MCDONNELL DOUGLAS

CORPORATION

Copy number

NASA CR-151931
Report number MDC A4551

LIFT/CRUISE FAN V/STOL
TECHNOLOGY AIRCRAFT
DESIGN DEFINITION STUDY

VOLUME I
TECHNOLOGY FLIGHT VEHICLE DEFINITION

Revision date

Revision letter

Issue date 18 November 1976

Contract number NAS2-9245

Prepared by Wm. J. O'Brien

MCDONNELL AIRCRAFT COMPANY

Box 516, Saint Louis, Missouri 63166 – Tel. (314)232–0232

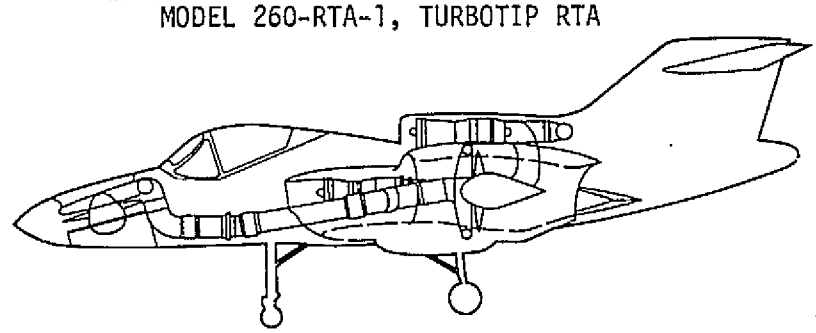
MCDONNELL DOUGLAS
CORPORATION

SUMMARY

This report presents results of Tasks I and II of a study by McDonnell Aircraft Company (MCAIR) for NASA Ames Research Center and the U.S. Navy to conceptually design two types of Lift/Cruise Fan Technology V/STOL Aircraft. The objective was to define two Research Technology Aircraft (RTA) - one using turbotip fans and the other using mechanically driven fans. A low risk, low cost philosophy was used during the design of both aircraft. The approach used was to refine the RTA designs previously accomplished under NASA Contract No. NAS2-5499 and reported in Report MDC A3440, Volume II.

The turbotip RTA reflects maximum usage of existing airframe components, i.e., T-39 wing and center fuselage, F-101 aft fuselage and empennage, A-6 cockpit and canopy, and A-4 landing gear The propulsion system consists of three General Electric (GE) LCF459 turbotip fans pneumatically interconnected to three GE YJ97 gas generators. Thrust modulation is accomplished by use of the MCAIR developed Energy Transfer and Control System and Thrust Reduction Modulation. This system can also be operated in the two engine/three fan mode. A large thrust to weight margin is available for both normal and emergency operation which equates to low risk. Engine-out VTO safety for the maximum VTOGW is available at a dry intermediate power setting.

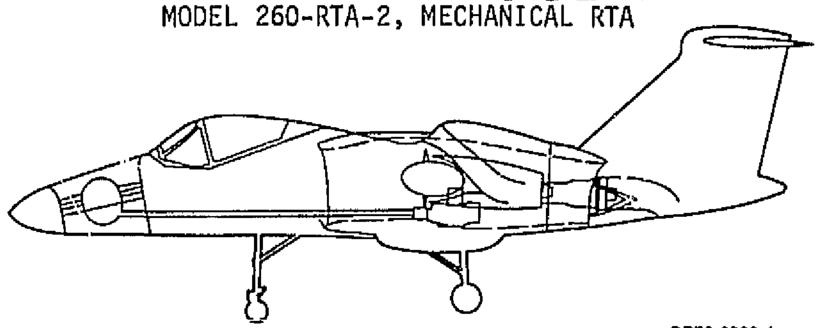
MODEL 260-RTA-1, TURBOTIP RTA



The mechanical RTA is virtually identical to the turbotip RTA with the exceptions that a different propulsion system and aft fuselage/tail are used. The T-39 aft fuselage and vertical tail are modified and a new horizontal tail is used instead of the F-101 assembly to achieve a weight reduction of approximately 544 pounds. This weight decrease was required so that engine-out vertical landing requirements could be met. The propulsion system consists

of three Hamilton Standard 62 inch diameter variable pitch fans mechanically interconnected to three Detroit Diesel Allison XT701 turboshaft engines. This aircraft was initially designed using 65 inch diameter fans; however, subsequent increases in thrust ratings of the 62 inch diameter fan provided adequate lift margins and it was incorporated into the design. The engines use a wet rating to achieve engine-out lift requirements.

MODEL 260-RTA-2, MECHANICAL RTA



GP76-0893-1

Both aircraft meet or exceed all of the mission performance guidelines specified in the Statement of Work and reflect a low cost, low risk approach. Additional STO mission performance is available at a reduced load factor, thereby providing operational mission demonstration capability. Budgetary estimates were prepared for each of the aircraft based on an austere development and flight test program and are presented in Volume III of this report.

TABLE OF CONTENTS

TABLE OF CONTENTS (Cont'd)

LIST OF FIGURES

LIST OF FIGURES (Cont'd)

LIST OF FIGURES (Cont'd)

LIST OF FIGURES (Cont'd)

LIST OF PAGES

Title

ii through xii
1 through 145
A-1 through A-17
B-1 through B-31
C-1 through C-29

SYMBOLS AND ABBREVIATIONS

ACS	active control system
AR	aspect ratio
a/g	acceleration/gravitational constant
BOT	burner out temperature
BTU	British thermal units
C_D	drag coefficient
C_{fg}	nozzle thrust coefficient
C.G.	center of gravity
C_L	lift coefficient
CNI	communication navigation identification
CPR	compressor pressure ratio
CSD	constant speed drive
deg.	degrees
ETaC	Energy Transfer and Control
EGP	exhaust gas pressure
EGT	exhaust gas temperature
f	drag area
ft	feet
F_N	net thrust
FPR	fan pressure ratio
g	gravitational constant, 32.2 ft/sec^2
GFE	Government Furnished Equipment
GG	gas generator
HP	horsepower
HPI	ideal horsepower
in.	inches
KEAS	equivalent airspeed, knots
kt	knots
L	lift, lb
L/C	lift/cruise
LGW	landing gross weight, lb
m	meters
M	Mach number
MAC	mean aerodynamic chord
N	Newtons

N_F	fan speed, rpm
N_{GG}	gas generator speed, rpm
NP	neutral point
nm	nautical miles
P_a	ambient pressure, psi
P_s	static pressure, psi
PCM	pulse control modulation
PR	pressure ratio
RTA	Research Technology Aircraft
S	area, ft^2
SCM	signal conversion mechanism
S.L.	sea level
S.M.	stall margin
STO	short takeoff
STOGW	short takeoff gross weight, lb
STOW	short takeoff weight
T	thrust
T/C	thrust center
t/c	airfoil thickness ratio
T/W	thrust/weight
TIT	turbine inlet temperature
T.O.	takeoff
TOS	time on station
TOGW	takeoff gross weight
TR	transformer-rectifier
TRM	thrust reduction modulation
V	velocity, vertical
VL	vertical landing
VLGW	vertical landing gross weight
V/STOL	vertical/short takeoff and landing
VTO	vertical takeoff
VTOGW	vertical takeoff gross weight, lb
W/S	weight/area, lb/ft^2
W_a	airflow, lb/sec
W_f	fuel flow, lb/hr
W_g	gas flow, lb/sec

α	angle of attack, deg.
β	fan blade angle, deg.
δ_F	flap angle, deg.
λ	taper ratio
$\Lambda_{c/4}$	sweep angle of quarter chord, deg.
θ_B	body angle, deg
θ_J	nozzle deflection angle, deg.

1. INTRODUCTION

Recent studies by the Navy and NASA have confirmed the future need for a high performance V/STOL aircraft for both military and civil applications. The Navy requires a multimission V/STOL aircraft in the 1980's capable of sea control operations from many platforms as well as ship-to-shore and shore-to-ship functions. The purpose of this study was to define two lift cruise fan Research Technology Aircraft (RTA) which can be used for propulsion/control system operational demonstration. The specific objectives of the study may be summarized as follows:

Task I: Conceptually design a turbotip lift cruise fan RTA

Task II: Conceptually design a mechanical lift cruise fan RTA

Task III: Design and analyze the transmission system components of the turbotip and mechanical RTA

Task IV: Evaluate program variants for cost reduction

The results of this study are reported in the following three volumes:

Volume I - Technology Flight Vehicle Definition

Volume II - Propulsion Transmission System Design

Volume III - Development Program and Budgetary Estimates

This volume defines the turbotip and mechanical RTA programs proposed to assess the benefits of the lift cruise fan V/STOL concept. The major test objectives of the technology aircraft program are to:

- Develop an integrated propulsion/control system for a V/STOL aircraft
- Evaluate this concept in powered lift and aerodynamic flight regimes
- Exploit the benefits of the lift/cruise fan system
- Define future V/STOL aircraft design requirements
- Obtain operational experience
- Develop operating techniques
- Serve as a facility for control/propulsion system tests
- Provide the capability to perform experiments related to terminal area operation with advanced stabilization, guidance, and navigation systems.

In accordance with the Statement of Work, the design definition study was directed toward a minimum cost research program consistent with providing maximum research productivity and proper attention to safety. The specified Design Guidelines are presented in Appendix A of the report.

The aerodynamic data and mission capabilities for the aircraft were determined and compared to the design guideline requirements and are presented in Section 2. The description and performance characteristics of the two propulsion systems are presented in Section 3. Section 4 summarizes the vehicle subsystem design characteristics, weight analysis and data base summary. An analysis of the aircraft control and handling qualities was performed for each aircraft and is summarized in Section 5. Excess control margins are provided in all axes for research purposes.

2. AERODYNAMICS

2.1 REQUIREMENTS AND APPROACH

The modified T-39 Sabreliner research and technology conceptual aircraft design, described in Report MDC A3440 was refined to achieve minimum risk and cost while retaining maximum R&D demonstration capabilities with engine out safety. Basic design guidelines and criteria for the design definition of the lift cruise fan research and technology V/STOL aircraft were specified by Attachment I of the Statement of Work wh :h is presented in Appendix A. Mission requirements, flight safety and operating criteria, handling qualities and engine out standards are prescribed. The aircraft are to be considered in the Class II category of MIL-F-83300. Level 1 handling qualities are to be provided for normal operation with no failures. Level 2 handling qualities are to be provided with a single reasonable failure of propulsion or control system. The cruise flight static stability margin without augmentation was defined to be 5-percent mean aerodyanmic chord at the critical center of gravity.

Two primary inputs to V/STOL aircraft safety and cost are the number of engines and the control system; the latter is discussed in Section 5. Specification of the modified Sabreliner design plus the J97 and PD370-24 propulsion systems including fan diameters established the aircraft gross weight and static thrust performance for both two and three engine configurations. The most critical engine-out criteria of the guidelines was the VL requirement with a T/W = 1.03 and the conversion requirement which states the maximum speed in the powered lift configuration shall be at least 10-percent greater than the power off stall speed in the converted configuration for Level 2 operation. These specifications and the previous RTA study resulted in the selection of a three engine propulsion system which provides the desired performance and safety levels. The aircraft operational gross weights are restricted to levels established by engine-out capability.

STO performance at gross weights greater than VTOGW is a function of the installed thrust level as defined by the design thrust/VTOGW ratio (T/W = 1.05). An RTA VTOGW selection based upon approximately two engine available thrust provides the desired safety and performance levels throughout the powered lift flight regime. The STO, transition, and conversion can be completed in the event of engine failure since the thrsut/weight ratios are maintained at levels equivalent to a two engine configuration. The technology aircraft are operated with three engines at part throttle during powered lift flight and with two engines during aerodynamic flight. The throttle is advanced to maintain thrust level in event of engine failure.

The lift/cruise fan aircraft configurations selected for the turbotip and mechanical fan technology aircraft are the culmination of extensive R&D efforts including wind tunnel test programs. Three interconnected fans are spaced longitudinally and laterally to maximize control, to provide symmetrical lift following an engine failure, and to compensate for suckdown while in ground effects. Lift/cruise fans are positioned on top of a low wing at the fuselage-wing root juncture to provide power induced lift in STO and transition, to reduce thrust trim moments, and to minimize the V/STOL structural penalty; i.e., the basic wing structure is maintained intact. A T-tail provides optimum stability and control contributions over the operational angles of attack, retains adequate control power at post-stall angles of attack, and minimizes stabilator trim changes with thrust vectoring.

Requisites for V/STOL aircraft configuration viability are compatible locations for aircraft thrust center (TC), weight center (CG), and aircraft neutral point (NP). A coincident TC and CG location forward of the NP is required to minimize the weight penalties associated with control provisions for all operating flight modes and to provide a static stability margin in the aerodynamic lift flight mode. The thrust center and most aft center of gravity of the selected configurations are established at 28 percent of the wing mean aerodynamic chord and the center of gravity travel with fuel use is minimized. The aircraft neutral point is positioned at 33 percent MAC by sizing the stabilator using wind tunnel test results.

2.2 BASIC AERODYNAMIC DATA

The basic aerodynamic data required for performance calculations were obtained through use of advanced design techniques, MCAIR lift/cruise fan aircraft technology base, technical data reports pertaining to existing aircraft components, and wind tunnel tests.

Mission performance capabilities of the technology aircraft are based on the estimated drag characteristics presented in Figures 2-1, 2-2 and 2-3. The minimum parasite drag consists of component skin friction drag modified for shape, roughness, and interference plus incremental drag for appendages and trim. Incremental appendage and trim drags are based on previous lift/cruise fan aircraft R&D efforts. The mechanical RTA exhibits the lowest aerodynamic minimum drag since the cruise engines are integral with the fans (part of

FIGURE 2-1
RTA PARASITE DRAG BREAKDOWN

AIRCRAFT COMPONENT	TURBOTIP RTA Δf $(FT)^2$	MECHANICAL RTA Δf $(FT)^2$
Fuselage + Engines	4.21	2.73
Wing	1.92	1.97
Nacelles	3.42	3.60
Stabilator	.64	.72
Fin	.60	.61
Trim + Appendages	1.29	1.29
Total	12.08	10.92
Wing Area $(Ft)^2$	342	342
Minimum C_D	0.0353	0.0320

FIGURE 2-2
TURBOTIP RTA
TRIMMED DRAG POLAR

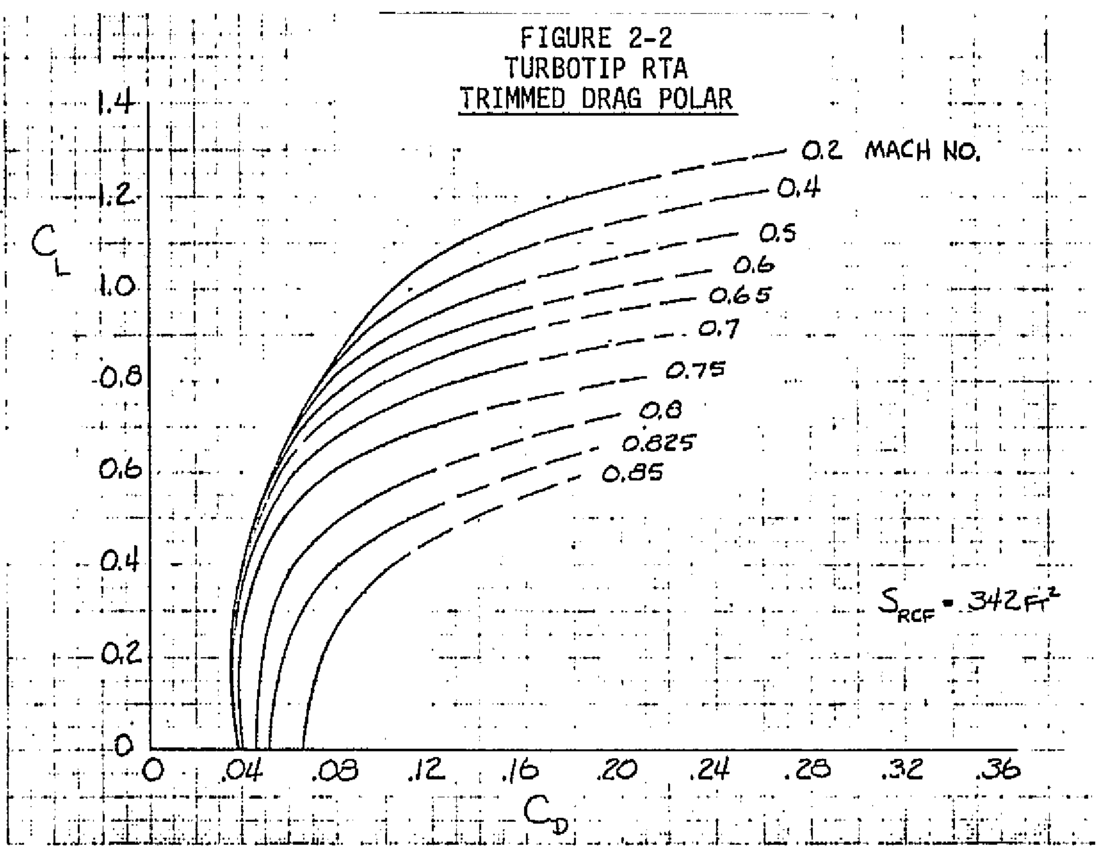
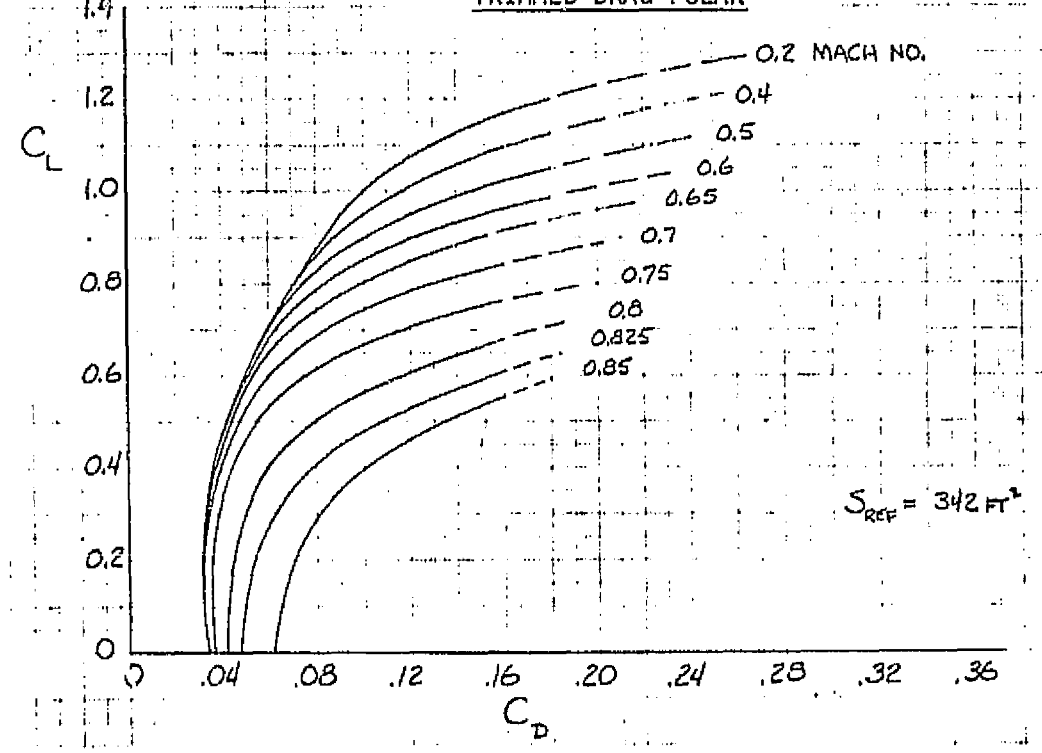


FIGURE 2-3
MECHANICAL RTA
TRIMMED DRAG POLAR



propulsion internal drag) rather than enclosed in fairings aside of the fuselage as for the gas drive version. The lift coefficient at minimum drag (C_{L_b}) as determined from model test is approximately 0.15. Figures 2-4 and 2-5 show the trimmed drag polars for the turbotip and mechanical fan aircraft at various flight Mach numbers. The drag polars are supplemented by basic aero-dyanmic data, Figures 2-6 through 2-9, prepared in previous NASA/Navy Design Definition Studies. These estimated aerodynamic data assume the Sabreliner wing leading edge slat is locked and sealed in its retracted position. For the high lift configuration estimated data, the Sabreliner flap is assumed modified to a plain flap of reduced span.

2.3 PERFORMANCE

2.3.1 TAKEOFF AND LANDING - The definitions for determining the RTA gross weights for VTOL, STOL and emergency engine-out conditions were established by the guidelines and by separate inputs from NASA.

VTOGW - Gross weight with fuel for a (5) circuit test mission, 2500 lb payload; thrust = 1.05 VTOGW.

STOGW - Gross weight with fuel for an (11) circuit test mission, 2500 lb payload.

FIGURE 2-4
TURBOTIP AND MECHANICAL RTA
TRIMMED LIFT COEFFICIENT

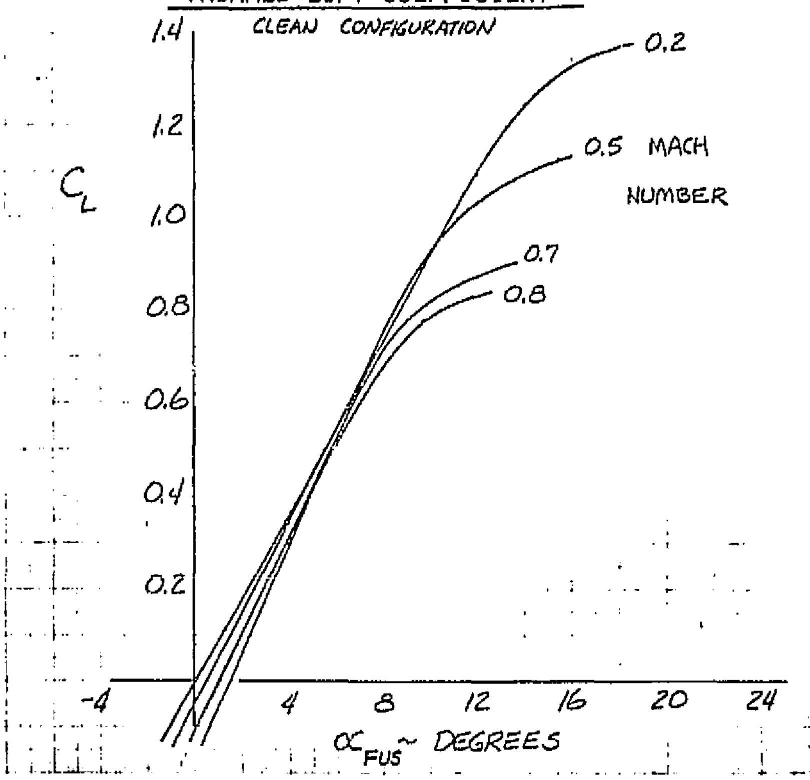


FIGURE 2-5
TURBOTIP AND MECHANICAL RTA
MAXIMUM USABLE LIFT COEFFICIENT

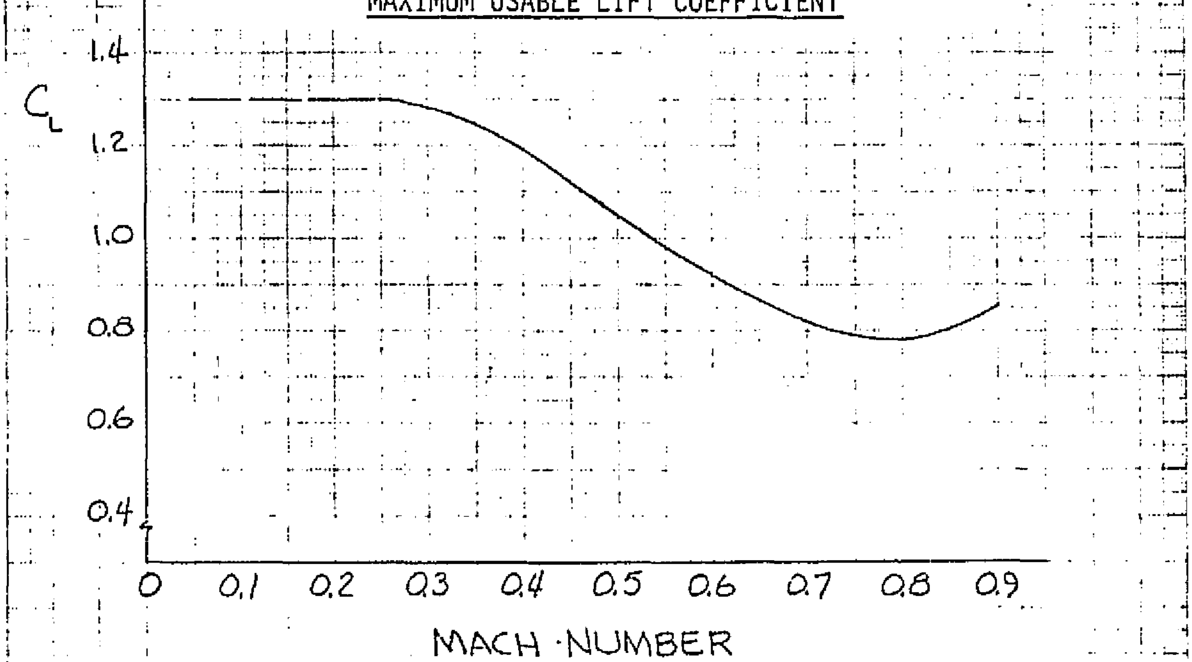


FIGURE 2-6
TURBOTIP RTA
MINIMUM DRAG

$S_{REF} = 342\ FT^2$

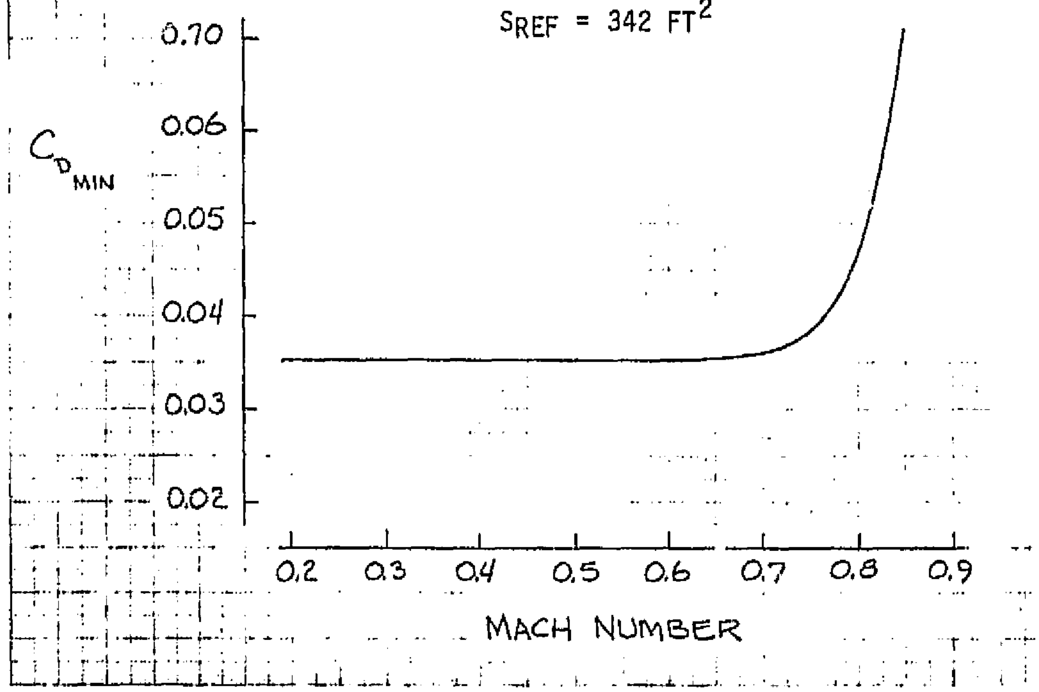
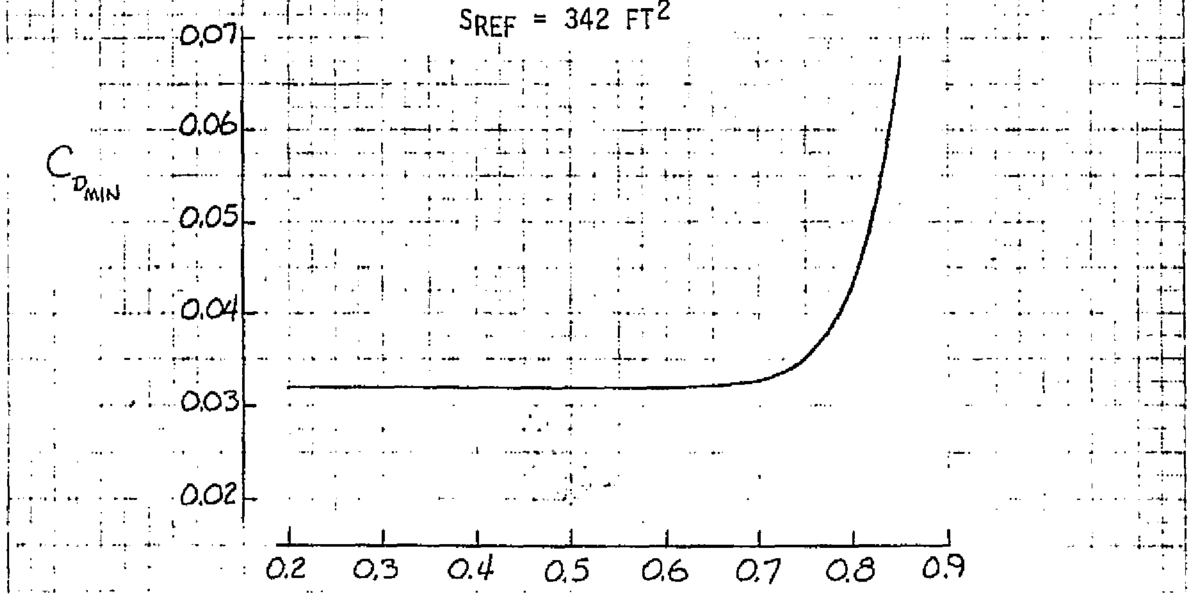


