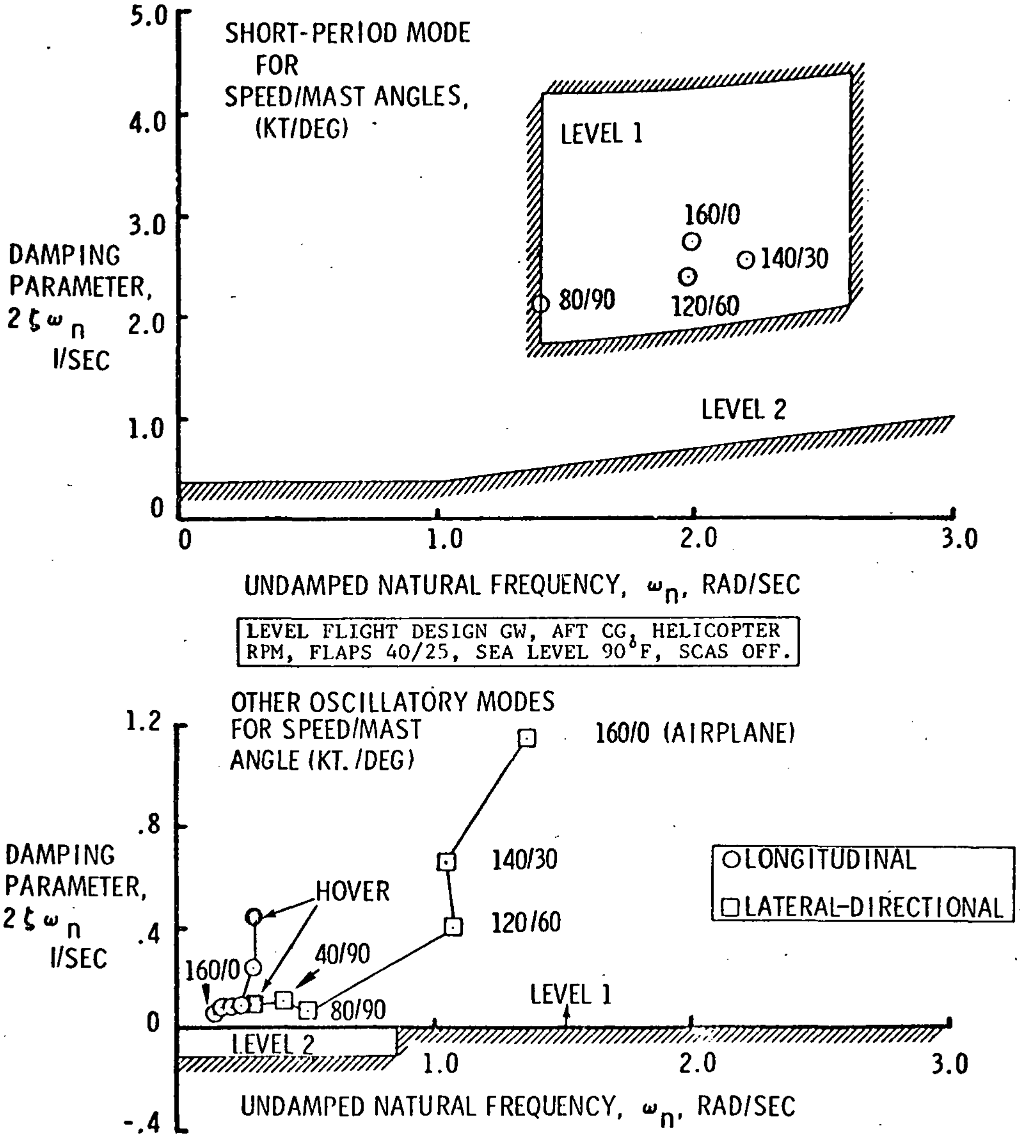


FIGURE 7.4-3
DYNAMIC STABILITY, +5 PNdB AIRCRAFT



baseline configuration (Figure 7.4-1) meet the requirement for single failure from hover through conversion to 160 knots in the airplane mode.

7.4.3 Airplane Cruise Stability - Results of cruise level flight stability at 11,000 ft (3353 m)/std. day conditions are also summarized in Tables 7.4-1, 7.4-2 and 7.4-3. Each configuration meets Level 1, Category B (Nonterminal without precision tracking) MIL-F-8785B (Reference 7-4) damping and frequency requirements without SCAS. Empennage sizing for each design point was based on meeting these criteria in the cruise phase of the mission.

7.5 CONTROL RESPONSE AND MANEUVER CAPABILITY

Attitude control power (determined from the trimmed cockpit control positions, total available control moments from the rotor and control surfaces, and the appropriate inertias) are analyzed for each configuration throughout the hover and low-speed envelope as shown in Tables 7.4-4 through 7.4-6. The study is made in each of the three principal axes for both conditions (a) and (b) (without and with a 25-knot (46 kph) crosswind, respectively) of paragraph 1.1.1 of the Design Criteria to determine the most critical condition which would satisfy the minimum Level 1 requirements. As discussed previously, the pitch control power (Table 7.4-4) is adequate throughout the speed and conversion angle range shown in Figure 7.3-1 for condition(a). Condition (b) is also not critical for maneuver capability in this axis since the F/A stick position is not significantly changed with the addition of a crosswind. Roll control power (obtained from differential collective pitch and ailerons) is more than adequate for meeting the minimum requirements of both conditions (a) and (b) throughout the speed/conversion angle range.

Yaw control power shown for helicopter mode in Table 7.4-6 is sufficient to meet the requirements for trimming in level flight (with or without a crosswind) from hover to 120 knots, and subsequently, for possessing enough pedal margin to accelerate to the Level 1 criteria in either speed range. Therefore, the yaw acceleration would be adequate while executing VTOL approach and landing or takeoff and climb-out.

Time histories of yaw attitude response to control inputs about each axis are shown in Figure 7.5-1 for the baseline configuration. The yaw angle response in one second meets the Level 1 criteria at the representative speeds shown. Typical examples of pitch and roll attitude response in hover with no crosswind are shown in Figure 7.5-2 indicating that they, too, exceed the Level 1 criteria.

TABLE 7.4-4
PITCH ACCELERATION CONTROL POWER
45-PASSENGER AIRCRAFT

FORWARD GRD. REF. SPEED, KNOTS	CONDITION MAST ANGLE, DEG.	a. NO CROSSWIND			b. 25 KNOT CROSSWIND		
		AIRCRAFT DESIGNS					
		-5 PNdB	BASELINE	+5 PNdB	-5 PNdB	BASELINE	+5 PNdB
(CRITERIA PAR. 1.1.1)		(LEVEL 1 / 2: .33/.2 RAD / SEC^2)			(LEVEL 1: .165 RAD / SEC^2)		
0	90	.86	.74	.66	.82	.73	.64
40	90	.86	.82	.82	.84	.75	.65
(CRITERIA PAR. 1.1.1)		(LEVEL 1 / 2: .3 / .2 RAD / SEC^2)			(LEVEL 1: .15 RAD / SEC^2)		
80	90	.98	1.00	1.00	.67	.66	.63
120	60	.56	.82	.90	.42	.75	.83
140	30	.76	1.04	1.15	.58	.89	1.00
160	0	1.98	1.79	1.70	1.84	1.62	1.52

DGW, AFT CG, FLAPS 40/25, HELO MODE RPM, SL 90^{0}F

TABLE 7.4-5
ROLL ACCELERATION CONTROL POWER
45-PASSENGER AIRCRAFT

FORWARD GRD. REF. SPEED, KNOTS	CONDITION MAST ANGLE, DEG.	a. NO CROSSWIND			b. 25 KNOT CROSSWIND		
		AIRCRAFT DESIGNS					
		-5 PNdB	BASELINE	+5 PNdB	-5 PNdB	BASELINE	+5 PNdB
(CRITERIA PAR. 1.1.1)		(LEVEL 1 / 2: .6 / .3 RAD / SEC^2)			(LEVEL 1: .3 RAD / SEC^2)		
0	90	.68	.77	.85	.35	.59	.73
40	90	.68	.80	.89	.35	.57	.72
(CRITERIA PAR. 1.1.1)		(LEVEL 1 / 2: .4 / .2 RAD / SEC^2)			(LEVEL 1: .2 RAD / SEC^2)		
80	90	.96	1.07	1.17	.62	.90	1.07
120	60	1.15	1.20	1.26	1.09	.88	.90
140	30	.92	.88	.87	.80	.84	.78
160	0	.51	.51	.50	.31	.34	.35

DGW, AFT CG, FLAPS 40/25, HELO MODE RPM, SL 90°F

TABLE 7.4-6
YAW ACCELERATION CONTROL POWER
45-PASSENGER AIRCRAFT

FORWARD GRD REF. SPEED, KNOTS	CONDITION MAST ANGLE, DEG.	a. NO CROSSWIND			b. 25 KNOT CROSSWIND		
		AIRCRAFT DESIGNS					
		-5 PNdB	BASELINE	+5 PNdB	-5 PNdB	BASELINE	+5 PNdB
(CRITERIA PAR. 1.1.1)		(LEVEL 1 / 2: .25 / .15 RAD / SEC^2)			(LEVEL 1: .125 RAD / SEC^2)		
0	90	.35	.34	.34	.13	.19	.22
40	90	.30	.29	.29	.23	.23	.24
(CRITERIA PAR. 1.1.1)		(LEVEL 1 / 2: .2 / .15 RAD / SEC^2)			(LEVEL 1: .10 RAD / SEC^2)		
80	90	.31	.31	.32	.29	.25	.24
120	.90	.62	.57	.56	.51	.40	.38