FIGURE 2-7
MECHANICAL RTA
MINIMUM DRAG

$S_{REF} = 342\ FT^2$



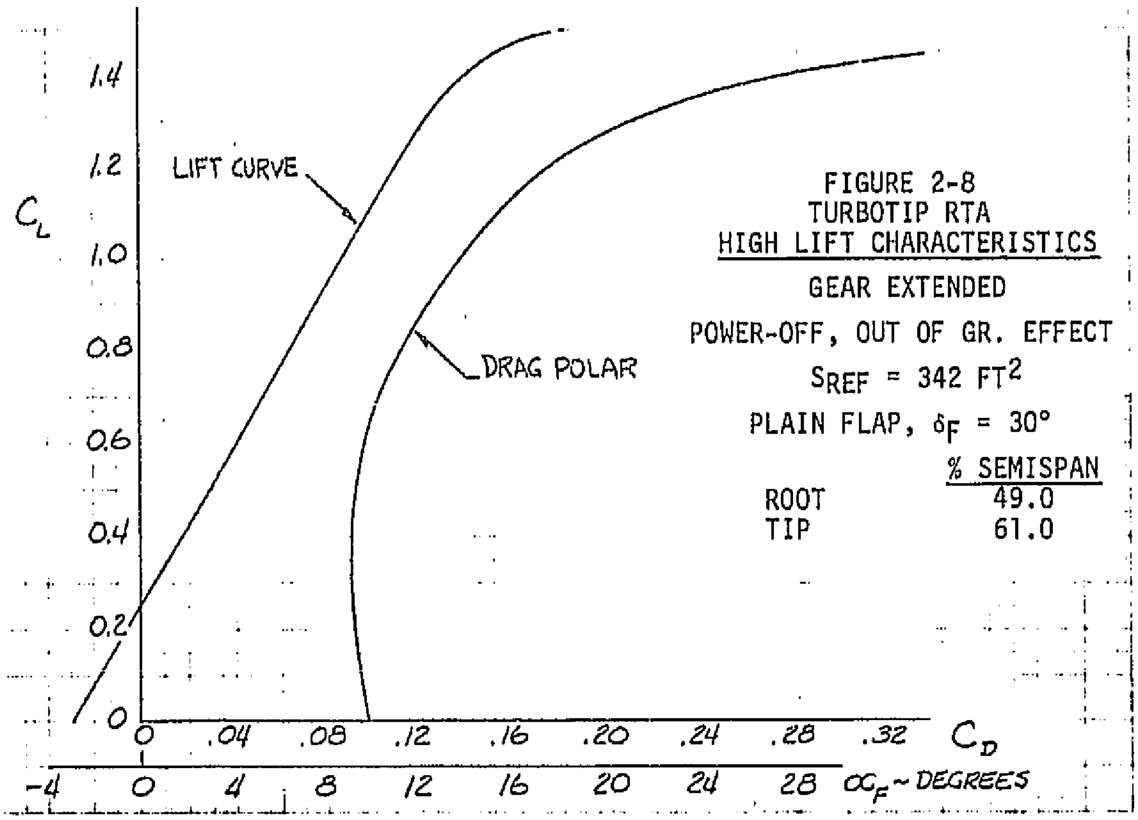


FIGURE 2-8
TURBOTIP RTA
HIGH LIFT CHARACTERISTICS

GEAR EXTENDED

POWER-OFF, OUT OF GR. EFFECT

S_{REF} = 342 FT^2

PLAIN FLAP, δ_F = 30°

	% SEMISPAN
ROOT	49.0
TIP	61.0

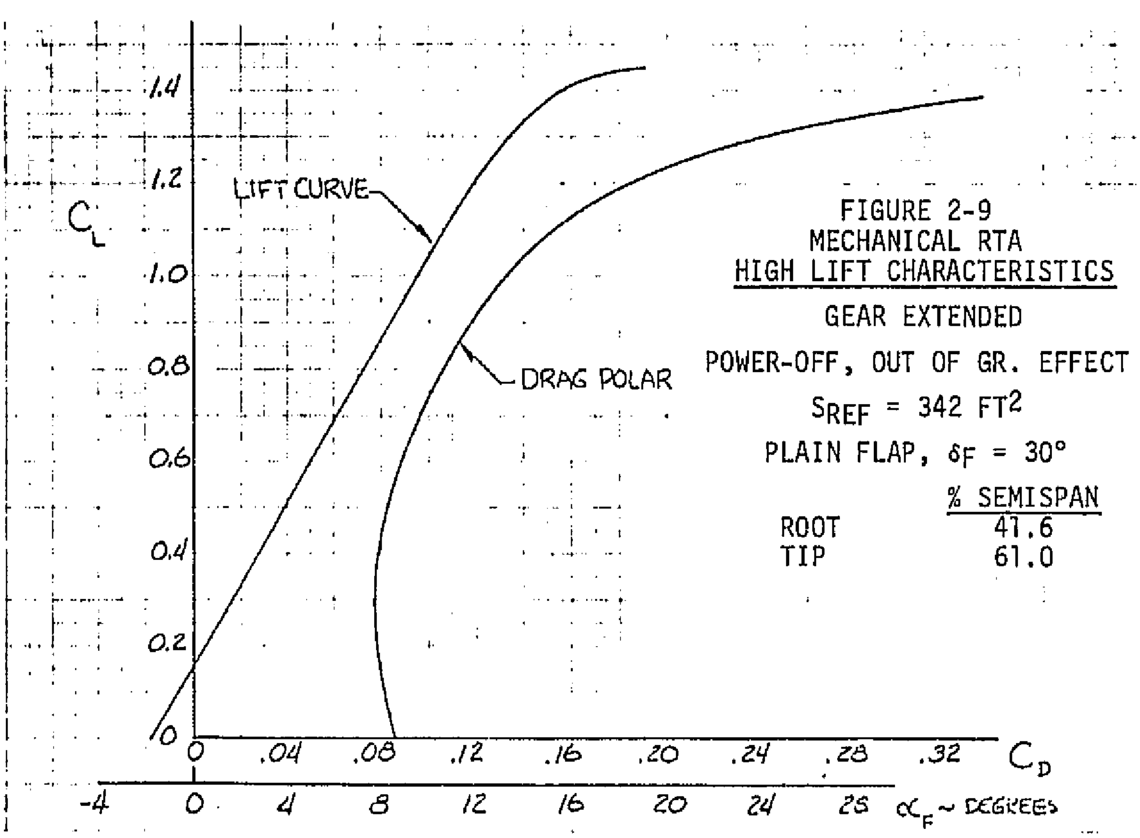


FIGURE 2-9
MECHANICAL RTA
HIGH LIFT CHARACTERISTICS

GEAR EXTENDED

POWER-OFF, OUT OF GR. EFFECT

S_{REF} = 342 FT^2

PLAIN FLAP, δ_F = 30°

	% SEMISPAN
ROOT	41.6
TIP	61.0

STOW (Alternate) - Gross weight with fuel for a (2) hour cruise/loiter test mission, 2500 lb payload.

VLGW - Gross weight with fuel for a (2) circuit test mission, 2500 lb payload; engine out thrust = 1.03 VLGW.

By NASA direction, the vertical takeoff and landing analyses assumed zero net ground effects; i.e. the fountain effects nullify any suckdown forces and the reingestion effects are negligible.

Both RTA's have more than adequate thrust margins during normal three engine operation. The engine out vertical landing requirement of a T/W = 1.03 at VLGW was the most critical sizing factor and established the need for a three engine configuration for both the turbotip and mechanical RTA.

The use of the YJ97 engines in the turbotip RTA resulted in a significant thrust/weight margin for both normal and engine out conditions. The engine out requirements can be met with the remaining engines at an Intermediate dry power setting. In fact, at this power setting the normal five circuit vertical mission gross weight at a T/W = 1.05 can be achieved with two engines thereby providing complete engine out safety during maximum VTOGW. This large thrust/weight for an RTA results in a low risk program. The turbotip RTA thrust/weight summary is shown in Figure 2-10.

FIGURE 2-10
TURBOTIP RTA THRUST/WEIGHT SUMMARY

	STO		VTO	
	11 STO CIRCUITS	2 HR ENDURANCE	5 "V" CIRCUITS	VL 2 CIRCUITS
OWE - LB	19451	19451	19451	19451
PAYLOAD - LB	2500	2500	2500	2500
FUEL - LB				
WARMUP	260	260	--	--
CIRCUITS	3905	4904	2710	1084
RESERVES	464	545	625	625
G.W. - LB	26580	27660	25286	23660
T/W REQ'D			1.05	1.03
LIFT REQ'D - LB (1)			26550	24370
THRUST AVAIL. - LB (2)				
3 ENG - INTER DRY			36584	
MAX DRY			37532	
2 ENG - INTER DRY				26687
EMER DRY				30266

COMPLETE VTO SAFETY WITH ENGINE-OUT @ INTER. DRY RATING

(1) NO ALLOWANCE FOR GRD EFFECTS/REINGESTION
(2) INCLUDES 3% ENGINE DERATE

The XT701 engines used in the mechanical RTA provide a large thrust margin for normal operation at an Intermediate dry power setting; however, water injection is required to meet the engine out vertical landing requirements. The engine out thrust shown in the thrust/weight summary, Figure 2-11, is for an Intermediate Wet rating plus 25°F temperature increase. It is apparent that the thrust margin required to meet the guidelines can probably be attained without using the 25°F temperature increase. If necessary, additional engine out thrust margin could be attained by (1) better engine/fan matching (2) increased engine temperature or (3) reduced payload. Both aircraft meet the design requirements of the study.

The STO performance of the RTA was determined and is presented in Figure 2-12 in terms of STOGW/VTOGW ratio as a function of ground roll distance and wind velocity. These estimates assumed a takeoff thrust equal to 1.05 VTOGW which is well below that available with three engines at intermediate thrust. The powered induced lift, Figure 2-13, used in the analysis reflects recent wing tunnel data; this factor is less than that used in previous studies. The STO technique used is as follows:

- Warmup plus cockpit checkout
- Thrust vector angle, $\theta_R \sim 17$ degrees
- Throttle advance, brake release to initiate ground roll acceleration
- Thrust vector increase to $\sim$ 55 degrees plus aircraft rotation to 0.8 $C_{L_{max}}$ in 1.5 seconds
- Aircraft liftoff, climbout, minimum a/g = 0.065

The 11 circuit STO and 2 hr Cruise/Endurance missions do not tax the aircraft's STO capabilities. With increased internal fuel, the RTA could have a STOGW of 32,000 lbs at a load factor of 2.0g. At this STOGW the STOGW/VTOGW ratio would be 1.265. At zero wind conditions, this weight would require a ground roll of only 325 feet. The ground roll required for the 2 hr cruise/endurance mission would be less than 150 feet. At small STOGW/VTOGW ratios and takeoff distances, pilot technique is the determining factor and vector angle increase without aircraft rotation would provide good STO performance.

2.3.2 FLIGHT ENVELOPES - The powered lift mode flight envelopes for the turbotip and mechanical RTA are shown in Figures 2-14 and 2-15 for both VTO and STO gross weights. The data is presented for thrust levels corresponding to both a 1.05 VTOGW and the three engine intermediate rating. The powered lift maximum speed corresponds to a lift/cruise and lift fan resultant vector angle of 40 degrees. At altitudes of 8,000 to 10,000 ft, conversion speed overlaps in

FIGURE 2-11
MECHANICAL RTA THRUST/WEIGHT SUMMARY

		STO		VTO	
		11 STO CIRCUITS	2 HR ENDURANCE	5 "V" CIRCUITS	VL 2 CIRCUITS
O.W.E.	LB	20,260	20,260	20,260	20,260
PAYLOAD	LB	2500	2500	2500	2500
FUEL	LB				
WARMUP		224	224	---	---
CIRCUITS		3366	4722	2365	946
RESERVES		405	468	545	545
G.W.	LB	26,755	28,174	25,670	24,251
T/W REQ'D				1.05	1.03
LIFT REQ'D (1)				26,953	24,978
THRUST AVAIL. (2)					
3 ENGINE - INTER. DRY				32,138	
2 ENGINE - INTER. WET +25° ΔT					25,756

(1) NO ALLOWANCE FOR ENGINE DERATE OR GRD EFFECTS/REINGESTION

(2) ASSUMES 100% SUPERCHARGING AND 2% HP LOSS [illegible]EAD ENGINE

FIGURE 2-12
MODEL 260 RTA STO PERFORMANCE

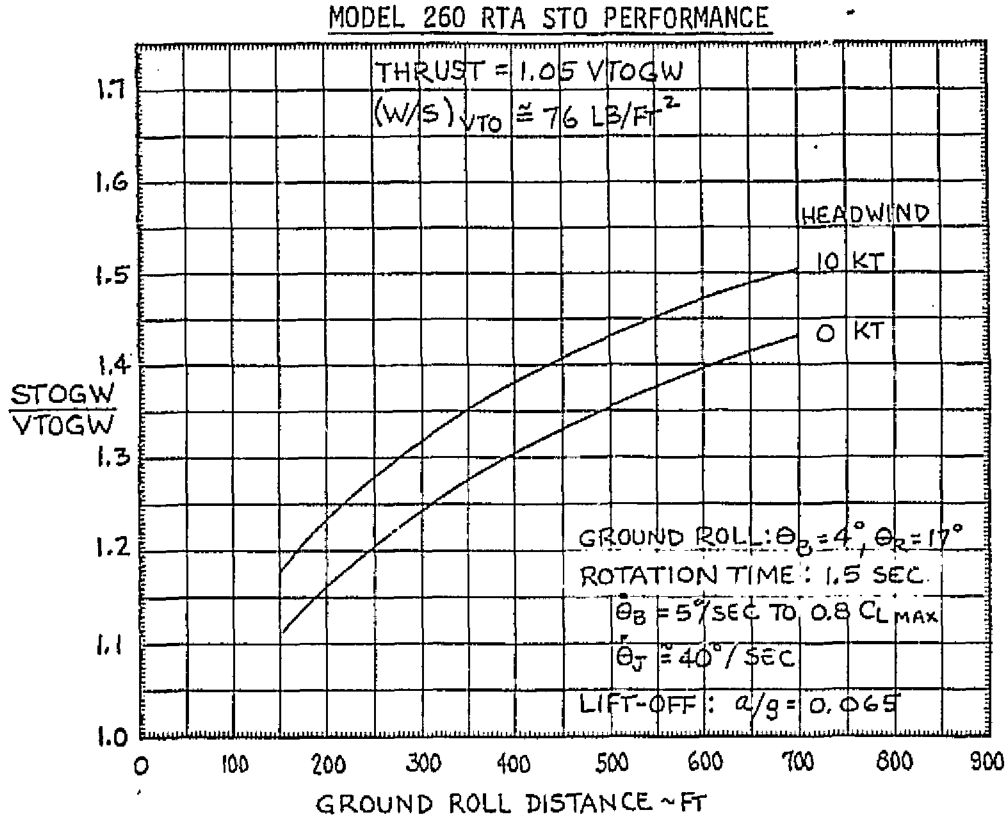


FIGURE 2-13
TURBOTIP AND MECHANICAL RTA
POWER INDUCED LIFT - (3) FANS

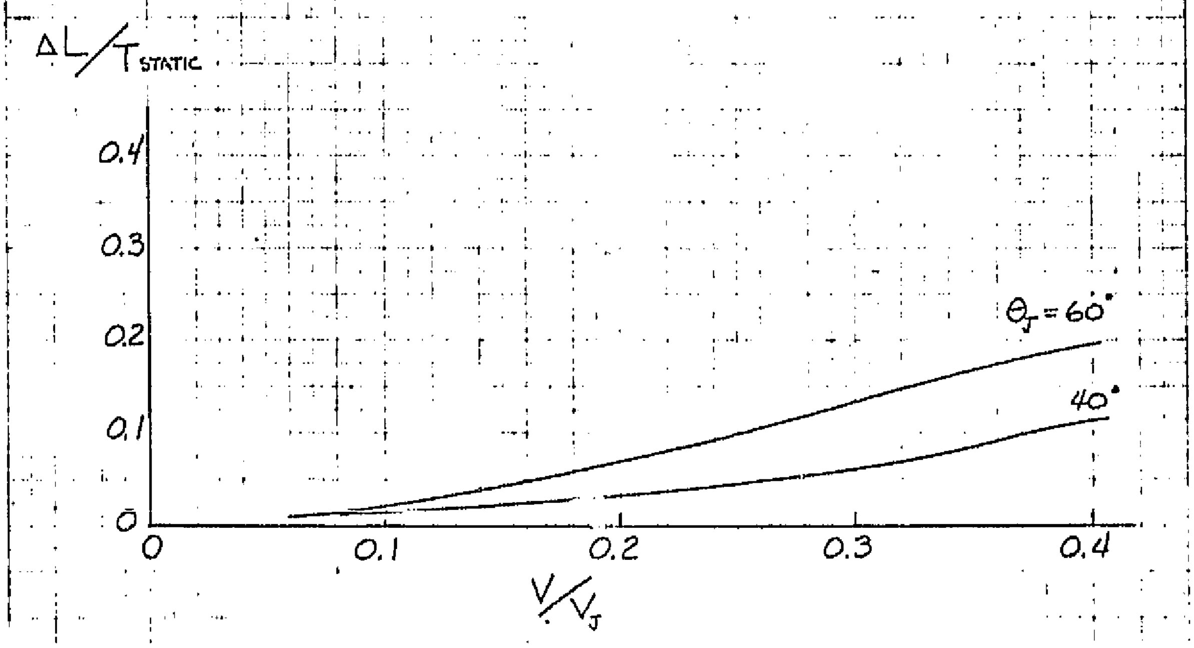


FIGURE 2-14
TURBOTIP RTA
FLIGHT ENVELOPE - POWERED LIFT MODE

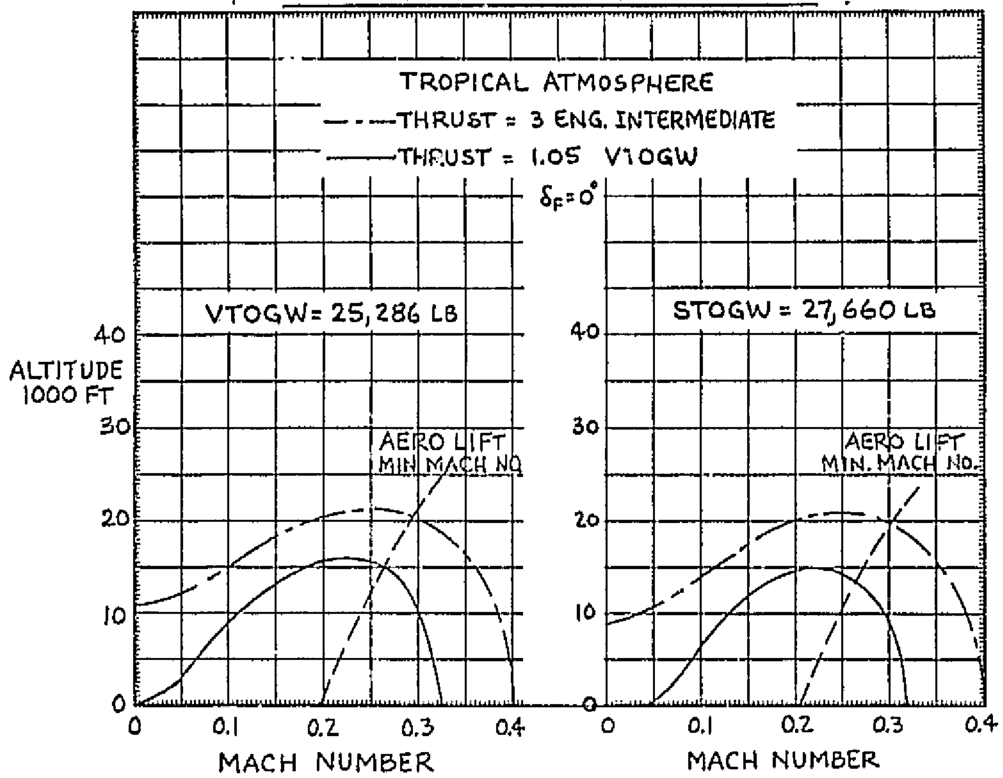
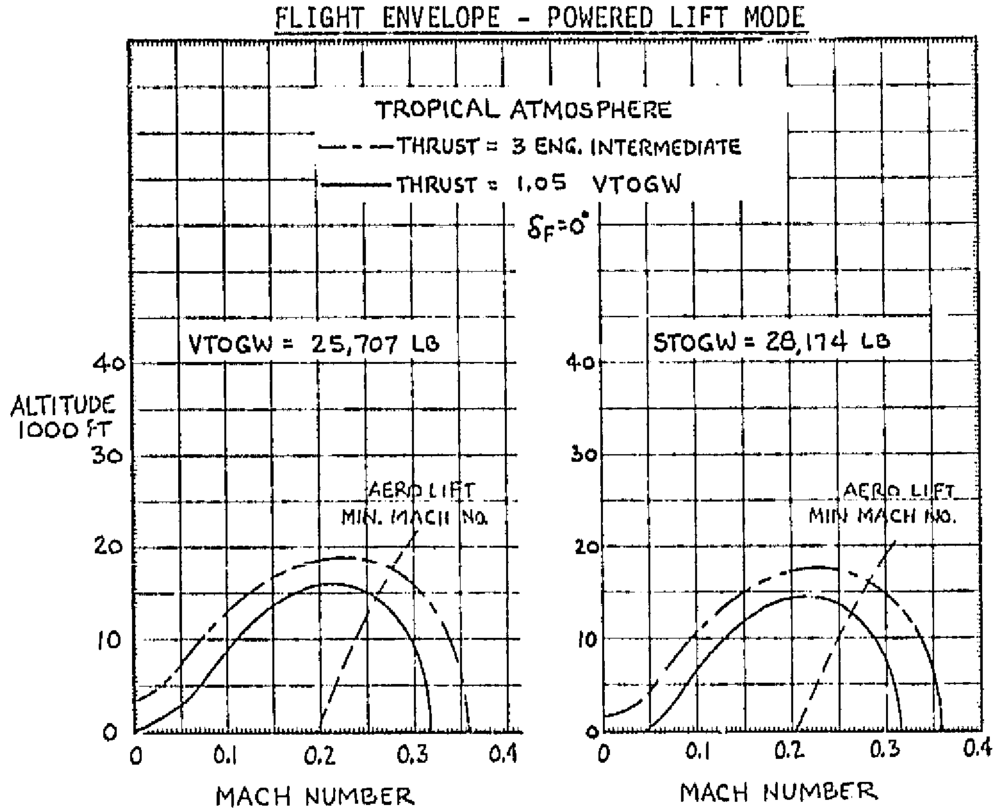


FIGURE 2-15
MECHANICAL RTA
FLIGHT ENVELOPE - POWERED LIFT MODE



excess of 20 percent are available at test mission STOGW and thrust = 1.05 VTOGW. The aircraft have a significant hover capability when using three engines at intermediate power.

The flight envelopes at a thrust level of 1.05 VTOGW (solid lines) show compliance with the forward flight performance requirements, S.L. 89.8°F atmosphere, in event of any reasonable failure of a power plant. The requirements at STOGW are:

- o Complete takeoff and continue accelerated flight with positive 1 1/2 degree climb angle, and
- o The maximum speed in the powered lift configuration shall be at least 10 percent greater than the power-off stall speed in the converted configuration.

Since the engine out thrust for the turbotip RTA exceeds that required for normal powered lift operation, Section 2.3.1, the solid lines on the powered lift flight envelopes can be used to assess engine out STOGW performance. The altitude and conversion velocity overlap capabilities at STOGW indicate engine out performance in excess of the requirements.

The mechanical RTA ratio of engine out thrust to that for 1.05 VTOGW is 0.955. Figure 2-15 shows that a change in thrust from three engine intermediate power to 1.05 VTOGW (ratio = 0.84) reduces the Sea Level maximum Mach number for the STOGW from 0.36 to 0.315. An additional thrust reduction to the engine out level (ratio = 0.955) results in M_{max} of about 0.3 which indicates an acceptable conversion speed overlap and climb performance for engine out STO operation.

Figures 2-16 and 2-17 present the aerodynamic lift mode flight envelopes for the turbotip and mechanical RTA at standard atmosphere, Basic Flight Design Gross Weight, and two engine Intermediate dry rating. In the aerodynamic lift mode two engines power the lift/cruise fans, the third engine is at idle or shut down, and the lift fan is shut down with its nozzle exit closed and faired. The minimum Mach number below 20,000 feet altitude is defined by the power off, maximum usable lift coefficient. Optimum loiter and cruise speeds are shown as a function of altitude. The guideline minimum cruise speed requirements, 300 KEAS at sea level and 0.7M at 25,000 feet, are attainable but exceed the

FIGURE 2-16
TURBOTIP RTA
FLIGHT ENVELOPE - AERO LIFT MODE

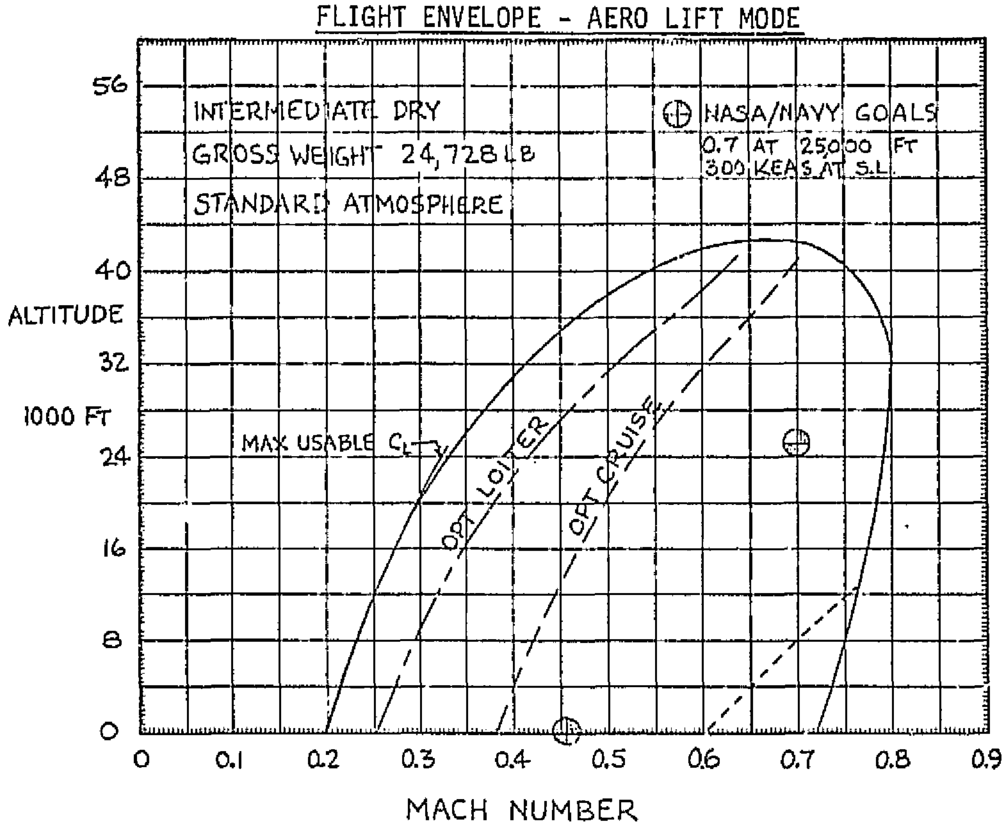
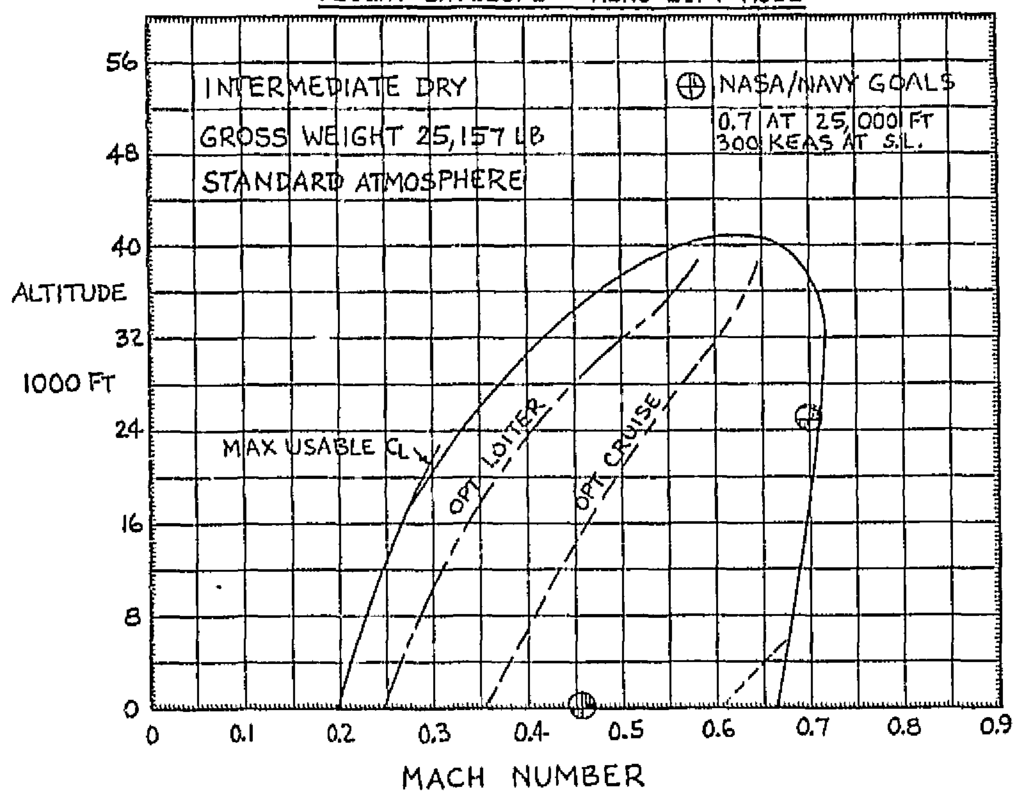


FIGURE 2-17
MECHANICAL RTA
FLIGHT ENVELOPE - AERO LIFT MODE



optimum cruise speed values. The maximum Mach number for the turbotip and mechanical fan aircraft is 0.8 and 0.7 at 30,000 feet, respectively. Each aircraft exhibits an absolute ceiling greater than 40,000 feet.

2.3.3 MISSION CAPABILITY - The RTA requirements for the VTOL, STOL, and cruise/endurance type missions with 2500 lb of payload were established by the design guidelines and are shown in Figure 2-18. The VTOL and STOL missions demonstrate takeoff, conversion, reconversion, and landing around an oval course. The cruise/endurance mission demonstrates the aircraft characteristics in aerodynamic flight with a minimum requirement of two hours mission time. For analysis purposes, the mission time on station was defined as loiter plus cruise at optimum altitude.

The mission weights for the turbotip and mechanical RTA are shown in Figures 2-10 and 2-11 and include O.W.E., payload and fuel for each mission. The fuel breakdown for VTO and STO circuits for each RTA is presented in Figures 2-19 through 2-22. The fuel breakdown includes warmup, circuit fuel

FIGURE 2-18
TYPICAL VTOL AND STOL TEST MISSIONS

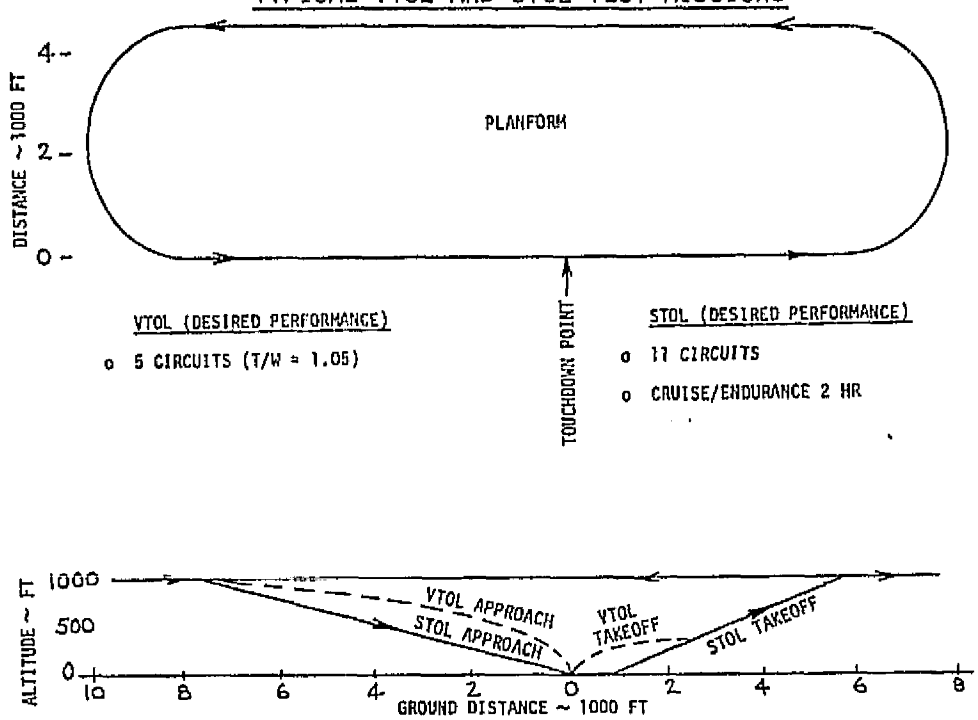
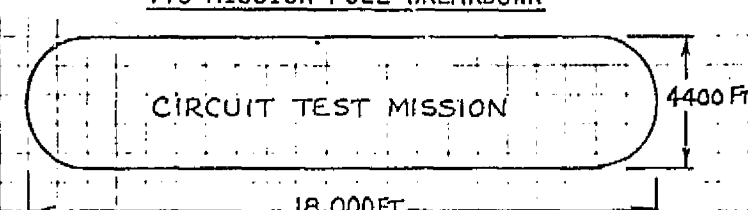


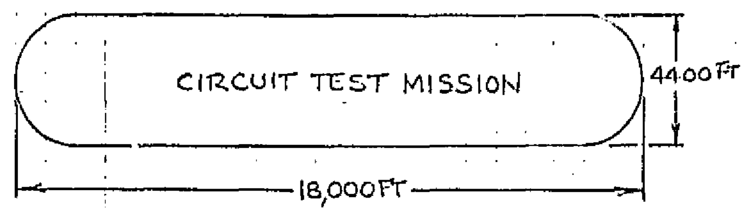
FIGURE 2-19
TURBOTIP RTA
VTO MISSION FUEL BREAKDOWN



MISSION PROFILE		TIME (MIN)	FUEL (LB)	
WARMUP, FUEL, THRUST = 1.05 VTOGW		1.5	260	
T.O., CLIMB ACCEL. TO 210 KT, THRUST = 1.05 VTOGW		1.0	173	542 LB PER CIRCUIT
180° TURN AT 210 KT	AERO LIFT MODE	0.33	22	
DOWN RANGE CRUISE	AERO LIFT MODE	0.66	29	
180° TURN AT 210 KT	AERO LIFT MODE	0.33	22	
3RD ENGINE IDLE FUEL	AERO LIFT MODE	(1.32)	16	
DECEL. DESCENT TO ZERO VEL., POWERED LIFT MODE		.7	115	
VERTICAL LANDING		1.0	165	
RESERVE: HOVER AT LGW		4.0	625	

5% SERVICE TOLERANCE ON FUEL FLOW

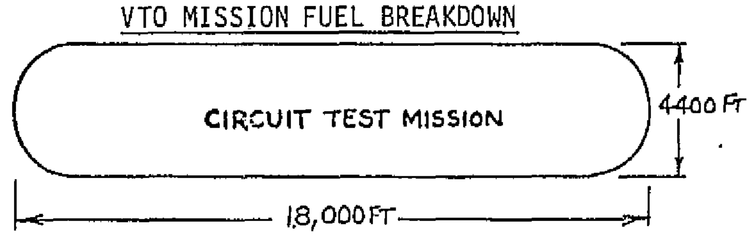
FIGURE 2-20
TURBOTIP RTA
STO MISSION FUEL BREAKDOWN



MISSION PROFILE	TIME (MIN)	FUEL (LB)	
WARMUP FUEL, THRUST = 1.05 VTOGW	1.5	260	
T.O., CLIMB ACCEL. TO 210 KT, THRUST = 1.05 VTOGW	1.0	173	355 LB PER CIRCUIT
180° TURN AT 210 KT (AERO LIFT MODE)	0.33	24	
DOWN RANGE CRUISE (AERO LIFT MODE)	0.66	29	
180° TURN AT 210 KT (AERO LIFT MODE)	0.33	24	
3RD ENGINE IDLE FUEL (AERO LIFT MODE)	(1.32)	16	
DECEL. DESCENT TO VTOUCHDOWN, POWERED LIFT MODE	.5	82	
GROUND DECEL. TO STOP	.2	7	
RESERVE: 10% INITIAL T.O. FUEL LOAD		464	

5% SERVICE TOLERANCE OF FUEL FLOW

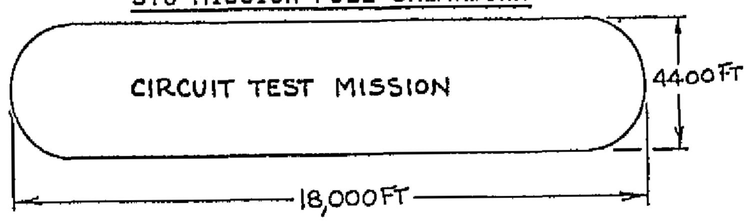
FIGURE 2-21
MECHANICAL RTA
VTO MISSION FUEL BREAKDOWN



MISSION PROFILE	TIME (MIN)	FUEL (LB)	
WARMUP FUEL, THRUST = 1.05 VTOGW	1.5	224	
T.O., CLIMB ACCEL. TO 210 KT, THRUST = 1.05 VTOGW	1.0	149	473 LB PER CIRCUIT
180° TURN AT 210 KT (AERO LIFT MODE)	0.33	19	
DOWN RANGE CRUISE (AERO LIFT MODE)	0.66	23	
180° TURN AT 210 KT (AERO LIFT MODE)	0.33	19	
3RD ENGINE IDLE FUEL (AERO LIFT MODE)	(1.32)	11	
DECEL DESCENT TO ZERO VEL., POWERED LIFT MODE	0.7	109	
VERTICAL LANDING	1.0	143	
RESERVE: HOVER AT LGW	4.0	545	

5% SERVICE TOLERANCE ON FUEL FLOW

FIGURE 2-22
MECHANICAL RTA
STO MISSION FUEL BREAKDOWN



MISSION PROFILE		TIME (MIN)	FUEL (LB)	
WARMUP FUEL, THRUST = 1.05 VTOGW		1.5	224	
T.O., CLIMB ACCEL. TO 210 KT, THRUST = 1.05 VTOGW		1.0	149	306 LB PER CIRCUIT
180° TURN AT 210 KT	AERO LIFT MODE	0.33	20	
DOWN RANGE CRUISE	AERO LIFT MODE	0.66	23	
180° TURN AT 210 KT	AERO LIFT MODE	0.33	20	
3RD ENGINE IDLE FUEL	AERO LIFT MODE	(1.32)	11	
DECEL DESCENT TO $V_{TOUCHDOWN}$, POWERED LIFT MODE		0.5	78	
GROUND DECEL. TO STOP		0.2	5	
RESERVE: 10% INITIAL T.O. FUEL LOAD			405	

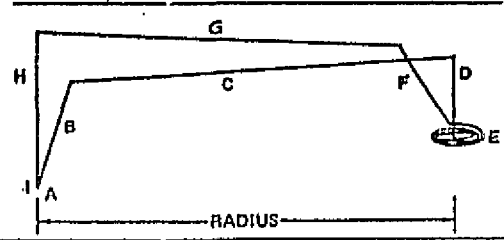
5% SERVICE TOLERANCE OF FUEL FLOW

and reserves. When determining fuel load for takeoff and landing gross weight estimates, the warmup fuel was assumed to be consumed prior to takeoff. The mission profile used for the cruise/endurance mission is shown in Figure 2-23.

The mission performance for the turbotip and mechanical RTA is presented in Figures 2-24 and 2-25 respectively. The fuel tankage in the RTA is sized for the two-hour mission; however, adequate volume is available for additional tanks. The mission performance is shown for a spectrum of weights from the 2 hour mission TOGW to a 32,000 lb TOGW which represents a load factor of 2.0 g. At the 32,000 lb TOGW with internal fuel, the loiter or combined cruise and loiter time on station is approximately four hours for either RTA. The maximum loiter time on station for a given TOGW occurs at the climb distance: 40 nm for the turbotip RTA and 90 nm for the mechanical RTA. Optimum cruise and loiter altitude, and Mach number performance as a function of average weight is also presented.

2.3.4 <u>CONTROL IN CROSSWIND</u> - The RTA were evaluated as to compliance with the requirement that at least 50% of the specified normal control power shall be available for maneuvering after the aircraft is trimmed in a 25 kt crosswind.

FIGURE 2-23
CRUISE/ENDURANCE MISSION PROFILE



COMPONENT	DEFINITION
A) WARMUP, T.O. ACCEL TO V_c, SL 89.8°F	2 1/2 MIN VTO THRUST
B) CLIMB, INTERMEDIATE THRUST	SL TO OPT ALT
C) CRUISE OUT, ALTITUDE/MACH	OPT/OPT
D) DESCEND, NO CREDIT	TO LOITER ALT
E) LOITER, ALTITUDE/MACH	OPT/OPT
F) CLIMB, INTERMEDIATE THRUST	TO OPT ALT
G) CRUISE RETURN, ALTITUDE/MACH	OPT/OPT
H) DESCEND, NO CREDIT	OPT TO SL
I) RESERVES	5% INITIAL FUEL + 10 MIN ENDURANCE, SL

All SFC Increased 5% for Service Tolerance

FIGURE 2-24
TURBOTIP RTA
STO CRUISE/LOITER MISSION

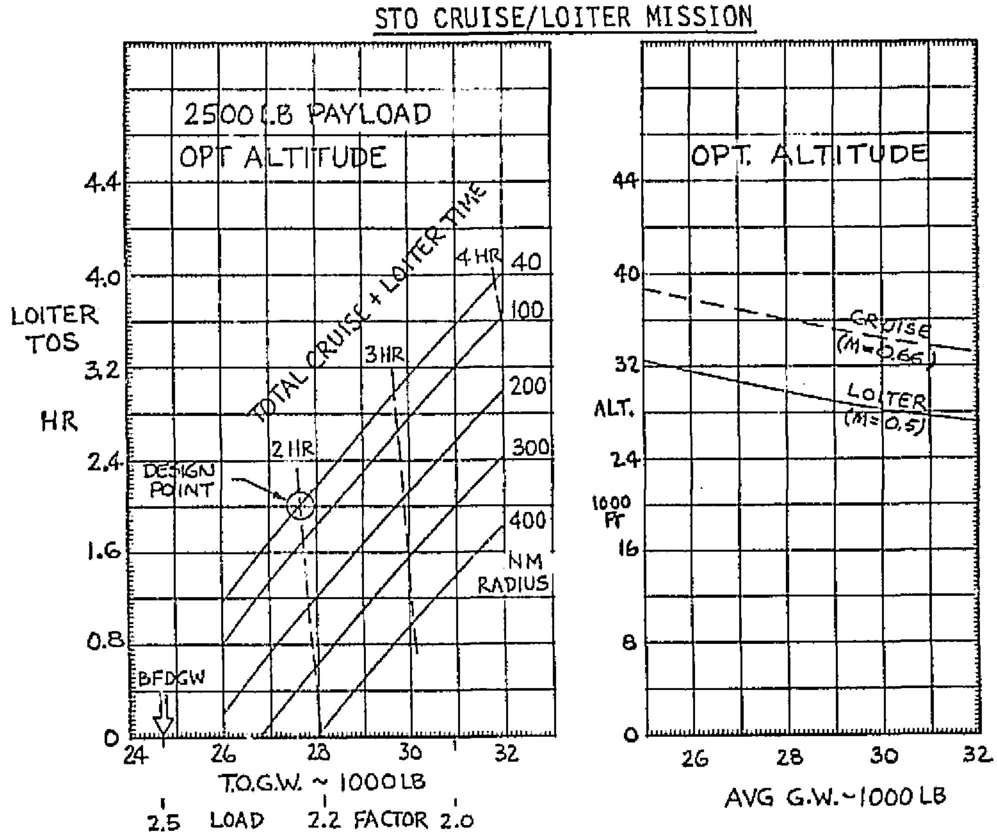
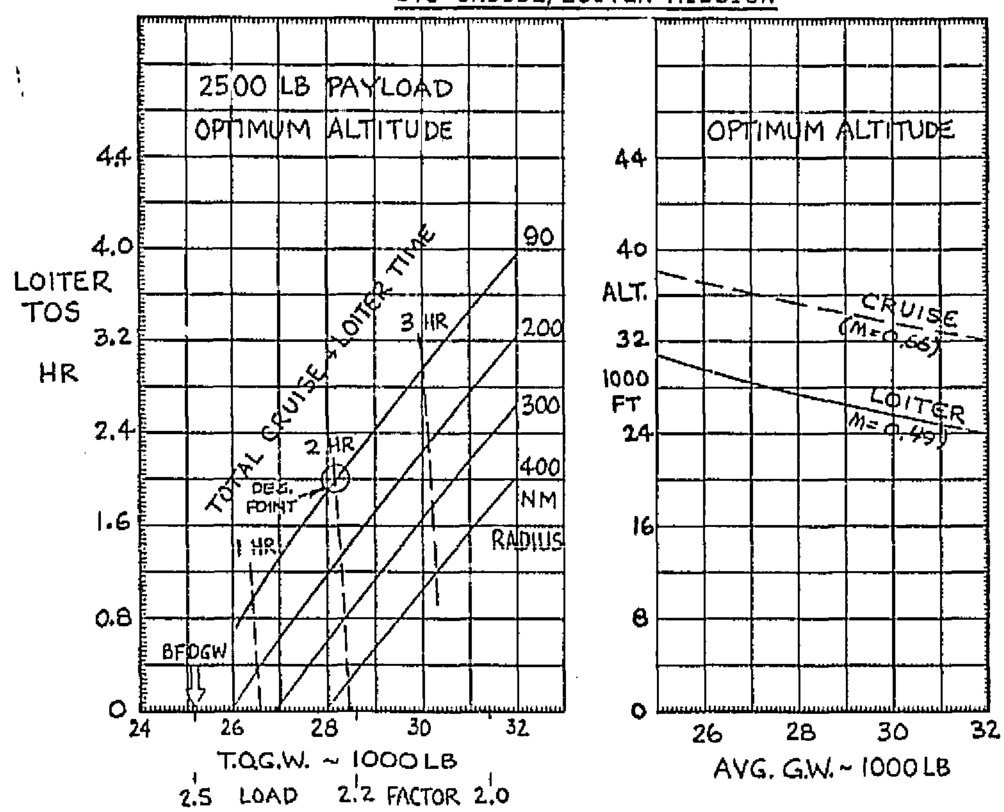


FIGURE 2-25
MECHANICAL RTA
STO CRUISE/LOITER MISSION



The primary sources of the forces and moments in a crosswind are the ram drag effect of inlet mass flows and aerodynamic loads on the fuselage and vertical tail. The maximum trim force and moment occur when the flow is normal to the aircraft plane of symmetry. Figure 2-26 presents the control required for trimming in a 25 kt crosswind at various hover gross weight levels. The yaw control is the most critical; the turbotip RTA control satisfies the requirement while the mechanical RTA shows 54 to 58 percent of the proposed control is used for trim. An increase in yaw control through a small increase in lift and lift/cruise vane deflection is required to fulfill the crosswind requirement. The mechanical RTA lift/cruise inlet location (approximately 5 ft forward of the aircraft C.G.) is the source of an unstable moment due to ram drag, a moment much larger than for the turbotip RTA since its lift/cruise inlet is relatively close to the C.G.

FIGURE 2-26
RTA CONTROL REQUIRED - HOVER IN 25 KT CROSSWIND

CONFIGURATION	AIRCRAFT GROSS WEIGHT	LIFT FAN			AIRCRAFT		LIFT CRUISE FANS	
		SIDE FORCE	YAW VECTOR ANGLE		TOT. SIDE FORCE	BANK ANGLE	Δ LIFT FOR ROLL CONTROL	THRUST MODULATION
	LB	LB	DEG	% TOT	LB	DEG	LB	%
Turbotip RTA	23,660	343	2.5	31	2424	5.9	149	1.9
	28,000	386	2.4	30	2611	5.4	154	1.7
	34,842	441	2.2	26	2850	4.7	161	1.4
Mechanical RTA	24,228	651	4.6	58	2639	6.3	90	1.1
	26,850	678	4.3	54	2719	5.8	89	1.0

Note: Lift Fan Maximum Yaw Vector Angle Assumed = ±8 Deg.;
Each Lift/Cruise Maximum Yaw Vector Angle Assumed = ±4 Deg.

3. PROPULSION SYSTEMS

3.1 TURBOTIP RTA

3.1.1 CONFIGURATION - The turbotip RTA propulsion system, illustrated in Figure 3-1, consists of three LCF459 turbotip fans driven by three interconnected YJ97 gas generators. These units provide both lift and aircraft control during VTOL, along with horizontal thrust for conventional wingborne flight. A fuselage nose lift fan and two over-the-wing mounted lift/cruise fans, each with their own thrust vectoring systems, produce the required lift for VTOL operation. Aircraft control during the powered lift mode is obtained by utilizing the Energy Transfer and Control (ETaC) concept, described in Report MDC A3440. During conventional flight, the third engine and the nose fan are shut down, and the lift/cruise fans provide thrust for horizontal flight.

3.1.2 ENGINE AND FAN DESIGN - The YJ97-GE-100 gas generator is a single-spool turbojet engine with a 14-stage axial compressor and a two-stage turbine. Geometric and installed performance characteristics are presented in Figure 3-2. The current status and availability of the YJ97 are described briefly in Figure 3-3. The engine has been tested in a full scale ETaC program at MCAIR, described in Report MDC A1588, which demonstrated the ETaC control principle and the compatibility of the YJ97 with the ETaC system.

The LCF459 turbotip fan is a 59 inch diameter, single-stage, fixed pitch fan with a design pressure ratio of 1.319. It has a single-stage turbine mounted on the fan tip, which extracts power to drive the fan directly from the gas generator exhaust gases. A schematic of the fan and uninstalled performance characteristics are presented in Figure 3-4.

3.1.3 ENERGY TRANSFER AND CONTROL (ETaC) SYSTEM - Energy transfer and control of the power generated by the three gas generators is accomplished with the gas interconnect ETaC system shown in Figure 3-5. The system is designed to distribute the available power to the lift and lift/cruise fans during all modes of operation. During STOL and VTOL operation the ETaC system delivers gas power to each of the three fans as necessary to produce balanced thrust (lift) and roll and pitch attitude control. Yaw control is achieved with lateral thrust deflection vanes in the thrust vectoring systems downstream of each fan. During wingborne or conventional flight the third engine and nose fan are shut down, and gas power is delivered only to the two lift/cruise fans. Attitude control is then accomplished with conventional aerodynamic aircraft

FIGURE 3-1
TURBOTIP RTA
PROPULSION SYSTEM

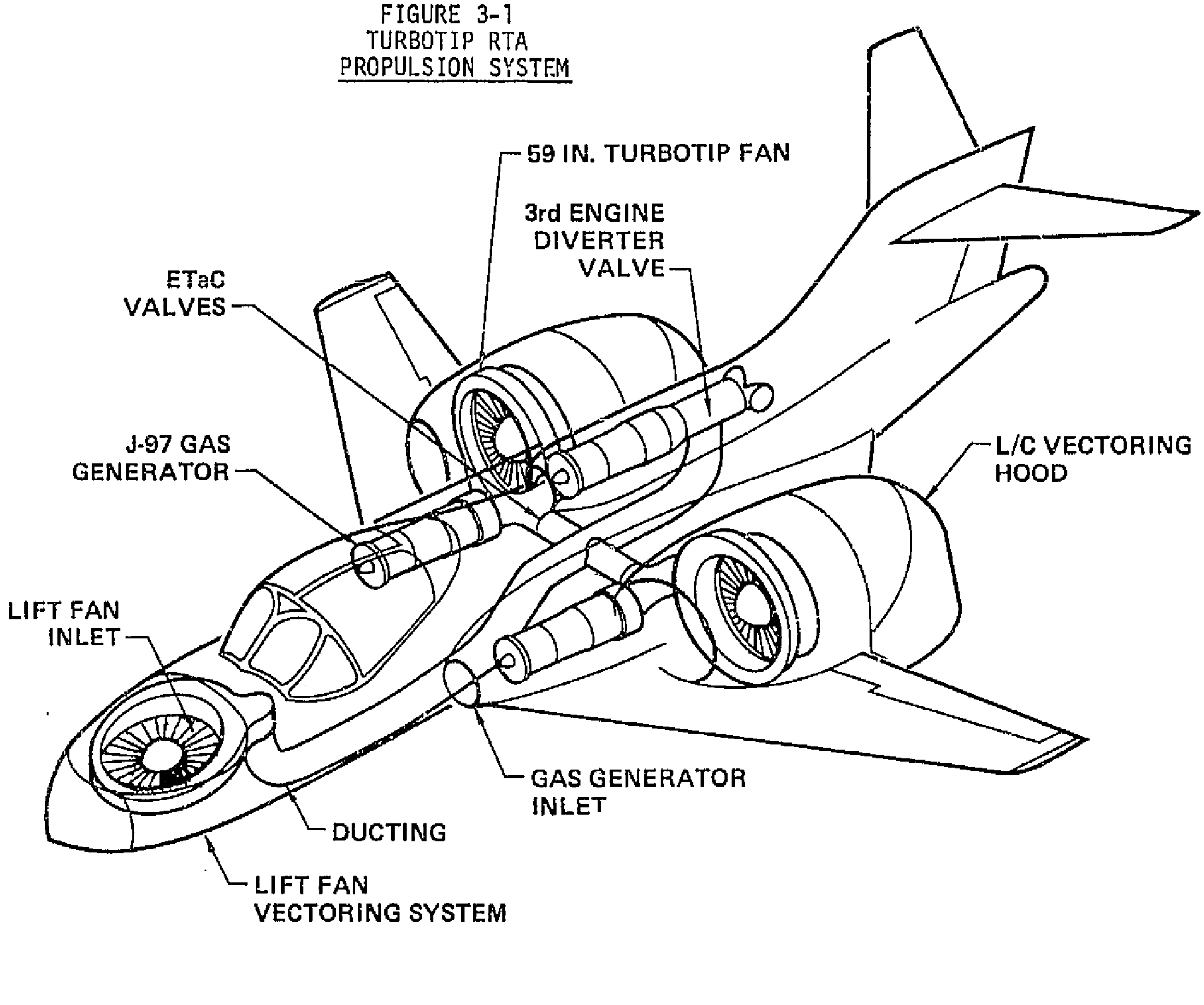
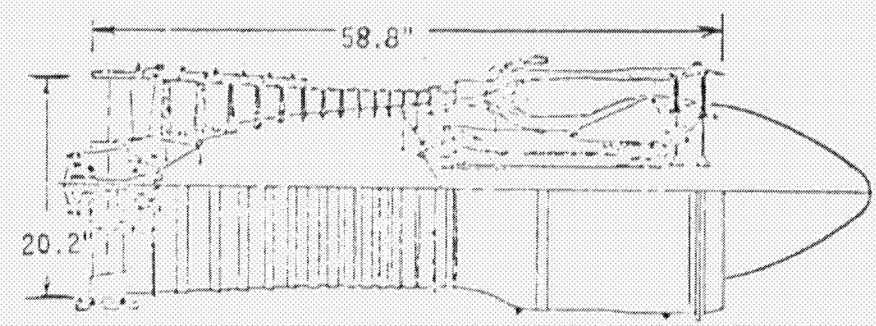


FIGURE 3-2

YJ97 CYCLE DESIGN CHARACTERISTICS



UNINSTALLED CHARACTERISTICS AT INTERMEDIATE
S.L. STATIC, STANDARD DAY

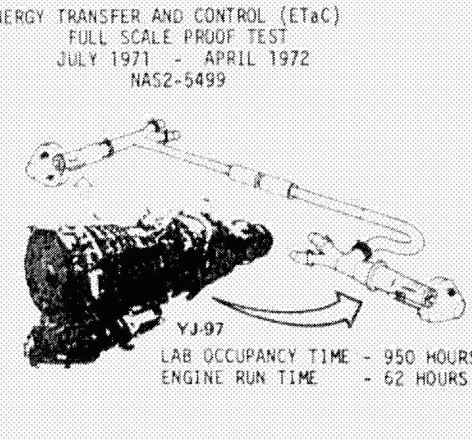
N_{GG}, %RPM (RPM)	101.5 (13855)
CPR	14.07
AVG. PR/STAGE	1.21
WA_{GG}, LB/SEC	69.2
TIT, °F	2030
EGT, °F	1375
EGP, PSIA	52.967
W_{GAS}, LB/SEC	70.54
W_F , LB/HR	4822
WEIGHT, LB	739
HPI	13700
HPI/WEIGHT	18.54
HPI SFC	.352

FIGURE 3-3

YJ97-GE-100 GAS GENERATOR

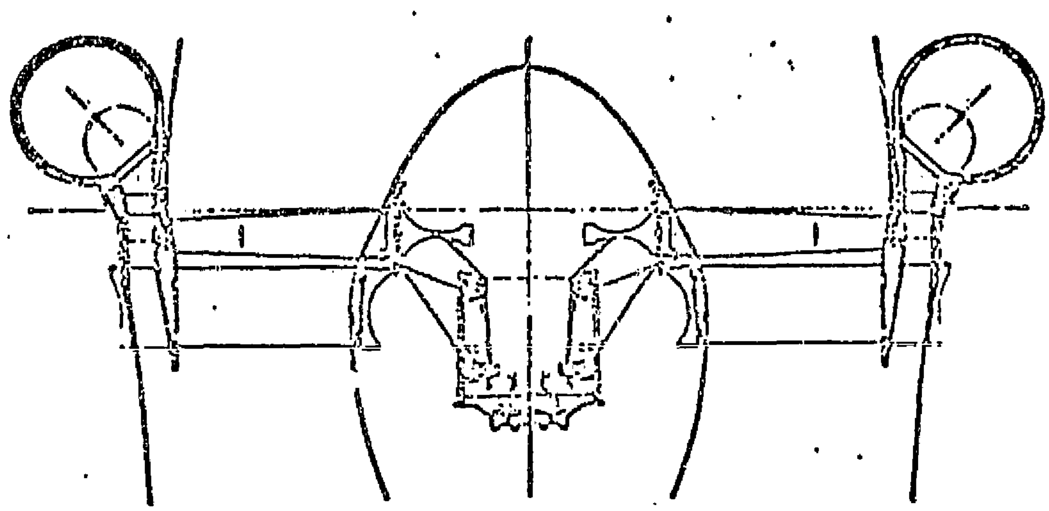
- PFRT TESTING COMPLETED SEPT 1969 ON YJ97-100
- YJ97 DEVELOPED TO A MODIFIED SPEC 5007-C
- 30 YJ97-GE-3 ENGINES SHIPPED
- ESTIMATED 22 YJ97-GE-3 ENGINES + SPARES STILL EXIST IN NAVY INVENTORY (18 SERV., 2 QUEST., 2 UNSERV.)
- 3 ENGINES + SPARES AT NASA AMES (2 SERV., 1 UNSERV.)
- MINOR CHANGES NECESSARY TO CONVERT -3 TO -100
- J97/ETaC PERFORMANCE AND COMPATIBILITY DEMONSTRATED DURING EXTENSIVE NORMAL AND ENGINE OUT TESTING