DGW, AFT CG, FLAPS 40/25, HELO MODE RPM, SL 90^{0}F

FIGURE 7.5-1
YAW ATTITUDE RESPONSE, BASELINE AIRCRAFT,
IN 25-KNOT (46 KM/HR) CROSSWIND

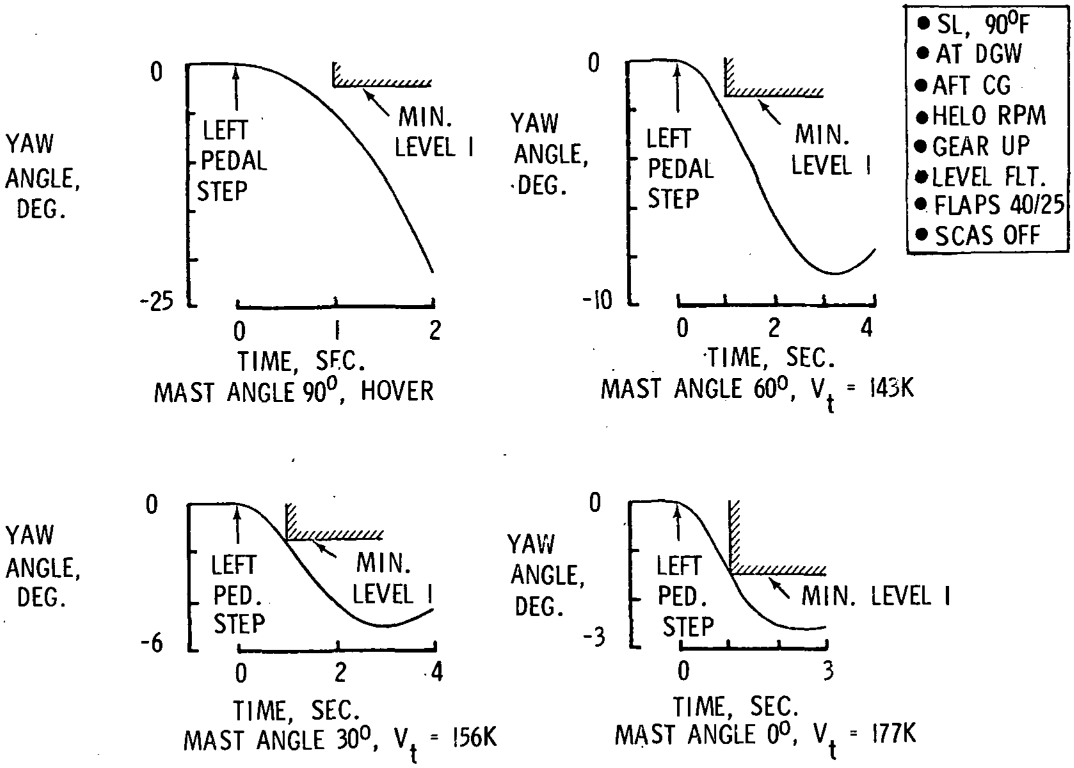
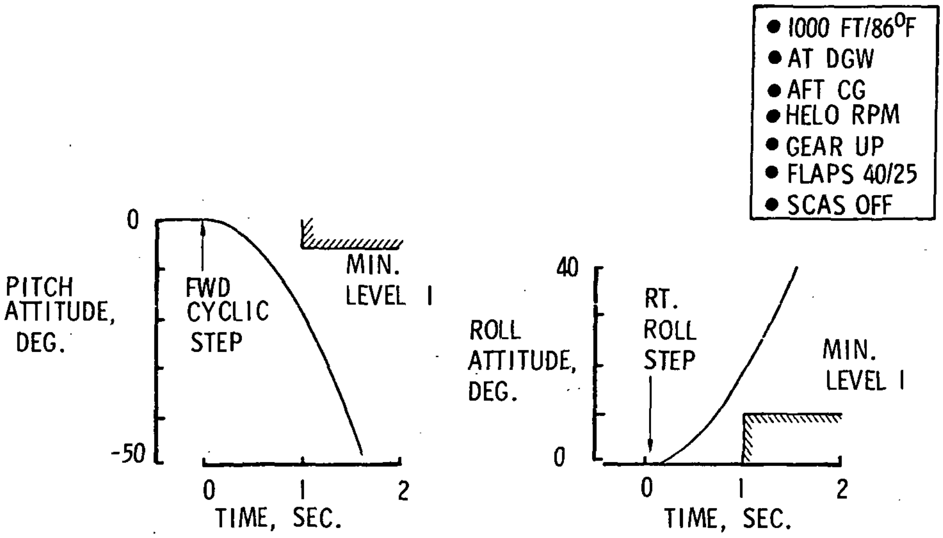


FIGURE 7.5-2
PITCH AND ROLL RESPONSE, BASELINE AIRCRAFT, IN HOVER, NO CROSSWIND



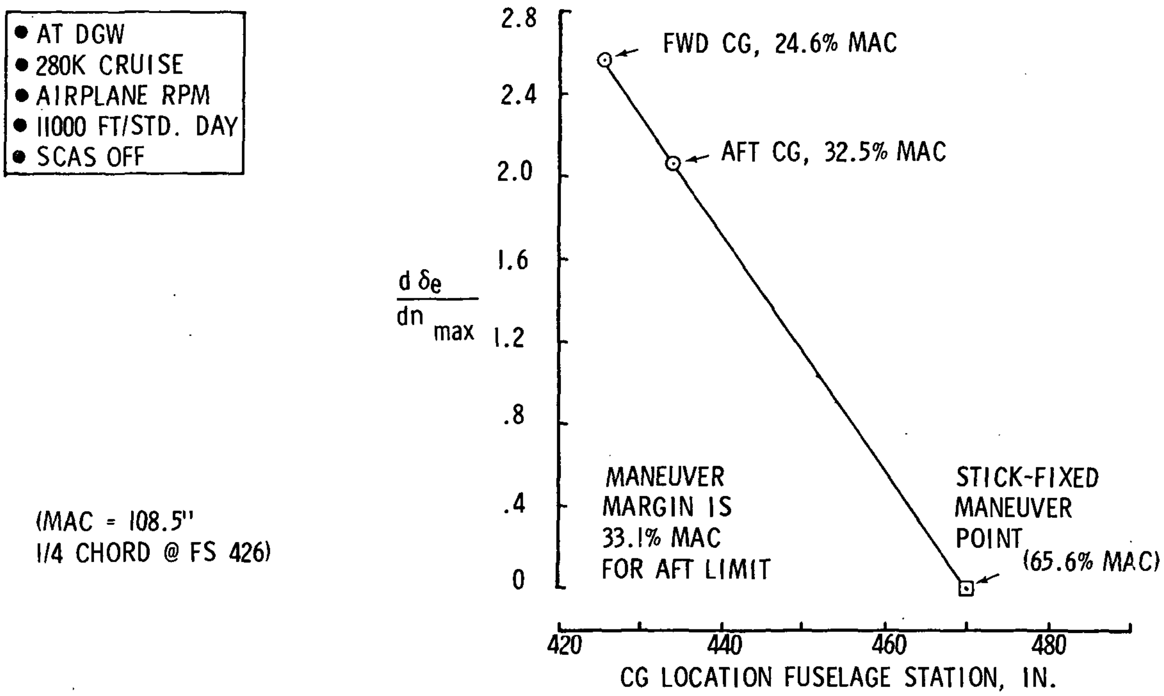
Flight path control power from hover to 40 knots is more than sufficient for achieving the Level 1 incremental vertical g requirements in paragraph 1.1.2.1 of the Design Criteria. The remaining collective pitch following a roll input (maximum angular acceleration specified) is adequate to produce $0.1\Delta g$ for OGE conditions and $+0.05$, $-0.1\Delta g$ for IGE conditions. The incremental horizontal acceleration capabilities of the aircraft also adequately meet the $0.15\Delta g$ requirement in both the longitudinal and lateral axes in this speed regime. The use of longitudinal cyclic to produce incremental normal accelerations above 0.1g is adequate above approximately 50 knots (93 kph) during a 2000 fpm descent with a 25-knot crosswind and,therefore, the use of collective pitch to arrest a high-sink rate could be limited to the VTOL flight regime (0-40 knots).

7.6 CRUISE FLIGHT MANEUVER STABILITY

The stick-fixed maneuver stability for the baseline design point at mission cruise conditions and design gross weight is shown in Figure 7.6-1. The forward center-of-gravity point (FS 425.6) is immediately forward of the wing quarter-chord (FS 426) while the aft c.g. point is located at 32.5% M.A.C. (FS 434.1). This c.g. range represents a payload shift of ±5% of the cabin length. Both of these limits possess positive maneuver stability without the use of SCAS. The stick-fixed maneuver point, i.e., that c.g. location at which the elevator deflection/g equals zero, is located at 65.6% M.A.C. (FS 470) providing a maneuver margin in this flight regime for the aft c.g. of 33.1% M.A.C. (35.9 in.)

Using the current XV-15 force-feel constants, (a stick-force gradient of 13.2 lbf/in. (23.1 N/cm) at 280 knots (519 kph)) provides values of 8.12 and 6.55 lbf (36 and 29 N)/g for the forward and aft c.g. limits, respectively. These results indicate that the center-of-gravity-envelope could be extended as the static stability margin is also more than adequate as shown in Figure 7.3-1.