ENERGY TRANSFER AND CONTROL (ETaC)
FULL SCALE PROOF TEST
JULY 1971 - APRIL 1972
NAS2-5499



LAB OCCUPANCY TIME - 950 HOURS
ENGINE RUN TIME - 62 HOURS

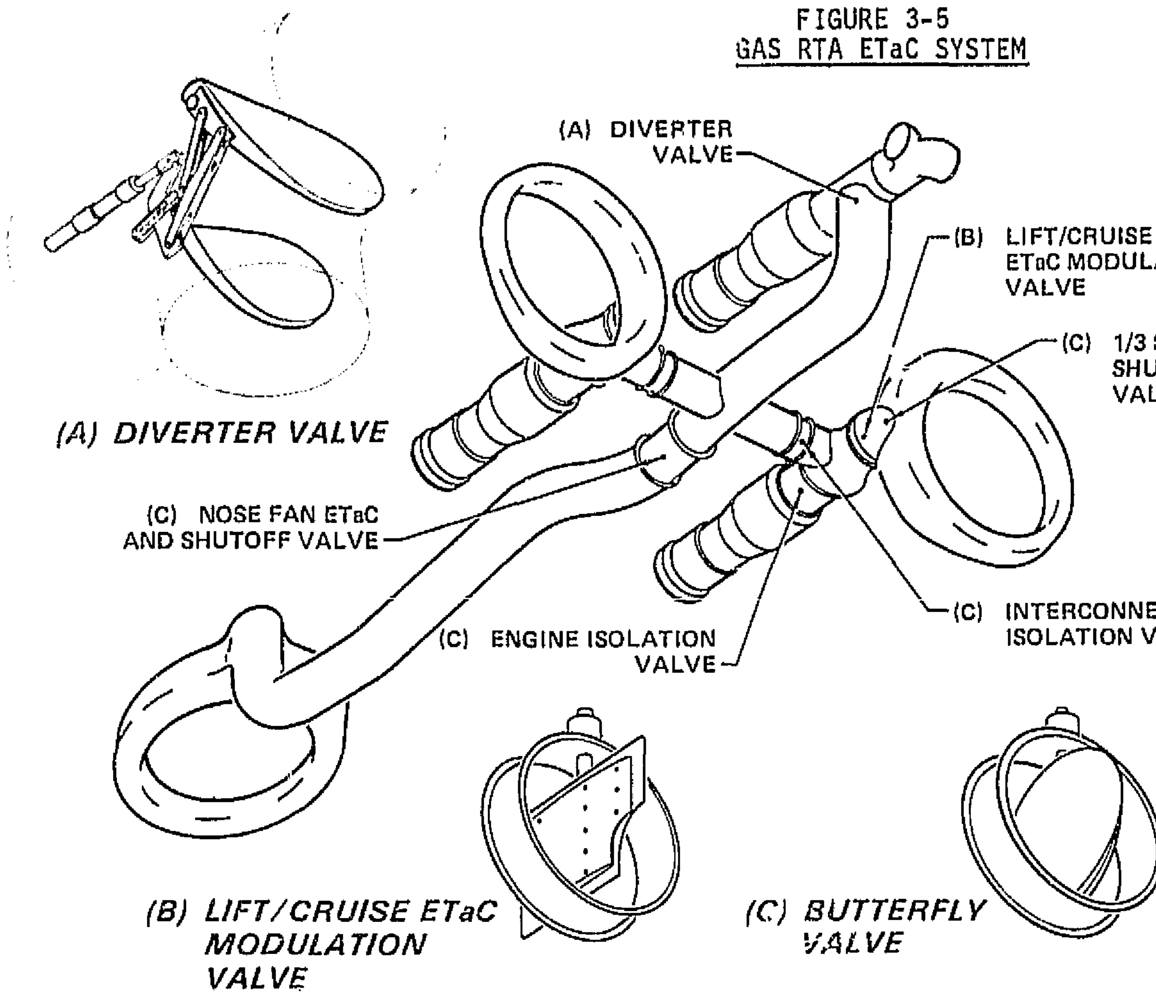
FIGURE 3-4
LCF 459 TURBOTIP FAN DESIGN CHARACTERISTICS



UNINSTALLED CHARACTERISTICS AT INTERMEDIATE
S.L. STATIC, STANDARD DAY

Aero Design FPR	1.319
Turbine Discharge Pressure Ratio	1.19
Fan Airflow, lb/sec	646
Turbine Gas Flow, lb/sec	70.54
Thrust, lb	14152
Fan Diameter, in.	59
Fan Weight, lb	885
Thrust/Weight	17:1

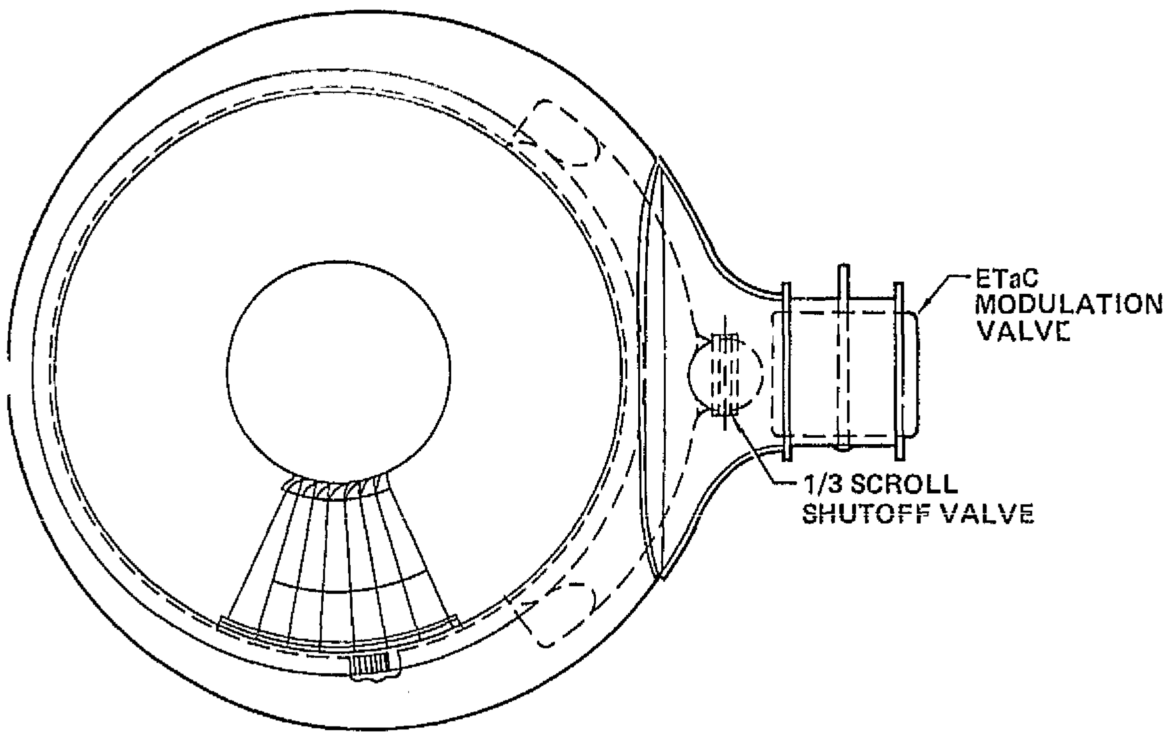
FIGURE 3-5
GAS RTA ETaC SYSTEM



control surfaces. In the event of engine failure or shutdown in either flight regime of any one of the gas generators, the ETaC system isolates the failed engine from the distribution system and continues to distribute the remaining gas horsepower to the fans.

The ETaC system used in this study consists of two primary flow paths: side gas generator to lift/cruise fan and third engine to lift fan, as illustrated in Figure 3-5. In the lift/cruise flow path, the engine exhaust first passes through an isolation valve (a simple butterfly valve) and then to the lift/cruise ETaC modulation valve. This valve is strictly a pressure control device and is not required to seal in the closed position. Downstream of the ETaC valve is a 1/3 scroll shutoff valve (butterfly), shown in Figure 3-6. The scroll employed in this study is referred to as the Scroll-in-Scroll, which

FIGURE 3-6
SCROLL-IN-SCROLL CONCEPT
(BASELINE DESIGN)



consists of an outer scroll (2/3 arc) and an inner scroll (1/3 arc). During normal operation both the ETaC and 1/3 scroll valves are open, to provide 100% arc admission. Following an engine failure, the 1/3 scroll shutoff valve is closed and only the remaining 2/3 portion of the scroll is utilized.

In the lift fan flow path, the third gas generator exhaust flow is directed down and forward by the diverter valve. This valve is the same design as the General Electric diverter valve used on the XV-5 except it is slightly larger. The gas flows forward past the interconnect duct to the lift fan ETaC/shutoff valve. This valve serves as both a pressure control device in the "V" mode and a shutoff valve in cruise. The lift fan scroll contains a 1/3 shutoff valve identical to the one in the lift/cruise fan scrolls. A simplified gas distribution system was devised just prior to the conclusion of this study, and is described in Section 3.1.8.

The total duct and scroll pressure loss to the nose fan is slightly greater than that to the lift/cruise fans; therefore, a larger nose fan tip turbine nozzle area is required to achieve equal flow distribution. Since identical lift and lift/cruise turbine nozzles were desired, discussions were held with General Electric which resulted in sizing all tip turbines to the required nose fan area with a preset on the lift/cruise ETaC valves to equalize the total pressure losses. Equal flow to all three fans was achieved but at a cost of approximately 2.4% thrust loss when compared to a gas distribution system without the valve preset. General Electric has recently developed a method, Section 3.1.8, which eliminates the need for the valve preset and the attendant thrust loss.

3.1.4 THRUST VECTORING AND THRUST MODULATION SYSTEMS - The lift and lift/cruise fans are each equipped with thrust vectoring and Thrust Reduction Modulation (TRM) systems. Thrust direction in the aircraft vertical plane is mechanically controlled such that during vertical takeoff through transition to wingborne flight, large thrust pitching moments are avoided by proper thrust vectoring, leaving full pitch control available at any powered flight condition. Thrust is also vectored transversely (side force) for yaw control during powered lift mode. Thrust modulation devices reduce the thrust at any one or two of the three fans during pitch or roll control demands only.

The lift fan nozzle and its associated functions are shown in Figure 3-7. Lateral louvers vector exhaust flow from 40° to 105° from horizontal. Longitudinal vanes vector the exhaust flow transversely for yaw control. TRM is achieved by rotating adjacent louvers in opposite directions, as was done on the XV-5, to spoil nose fan thrust. The longitudinal vanes function as closure doors after conversion to aerodynamic flight. The fan is tilted forward 15° to improve inlet performance and to reduce the peak thrust deflection required in the vertical plane. MCAIR studies show that reduced time, fuel, and exposure to engine failure during takeoff and landing result with this vectoring system when compared to a similar aircraft without a nose fan vectoring system.

A significant amount of in-house and NASA-funded developmental testing has been conducted on the MCAIR design "D" vented thrust vectoring nozzle. This nozzle configuration is illustrated in Figure 3-8 in each of its primary functional modes, high speed cruise, low speed cruise, longitudinal vectoring, yaw vectoring, and thrust reduction modulation. The designation, "D" vented, is used primarily to identify the shape of the nozzle exit in its cruise and VTO positions and the open or "vented" lower elbow corner which is formed in the VTO position. The venting feature, on the basis of MCAIR small scale and

FIGURE 3-7
TURBOTIP RTA
FORWARD FAN NOZZLE FUNCTIONS

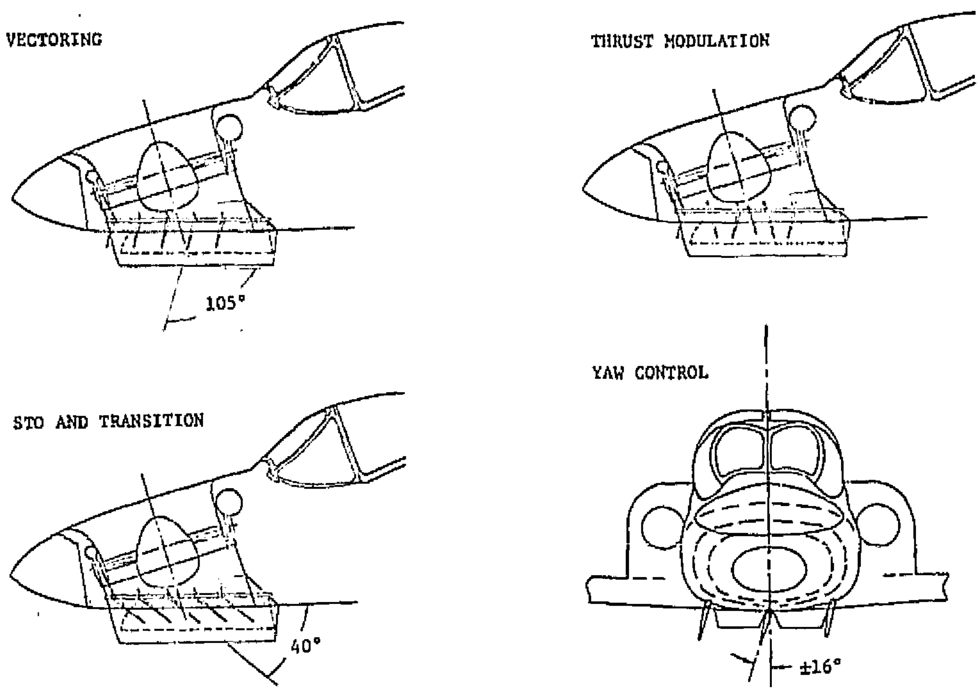
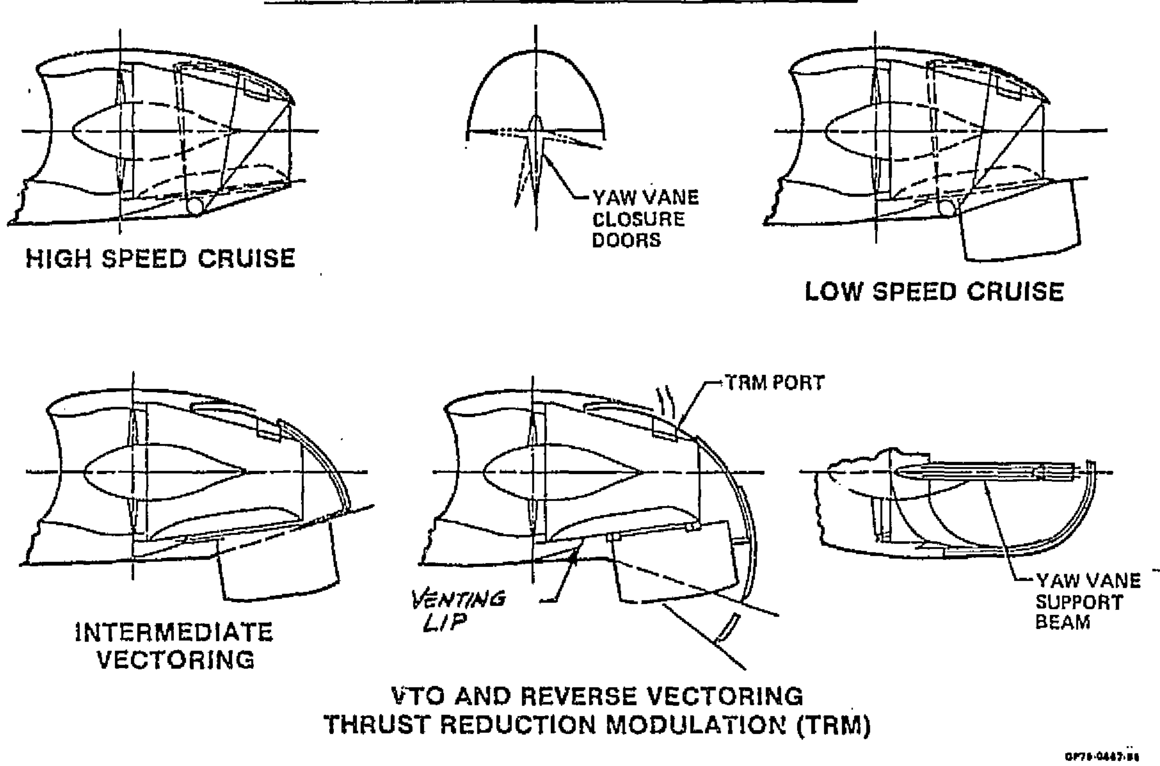


FIGURE 3-8
"D" VENTED LIFT/CRUISE VECTORING NOZZLE



NASA-funded large scale test results, provides entrainment of ambient air into the lower elbow corner region of the ozzle flow and results in improved 90° vectoring performance over that of an unvented design.

3.1.5 PROPULSION SYSTEM PERFORMANCE

VTO Performance - Figure 3-9 summarizes three engine, normal powered lift performance. These data indicate that

(1) The thrust/weight ratio at intermediate power, based on 5 VTO circuits of fuel, is 1.45. This means that the engines and fans can be run at substantially reduced power settings for a T/W = 1.05 takeoff condition.

(2) When operating at these reduced power settings, the resulting exhaust gas temperatures (1200 - 1250°F) are similar to the operating temperatures used in the XV-5 program.

Engine out performance is shown in Figure 3-10. Again, as with the normal ratings, intermediate power provides excess thrust margin. Therefore, at the required thrust to weight of 1.03, reduced power and reduced exhaust gas temperatures result. In addition, the Intermediate Dry rating provides adequate lift to insure complete engine out safety for a 5 circuit VTO mission. Additional VTOL and cruise performance, ram drag characteristics, component geometric descriptions, and other detailed data is provided in Appendix B.

Control Performance - A summary of control performance is presented in Figures 3-11 and 3-12. Three engine VTO control performance, Figure 3-11, shows the considerable excess control margin available. The cross-hatched areas designate required control margin and, as shown, additional control margin is available, if desired, for more extensive aircraft flight research. Also, the excess available thrust means that the aircraft can be loaded up to approximately 32,000 lb (2 g limit) for additional STOL research. Similarly, for engine out operation, Figure 3-12 illustrates significant excess control margin at T/W = 1.03. Also, the application of required control results in only a very small change in fan speed for both normal and engine out operation.

Engine and Fan Operating Conditions - Figure 3-13 shows engine operating characteristics as represented by the YJ97 compressor map. At hover, a 22.5% engine stall margin exists, dropping to 20.0% for maximum control operation. Additional stall margin is available by lowering the operating line, which would result in a slight decrease in lift.

Fan operating characteristics are displayed on the LCF459 fan map, Figure 3-14. General Electric selected the operating line to provide a constant 18% stall margin. For normal and engine out operation, the fans are operating at considerably reduced speed. As with the engine operating line, a slight decrease in lift can be exchanged for a large increase in stall margin.

3.1.6 COMPONENT DESIGN GUIDELINES - Internal and external geometry, inlet areas, and total pressure recovery for the side and top gas generator inlets are shown in Figures 3-15 and 3-16, respectively. Figure 3-17 presents similar design guidelines for the nose fan unit, and Figure 3-18 for the lift/cruise unit. All inlets were sized to provide a total pressure recovery ≥.988 for static operation. Auxiliary inlets were not required for the RTA.

FIGURE 3-9
TURBOTIP RTA
(3) YJ97-GE-100/(3) LCF 459 FANS
NORMAL POWERED LIFT
INSTALLED VTOL PROPULSION SYSTEM PERFORMANCE
SLS, 89.8°F

PARAMETERS		POWER RATINGS MAX DRY (T/W = 1.48)*	INT. DRY (T/W = 1.45)*	REDUCED POWER (T/W = 1.05)*
GAS GENERATOR				
RPM	% N_{GG}	$\boxed{102.3}$	$\boxed{101.5}$	$\boxed{96.3}$
TIT	°F			
EGT	°F	$\boxed{1452}$	$\boxed{1431}$	$\boxed{1214}$
EGP	psia	51.19	50.42	41.11
W_g	LB/SEC	66.41	65.77	56.9
W_f	LB/HR	4754	4627	3319
W_a	LB/SEC	65.42	64.81	560
H_2O/W_a	%	0.0	0.0	0.0
FANS				
RPM (FWD)	% N_F	94.7	93.7	80.3
RPM (L/C)	% N_F	$\boxed{94.7}$	$\boxed{93.7}$	$\boxed{80.3}$
W_a (FWD)	LB/SEC	571.84	565.39	481.0
W_a (L/C)	LB/SEC	571.84	565.39	481.0
F_N (FWD)	LB	12560	12276	8910
F_N (L/C)	LB	12436	12154	8820
TOTAL PROPULSION SYSTEM				
LIFT	LB	$\boxed{37432}$	$\boxed{36584}$	$\boxed{26550}$
W_f	LB/HR	14262	13881	9957
LIFT SFC	LB/HR-LB	.381	.379	.375

*5 CIRCUITS FUEL

FIGURE 3-10
TURBOTIP RTA
(3) YJ97-GE-100/(3) LCF 459 FANS
ENGINE OUT LIFT RATINGS
INSTALLED VTOL PROPULSION SYSTEM PERFORMANCE
SLS, 89.8°F

Parameters		Power Ratings		
		Emergency Dry (T/W = 1.28)*	Intermediate Dry (T/W = 1.13)*	Reduced Power (T/W = 1.03)*
Gas Generator				
RPM	% N_{GG}	107.0	101.5	99.5
TIT	°F			
EGT	°F	1600	1418	1349
EGP	psia	54.47	49.79	46.92
W_g	lb/sec	68.08	65.70	63.04
W_f	lb/hr	5468	4570	4147
W_a	lb/sec	66.90	64.76	61.89
H_2O/W_a	%	0.0	0.0	0.0
H_2O	lb/sec	0.0	0.0	0.0
Fans				
RPM (Fwd)	% N_F	86.1	81.6	78.0
RPM (L/C)	% N_F	86.1	81.6	78.0
W_a (Fwd)	lb/sec	520.82	496.2	474.3
W_a (L/C)	lb/sec	520.82	496.2	474.3
F_N (Fwd)	lb	10,156	8955	8178
F_N (L/C)	lb	10,055	8866	8096
Total Propulsion System				
Lift	lb	30,266	26,687	24,370
W_f	lb/hr	10,936	9140	8294
Lift SFC	lb/hr-lb	.361	.342	.340

*2 Circuits Fuel

FIGURE 3-11
TURBOTIP RTA
INSTALLED CONTROL PERFORMANCE FOR THREE ENGINE VTO OPERATION
(3) YJ97-GE-100/(3) LCF 459 FANS
SEA LEVEL, 89.8°F

GROSS WEIGHT = 25286 LB (5 CIRCUITS FUEL)

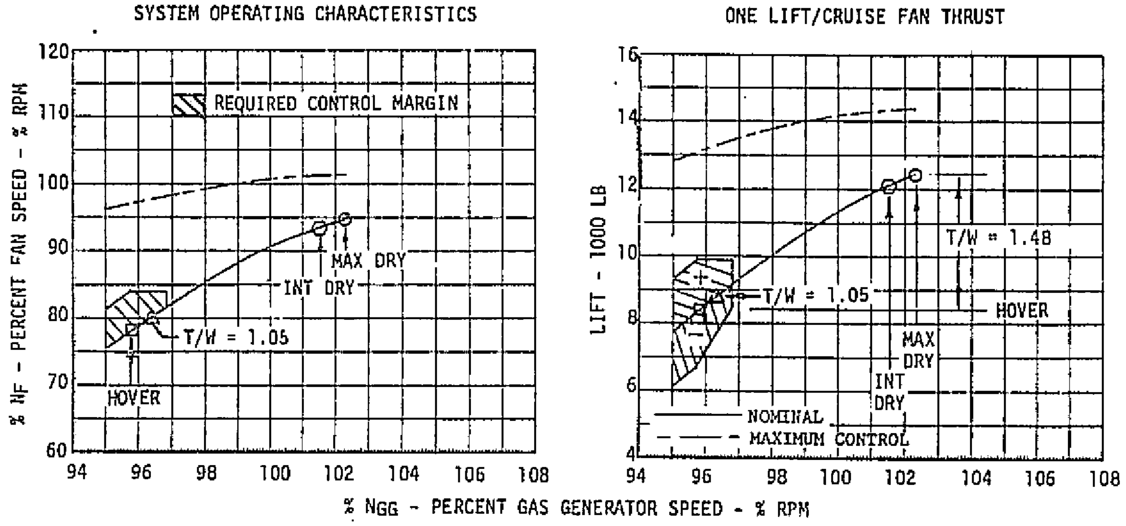


FIGURE 3-12
TURBOTIP RTA
INSTALLED CONTROL PERFORMANCE FOR ENGINE OUT VTO OPERATION
(3) YJ97-GE-100/(3) LCF 459 FAN SYSTEM
SEA LEVEL, 89.8°F
GROSS WEIGHT = 23660 LB (2 CIRCUITS FUEL)

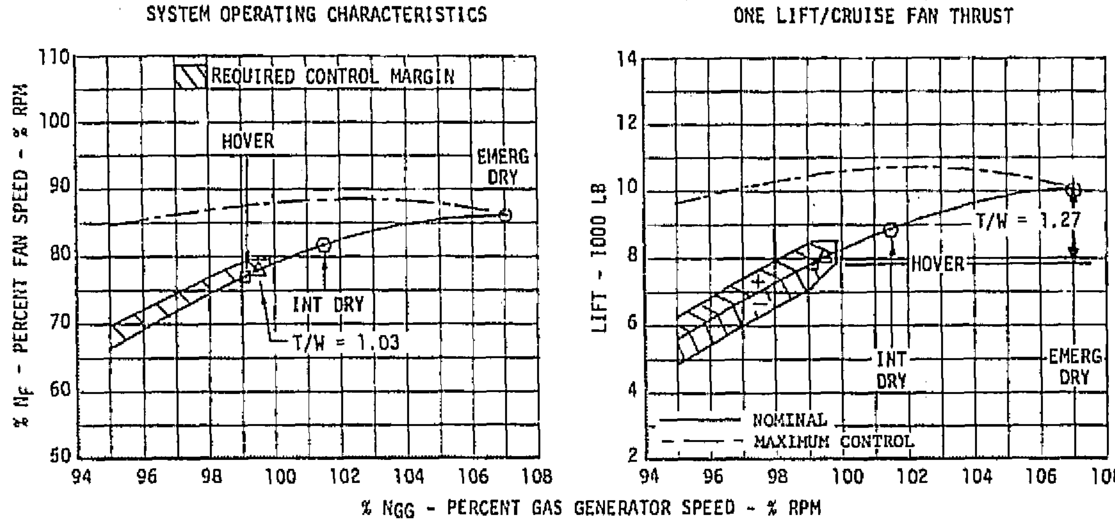
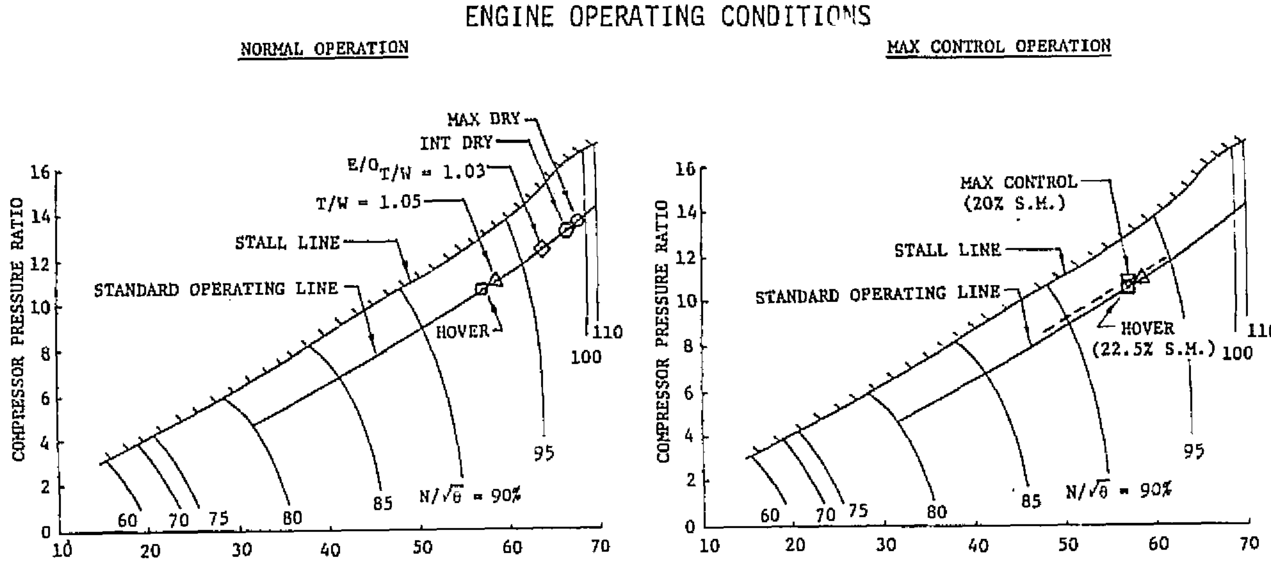


FIGURE 3-13
TURBOTIP RTA
ENGINE OPERATING CONDITIONS

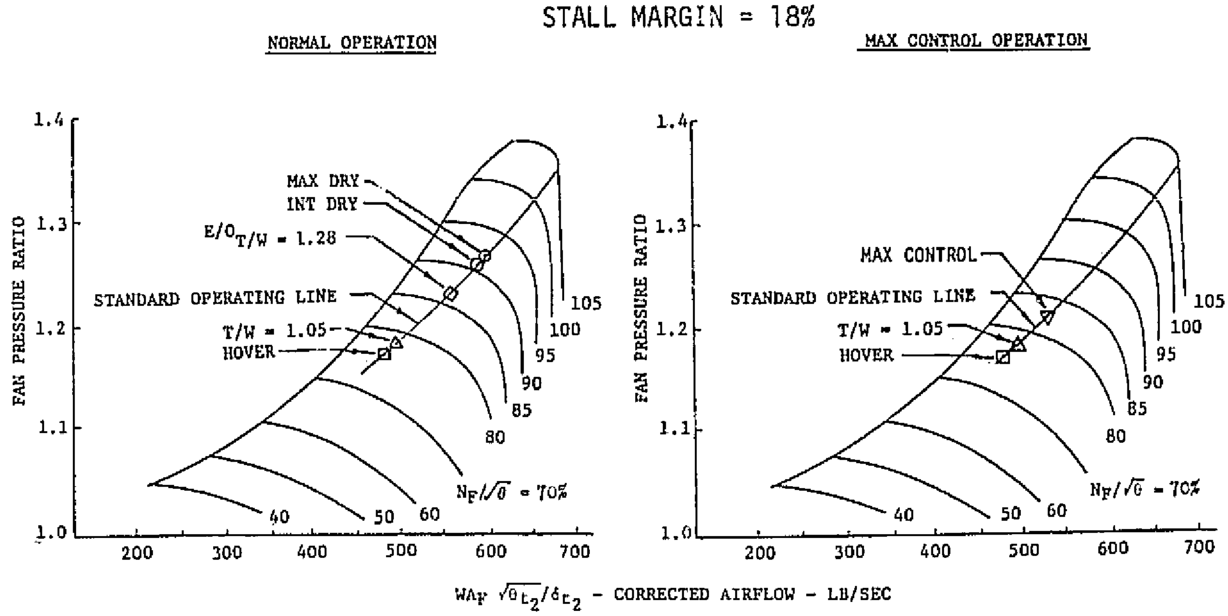


$$\text{STALL MARGIN} = \left(\frac{PR_{stall} - PR_{operating}}{PR_{operating}}\right)_{N/\sqrt{\theta}}$$