FIGURE 7.6-1
MANEUVER STABILITY, BASELINE AIRCRAFT,
CRUISE MODE



8. AEROELASTIC STABILITY AND RIDE COMFORT

An important design consideration for the tilt rotor is the provision of adequate aeroelastic stability margins of the rotor-wing combination for the speed-altitude envelope capability of the aircraft. To check the credibility of the three 45-passenger point-designs, analysis of the aeroelastic boundaries was conducted. The same BHC computer program, DYN4, may be used to assess ride comfort based on a Von Karman turbulence field. Ride comfort levels for the three point designs were analyzed in this way, both for vertical response to vertical gusts and longitudinal response to head-on gusts.

8.1 AEROELASTIC STABILITY

8.1.1 METHOD ANALYSIS - The parameters defining kinematics and structural quantities were generally obtained from the Tilt Rotor Aircraft Design Synthesis program (OMSW02). However, the parameters of wingtip beamwise spring rate, chordwise spring rate, wing effective mass, wing chord effective hinge location, pylon pitch and yaw spring rates, and the offset from pylon conversion axis to wing elastic axis were scaled from the XV-15. The aircraft rigid body stability derivatives were also scaled from the XV-15. Studies of aeroelastic stability were made by treating symmetric modes about the fuselage longitudinal centerline separately from those antisymmetrical about the centerline. For the symmetric or antisymmetric modes, the DYN4 math model consists of the following degrees-of-freedom.

a. Two rigid-body flapping modes, one involving backward precession in the rotating system; the other, forward precession. These are both symmetric and antisymmetric modes.

b. Three rigid-body airframe modes: plunging, pitching and longitudinal translation in the symmetric case; and roll, yaw, and lateral translation in the antisymmetric case.

c. Five wing-pylon elastic degrees of freedom: wing beamwise bending, chordwise bending, and torsion; and pylon pitch and yaw with respect to the wing. These are for both symmetric and antisymmetric modes.

These ten degrees-of-freedom for each set of modes, which are completely coupled in the analysis, were considered to be adequate to represent the coupled natural modes of the point designs.

8.1.2 RESULTS - The criterion for aeroelastic stability for the commercial transport is taken from the FAA Airworthiness Standards: Transport Category Airplanes, Part 25, Section 25.629 (Reference 8-1). FAR Part 25 requires that the aircraft be designed to be free from flutter and divergence for all combinations of altitude and speed encompassed by the dive speed (V_D) versus altitude envelope, enlarged by an increase of 20% in equivalent airspeed. Based on this criterion, and defining V_D as 1.15 times the speed at maximum continuous power, V_{MCP}, the three 45-passenger point designs all have sufficient margins for aeroelastic stability, as shown in Table 8.1-1. The speed margins for the baseline aircraft, versus altitude, are shown in Figure 8.1-1.

8.2 ANALYSIS OF RIDE COMFORT

8.2.1 METHOD OF ANALYSIS - BHC computer program DYN4 was used to analyze ride confort of the three 45-passenger point design aircraft in the cruise mode. An analytical method based on a statistical representation of turbulence was math modeled in this analysis to calculate the aircraft response to atmospheric turbulence. A Von Karman turbulence field power spectral density with a scale of 2500 ft (762 m) was used. This spectrum is considered to be a reasonable analytical representation for atmospheric turbulence, Reference 7-4. The assumption of a one-dimensional gust field was made to simplify the analysis.

By calculating the rms value of a response parameter (such as the vibration level in g's), a scalar measure of the response is obtained for the aircraft encountering turbulence consisting of excitation over a wide range of frequencies.

8.2.2 RESULTS - The gust response of the three aircraft are compared to the NASA criteria in Figure 8.2-1. At an altitude of 11,000 ft (3353 m) all three aircraft essentially meet the longitudinal gust response criteria. However, the vertical gust response exceeds the criteria boundary taken from Figure 3 of the Study Guidelines.

Higher cruise altitudes were then investigated for the baseline aircraft and at 20,000 ft (6096 m) the NASA criterion of 0.03 g/fps (0.098 g/m per sec) was met for the vertical gust response. This NASA criterion indicates that cruise altitudes of approximately 20,000 ft should be considered from the viewpoint of ride comfort.

8.3 WEIGHT INCREMENT FOR RIDE-COMFORT CRITERIA COMPLIANCE

As discussed in the preceding section the baseline aircraft,

TABLE 8.1-1

AEROELASTIC SPEED MARGINS AT 11,000-FEET ALTITUDE

AIRCRAFT	V_{MCP}, KT/FEET	$V_D = 1.15\ V_{MCP}$ KT	$1.2\ V_D$ KT	$V_{FLUTTER}$ KT
-5 PNdB	296/11000	340	408	432
BASELINE	310/11000	357	428	487
+5 PNdB	320/11000	368	442	500 +

FIGURE 8.1-1
AEROELASTIC MARGINS
45-PASSENGER BASELINE AIRCRAFT

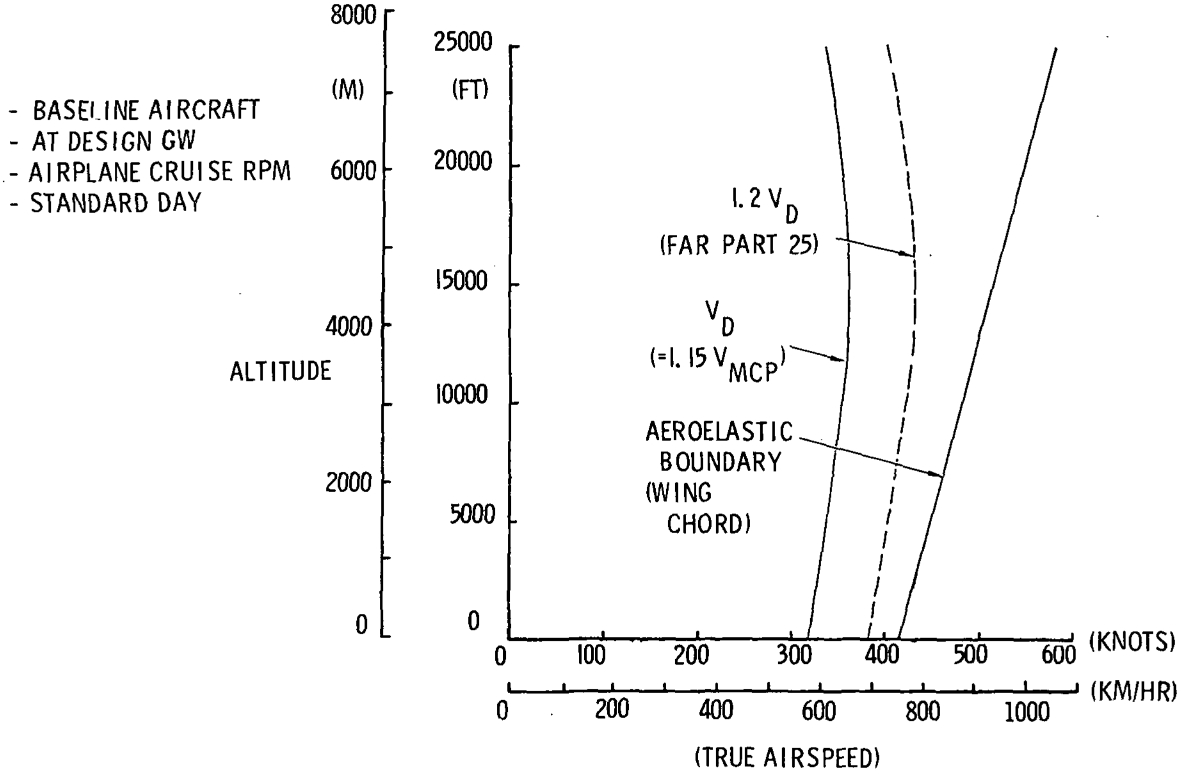
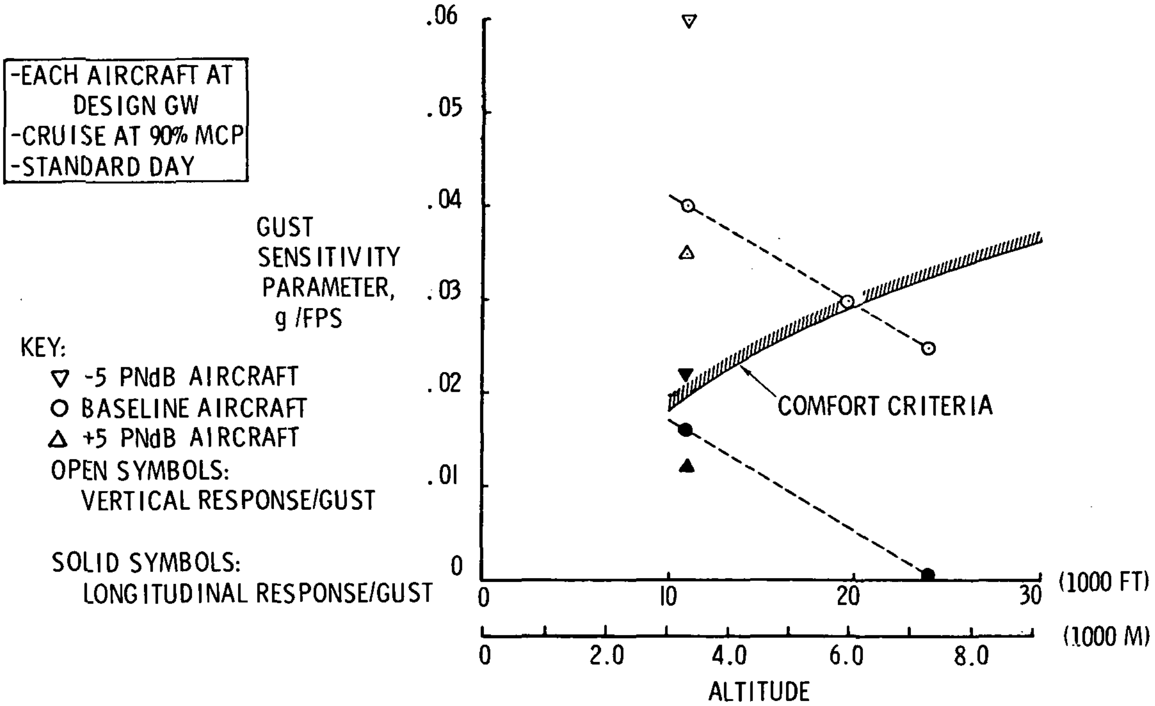


FIGURE 8.2-1
RIDE COMFORT ANALYSIS IN CRUISE MODE
45-PASSENGER AIRCRAFT



with a design gross weight of 44,848 lbf (199 493 N), did not meet the NASA ride-comfort criteria for response to a vertical gust.

To meet the criteria, cruise altitude was increased to 20,000 ft and a point design aircraft was synthesized to meet the mission. Design gross weight was 44,530 lbf (198 078 N). At the cruise speed of 287 knots (531 kph) this point design aircraft essentially met the gust response requirement of .0292 g/fps. However, the DOC increased from 4.66 ¢/assm to 4.76 ¢/assm. This small increase, for a fleet of 300 aircraft flying 2500 block hours/year over 12 years, represents an increase in operating costs of $108.6 million dollars. A point design aircraft was then synthesized with the capability to meet the ride comfort criteria at 20,000 ft and also with the larger fuel system required to fly the mission at 11,000 ft. This dual-mission aircraft has:

- An extra 103 lbf (458 N) of fuselage pressurization weight to increase design cruise altitude from 11,000 to 20,000 ft.

- A fuel system large enough to meet the low altitude mission.

Design gross weight for this "dual-mission" aircraft increased from the 44,848 lbf of the baseline aircraft to 45,078 lbf (200 516 N). When atmospheric turbulence is low, it can fly the 11,000 ft mission profile at a DOC of 4.68 ¢/assm and when required, can fly the 20,000 ft mission profile at a DOC of 4.79 ¢/assm. Results are tabulated in Table 8.3-1.

TABLE 8.3-1
DOC - RIDE COMFORT TRADES

DESIGN CRITERIA	SPEED/ ALTITUDE Kt./FEET (KPH/M)	DGW LBF (N)	FUEL CAPACITY LBF (N)	DOC ¢/ASSM (¢/ASKM)	COMFORT g/FT/SEC (g/M/SEC)
MIN DOC	296/11000 (548/3353)	44848 (199492)	2858 (12712)	4.66 (2.89)	.04 (.131)
RIDE COMFORT	287/20000 (531/6096)	44530 (198078)	2598 (11556)	4.76 (2.96)	.03* (.098)
CAPABILITY FOR BOTH (MIN. DOC AND RIDE COMFORT)	296/11000 287/20000	45078 45078 (200515)	2872 2872** (12775)	4.68 (2.91) 4.79 (2.98)	.04 (.131) .03* (.098)

* MEETS RIDE COMFORT CRITERIA OF DESIGN GUIDELINES.
** RESERVE-LEG LOITER TIME CAN BE EXTENDED 53%.

9. NOISE CHARACTERISTICS

Tilt rotor noise levels are calculated with the BHC rotorcraft noise prediction computer program KA9701. This procedure uses the analytical formulation of Lawson and Ollerhead (Reference 9-1) and also correlation with experimental data. For this study whirl test data of the BHC Model 300 tilt rotor at Wright-Patterson Air Force Base (Reference 9-2) were used for correlation. This rotor is identical to the right-hand rotor of the XV-15.

9.1 EXPERIMENTAL TEST DATA

Figure 9.1-1 is a 1.5 Hz narrow band frequency spectrum of the 25 ft (7.61 m) diameter tilt rotor as measured on the whirl stand. This spectrum is typical of the various test conditions and microphone locations. Rotational sound harmonics are distinguishable beyond the 50th, whereas the presence of the broad band noise component is not obvious. The rate of harmonic decay appears to be somewhat less than conventional rotors. These decay characteristics were assumed to be typical for the larger tilt rotors used in this study and the prediction method was calibrated accordingly.

9.2 PREDICTED NOISE LEVELS

The predicted 500 ft (152 m) sideline perceived noise levels (PNL) of the three 45-passenger point designs are shown in Figure 9.2-1. The variation of PNL with hovering altitude for each configuration is a result of the basic directivity pattern of rotor noise. The increase in tipspeed from the baseline -5 PNdB to the baseline +5 PNdB configuration also affects directivity as indicated by the variation of hovering altitude at which the peak perceived noise level is calculated.

9.3 TYPICAL BELL D312 NOISE CONTOURS AT TAKE-OFF AND LANDING

The Bell D312 point designs have four engines installed and are designed to have one engine-out hover capability at sea level 90^{o}F (32.2^{o}C). The design power loading for the baseline aircraft is 4.94 lbf/hp (29.5 N/kw). The combination of this adequate power loading and the control capability of rotors enabled selection of takeoff and landing contours which provide optimum combinations of pilot work load and noise exposure contours. The relatively complex shapes of the footprints are due to the directivities of the rotor systems as a result of altitude and tip-path-plane angle to the observer.

FIGURE 9.1-1
TILT ROTOR NOISE TEST DATA,
MODEL 300 ROTOR, MARCH 1973

Rotor Thrust/σ'	7840 LBF	(34874 N)
Tip Speed	785 FT/SEC	(239 M/SEC)
Disc Loading/σ'	15.97 LBF/PSF	(764.5 N/M^2)
C_T/σ	0.122	

Noise at 500' 94 PNdB (MIC 7 @ 74 Ft)

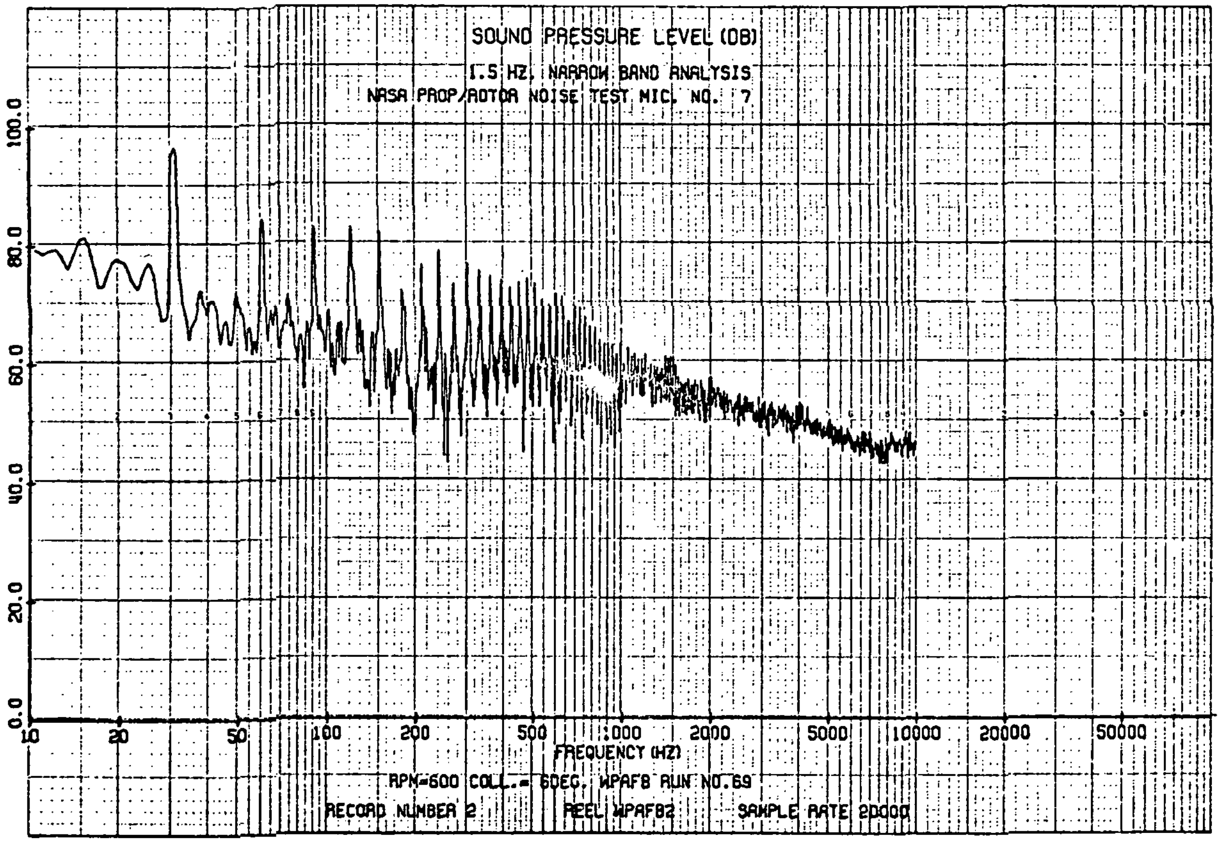
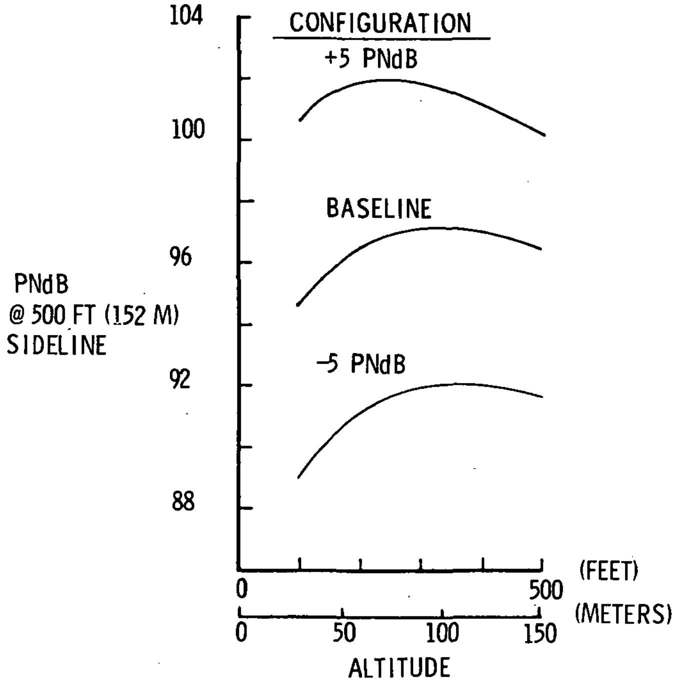