PR = Compressor pressure ratio

FIGURE 3-14
TURBOTIP RTA
FAN OPERATING CONDITIONS

STALL MARGIN = 18%



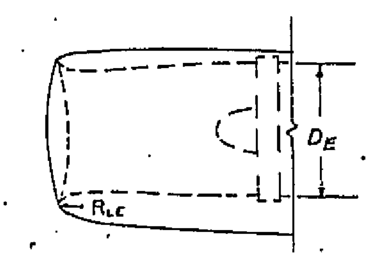
$$\text{STALL MARGIN} = \left(\frac{(PR/Wa_{corr})_{stall}}{(PR/Wa_{corr})_{oper.}} - 1 \right)_{N/\sqrt{\theta}}$$

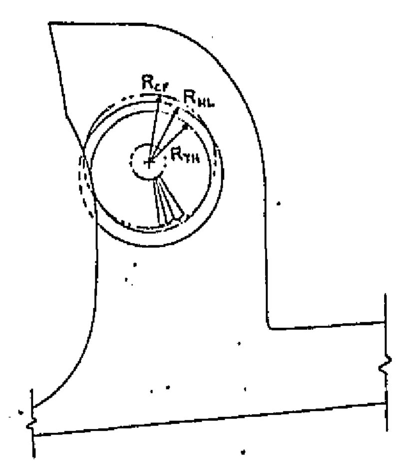
PR = Fan pressure ratio

$W_{a_{corr}} = W_{a_F} \sqrt{\theta_{t_2}}/\delta_{t_2}$

FIGURE 3-15
GAS GENERATOR SIDE INLET DESIGN GUIDELINES

- INTERNAL GEOMETRY
  - MAX INTERNAL WALL ANGLE (θ_{MAX}) = 5° at .5 L_D
  - CONTRACTION RATIO $(R_{HL}/R_{TH})^2$ = 1.40
  - INLET LIP CONTOUR = 2:1 ELLIPSE
  - INTERNAL DUCT CONTOUR = CUBIC CONTOUR
  - DIFFUSER LENGTH RATIO (L_D/D_E) = 1.39
  - ENGINE FACE DIAMETER (D_E) = 20.15 IN.
- EXTERNAL GEOMETRY
  - LIP LEADING EDGE RADIUS (R_{LE}) = b^2/a
  - COWL CONTOUR = DAC-3 SHAPE
- INLET AREAS
  - A_{HL} = 2.688 FT^2
  - A_{TH} = 1.920 FT^2
- INLET STATIC PERFORMANCE
  - PRESSURE RECOVERY ≥ .988





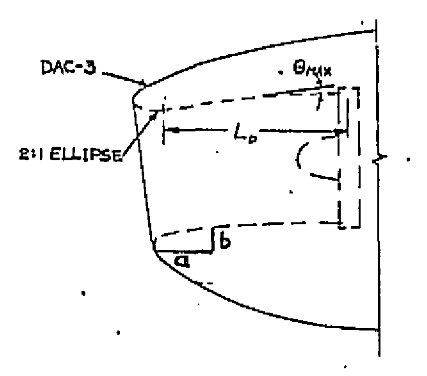
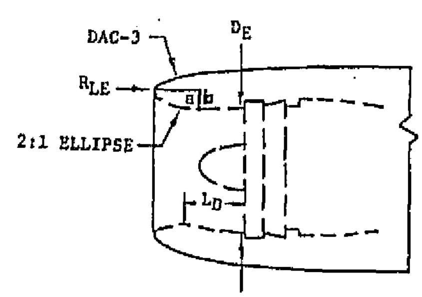
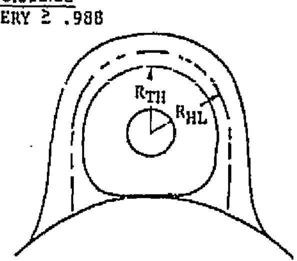


FIGURE 3-16
GAS GENERATOR TOP INLET DESIGN GUIDELINES
(FOR 3RD ENGINE)

- o INTERNAL GEOMETRY
  - MAX INTERNAL WALL ANGLE (θ_{MAX}) = 9° at .5 L_D
  - CONTRACTION RATIO $(R_{HL}/R_{TH})^2$ = 1.40
  - INLET LIP CONTOUR = 2:1 ELLIPSE
  - INTERNAL DUCT CONTOUR = CUBIC CONTOUR
  - DIFFUSER LENGTH RATIO (L_D/D_E) = .50
  - ENGINE FACE DIAMETER (D_E) = 20.15 IN.
- o EXTERNAL GEOMETRY
  - LIP LEADING EDGE RADIUS (R_{LE}) = b^2/a
  - COWL CONTOUR = DAC-3 SHAPE
- o INLET AREAS
  - A_{HL} = 2.710 FT^2
  - A_{TH} = 1.920 FT^2
- o INLET STATIC PERFORMANCE
  - PRESSURE RECOVERY ≥ .988





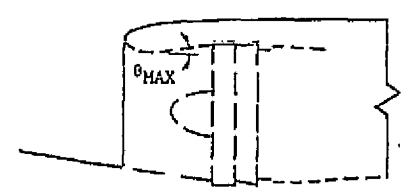
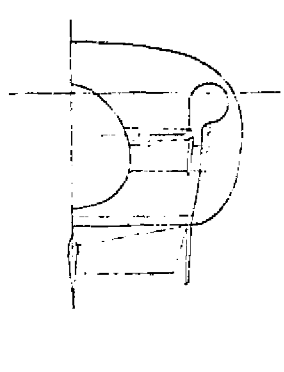


FIGURE 3-17
TURBOTIP RTA
NOSE FAN UNIT DESIGN GUIDELINES

- o INTERNAL GEOMETRY
  - INLET LIP CONTOUR = 1.41:1 ELLIPSE
  - CONTRACTION RATIO = 1.84
- o INLET AREAS
  - A_{HL} = 4185 $IN.^2$
  - A_{TH} = 2278 $IN.^2$
- o INSTALLED STATIC PERFORMANCE
  - INLET RECOVERY ≥ .988
  - NOZZLE THRUST COEFFICIENT = 0.95
- o VECTORING REQUIREMENTS
  - ARTICULATING VANES: 40° ≤ θ ≤ 105°
  - YAW VANES: ±16°



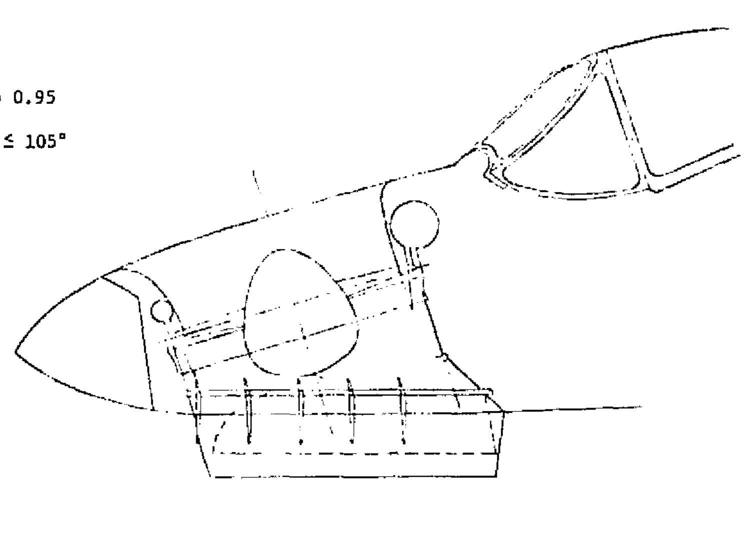


FIGURE 3-18
TURBOTIP RTA
LIFT/CRUISE UNIT DESIGN GUIDELINES

o INTERNAL GEOMETRY

- MAX INTERNAL WALL ANGLE (θ_{MAX}) = 7° @ .5 L_D
- INLET LIP THICKNESS RATIO (Y/R_{HL}) = .10
- INLET LIP CONTOUR = 2:1 ELLIPSE
- LIP LEADING EDGE RADIUS (R_{LIP}) = .05 R_{HL}
- INTERNAL DUCT CONTOUR = CUBIC CONTOUR

o INSTALLED STATIC PERFORMANCE

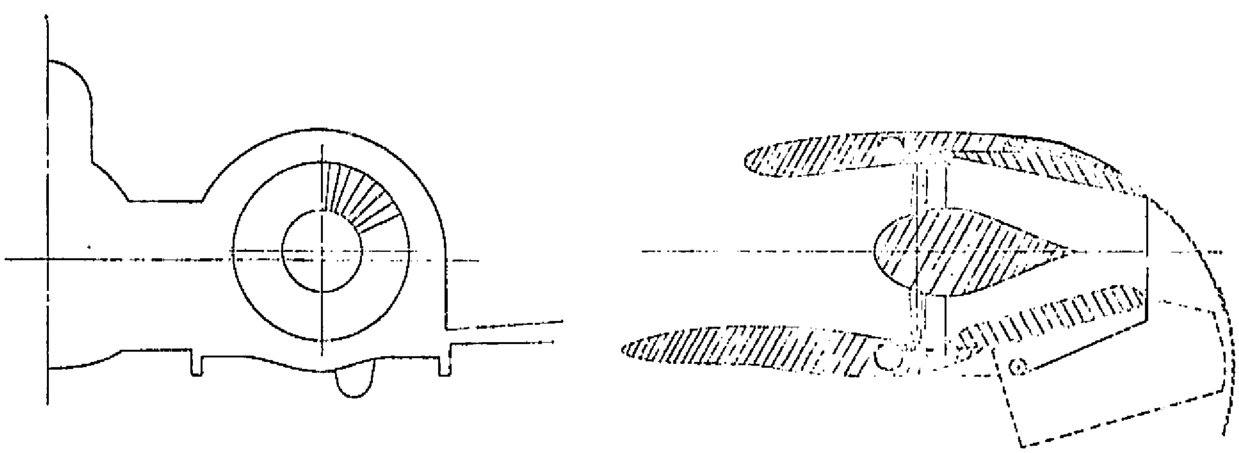
- INLET RECOVERY ≥ 0.988
- NOZZLE THRUST COEFFICIENT = 0.94

o EXTERNAL GEOMETRY

- COWL THICKNESS RATIO (R_{HL}/R_{MAX}) = .85
- COWL FINENESS RATIO [$L_C/(R_{MAX} - R_{HL})$] = 6.0
- COWL CONTOUR = DAC-3
- DRAG RISE MACH NUMBER = 0.80

o INLET AREAS

- THROAT = 14.5 FT^2
- HIGHLIGHT = 18.13 FT^2



3.1.7 INSTALLATION FACTORS - A summary of propulsion system installation factors for VTO operation is shown in Figure 3-19, along with the lift allowance associated with each factor. In addition to previously discussed allowances, General Electric included a 3% net thrust derate in their data. By direction of NASA, ground effects and reingestion allowances were not considered for this study.

Conventional flight installation factors are presented in Figures 3-20 and 3-21. The lift/cruise ETaC valve preset was removed for conventional flight, thus dropping the compressor operating line, and resulting in a slight decrease in intermediate power thrust, relative to a system with a smaller tip turbine area and no valve preset.

3.1.8 GENERAL ELECTRIC DESIGN CHANGES - Just prior to the conclusion of this study, General Electric simplified the scroll design, revised the method for achieving equal flow distribution, and revised fan and engine weights. However, these changes were not received in time to be included in this study. The changes and their impact on performance and weight are discussed below.

- o The new scroll design, referred to as the Valve-in-Scroll, is shown schematically in Figure 3-22. When compared with the baseline design,

FIGURE 3-19
TURBOTIP RTA
PROPULSION SYSTEM INSTALLATION FACTORS
VTO OPERATION

COMPONENT INSTALLATION LOSS	INSTALLATION FACTOR	TOTAL LIFT ALLOWANCE (3)
GAS GENERATORS		
INLET PRESSURE RECOVERY	0.988	1.52%
COMPRESSOR BLEED	1/2%	1.06
HORSEPOWER EXTRACTION	25 HP/ENG.	0.25
L/C FANS		
INLET PRESSURE RECOVERY	0.988	2.60
HORSEPOWER EXTRACTION	50 HP/FAN	0.31
NOZZLE THRUST COEFFICIENT	0.940	4.00
NOSE FAN		
INLET PRESSURE RECOVERY	0.988	1.30
NOZZLE THRUST COEFFICIENT	0.950	1.67
INTERCONNECTING DUCTING		
L/C FAN DUCT PRESSURE LOSS	10.3% (1)	4.12
NOSE FAN DUCT PRESSURE LOSS	10.3%	2.06
SCROLL PRESSURE LOSS	5%	3.00
ADDITIONAL PERFORMANCE ALLOWANCES		
THRUST DERATE	3% (2)	3.00
GROUND EFFECTS/REINGESTION	0%	0.00
		24.89%

(1) BASED ON PRESET ETaC VALVE ANGLE (ΔP = 6.1%) APPROACH FOR TRIMMING LIFT

(2) SUPPLIED BY ENGINE CO.

(3) BASED ON INSTALLED LIFT OF 26550 LB

FIGURE 3-20
TURBOTIP RTA
PROPULSION SYSTEM INSTALLATION FACTORS
CONVENTIONAL FLIGHT

COMPONENT INSTALLATION LOSS	NOMINAL INSTALLATION FACTOR
GAS GENERATORS	
INLET PRESSURE RECOVERY	SEE FIGURE 3-21
COMPRESSOR BLEED	1%
HORSEPOWER EXTRACTION	25 HP/ENG
L/C FANS	
INLET PRESSURE RECOVERY	SEE FIGURE 3-21
HORSEPOWER EXTRACTION	25 HP/FAN
NOZZLE THRUST COEFFICIENT	.98
INTERCONNECTING DUCTING	
L/C FAN DUCT PRESSURE LOSS	4.2%
NOSE FAN DUCT PRESSURE LOSS	--
SCROLL PRESSURE LOSS	5%
ADDITIONAL PERFORMANCE ALLOWANCES	
THRUST DERATE	3%
ENGINE BAY VENTILATION AND ECS DRAGS	(a)

NOTE: (a) 10% OF INTERMEDIATE POWER GAS GENERATOR RAM DRAG AT ALL MACH/ALTITUDE/POWER SETTINGS

FIGURE 3-21
GAS RTA
ESTIMATED INLET TOTAL PRESSURE RECOVERY
CONVENTIONAL FLIGHT

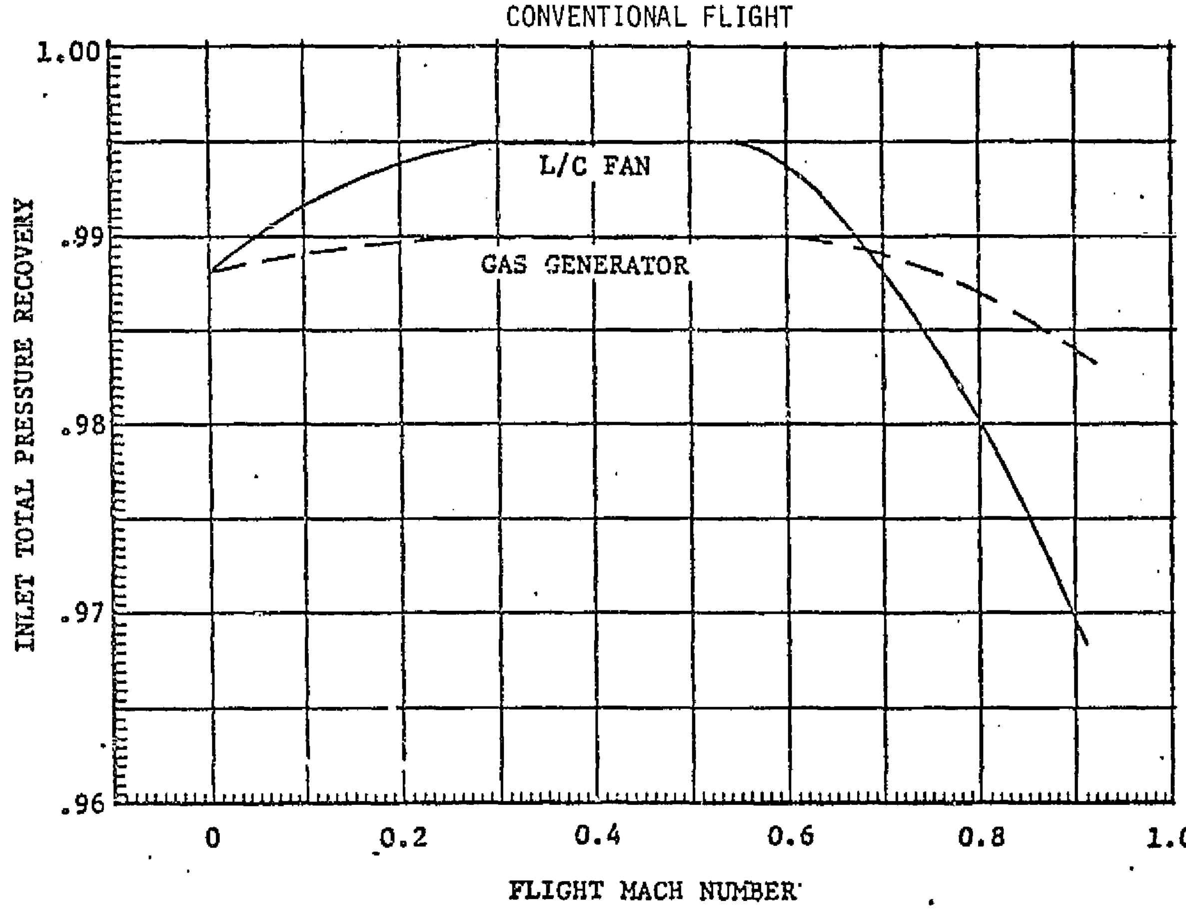
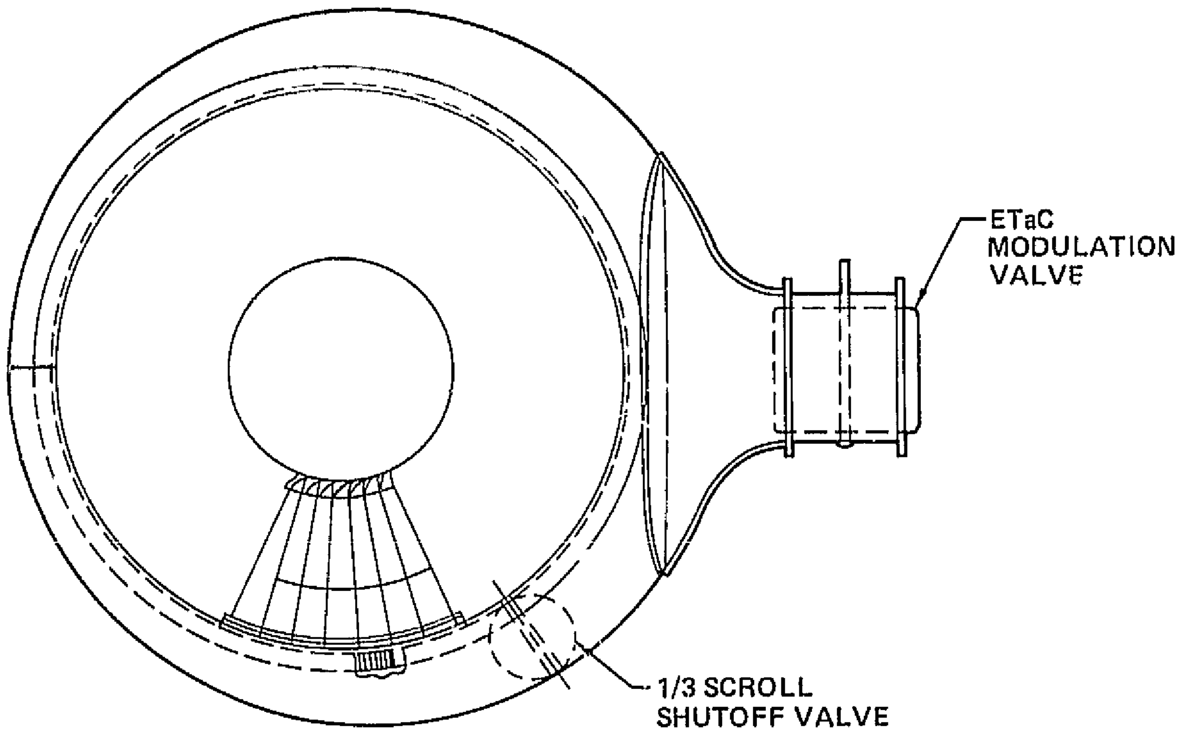


FIGURE 3-22
VALVE-IN-SCROLL CONCEPT



the inner scroll is replaced by a 120° arc with a wall on one end and a shutoff valve on the other. The ETaC valve remains a pressure control device only.

- o General Electric also devised a new method for obtaining equal flow distribution. By installing the turbine stator nozzles in the lift fan at ˒ 'ightly different angle than those in the lift/cruise fan, a larg˛ ˑ ˑit fan turbine stator nozzle effective area can be attained. This method obviates the need for the lift/cruise ETaC valve preset and removes its accompanying thrust penalty.
- o The fan weight increased because General Electric selected less expensive materials to reduce cost.

In addition to the items listed above, the engine weight decreased because the original weight included the exhaust nozzle, which is not required in MCAIR's installation. The impact of these changes on performance has been assessed and is shown in Figures 3-23 and 3-24. The most significant change is a 4.24% increase in engine out lift, which translates to an increase of 1132 pounds at Intermediate power.

FIGURE 3-23
TURBOTIP RTA
EFFECT OF GE CHANGES ON AIRCRAFT PERFORMANCE AND WEIGHT

PRESSURE LOSS FACTOR	VALUE (%)		LIFT ALLOWANCE (%)			
			NORMAL		ENG. OUT	
	BASELINE	CURRENT	BASELINE	CURRENT	BASELINE	CURRENT
o L/C ETaC VALVE	7.3	1.2	2.92	.48	2.92	.48
o SCROLL						
- NORMAL 360° OPER.	5.0	2.9	3.0	1.74	----	---
- ENG. OUT 240° OPER.	5.0	2.0	---	----	3.0	1.20
		TOTAL LIFT ALLOW. =	5.92	2.22	5.92	1.68
		TOTAL LIFT CHANGE =	+3.70		+4.24	

COMPONENT	WEIGHT PER UNIT (LB)	
	BASELINE	CURRENT
YJ97	739	719
LCF459	885	915

TOTAL WEIGHT CHANGE = +30 LB

FIGURE 3-24
TURBOTIP RTA
EFFECT OF GE CHANGES ON AIRCRAFT PERFORMANCE
INTERMEDIATE POWER, DRY

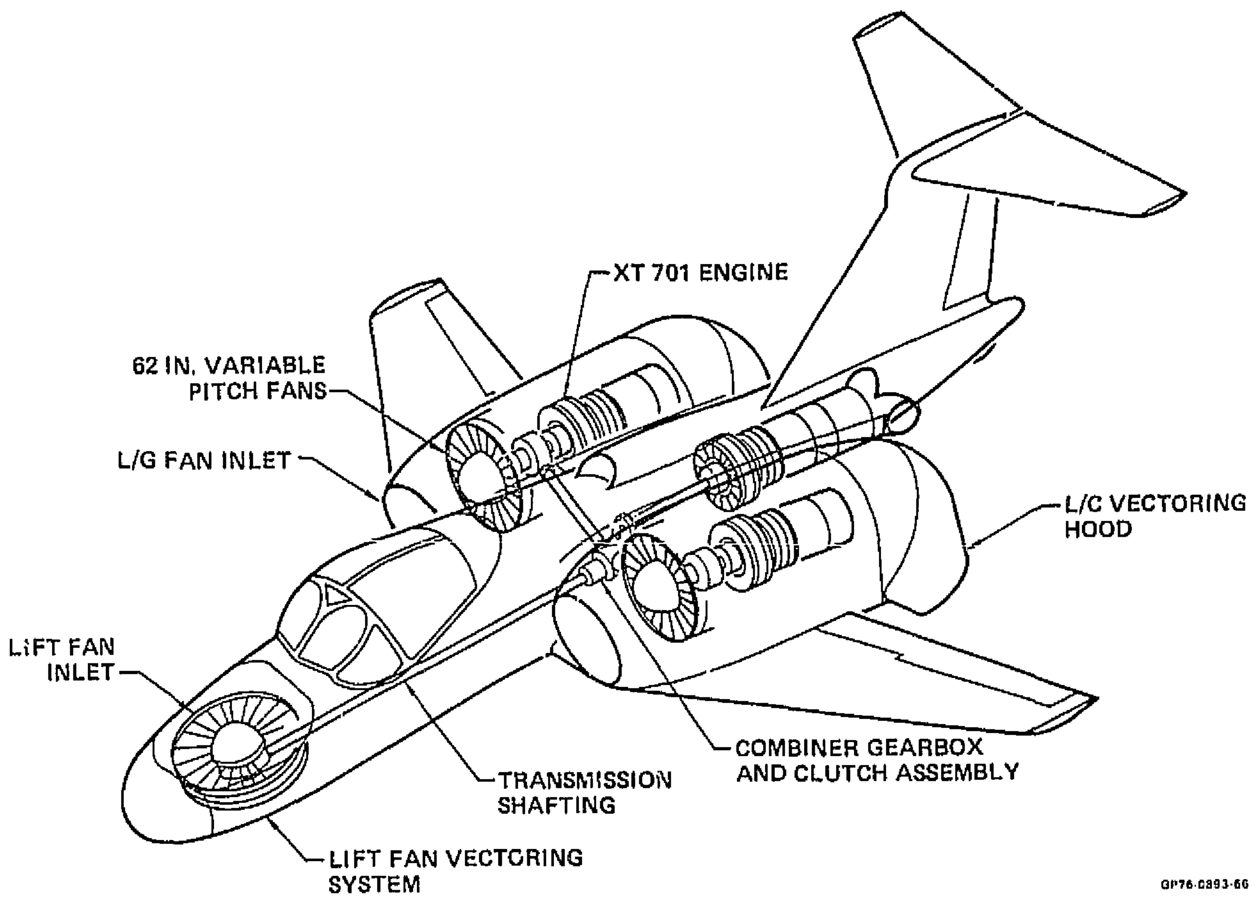
OPERATING CONDITION	BASELINE DESIGN		CURRENT GE DESIGN		THRUST INCREASE (LB)
	LIFT (LB)	T/W	LIFT (LB)	T/W	
NORMAL	36,584	1.45*	37,938	1.50*	1354
ENGINE OUT	26,687	1.13**	27,819	1.18**	1132

*5 CIRCUITS FUEL
**2 CIRCUITS FUEL

3.2 MECHANICAL RTA

3.2.1 CONFIGURATION - The mechanical RTA propulsion system, shown in Figure 3-25, consists of three Detroit Diesel Allison (DDA) XT701 turboshaft engines, two supercharged and one non-supercharged, and three variable pitch mechanically driven fans. As with the turbotip RTA, the nose lift fan and two over-the-wing mounted lift/cruise fans produce the required lift for VTOL operation. Aircraft control during the powered lift mode is achieved by utilizing variable fan blade pitch and power transfer between fans. During conventional flight, the third engine and the nose fan are shut down and the lift/cruise fans provide thrust for horizontal flight.

FIGURE 3-25
MECHANICAL RTA PROPULSION SYSTEM

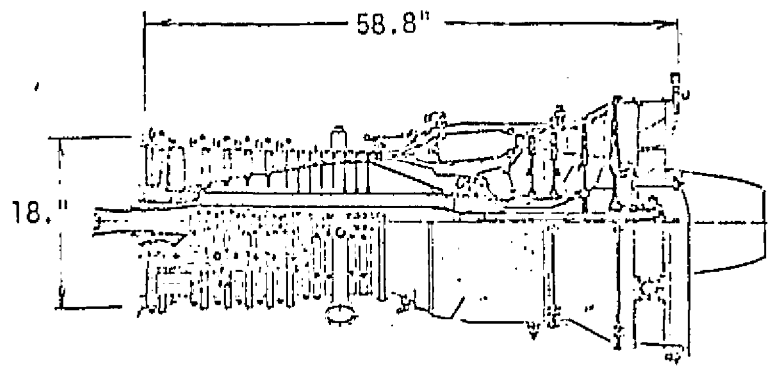


3.2.2 ENGINE AND FAN DESIGN - The XT701 turboshaft engine is a single-spool design with a 12-stage axial compressor, a two-stage high pressure turbine, and a two-stage low pressure turbine. Physical characteristics and uninstalled performance characteristics are presented in Figure 3-26. Engine testing accomplishments and availability are described in Figure 3-27.

The 62-inch Hamilton Standard (HS) variable pitch fan is a single stage, 1.20 pressure ratio design. The variable pitch capability allows fan thrust to be modulated without changing fan speed. A schematic of the nose fan along with installed, tropic day, static performance is shown in Figure 3-28.

The lift/cruise propulsion unit, referred to by DDA as PD370-25A, consists of a 62-inch variable pitch fan connected by a reduction gear to an XT701 turboshaft engine. Figure 3-29 includes a schematic of the lift/cruise unit together with available uninstalled performance characteristics.