FIGURE 9.2-1
HOVER NOISE VERSUS ALTITUDE
45-PASSENGER AIRCRAFT



9.3.1 TAKE-OFF PROFILE AND NOISE CONTOURS

All three point designs can achieve a 40 knot (74 kph), 1900 fpm (579 m/min) climb as shown in the typical take-off profile of Figure 9.3-1. The climb gradient is 28°, and the aircraft reaches an altitude of 2000 ft (610 m) at a horizontal distance of 4200 ft (1280 M) from the initial vertical take-off. The fuselage attitude during climb is +10.3° (for 75° mast angle), resulting in adequate visibility for the pilot.

Perceived noise footprints and contours for each of the three point designs are shown in Figure 9.3-1. The area within the 95 PNdB contour for the baseline aircraft is estimated to be 49.0 acres (.198 sq km), and the contour could be enclosed in a rectangle 2400 ft (732 m) long by 1200 ft (366 m) wide.

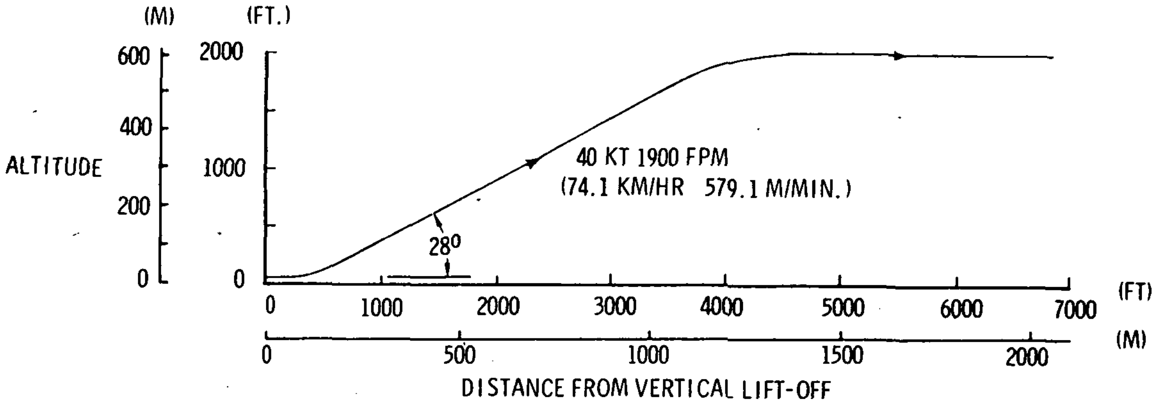
9.3.2 LANDING PROFILE AND NOISE CONTOURS

All three point design aircraft would typically make a two segment approach as shown in Figure 9.3-2. In helicopter mode at an altitude of 2000 ft and a horizontal distance of 6400 ft (1951 m) from touchdown, the pilot would begin a descent rate of 1000 fpm (305 m/min) at a 40-knot speed (13.8° glideslope). At an altitude of 1000 ft (304 m) and 2200 ft (671 m) horizontally from touchdown a gentle cyclic flare reduces speed to 21 knots (39 kph) and steepens the glideslope to 25°. Below 500 ft (152 m), ground speed and rate of descent are gradually reduced to maintain slope. The fuselage attitude is approximately 2° nose-down. Since the cockpit is designed for a downward visibility of 25°, the final hover and touchdown point can be in sight throughout the final approach.

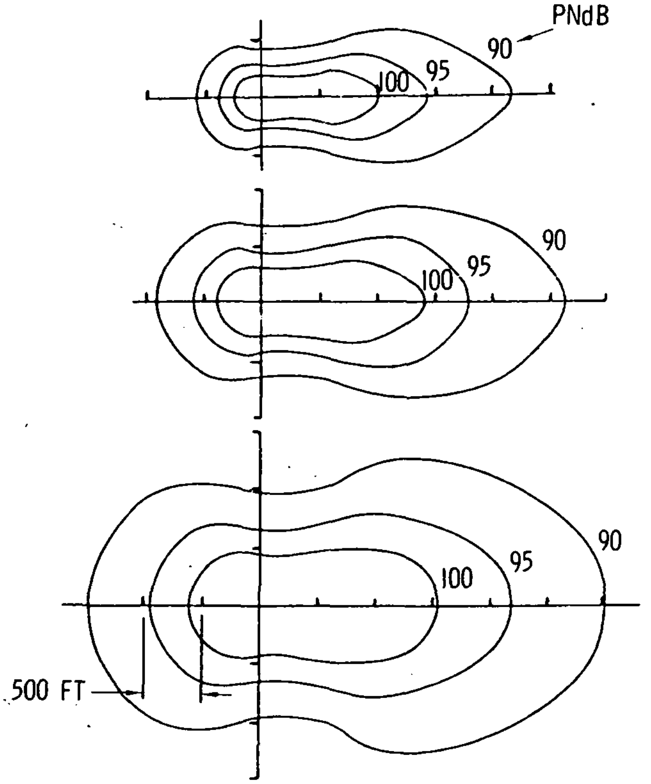
Perceived noise footprints and contours for this landing profile for each of the three point designs are shown in Figure 9.3-2. The area within the 95 PNdB contour for the baseline aircraft is estimated to be 46.5 acres (.188 sq km), and the contour could be enclosed in a rectangle 2200 ft (671 m) long by 1100 ft (335 m) wide.

FIGURE 9.3-1
TAKEOFF PROFILE AND NOISE FOOTPRINTS

TAKEOFF PROFILE



TAKEOFF NOISE FOOTPRINTS
45 - PASSENGER CLASS



CONFIGURATION

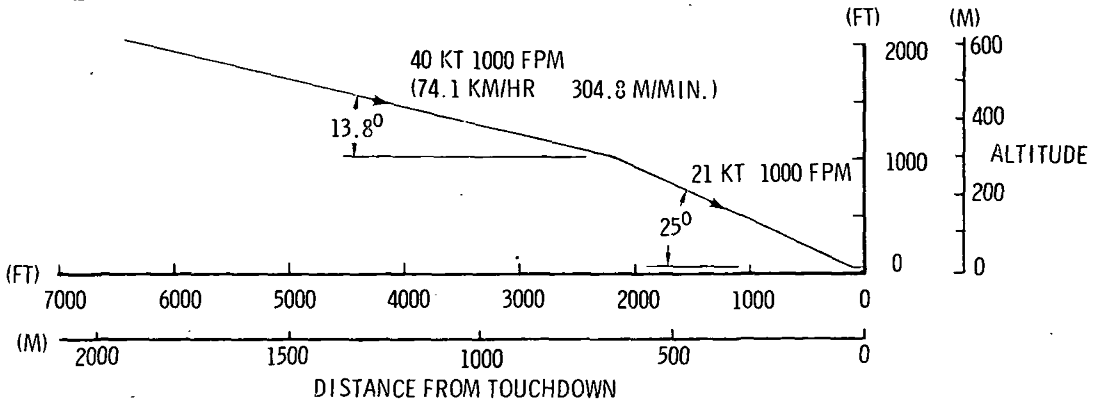
BASELINE -5 PNdB
(ACRES WITHIN 95 PNdB = 23.2)
(= .094 KM^2)

BASELINE
(ACRES WITHIN 95 PNdB = 49.0)
(= .198 KM^2)

BASELINE +5 PNdB
(ACRES WITHIN 95 PNdB = 92.5)
(= .374 KM^2)

FIGURE 9.3-2
LANDING PROFILE AND NOISE FOOTPRINTS

LANDING PROFILE



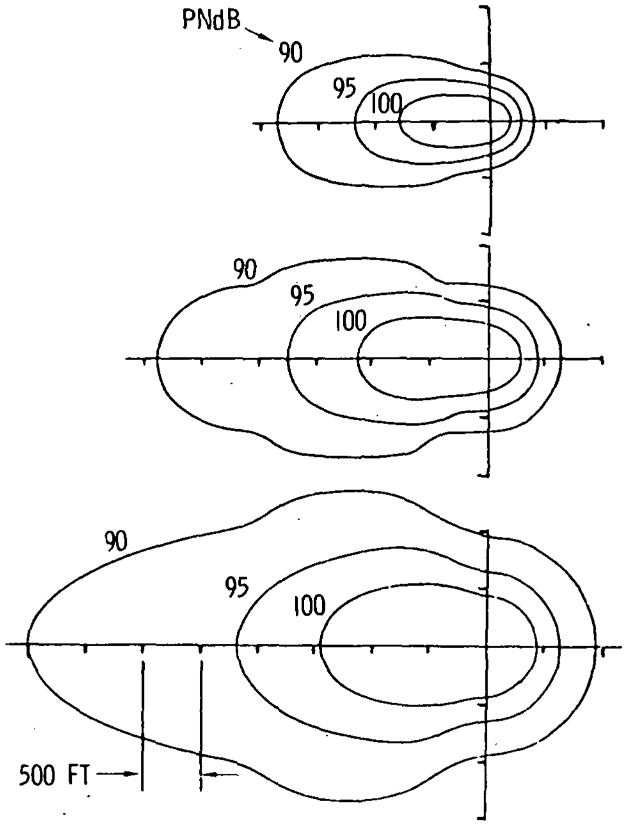
LANDING NOISE FOOTPRINTS
45 - PASSENGER CLASS

CONFIGURATION

BASELINE -5 PNdB
(ACRES WITHIN 95 PNdB = 21.4)
(= .086 KM^2)

BASELINE
(ACRES WITHIN 95 PNdB = 46.5)
(= .188 KM^2)

BASELINE +5 PNdB
(ACRES WITHIN 95 PNdB = 83.6)
(= .338 KM^2)



10. SAFETY ASPECTS

This section covers the safety aspects of one engine-out performance, low speed gust response and critical component redundancy.

10.1 ONE ENGINE INOPERATIVE PERFORMANCE IN THE HELICOPTER MODE

Since the rotors are mechanically interconnected and any engine can drive either rotor, there is no critical engine. Engine-out performance in helicopter mode, Figure 10.1-1, shows the three-engine hover design point at sea level 90°F (32.2°C). Also shown is that the required climb rate, 300 fpm (91.4 m/min) of Reference (10-1), for a four engined aircraft, can be met by three engines at the 30 minute (IRP) rating at all speeds above 25 knots (46 kph). At the recommended climb-out speed of 40 knots (74 kph), a climb rate of 800 fpm (244 m/min) can be achieved. Engine failure on the approach is not critical.

10.2 ONE ENGINE INOPERATIVE PERFORMANCE IN THE AIRPLANE MODE

Figure 10.2-1 shows one engine-out performance in airplane mode for all three point designs. At an altitude of 11,000 ft (3353 m), cruise speeds of 170 knots (315 kph) to over 250 knots (463 kph) are possible.

10.3 LOW SPEED GUST RESPONSE

Aircraft response to four discrete sharp-edged gusts during a typical VTOL initial approach to landing are presented in Figure 10.3-1 (longitudinal gust) and Figure 10.3-2 (lateral gust) for the baseline design point. These horizontal gusts are of 15 fps (4.6 m/sec) amplitude for a duration of 5 seconds, originating laterally from the left and right, and longitudinally from the forward and aft directions, in the earth-based coordinate system. The aircraft is initially trimmed (at the 2 second point) in a 25 knot steady-state crosswind from the right with a 1000 fpm (305 m/min) descent rate and a 40 knot forward (ground reference) speed with flaps 40/25° and gear up. This descent condition is the same as used in the landing profile of Section 9. No corrective action by the pilot nor any Stability and Control Augmentation System (SCAS) inputs are present in these time history analyses. All of the longitudinal and lateral/directional stability modes are stable in this configuration with the exception of the spiral mode which has a time to double-amplitude of 32 seconds. In each case it can be seen that the basic aircraft contains sufficient attitude and velocity damping to continue sustained flight without SCAS or pilot corrective action during the gust duration. Following the removal of the gust, some