FIGURE 3-26
XT701 TURBOSHAFT ENGINE



UNINSTALLED ENGINE CHARACTERISTICS
S.L.S., 59°F
INTERMEDIATE

N_{GG}, %RPM (RPM)	100.0 (15049)
CPR	12.3:1
AVG. PR/STAGE	1.21
WA_{GG}, LB/SEC	44.3
TIT, °R	
EGT, °R	2097
SHP	8079
W_{GAS}, LB/SEC	45.4
W_F, LB/HR	3780
WEIGHT, LB	765

FIGURE 3-27
XT701-AD-700 ENGINE DEVELOPMENT TESTING ACCOMPLISHMENTS

COMPONENT TESTS:	OVER 50 DIFFERENT TESTS	
COMPRESSOR RIG TESTS:	2 CONFIGURATIONS -	104 HR.
TURBINE HOT CASCADE:	1 CONFIGURATION -	8 HR. BURN 15 HR. AIR
COMBUSTOR RIG TESTS:	8 CONFIGURATIONS -	111 HR. BURN 349 HR. AIR
ENGINE TESTS:	1432 HR. TOTAL	581 HR. ENDURANCE

COMPLETED 30 HOUR PROTOTYPE PRELIMINARY FLIGHT RATING TEST	MARCH 1975
COMPLETED 60 HOUR SAFETY DEMONSTRATION TEST	AUGUST 1975

501-M62 ENGINE DEVELOPMENT TESTING ACCOMPLISHMENTS

ENGINE DEVELOPMENT AND ACCEPTANCE TESTING	496 HR.
BOEING VERTOL DSTR TESTING	418 HR.

TOTAL 501M62 & XT701-AD-700 ENGINE TEST TIME	2346 HR.
ENGINES AVAILABLE FOR RTA PROGRAM:	5 (IN ARMY INVENTORY)

FIGURE 3-28
MECHANICAL RTA
NOSE FAN UNIT

INSTALLED DESIGN CHARACTERISTICS
S.L.S., 90°F
INTERMEDIATE DRY

FAN DIA., IN	62
N_{FAN}, %RPM	100
N_{FAN}, RPM	
V_{TIP}, FT/SEC	932
W_{FAN}, LB/SEC	597.8
WEIGHT, NOSE FAN, LB	1047
FAN THRUST, LB	9461
THRUST/WEIGHT	9.04
$\Delta\beta$ FAN, DEGREES	-4
HP REQ'D	6159

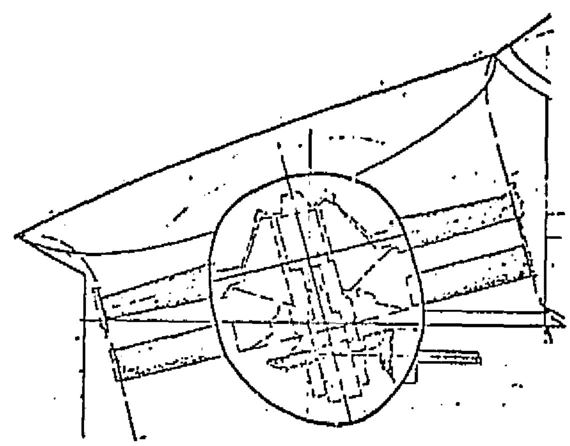
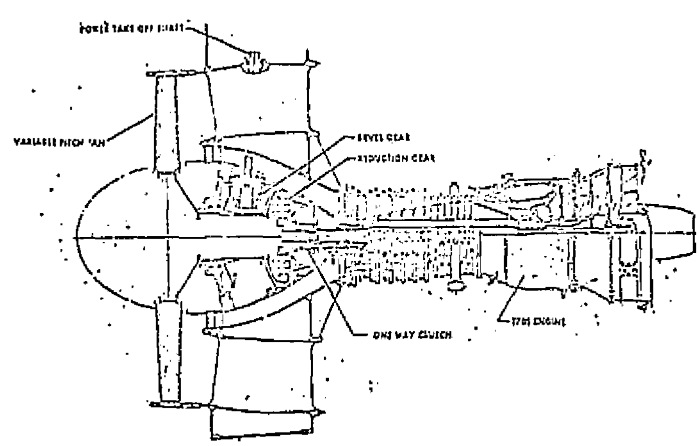


FIGURE 3-29
MECHANICAL RTA PROPULSION SYSTEM
LIFT/CRUISE UNIT

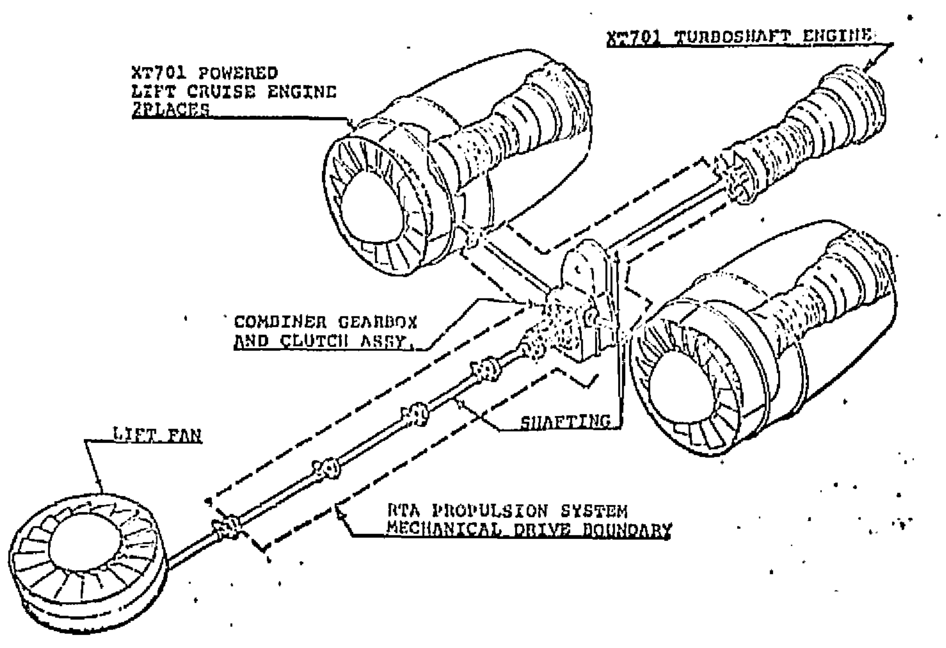
UNINSTALLED PERFORMANCE CHARACTERISTICS
S.L.S., 59°F
INTERMEDIATE

COMPRESSOR PRESSURE RATIO	12.3
OVERALL PRESSURE RATIO	14.8
WEIGHT, (L/C UNIT), LB	2361



3.2.3 POWER TRANSMISSION SYSTEM - The power transmission system transfers horsepower between the engines and fans, via mechanically linked shafts and gearboxes, as shown in Figure 3-30. The supercharged turboshaft engines provide horsepower to the lift/cruise fans through a reduction gear, and to a central combiner gearbox and clutch assembly through the same reduction gear and a right angle, power takeoff gear. The center, non-supercharged engine, also feeds into the combiner gearbox. A shaft from the combiner box runs forward to the nose, then through a right angle gearbox to drive the lift fan.

FIGURE 3-30
MODEL 260-RTA-2
PROPULSION SYSTEM



The transmission system is designed to distribute the available power to the lift and lift/cruise fans during all modes of operation. During STOL and VTOL operation the system distributes power to each fan as necessary to produce balanced lift and pitch and roll attitude control. Yaw control is achieved with lateral thrust deflection vanes in the thrust vectoring systems downstream of each fan. After conversion to wingborne flight, the third engine and nose fan are shut down, and power is delivered only to the two lift/cruise fans. Attitude control is then accomplished with conventional aerodynamic control surfaces. In the event of engine failure or shutdown of any one of the engines, during VTO or conventional flight, the transmission system distributes the remaining horsepower proportionately to each of the fans to maintain balanced lift and control.

3.2.4 <u>THRUST VECTORING AND THRUST MODULATION SYSTEMS</u> - The lift and lift/cruise fan thrust vectoring systems are essentially the same as those previously described for the turbotip propulsion system and illustrated in Figures 3-7 and 3-8. The primary difference is that TRM is not required on the mechanical system, due to the variable pitch capability.

3.2.5 PROPULSION SYSTEM PERFORMANCE

VTO Performance - Installed VTOL performance is summarized in Figure 3-31. For normal, three engine operation, intermediate power provides considerable excess thrust (T/W = 1.25) for a five circuit VTO mission. However, for engine out operation, water injection is necessary to produce enough fan thrust to provide "two circuit" engine out capability. Additional VTOL and cruise performance, ram drag characteristics, component geometric descriptions, and other detailed data on the mechanical RTA propulsion system are included in Appendix C.

Control Performance - As previously discussed, aircraft control during powered lift is achieved by varying fan blade pitch. Figure 3-32 provides a summary of powered lift control performance and indicates that a significant control margin is available for a single lift/cruise fan running at constant speed. The data available at this time was based on partial fan maps and did not encompass most of the operational points. Therefore, a large amount of the performance data used in the analysis was determined by extrapolation. The excess thrust available for normal operation would permit the RTA to be loaded to a higher gross weight, if this was desired for additional STOL research.

Fan Operating Conditions - Approximately one week prior to the conclusion of the study, variable pitch fan maps were received from Hamilton Standard for a range of blade angles from a $\Delta\beta$ of +7.3° to -17°. The fan maps for two blade angle positions, $\Delta\beta$ = 0° and $\Delta\beta$ = +7.3°, with the selected fan operating line superimposed are shown in Figures 3-33 and 3-34. Based on the map data, fan lift and stall margin characteristics as a function of $\Delta\beta$ at a corrected fan speed of 100 percent were determined for both normal operation, Figure 3-35, and engine out operation, Figure 3-36. The available lift line is shown on these curves and varies somewhat from that shown in Figure 3-32. During maximum control, stall margins of 23% during normal operation and 27% during engine out operation are available. These curves can also be used to interpret the effect on stall margin as VTOGW increases. For example, if a 31,000 lb VTOGW were used with the engines operating at intermediate power the maximum control point would be at a $\Delta\beta$ of +7.3° resulting in a stall margin of less than 10 percent, Figure 3-37.

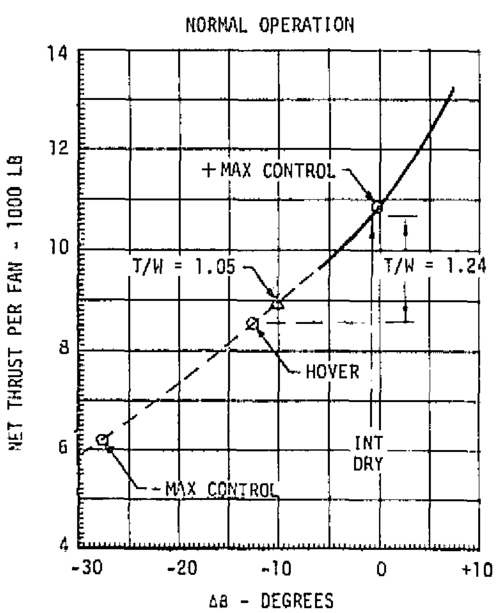
3.2.6 COMPONENT DESIGN GUIDELINES - Nose fan unit, lift/cruise unit, and third engine inlet design guidelines are presented in Figures 3-38, 3-39, and 3-40, respectively. The lift and lift/cruise inlets were sized to achieve 0.988

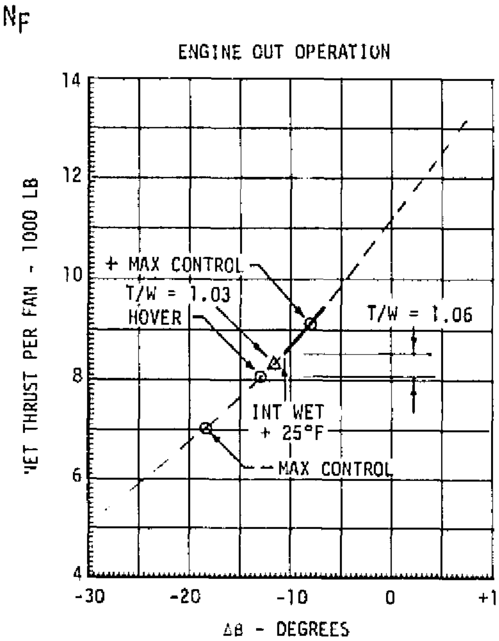
FIGURE 3-31
MECHANICAL RTA
INSTALLED VTOL PROPULSION SYSTEM PERFORMANCE
SLS, 89.8°F

PARAMETERS		POWER RATINGS			
		NORMAL INT. DRY (T/W=1.25)(1)	NORMAL (MCAIR EST) (T/W=1.05)(1)	ENGINE OUT INT. WET+25° (T/W=1.06)(2)	ENGINE OUT (MCAIR EST) (T/W=1.03)(2)
SUPERCHARGED GAS GENERATOR (CORE)					
RPM	% N_{GG}	104.6	101.1	108.1	106.5
BOT	°F	2290	2077	2375	2297
W_A	LB/SEC	44.9	40.1	45.9	45.2
W_F	LB/HR	3618	2846	3887	3739
W_{GAS}	LB/SEC	45.9	40.9	47.0	46.2
W_{H2O}	LB/SEC	0	0	1.4	1.36
UNSUPERCHARGED GAS GENERATOR (CORE)					
RPM	% N_{GG}	100.8	97.2	108.0	106.4
BOT	°F	2290	2077	2375	2297
W_A	LB/SEC	41.3	38.5	41.7	41.1
W_F	LB/HR	3346	2741	3579	3430
W_{GAS}	LB/SEC	42.2	39.3	42.7	42.1
W_{H2O}	LB/SEC	0	0	1.3	1.23
FANS					
RPM (LIFT & L/C FANS)	% N_F	103.0	103.0	103.0	103.0
W_A (L/C FAN)	LB/SEC	638.1	581.8	563.5	554.2
W_A (LIFT FAN)	LB/SEC	641.8	583.3	571.5	562.1
$\beta_{L/C}/\beta_{LIFT}$	DEGREES	-0.40/-0.10	-10.0/-9.85	-10.60/-9.30	-13.20/-11.90
TOTAL PROPULSION SYSTEM					
LIFT	LB	32,138	26,953	25,756	24,978
W_F	LB/HR	10,582	8433	7466	7169
LIFT SFC	LB/HR/LB	.329	.313	.290	.287

(1) 5 CIRCUITS FUEL
(2) 2 CIRCUITS FUEL

FIGURE 3-32
MECHANICAL RTA CONTROL PERFORMANCE
LIFT/CRUISE FAN
SLS, 89.8°F
100% N_F





——— DDA DATA
----- MCAIR EXTRAPOLATION

FIGURE 3-33
VARIABLE PITCH FAN MAP (62" DIA.) - Δβ = 0°

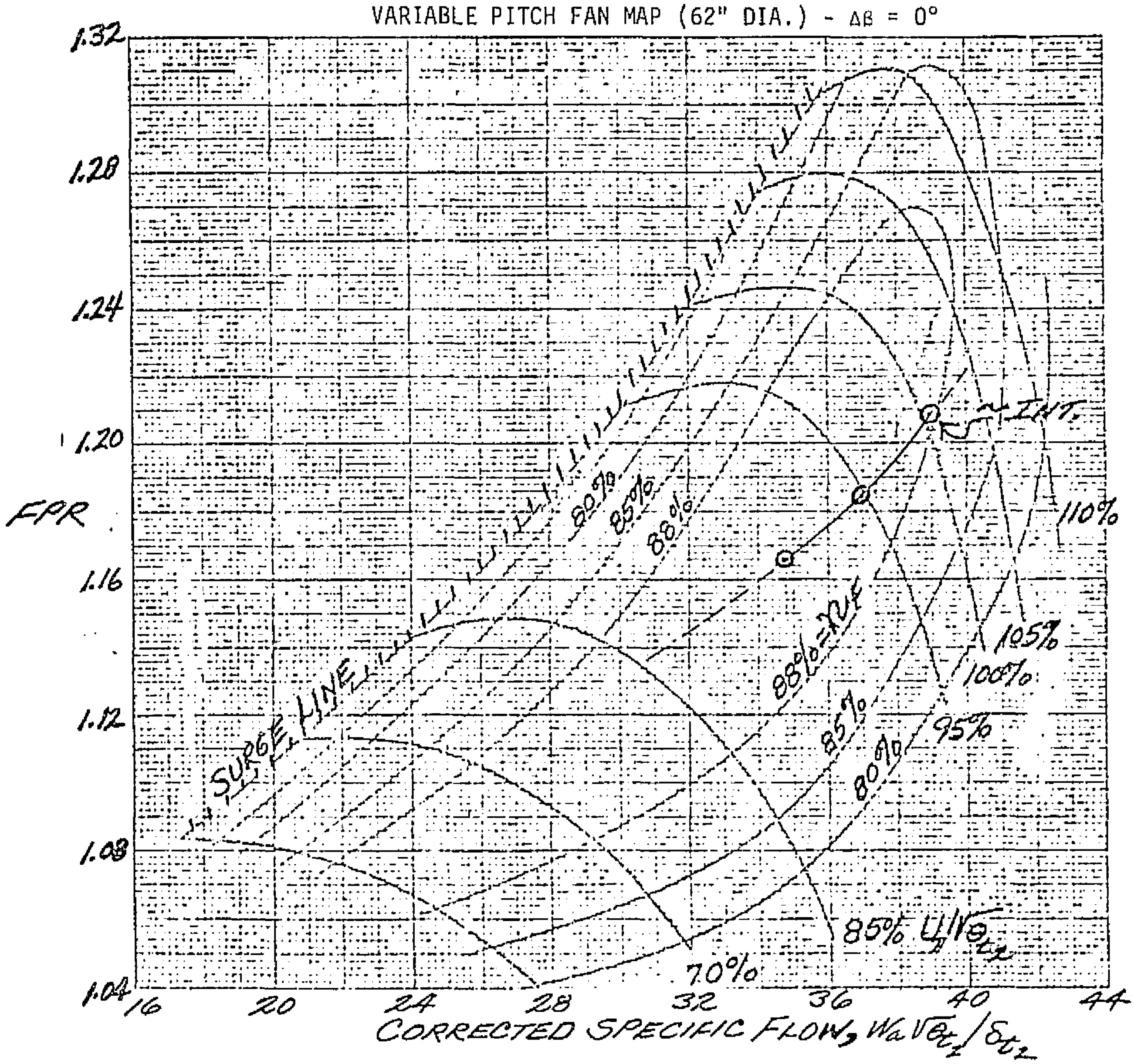


FIGURE 3-34
VARIABLE PITCH FAN MAP (62" DIA.) - $\Delta\beta$ = 7.3°

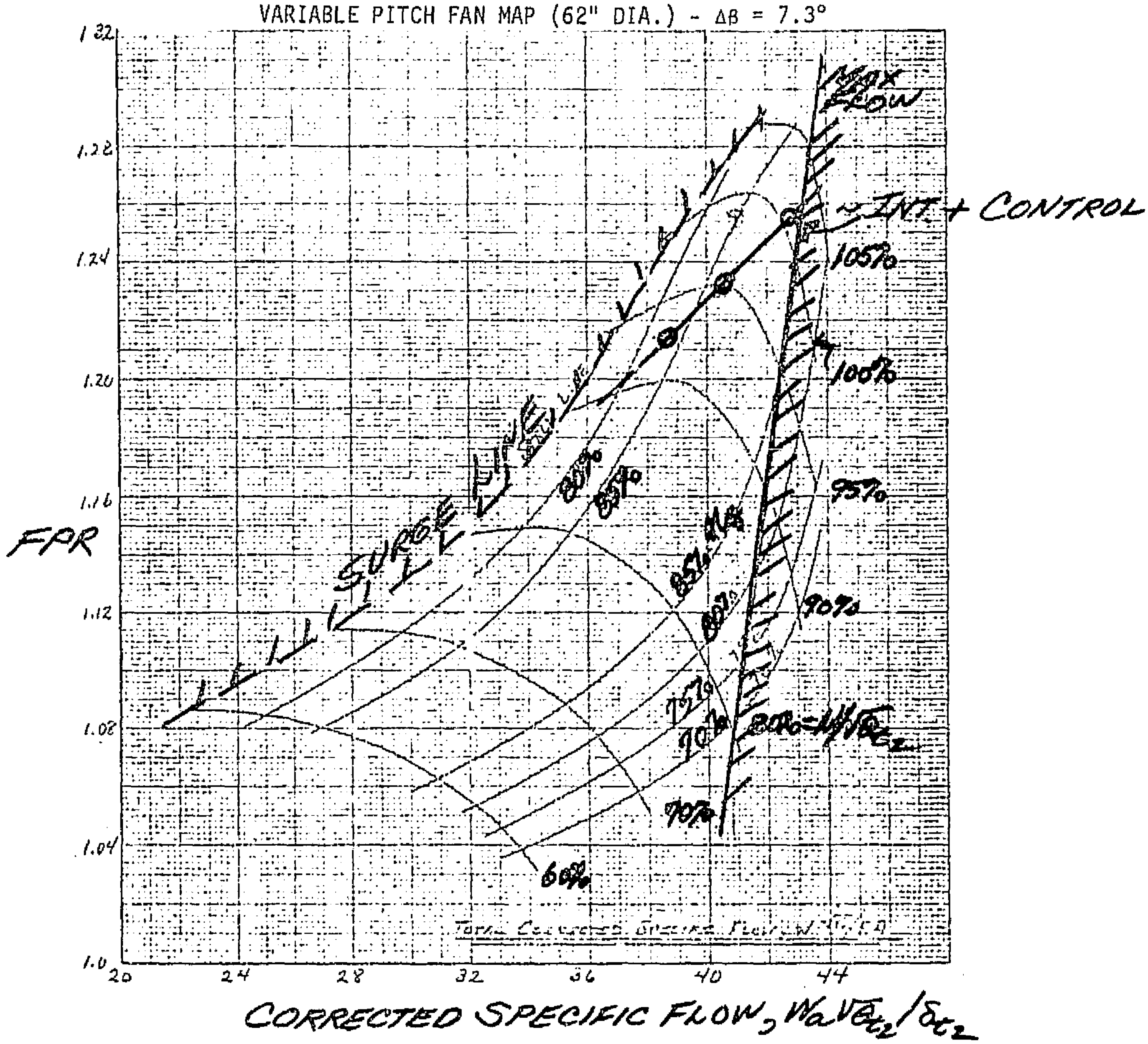


FIGURE 3-35
MECHANICAL RTA
62" FAN STALL MARGIN, NORMAL OPERATION
SLS, 89.8°F

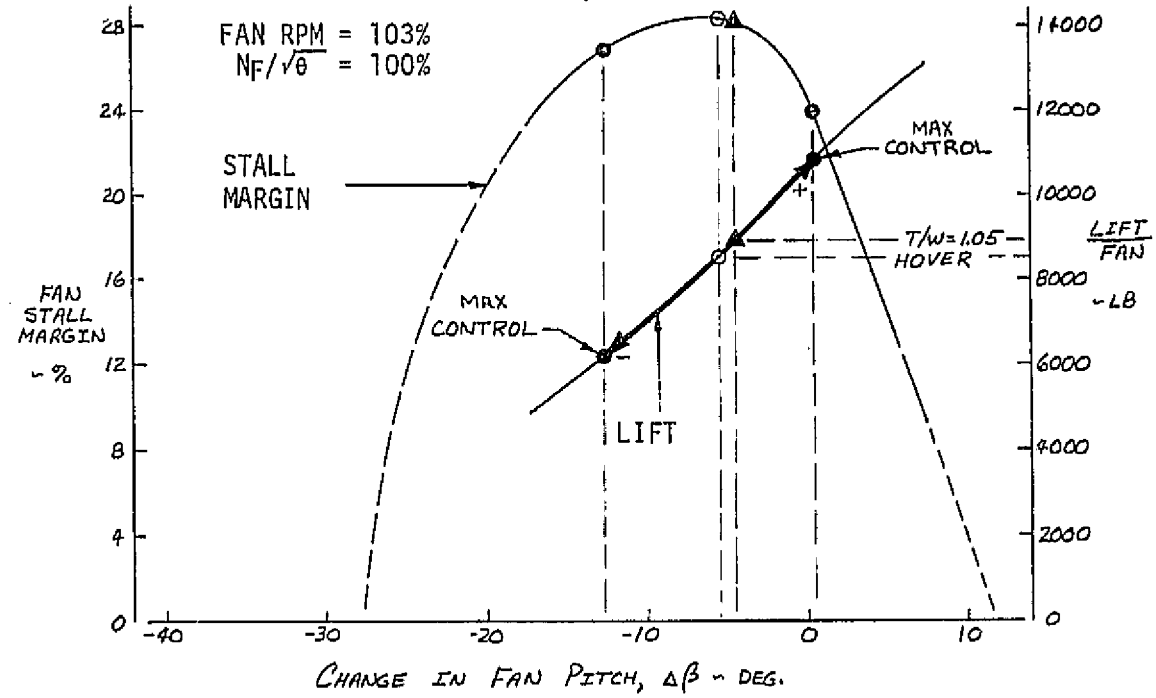


FIGURE 3-36
MECHANICAL RTA
62" FAN STALL MARGIN, ENGINE OUT OPERATION
SLS, 89.8°F

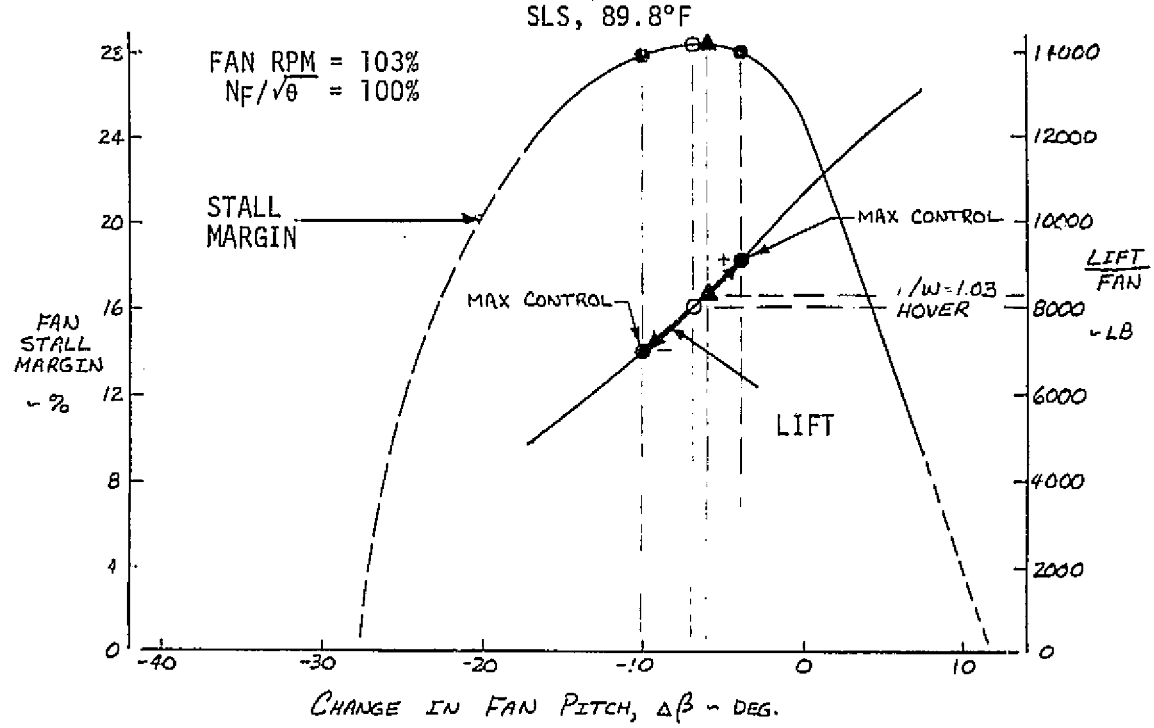


FIGURE 3-37
MECHANICAL RTA
62" FAN STALL MARGIN, INTERMEDIATE POWER (T/W = 1.05)
SLS, 89.8°F

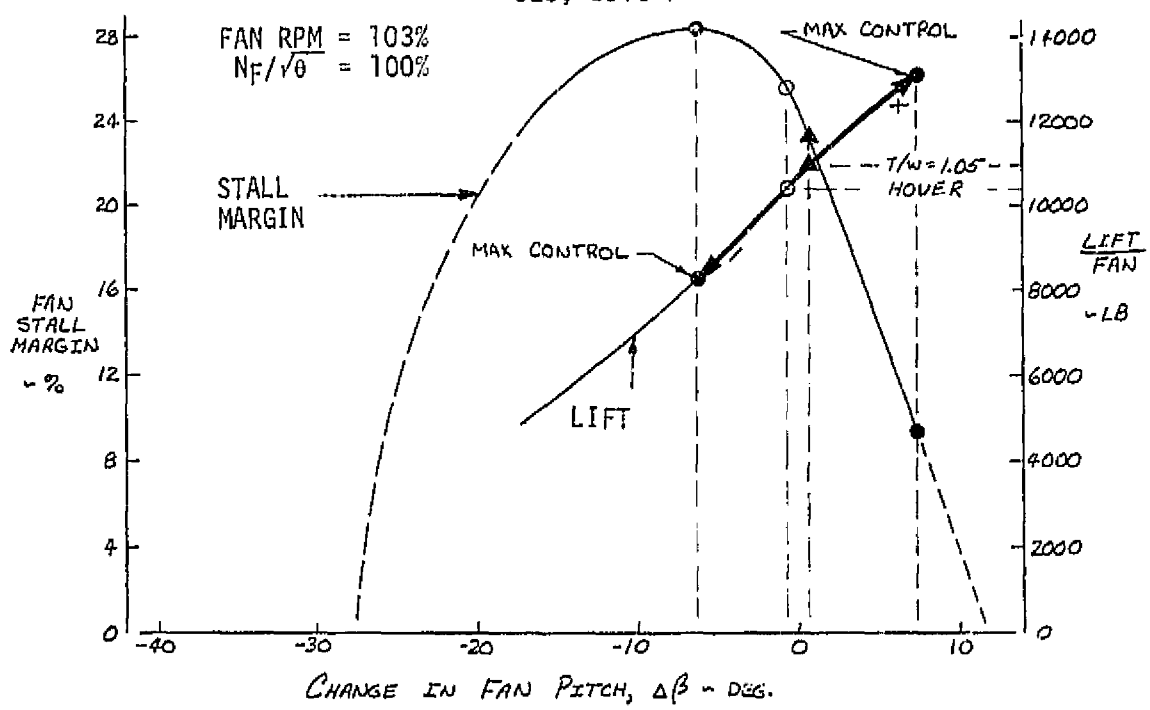


FIGURE 3-38
MECHANICAL RTA
NOSE FAN UNIT DESIGN GUIDELINES

- <u>INTERNAL GEOMETRY</u>
  - INLET LIP CONTOUR = 2:1 ELLIPSE
  - CONTRACTION RATIO = 1.45
- <u>INLET AREAS</u>
  - A_{HL} = 3584 IN.^{2}
  - A_{TH} = 2472 IN.^{2}
- <u>INSTALLED STATIC PERFORMANCE</u>
  - INLET RECOVERY = .988
  - NOZZLE THRUST COEFF. = 0.95
- <u>VECTORING REQUIREMENTS</u>
  - ARTICULATED VANES: 40° ≤ δ ≤ 105°
  - YAW VANES: ±16°