FIGURE 10.1-1
CLIMB PERFORMANCE, ONE-ENGINE INOPERATIVE,
45-PASSENGER BASELINE AIRCRAFT

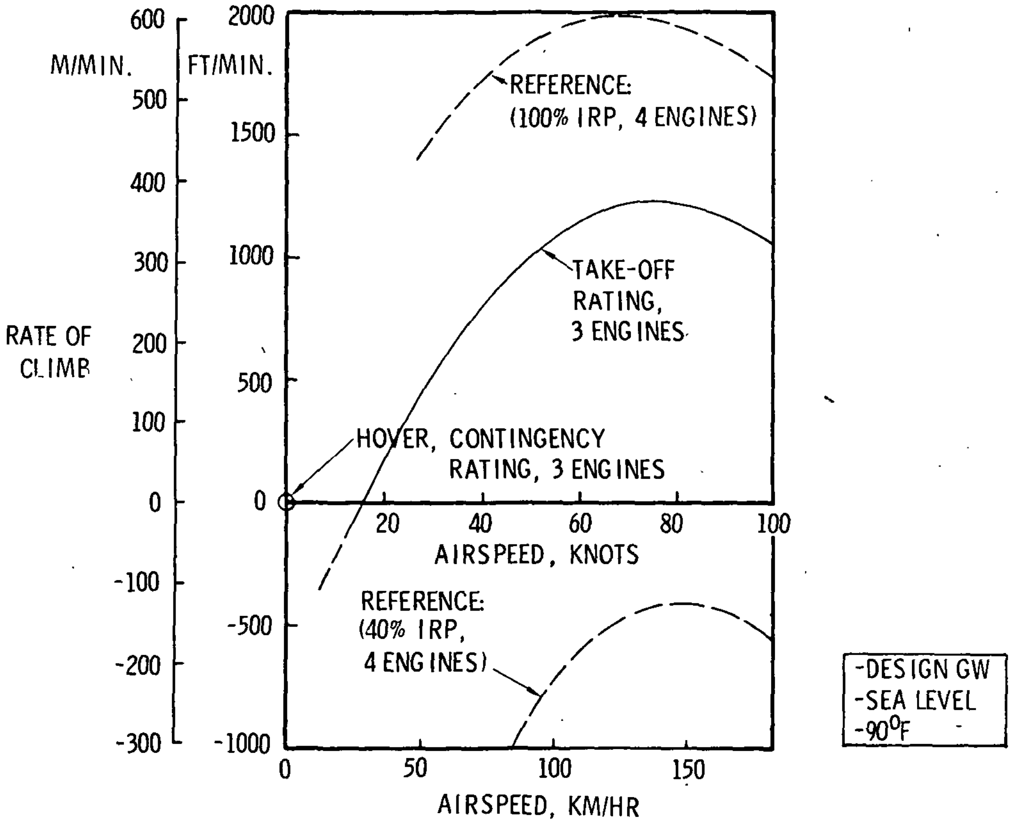


FIGURE 10.2-1
CRUISE PERFORMANCE, ONE-ENGINE INOPERATIVE,
45-PASSENGER AIRCRAFT

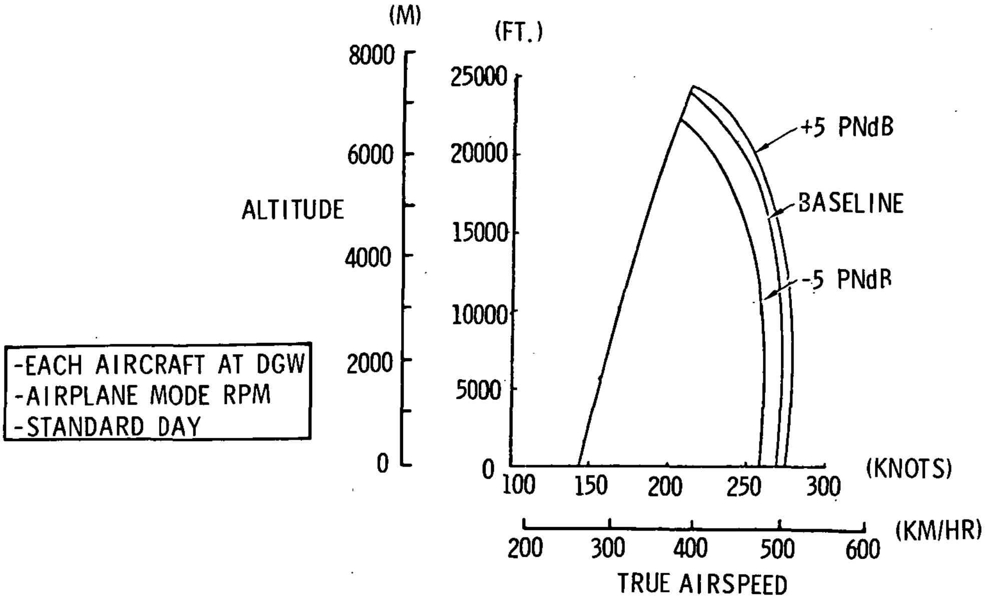


FIGURE 10.3-1
RESPONSE TO LONGITUDINAL GUSTS, HELICOPTER MODE,
45-PASSENGER BASELINE AIRCRAFT

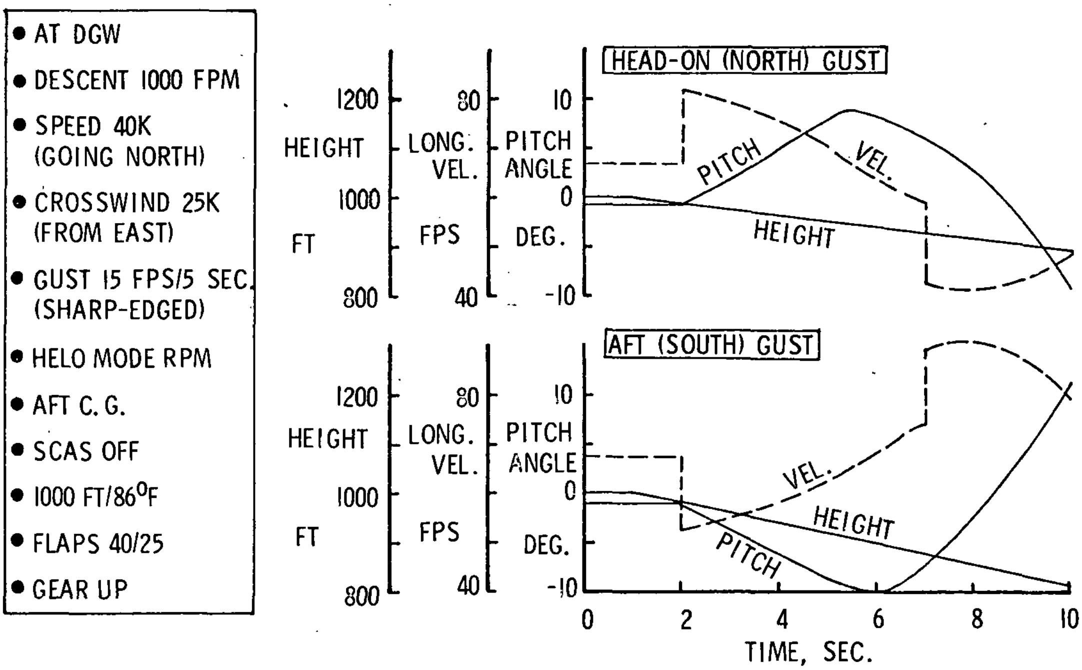
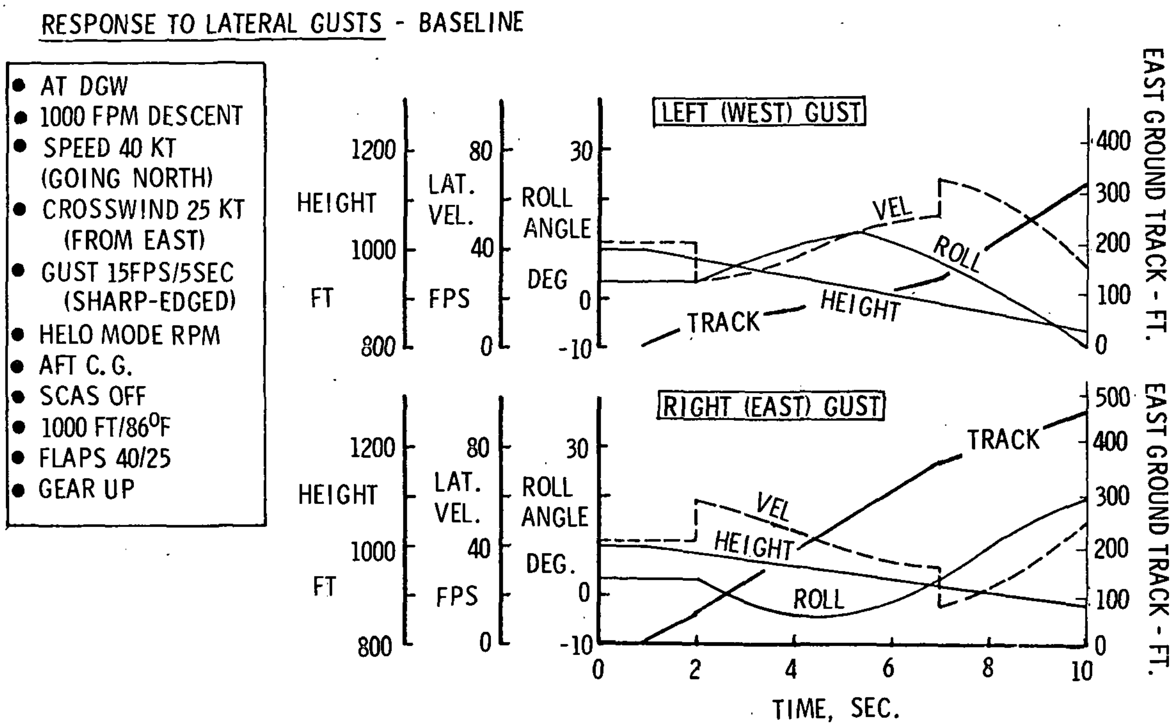


FIGURE 10.3-2
RESPONSE TO LATERAL GUSTS, HELICOPTER MODE
45-PASSENGER BASELINE AIRCRAFT



corrective action appears necessary to eliminate excessive pitch or roll attitudes. Additional analyses at slower forward ground speeds closer to the touchdown point, with full flaps (75/45°) and gear down, indicates that SCAS would be required to maintain continued flight path equilibrium following a gust disturbance. SCAS would also be necessary during a low speed, high angle, 2000 fpm (610 m/min) climb following takeoff in order to remain stable during a gust input. It is recommended that these effects be systematically investigated with pilot-in-the-loop flight simulation using the tilt rotor math modeling available on NASA-AMES simulators.

10.4 GENERAL SAFETY CHARACTERISTICS

The two low disc loading rotors provide autorotation capability for a reduced descent rate emergency landing in case of fuel exhaustion or total loss of power. Adequate collective pitch range and rotor solidity (total blade planform) permit rotor speed control during descent and provide flare thrust to reduce rate-of-sink. The landing gear is designed to withstand a vertical sink rate of 10 fps at the design gross weight.

The rotors are driven by wingtip mounted turbine engines. An interconnecting shaft system between the rotors (cross-shafting) allows any engine to power both rotors in the event of an engine or engine gearing failure. Driving each of the rotors independently is also possible in the case of a cross-shaft failure. Rotor desynchronization due to a cross-shaft failure will not cause rotor intermeshing problems (as on some tandem helicopers) because the rotors do not overlap.

Overrunning clutches in the engine reduction-gearing automatically disconnect a failed engine from the drive system, thus allowing the effective use of available power. Redundant transmission housing mounting-lugs prevent a catastrophic single bolt or lug failure. The drive system strength requirements allow for uneven power distribution (such as a double engine failure on one side) and maneuver or gust transient loads and torques. For normal operation, torque limitations will be placarded and are a pilot-control function.

The Bell stiff-in-plane proprotor design philosophy, as used for the XV-15, is considered to be a major design parameter to ensure flight safety. With an inherently stable dynamic system, the failure of the stability and control augmentation or a gust alleviation feedback-system will not lead to a catastrophic instability.

The conversion (nacelle tilt) mechanism is provided with dual hydraulic actuation and redundant control subsystems to enable full range operation after any single failure. In the event of complete hydraulic failure, the nacelles can be converted slowly by the use of an electrically powered drive system. A nacelle synchronization feature is also provided.

Three separate hydraulic systems would be typically installed in a four engine transport; two primary flight control systems and a utility system. The primary systems would be powered by a hydraulic pump driven from each main-rotor transmission. The utility system would be powered by a hydraulic pump driven from the interconnect shaft, adjacent to the fuselage so that hydraulic power is available as long as the rotors are rotating. In addition, the auxiliary power unit (APU) and the electrical system drive additional pumps which would power the primary and utility systems for ground checkout, and as desired by the pilot in flight.

Critical components of the separate systems will be physically isolated, where possible, to prevent concurrent failure due to local damage. The flight controls will be irreversible and include a force-feel and a stability and control augmentation system. Controls that are not safety-of-flight items may be powered by single actuators. Built-in test equipment (BITE) will be provided. Fire resistant hydraulic fluid will be used to reduce the fire potential of the hydraulic system.

The electrical system follows the same design approach as for the hydraulics; three completely independent systems, of which one generator is driven by each rotor transmission and the remaining generator by the interconnect shaft. In addition the APU and the batteries provide electrical power on the ground and as desired by the pilot in flight. Adequate electrical power for the critical flight-required equipment will be available after the loss of any two of the elctrical systems.

An engine fire detection and pilot actuated fire extinguishing system will be incorporated. Engine inlet icing detection and anti-icing is also provided. Fuel is stored in the wings, outboard of the fuselage, in integral spray-in cells. Breakaway fittings are utilized to eliminate fuel spillage from fuel lines separated in a crash. The remote location of the engines from the fuselage reduce the hazard, to the passengers and crew, of engine fire and the resulting smoke and heat.

Nose gear swiveling and differential braking are provided for ground operation. For the 45-passenger aircraft, the rotor disc in the VTOL takeoff configuration will be over twenty feet above ground level at the design gross weight. The crew members will have an unobstructed view of the out-board rotor tippath to reduce the hazard of rotor tip collision with ground objects during taxi or ground maneuvering.

Flight operation will display safety characteristics similar to helicopters or conventional aircraft. High hover mode thrust weight ratios coupled with control powers and sensitivities greater than the minimum levels recommended in AGARD Report No. 577 will permit hover, in and out of ground effect, with adequate control about all axes.

Transition to cruise flight is performed within the boundaries established by wing stall, the torque limit, or rotor/hub endurance limits. The allowable corridor is broad (generally greater than 80 knots).

The general flight characteristics in cruise are those of a turboprop airplane. Conventional aircraft control surfaces are employed.

A pilot caution and warning system will provide visual and/or audible indications of detectable system malfunctions, such as hydraulic system pressure loss, rotor control discrepancies, engine fire, etc.

11. CONCLUSIONS

A conceptual design study of 1985 commercial tilt rotor transports, based on the NASA VTOL mission, has been completed. The conclusions are as follows:

1. No technical limit on the size of tilt rotor aircraft was identified in this study. For reference, the 100-passenger candidate baseline point design has a sideline noise level in hover of 99 PNdB. The direct operating cost is 3.01 ¢/assm (1.87 ¢/askm) at a utilization of 2500 hr/yr and mission fuel consumed is 35.44 lbf (157.6 N) per available seat.