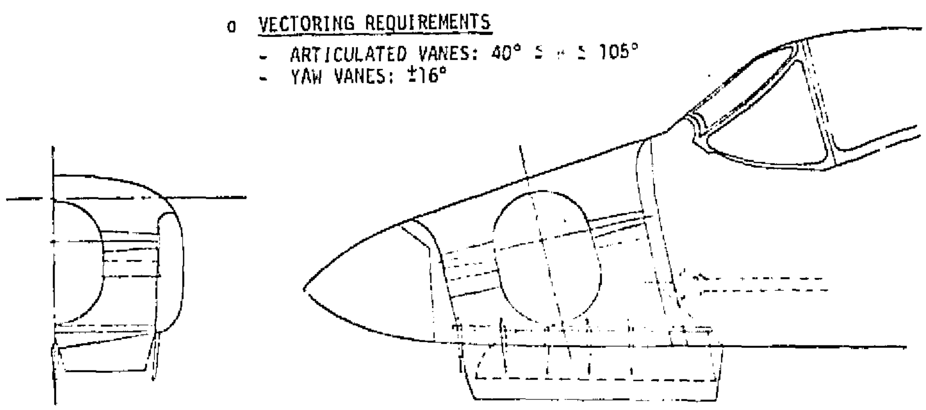


FIGURE 3-39
MECHANICAL RTA
LIFT/CRUISE UNIT DESIGN GUIDELINES

- o <u>INTERNAL GEOMETRY</u>
  - MAX INTERNAL WALL ANGLE (θ_{MAX}) = 7° @ .5 L_D
  - INLET LIP THICKNESS RATIO (Y/R_{HL}) = .10
  - INLET LIP CONTOUR = 2:1 ELLIPSE
  - LIP LEADING EDGE RADIUS (R_{LIP}) = .05 R_{HL}
  - INTERNAL DUCT CONTOUR = CUBIC CONTOUR
- o <u>EXTERNAL GEOMETRY</u>
  - COWL THICKNESS RATIO (R_{HL}/R_{MAX}) = .85
  - COWL FINENESS RATIO [$L_C/(R_{MAX} - R_{HL})$] = 6.0
  - COWL CONTOUR = DAC-3 SHAPE
- o <u>INLET AREAS</u>
  - THROAT = 16.63 FT^2
  - HIGHLIGHT = 20.79 FT^2
- o <u>INSTALLED STATIC PERFORMANCE</u>
  - INLET RECOVERY = .988
  - NOZZLE THRUST COEFFICIENT = .940

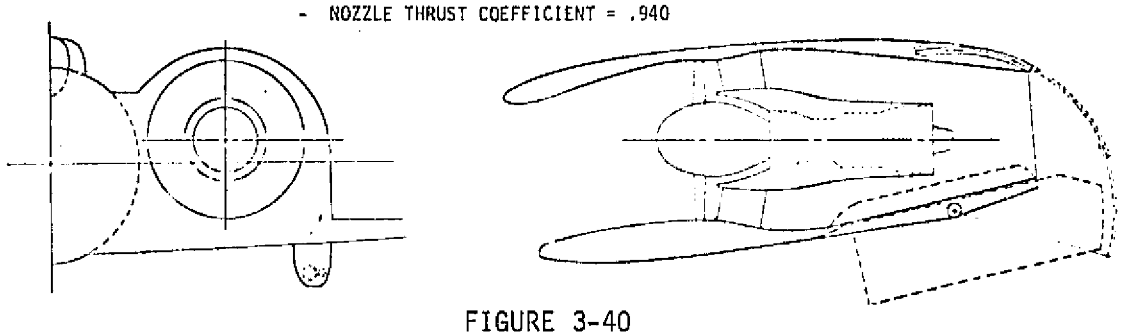
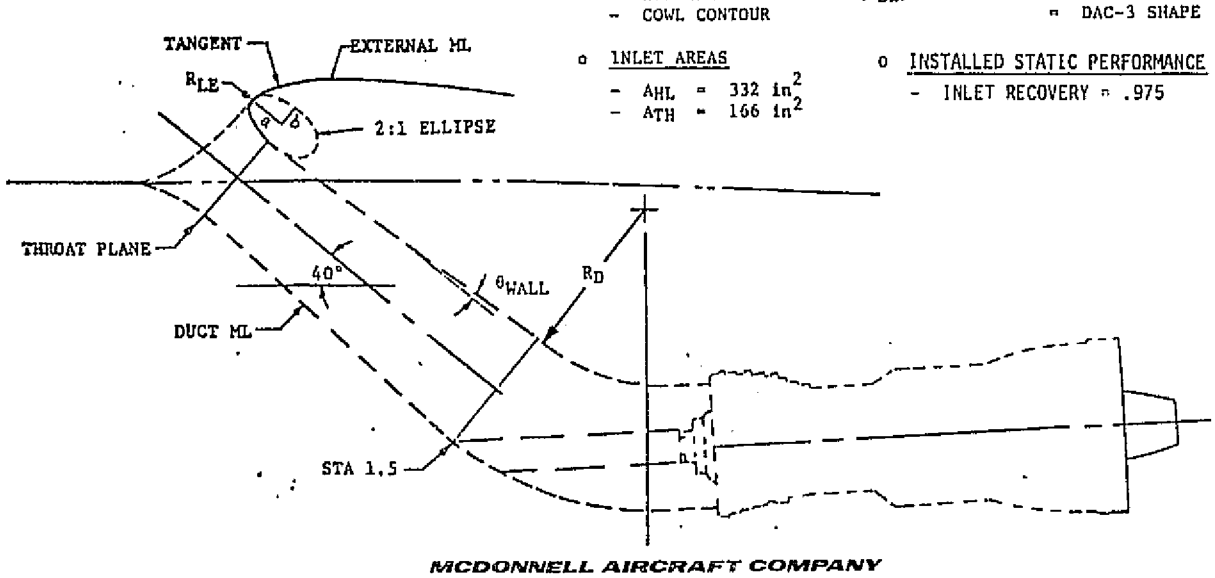


FIGURE 3-40
MECHANICAL RTA
THIRD ENGINE INLET DESIGN GUIDELINES

- o <u>INTERNAL GEOMETRY</u>
  - CONTRACTION RATIO $(R_{HL}/R_{TH})^2$ = 2.0
  - INLET LIP CONTOUR = 2:1 ELLIPSE
  - ENGINE FACE DIAMETER (D_E) = 18.4 in.
  - DIFFUSER WALL ANGLE ($2\theta_{WALL}$) = 7°
  - DUCT INSIDE TURN RADIUS (R_D) = 27 in.
  - TURN CONTRACTION RATIO $(R_{STA\ 15}/R_E)^2$ = 1.18
- o <u>EXTERNAL GEOMETRY</u>
  - LIP LEADING CONTOUR (R_{LE}) = b^2/a
  - COWL CONTOUR = DAC-3 SHAPE
- o <u>INLET AREAS</u>
  - A_{HL} = 332 in^2
  - A_{TH} = 166 in^2
- o <u>INSTALLED STATIC PERFORMANCE</u>
  - INLET RECOVERY = .975



total pressure recovery during static operation. The third engine inlet recovery is slightly lower, 0.975, because of the inlet ducting required for this engine location. As with the turbotip propulsion system, to keep costs as low as possible, auxiliary inlets were not employed on the mechanical system.

3.2.7 INSTALLATION FACTORS - A summary of propulsion system installation factors for VTO operation and the lift allowance associated with each factor are shown in Figure 3-41. No engine company derate was included in the data received; and, as stated previously in the turbotip system description, no ground effects or reingestion allowances were used in this study. DDA assumed 100% supercharging for all performance, and included a 2% fan horsepower allowance due to flow past a dead engine during engine out operation.

One other installation factor, lift/cruise nozzle thrust coefficient, deserves additional comment. In the fan performance data received, for operating conditions of interest, the fan stator exit static pressure was less than ambient static pressure. Entering the "D" vented nozzle performance map, Figure 3-42, at a static pressure ratio corresponding to the engine out operating condition, would result in a VTO thrust coefficient that would not provide sufficient lift for engine out safety. Since the nozzle performance is sensitive to entrance static pressure, a procedure was required to increase this pressure, and thus improve the thrust coefficient. The method selected involved diffusing the flow in both the fan stators and the fan duct downstream, before reaching the nozzle venting lip. Hamilton Standard provided data indicating the amount of diffusion which could be accomplished in the stators and the associated fan performance and stall margin penalties. Performance with and without this stator diffusion is shown in Figure 3-43. Diffusing from 0.96 to 1.03 static pressure ratio resulted in approximately 1% loss in gross fan thrust and a 7% loss in fan stall margin, from 27% to 20%. However, with the higher fan exit static pressure plus additional duct diffusion, the thrust coefficient was raised to 0.95, Figure 3-42. When the 1% loss in gross thrust is taken into consideration, the effective thrust coefficient reduced to 0.94. The nose fan nozzle thrust coefficient of 0.95 was achieved in a similar manner employing additional stator diffusion.

Conventional flight installation factors are presented in Figures 3-44 and 3-45.

FIGURE 3-41
MECHANICAL RTA
PROPULSION SYSTEM INSTALLATION FACTORS
VTO OPERATION

COMPONENT	NOMINAL INSTALLATION FACTOR	TOTAL LIFT[e] ALLOWANCE %
TURBOSHAFT ENGINES		
PRESSURE RECOVERY[d]	.975	1.04
COMPRESSOR AIRBLEED[a] (LB/SEC)	0.10	0.52
HORSEPOWER EXTRACTION[a]	125[c] 0[d]	1.23
L/C FAN AND NACELLE		
PRESSURE RECOVERY	.988	4.00
DUCT PRESSURE LOSS (FAN COLD STREAM)	(b)	3.34
NOZZLE THRUST COEFFICIENT	.94	4.00
LIFT FAN SYSTEM		
PRESSURE RECOVERY	.988	2.00
NOZZLE THRUST COEFFICIENT	.95	1.67
CENTER GEARBOX		
HORSEPOWER EXTRACTION	0	0
HORSEPOWER LOSS	(b)	1.33
ADDITIONAL PERFORMANCE ALLOWANCES		
NET THRUST DERATE	0	0
SUPERCHARGING	100%	0
FAN HP ALLOWANCE DUE TO DEAD ENGINE	2%	1.33
GROUND EFFECTS/REINGESTION	0	0
		20.46

NOTES:
(a) PER ENGINE
(b) FURNISHED BY ENGINE CO.
(c) SUPERCHARGED ENGINE
(d) NON-SUPERCHARGED ENGINE
(e) BASED ON INSTALLED LIFT OF 26953 LB

FIGURE 3-42
MECHANICAL RTA
"D" VENTED NOZZLE PERFORMANCE
90° VECTOR

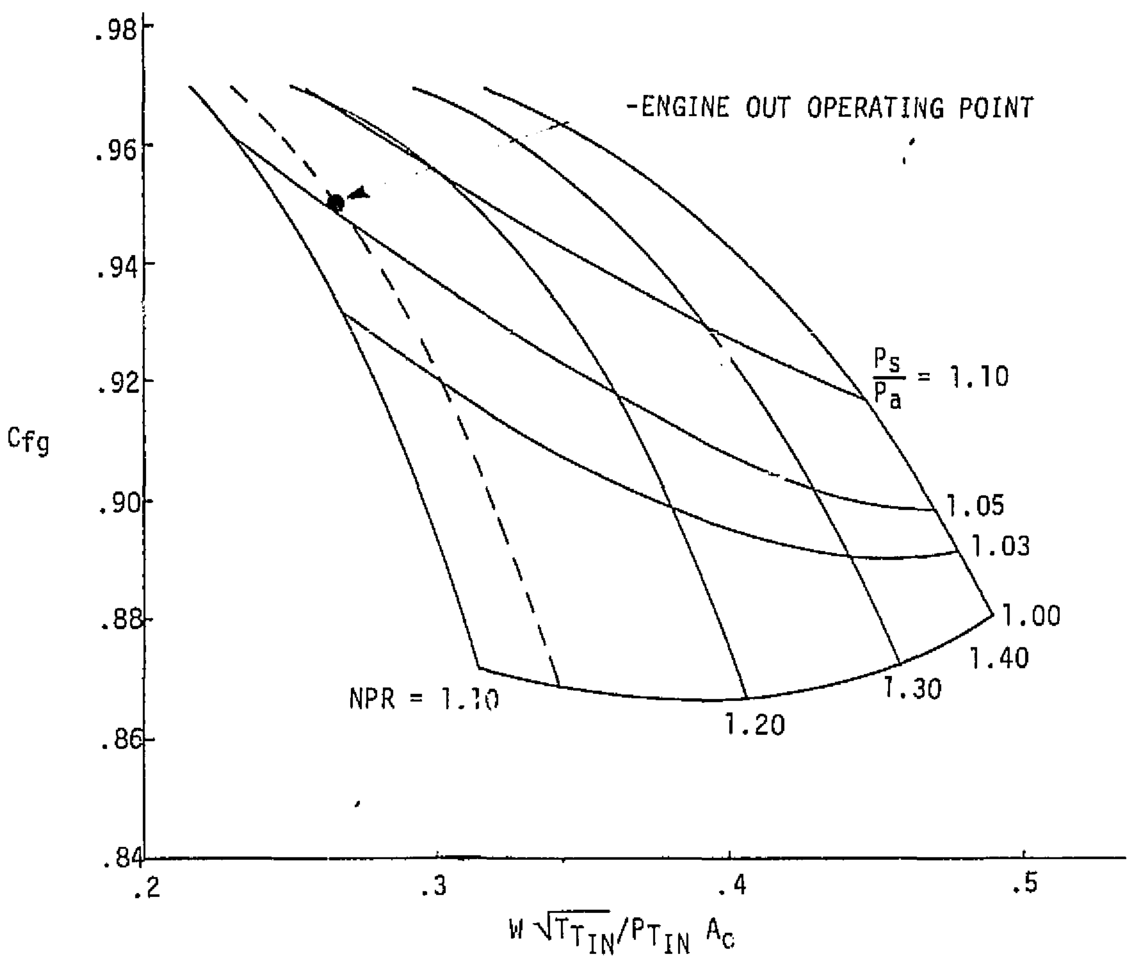
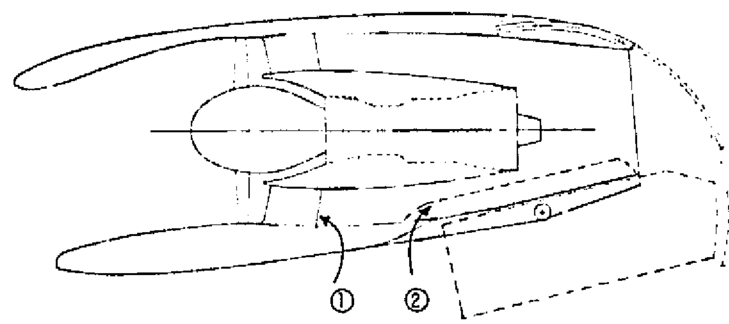


FIGURE 3-43
MECHANICAL RTA
EFFECT OF STATOR DIFFUSION OF FAN & NOZZLE STATIC PERFORMANCE

WITHOUT DIFFUSION IN STATOR

- o STATION ①
  P_s/P_a = .96
  M_{EXIT} = .501
- o STATION ②
  P_sP_a = 1.0
  M_{EXIT} = .427
- o $Cf_{g_{EFF.}}$ = .879
- o A_{NOZ} = 5780 IN.^{2}



WITH DIFFUSION IN STATOR PLUS DOWNSTREAM

- o STATION ①
  P_s/P_a = 1.03
  M_{EXIT} = .401
- o STATION ②
  P_s/P_a = 1.07
  M_{EXIT} = .282
- o $Cf_{g_{EFF.}}$ = .940
- o A_{NOZ} = 5370 IN.^{2}

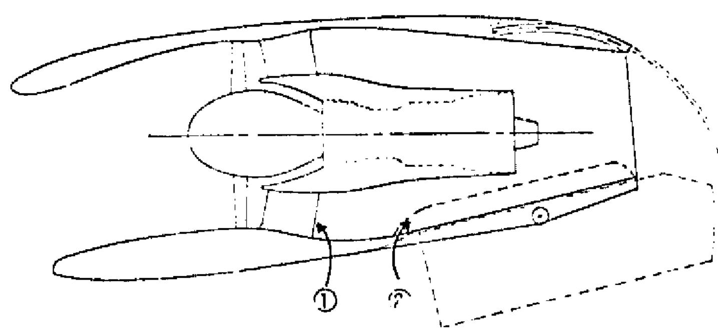
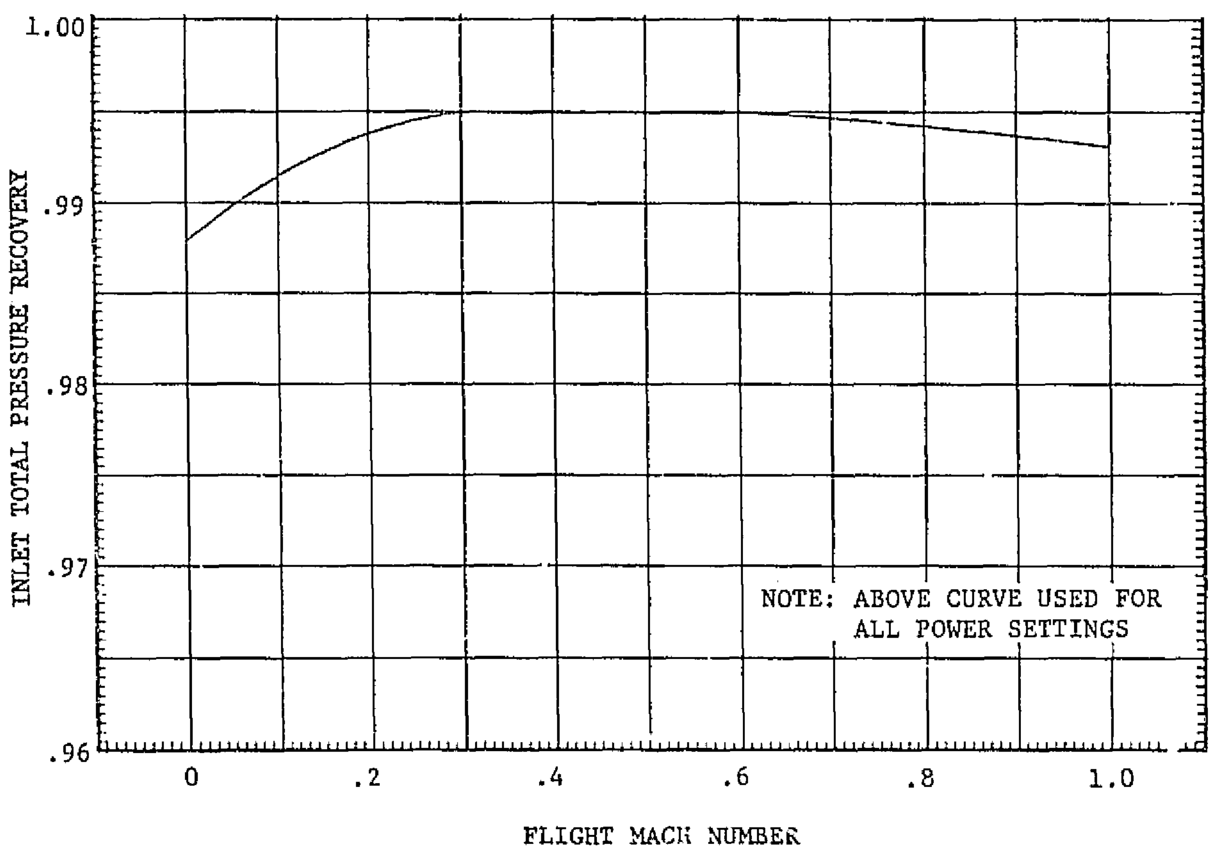


FIGURE 3-44
MECHANICAL RTA
PROPULSION SYSTEM INSTALLATION FACTORS
CONVENTIONAL FLIGHT

Component	Cruise
Engine	
Pressure Recovery[d]	See Figure 3-45
Compressor Airbleed[a] (lb/sec)	0.10
Horsepower Extraction[a]	40[c]
Nozzle Thrust Coefficient	.98
L/C Fan and Nacelle	
Pressure Recovery	See Figure 3-45
Duct Pressure Loss (Fan Cold Stream)	(b)
Nozzle Thrust Coefficient	.98
Additional Performance Allowances	
Net Thrust Derate	0
Center Gearbox	
Horsepower Extraction	0
Horsepower Loss	(b)

Notes: (a) Per Engine
(b) Included in Engine Co. Performance
(c) Supercharged Engine
(d) Non-Supercharged Engine

FIGURE 3-45
MECHANICAL RTA
ESTIMATED INLET TOTAL PRESSURE RECOVERY
CONVENTIONAL FLIGHT
LIFT/CRUISE FANS



4. FLIGHT VEHICLE DESIGN

4.1 TURBOTIP RTA

4.1.1 GENERAL ARRANGEMENT - The turbotip RTA airframe reflects a maximum utilization of GFE components as illustrated in Figure 4-1. The airframe consists of a modified T-39 center fuselage and wing, a modified F-101 aft fuselage and empennage, a new airframe forward fuselage, a modified A-6 cockpit, and an A-4 landing gear and brakes. The propulsion system consists of three LCF459 turbotip fans powered by three YJ97 engines. They are interconnected by the ETaC system. The propulsion and ETaC systems are described in Section 3.1 and Volume II of this report respectively. The principal weights and geometric characteristics of the vehicle are shown in Figure 4-2. The general arrangement, wetted areas, and cross-sectional areas of the aircraft are shown in Drawing M260-RTA-1, Sheets 1, 2, and 3.

4.1.2 AIRFRAME

Fuselage - The T-39 center fuselage was modified to accommodate the installation of three YJ97 gas generators, the gas distribution system, two LCF459 lift/cruise fans, two bladder fuel tanks and to adapt to the F-101 aft fuselage.

The F-101 aft fuselage was modified to accommodate the top mounted gas generator and to mate with the T-39 center fuselage. The stainless steel and titanium fairings and shingles were replaced with aluminum fairings to reduce the weight.

The forward fuselage is a new all metal construction. It was designed to accommodate the A-6 cockpit, the LCF459 lift fan assembly, the ETaC system, and the A-4 nose landing gear. It was also designed to mate with the T-39 center fuselage.

Empennage - The F-101 vertical tail was modified to incorporate a fuel vent tank. The horizontal tail was modified by adding a small glove to each outboard tip for increased area. This area was increased to meet the 5 percent static margin requirements of the aircraft. The overall span of the horizontal tail was increased by 35 inches.

Wing - Drawings of the T-39 were not available; however, weight allocations were made for the required modifications. These modifications consisted of (1) "beef-up" to meet the 2.5 g load factor requirement, (2) mounting of the A-4 main gear and required fairings, (3) removal of the inboard section of the flaps and conversion of the remaining outboard section to a hinged flap, and

FIGURE 4-1
TURBOTIP RTA
MODEL 260-RTA-1

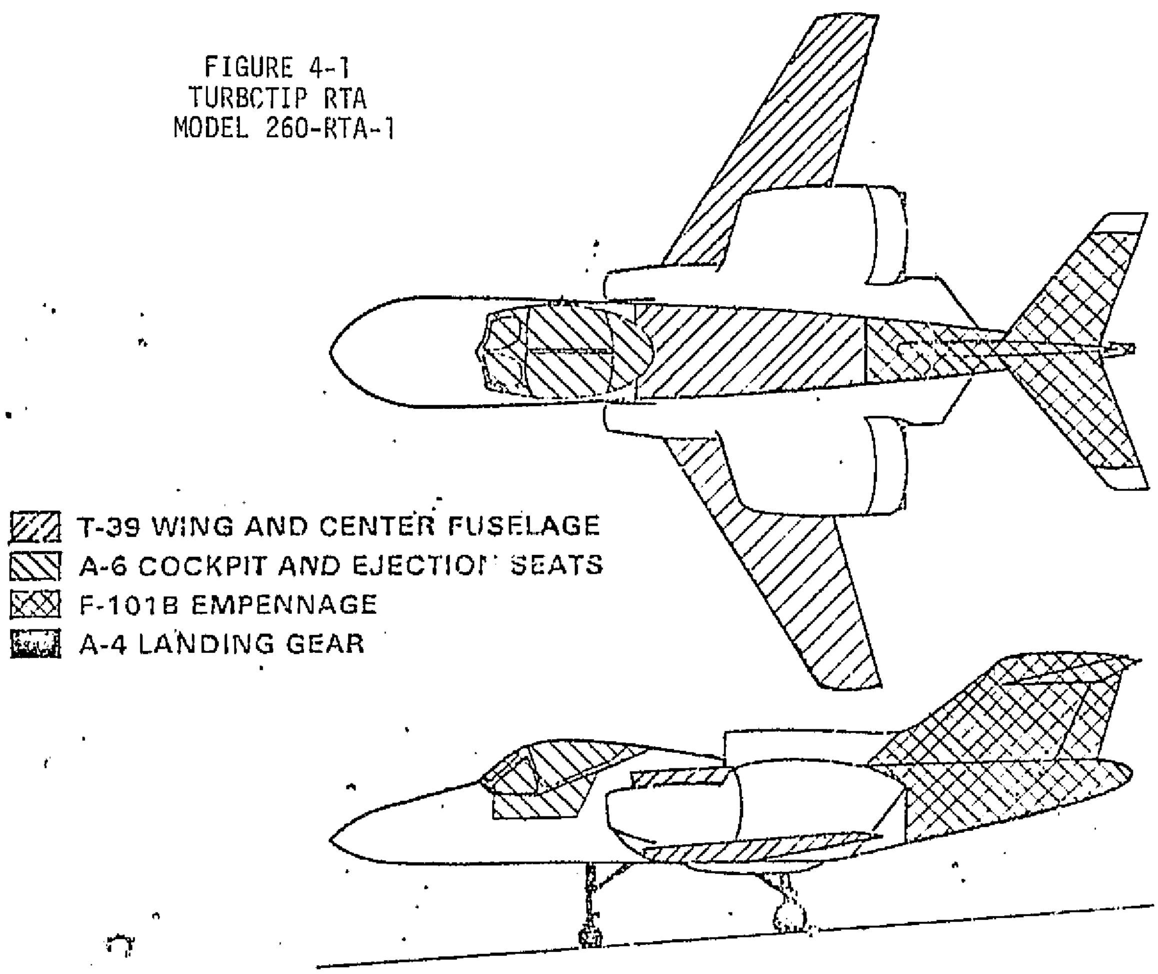


FIGURE 4-2
DIMENSIONAL AND DESIGN DATA
TURBOTIP RTA

STOGW	(lb)	27,400
	(N)	121,875
VTOGW	(lb)	25,286
	(N)	112,472
OWE	(lb)	19,451
	(N)	86,518
Overal Length	(ft)	51.9
	(m)	15.82
Wing Span	(ft)	44.43
	(m)	13.54
Height	(ft)	17.17
	(m)	5.23
Crew Provisions	(No.)	2
Max. Internal Fuel	(lb)	5,750
	(N)	25,576

	Wing	Horizontal Tail	Vertical Tail
S (ft^2)	342.05	82.02	84.88
(m^2)	31.78	7.62	7.89
AR	5.77	3.959	.663
λ	.323	.3703	.509
b (ft)	44.43	18.02	7.5
(m)	13.54	5.49	2.29
$\Lambda_{c/4}$	28.56	31.25	46
t/c (% Root/Tip)	11.3/9.36	7.0/ -	7.0/7.0
Airfoil (root)	64A series (MOD)	NACA 65A007 (MOD)	NACA 65A007
Airfoil (tip)	64A series (MOD)	Modified	NACA 65A007

(4) fixing the leading edge slats to a nonfunctional configuration. Fairings were added to provide a smooth, aerodynamic interface with the lift/cruise fan nacelles, gas generators, and A-4 landing gear. Since the fuel tanks are fuselage mounted, there was no requirement for modification of the integral wing fuel tanks, thereby reducing cost.

4.`.3 COCKPIT - A study was conducted to determine if the T-39 cockpit should be modified to accept ejection seats or if an A-6 cockpit should be used. Consideration was given to vision, access/egress, arrangement and effects on the aircraft configuration.