2. Based on the study ground rules and predicted characteristics of the point designs generated in this study, the largest size commercial tilt rotor transport that would be feasible and practical if fabrication would begin in 1980 has a capacity of 45 passengers.

3. The selected baseline 45-passenger point design has a predicted sideline noise level in hover of approximately 97 PNdB. The area enclosed by the 95 PNdB footprint contour is 49 acres (0.198 sq km) during takeoff and 5% less during landing. The direct operating cost is 4.66 ¢/assm (2.90 ¢/askm) at 2500 hr/yr utilization and mission fuel consumed is 44.77 lbf (199.1 N) per available seat.

4. The -5 PNdB and +5 PNdB 45-passenger point designs have areas enclosed by the 95 PNdB footprint contours of 23.2 acres (0.094 sq km) and 92.5 acres (0.374 sq km) respectively, during takeoff. The direct operating costs are 5.24 ¢/assm (3.26 ¢/askm) and 4.52 ¢/assm (2.81 ¢/askm), and fuel quantities consumed are 49.33 lbf (219.4 N) and 42.49 lbf (189 N) per available seat, respectively.

5. Achieving the predicted characteristics of the baseline point design is dependent on the applicable technology programs taking place in the 1975-1979 time period. These include tilt rotor flight simulation, flight research with the XV-15, and advanced technology components.

6. The strength/density ratio of the current technology, steel and titanium rotor of the XV-15 meets the rotor weight fraction predicted for the 45-passenger aircraft but not that of the 100-passenger design. An advanced design composite rotor is identified as an important component to increase productivity in the performance, weight and service life, areas of the commercial transport. An

advanced composite wing, with its unique strength and stiffness requirements, is identified as another important component for meeting the airframe component weight fractions (25% reduction) assumed in this study.

7. These advanced components, scaled to preserve the technological factors for the 1985 transport, should be planned for flight research on the XV-15 to demonstrate that the required technology is in hand by 1979.

8. For the aircraft size selected no new engine development is required and no unusual drive system characteristics were identified.

APPENDIX - ADDITIONAL YAW CONTROL CHARACTERISTICS

GENERAL

This section presents the results of analyses of yaw control power characteristics of the three point designs (45-passenger baseline configuration and the -5 PNdB and +5 PNdB versions) between the helicopter and airplane modes of flight. The results are compared with the Design Criteria and areas are identified where the yaw control power are inadequate to meet the criteria. If the criteria are intended to govern the takeoff and landing phases of a mission, then they are not directly applicable to the conversion mode since the tilt rotor aircraft normally lands in the helicopter configuration where the yaw control is adequate. For this reason, and for comparisons with other concepts in the flight regime between landing and cruise configurations, the results have been compiled in this appendix.

RESULTS

Yaw control power (obtained from differential longitudinal cyclic pitch and rudder) during crosswinds in the conversion mode resulted in the most critical combination for satisfying all the Level 1 attitude control power requirements. Figure A-1 shows that in helicopter mode the remaining directional control moment following trim are sufficient from hover to 120 knots (222 kph) with a 25 knot (46 kph) crosswind as discussed previously. However, as the rotors are tilted the differential F/A cyclic is gradually reduced to zero in airplane mode, and, therefore, until sufficient dynamic pressure is obtained on the vertical stabilizer and rudder, the remaining yaw moment following trim is insufficient to meet the minimum Level 1 acceleration after a step input of the pedals to the nearest stop. The minimum speed (based on the Design Criteria for flying below V_{CON} in a crosswind) as a function of conversion angle for each design point is shown in Figure A-2.

This flight condition, although critical from the standpoint of comparison to the Study Guidelines and Design Criteria, is not considered critical from the operational standpoint of a tilt rotor in that during airplane and conversion mode flight with a crosswind, yawed flight into the wind would be more feasible than a sideslipped condition that may be necessary in a helicopter mode approach. Simultaneous control inputs at these speeds are more than adequate for meeting the 100% yaw moment plus 30% pitch and roll moment occurring simultaneously due to the moderate amount of roll-yaw cross coupling that exists in conversion mode with a crosswind.

RECOMMENDATION

A supplementary investigation is in order to reconcile the yaw characteristics in the conversion mode with the Design Criteria. Flight simulation studies with the existing tilt rotor math models could evaluate the net handling qualities characteristics (i.e., in the presence of roll-yaw cross coupling) for realistic tasks in the conversion mode. This could be accompanied by analysis of the variation of yaw acceleration capability in this mode with variations in key tilt rotor design variables such as: C_T/σ, wing loading, tail volume coefficients, and/or control rigging. It is possible that minor adjustments in some of the design parameters used in this study would help considerably in the reconciliation process. Specific areas would be identified for flight research verification with the XV-15.

FIGURE A-1
YAW CONTROL POWER IN CROSS WIND
45-PASSENGER AIRCRAFT

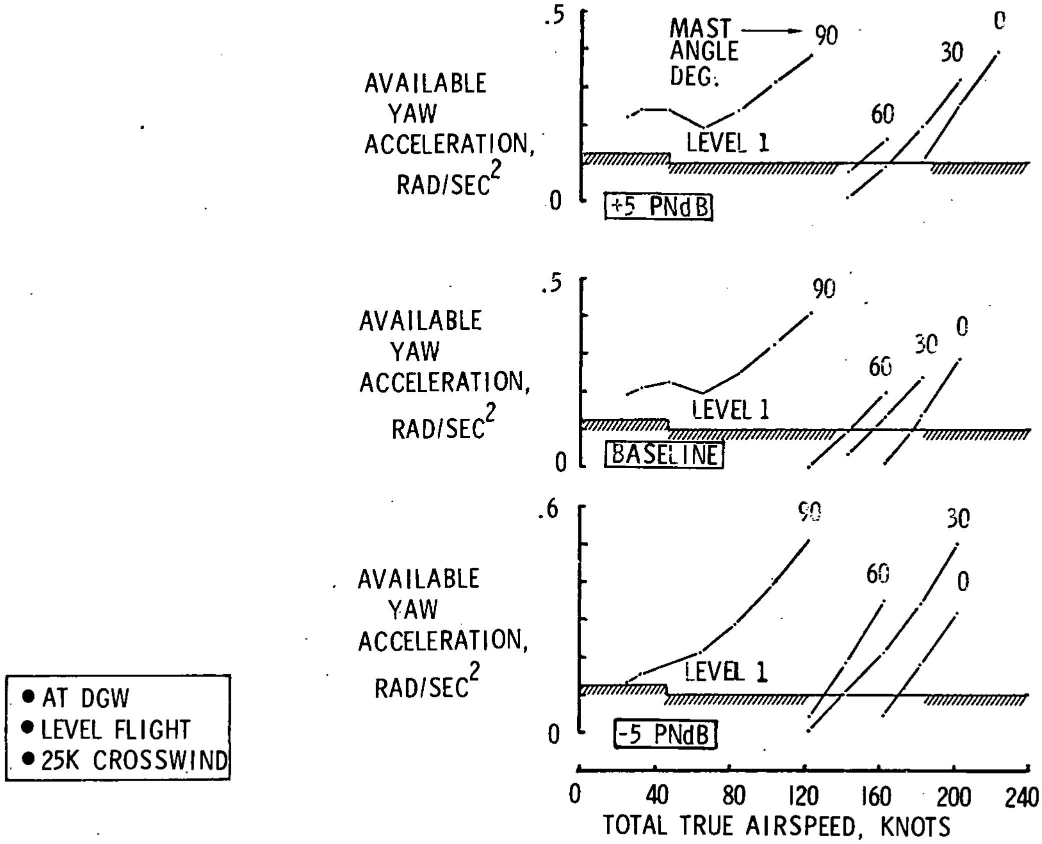
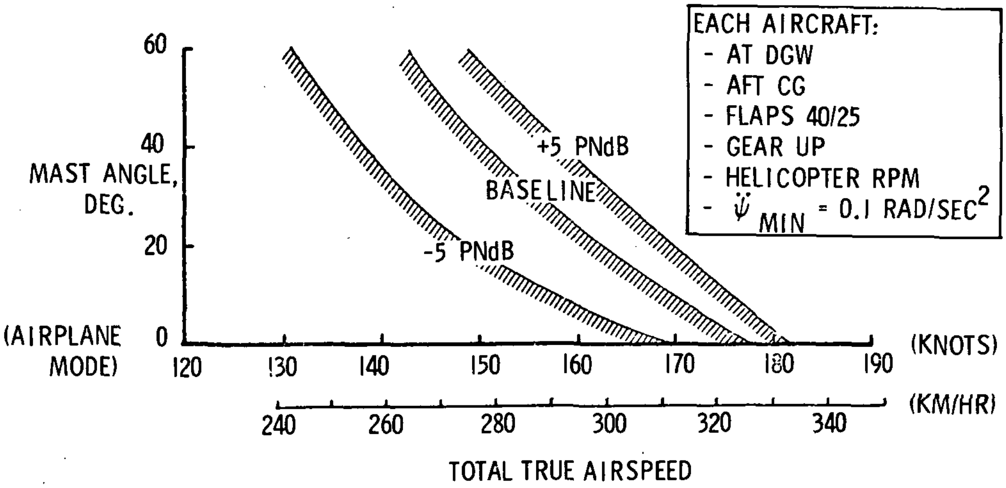


FIGURE A-2
YAW CONTROL POWER, MINIMUM SPEEDS
45-PASSENGER CLASS AIRCRAFT

MINIMUM SPEED FOR MEETING LEVEL 1 IN CONVERSION WITH 25 - KNOT CROSSWIND (46 KM/HR)



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