It was determined that modification of the T-39 cockpit for access, egress, and ejection seats would require extensive development and testing. A comparison of the modifications required for each concept is presented in the following:

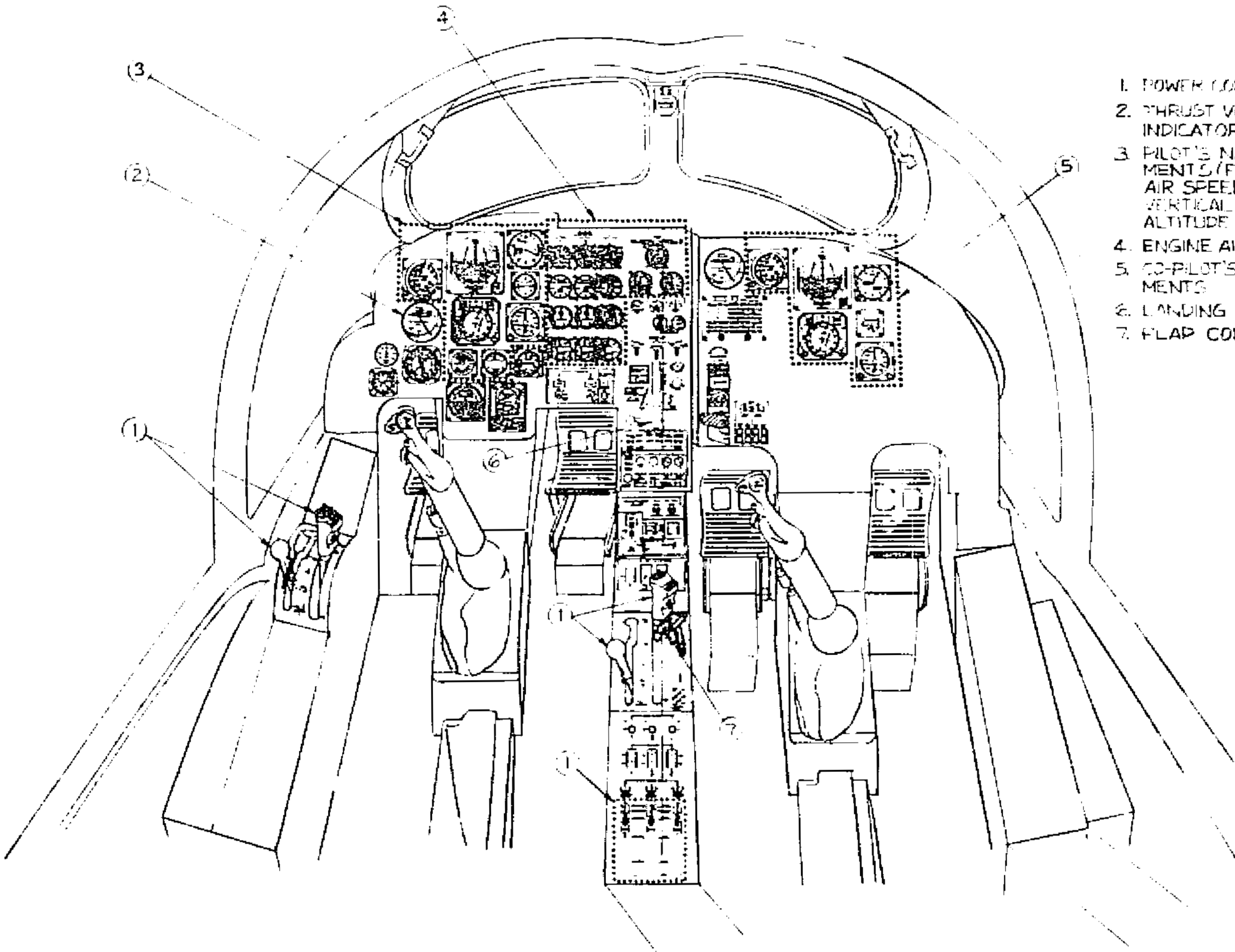
COMPARISON OF REQUIRED COCKPIT MODIFICAT

	A-6	[illegible]
Access	Existing	Lengthen Aircraft (Using Existing Door) or New Development
Egress	Existing	New Development
Ejection Seats	Existing	New Development
Testing	None	Extensive (~$2M)
Consoles	Existing	New
Instruments	New	New
Stick/Rudder Pedals	Mod/Existing	New/Existing
Visibility	Satisfactory	New Side Panel

Based on this analysis, the use of the A-6 cockpit was selected as being the most cost effective approach even though it is approximately 252 lb heavier. The A-6 canopy, canopy actuation system, jettison system, oxygen system, and Martin-Baker seats are also used.

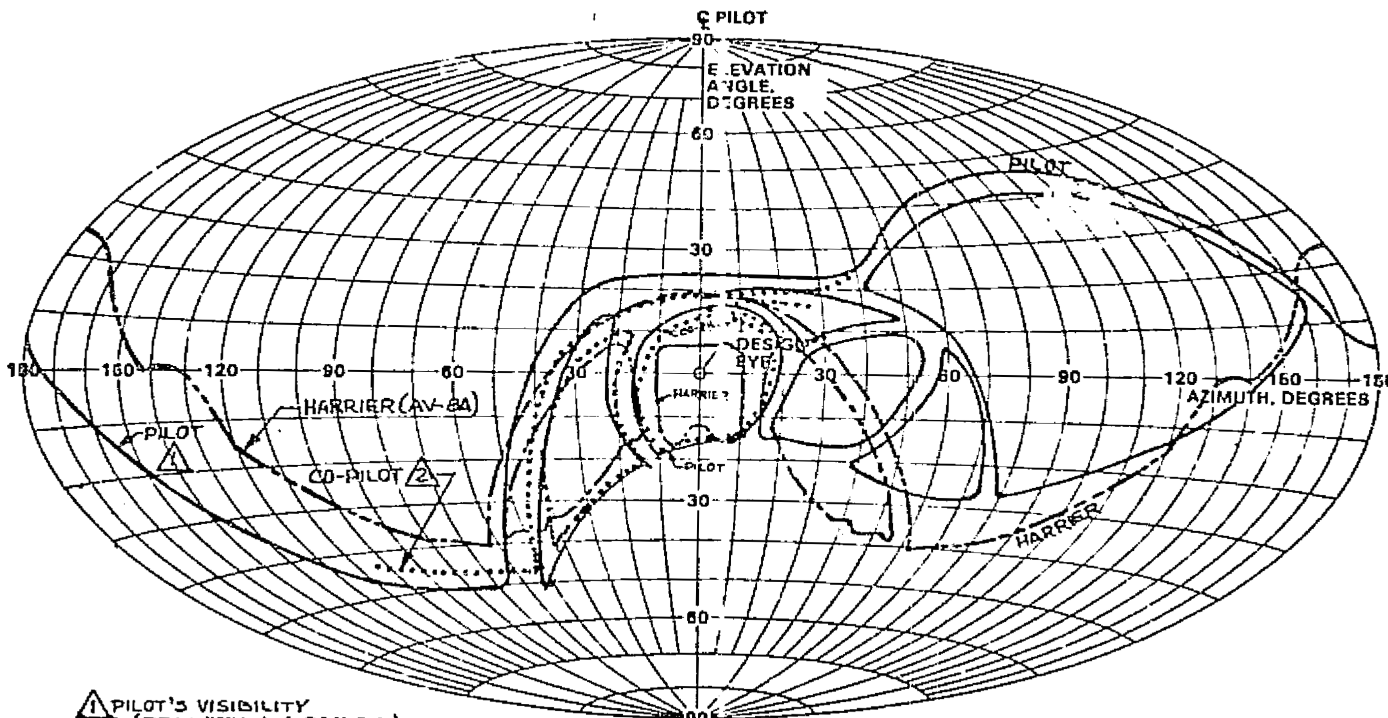
The A-6 cockpit modifications consisted of (1) raising the copilot's seat approximately 4 inches, (2) adding control elements and instruments for both crewmen (the consoles are used as is), and (3) modification of the upper portion of the stick to incorporate new switches. The resulting cockpit arrangement is shown in Figure 4-3. The cockpit is not pressurized for the RTA. The visibility for both the pilot and copilot is better than that available on the Harrier. A visibility diagram is shown in Figure 4-4.

FIGURE 4-3
M260-RTA
COCKPIT ARRANGEMENT



1. POWER CONTROL LEVERS
2. THRUST VECTOR ANGLE INDICATORS
3. PILOT'S NAVIGATIONAL INSTRUMENTS (FLIGHT DIRECTORS AND AIR SPEED, PRESSURE ALTITUDE, VERTICAL SPEED AND RADAR ALTITUDE INDICATORS
4. ENGINE AND FAN INSTRUMENTS
5. CO-PILOT'S NAVIGATIONAL INSTRUMENTS
6. LANDING GEAR CONTROL
7. FLAP CONTROL

FIGURE 4-4
PILOT'S EXTERNAL VISIBILITY



1 PILOT'S VISIBILITY
(RTA WITH A-6 COCKPIT)

2 CO-PILOT'S VISIBILITY
(RTA WITH A-6 COCKPIT, CO-PILOT'S SEAT ON SAME LEVEL AS PILOT'S)

4.1.4 LANDING GEAR/BRAKES

Analysis - Prior to selection of the A-4 landing gear and brake system, an analysis was conducted to assess their compatibility with the aircraft. The gear and brakes were evaluated for the loading conditions shown below.

A-4 LANDING GEAR/BRAKE EVALUATION

Landing Loads
- Two Point Level
- Three Point Level
- Tail Down
- One Wheel Nose Down
- One Wheel Tail Down
- Side Drift
- Side Load

Spin Up, Spring Back and Maximum Strut Reaction

Taxiing Loads
- Braked Roll - Two Point
- Unsymmetrical Braking
- Turning

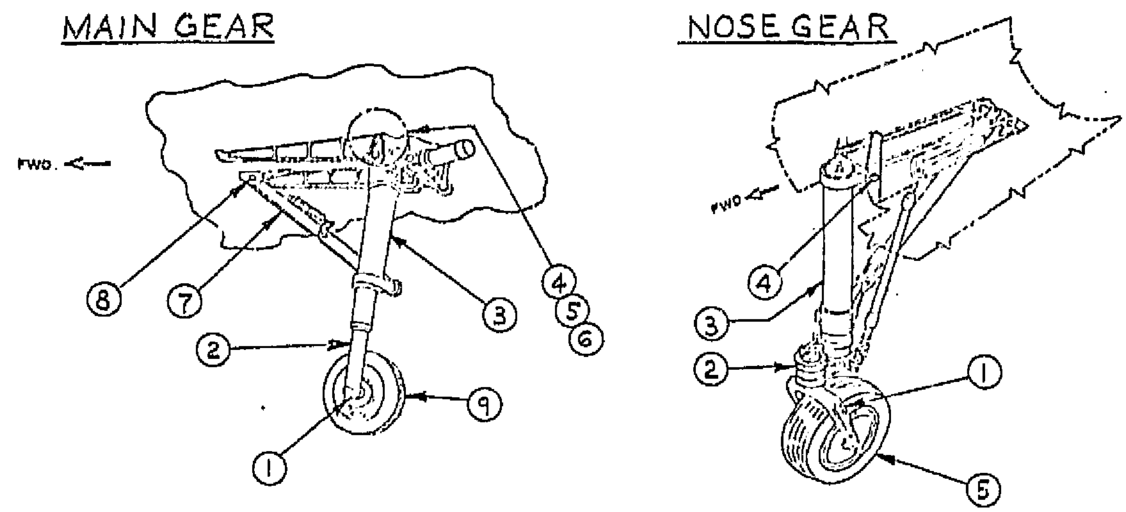
Handling Loads
- Towing
- Jacking

Thirty-two sections in the main gear and ten sections in the nose gear, Figure 4-5, were analyzed for this loading. It was determined for the main gear that thirty sections had positive margins at 12 fps sink rate for all conditions. One section was found that had a positive margin at 12 fps for all conditions except for the two point level landing spin-up. This section had a positive margin at 11 fps. This section could be retested for this loading condition; however, normal landing for the RTA will probably always be less than at 12 fps sink rate and would not justify the cost of retesting.

The nose gear was found to have seven sections which had positive margins at 12 fps for all conditions. The trunnion bolt was found to require a change for positive margin during unsymmetrical braking. This minor change would consist of a slightly larger bolt with higher heat treat.

The wheels and brakes were found to provide adequate braking for STO landing, STO RTO, and taxiing. Adequate braking for CTOL operation is also provided when an increased stopping distance is used. Based on this analysis, the A-4 gear and brakes were selected for use on the RTA.

FIGURE 4-5
STRUCTURAL SECTIONS EVALUATED



MAIN GEAR

(1) AXLE - 4 SECTIONS
(2) PISTON - 3 SECTIONS
(3) JACKET - 8 SECTIONS
(4) ROTATING ARM - 2 SECTIONS
(5) ROTATING LINK - 5 SECTIONS
(6) ROTATING SWIVEL - 3 SECTIONS
(7) DRAG BRACE - 5 SECTIONS
(8) DRAG LINK SWIVEL PIN - 1 SECTION
(9) WHEEL

32 SECTIONS EVALUATED

NOSE GEAR

(1) FORK - 2 SECTIONS
(2) SPINDLE - 4 SECTIONS
(3) BARREL - 1 SECTION
(4) TRUNNION BOLT - 2 SECTIONS
(5) WHEEL

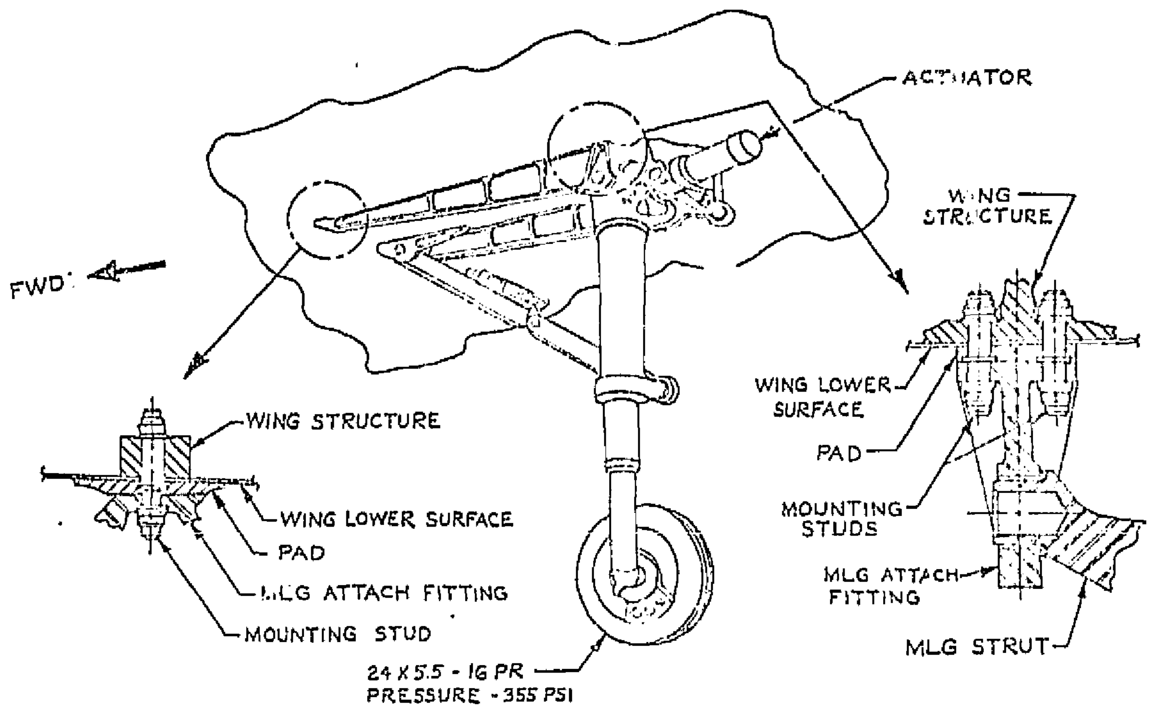
10 SECTIONS EVALUATED

Main Landing Gear - The A-4 main landing gear assembly is attached to the lower surface of the wing as shown in Figure 4-6. A fairing and appropriate doors are also installed on the wing lower surface to provide a well for enclosing the gear in the retracted position.

The main gear retracts forward, and during retraction, the shock strut rotates 90 degrees so that the wheel is stowed horizontally in the main gear well. Normal operation of the gear and doors is hydraulic, using power supplied by the aircraft hydraulic system. Should a hydraulic system failure occur, the gear can be lowered by pulling the Emergency Landing Gear handle in the cockpit to actuate cables which unlatch the doors and permit the landing gear to free-fall with the aid of the airstream.

The main gear is equipped with a hydraulically operated double disc brake, a forged aluminum wheel, and a 24 x 5.5 - 16 PR tube type tire.

FIGURE 4-6
MLG ATTACHMENT TO WING



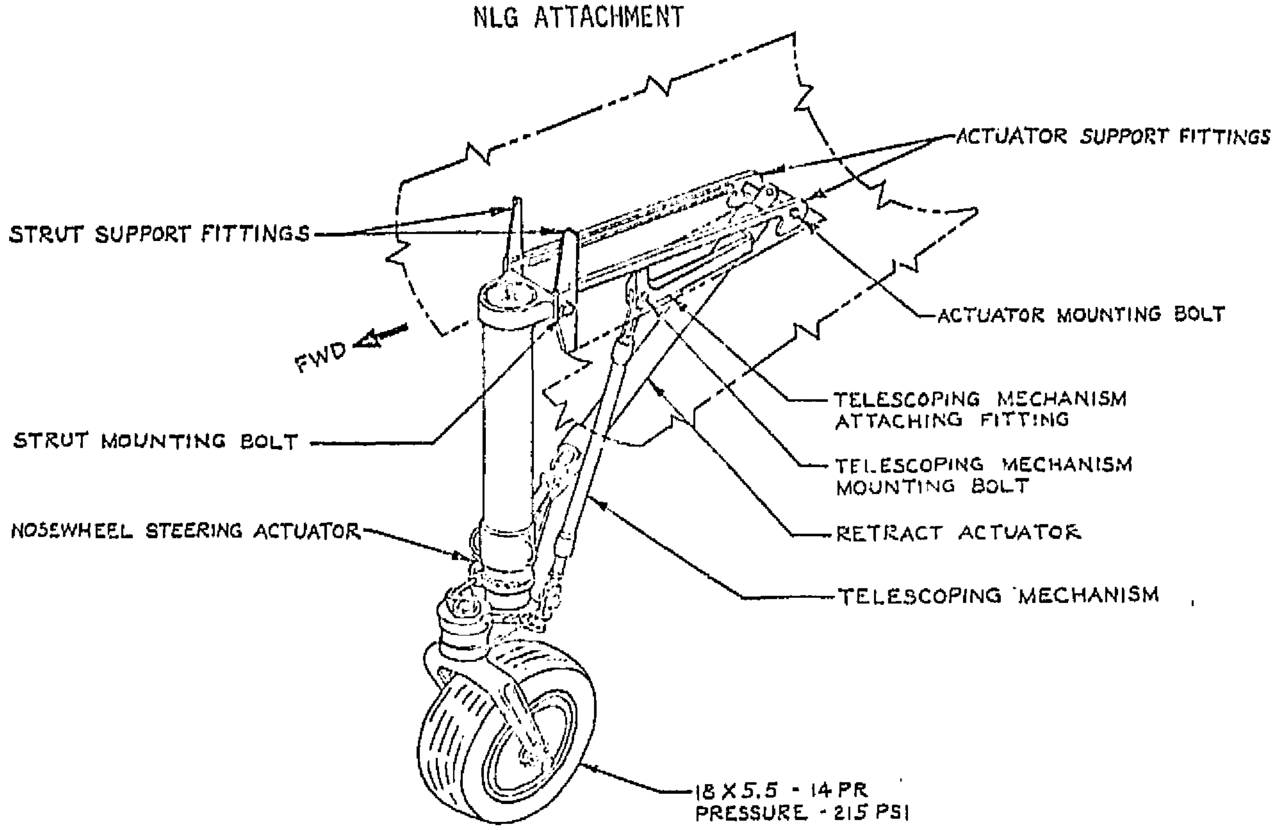
Nose Landing Gear - The A-4 nose landing gear assembly is attached to the fuselage as shown in Figure 4-7.

The nose gear retracts forward into the nose gear well in the forward fuselage. As the gear retracts, a telescoping mechanism compresses the shock strut sufficiently to fit the gear into the nose gear well. Power supplied by the aircraft hydraulic system is used for normal operation of the gear and doors. Emergency extension of the gear is accomplished by pulling the Emergency Landing Gear handle to unlatch the doors and permit the nose gear to free-fall with the aid of the airstream.

The nose gear is equipped with a forged aluminum wheel and an 18 x 5.5 - 14 PR tubeless tire.

The nose gear is also equipped with a nosewheel steering system that is actuated by movement of the rudder pedals. The actuator, which operates from the landing gear down hydraulic pressure, has the capability of turning the nosewheel 45 degrees either side of center.

FIGURE 4-7
NLG ATTACHMENT



Brake System - The main landing gear brake system is an independent hydraulic system operated from the cockpit. The principal components of the system are as follows:

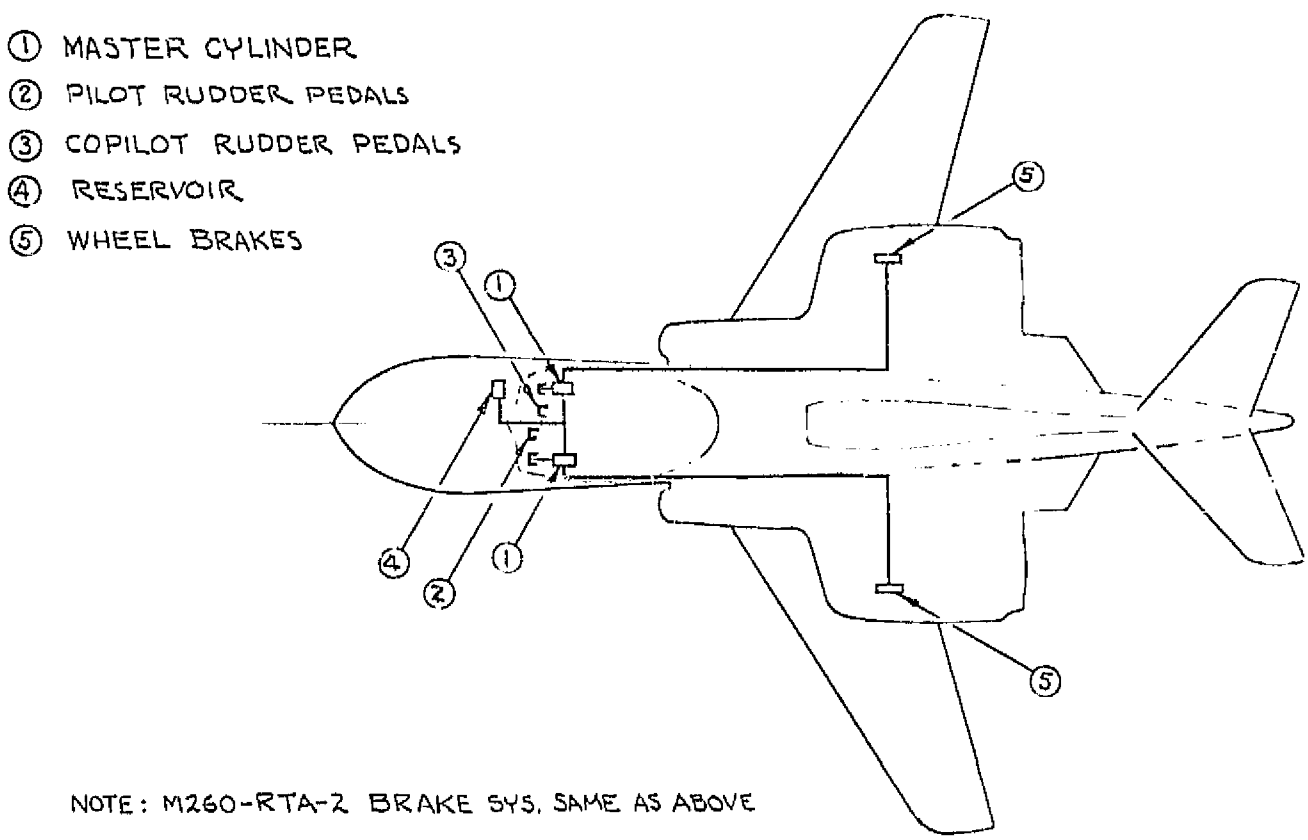
A-4 brake reservoir

A-4 brake master cylinders

A-4 main gear brakes

The hydraulic fluid in the brake system is supplied by the brake reservoir and is pressurized by the master cylinders for application to the brakes. The master cylinders are controlled by toe pressure on the rudder pedals. The pilot's and copilot's brake pedals are mechanically connected so that the operation of either left pedal will operate the left brake master cylinder and the operation of either right hand pedal will operate the right brake master cylinder. The location of the brake system is illustrated in Figure 4-8.

FIGURE 4-8
M260-RTA
LANDING GEAR BRAKE SYSTEM



4.1.5 HYDRAULIC SYSTEM - Two independent, 3000 psi hydraulic systems which incorporate Reservoir Level Sensing (RLS) are provided. The RLS separates each system into two independent circuits, thereby providing four circuits which is the same approach as used in the F-18. This redundancy feature is incorporated to provide adequate safety for the triplex fly-by-wire control system. Switching valves are used at each flight critical actuator to assure hydraulic power system redundancy for fail-safe operation.

The 90 hp pumps are driven by gearboxes mounted on the aft side of the lift/cruise fans. The existing YJ97 engine did not have adequate power extraction capability to mount the pumps on the engine. The selected mounting provides for normal hydraulic power after loss of one or two engines and some power after loss of all engines since the fans will function as ram air turbines. Standard tubing materials and MIL-H-5606 fluid are used to assure a low cost, low risk system.

Dual tandem actuators with "rip-stop" construction are used to power all the flight critical items; i.e., stabilator, ailerons, rudder, ETaC valves, TRM doors, and lift fan louvers and yaw doors. Hydraulic motors are used to power the lift/cruise fan vector hoods and yaw doors through appropriate gearboxes. The mechanical interconnect between the thrust vector hoods provides for symmetry during operation. Triplex electrohydraulic Signal Conversion Mechanisms (SCM) are used to convert the electronic signals from the AFCS to mechanical outputs which are then used as inputs to the flight control actuator mechanical servo. The schematic of the hydraulic system is shown in Figure 4-9. The locations of the major power generation components and primary flight control components are shown in Figures 4-10 and 4-11 respectively.

4.1.6 FLIGHT CONTROLS - Pitch, roll, and yaw are accomplished by conventional control surfaces during aerodynamic flight, and by fan thrust modulation and vectoring during powered lift flight. Fan thrust is modulated by ETaC valves in the gas distribution system to each fan and by Thrust Reduction Modulation (TRM) at each fan. These functions are powered by dual hydraulic power cylinders and are controlled by a triplex control-by-wire system. The ETaC, TRM, and control-by-wire systems are discussed in Sections 3.1 and 5.4. The electronic signals from the control-by-wire system are converted to mechanical inputs to each power actuator by triplex Signal Conversion Mechanisms (SCM). The lift/cruise fan yaw louvers are powered by rotary mechanical actuators driven by hydraulic motors through a system of gearboxes and torque shafts.

The power management control system consists of mechanical throttle systems to modulate the gas generator fuel controls, and a mechanical control system for thrust vectoring and flight mode conversion. The thrust of the three gas generators is controlled collectively by a master power lever and a set of individual engine throttle levers is provided for engine trimming. The transition controls are positioned by an electrical actuation device through a thumb switch on the master power lever. The actuation device can be overridden manually in the event of an error signal. The lift fan thrust vector system is powered by hydraulic actuators, and the lift/cruise fan vectoring hoods are powered by rotary mechanical actuators driven by hydraulic motors through a system of gearboxes and torque shafts.

The A-6 cockpit pilot's controls were modified for compatibility to a fly-by-wire control system and a duplicate set of controls was added for the copilot. A transition lever and master power lever are provided for both

FIGURE 4-9
MODEL 260-RTA-1
HYDRAULIC SYSTEM

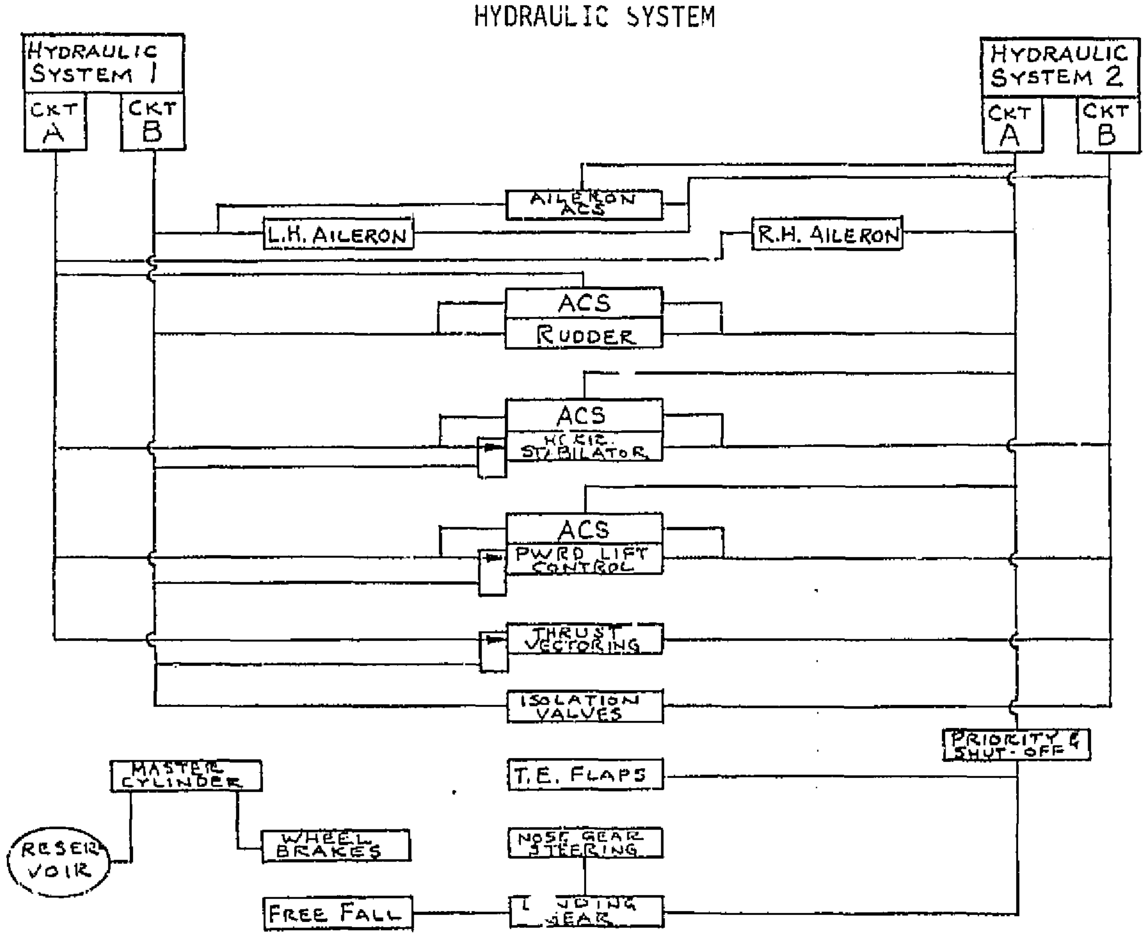


FIGURE 4-10
MODEL 260-RTA-1
MAJOR HYDRAULIC POWER GENERATION AND UTILITY COMPONENTS

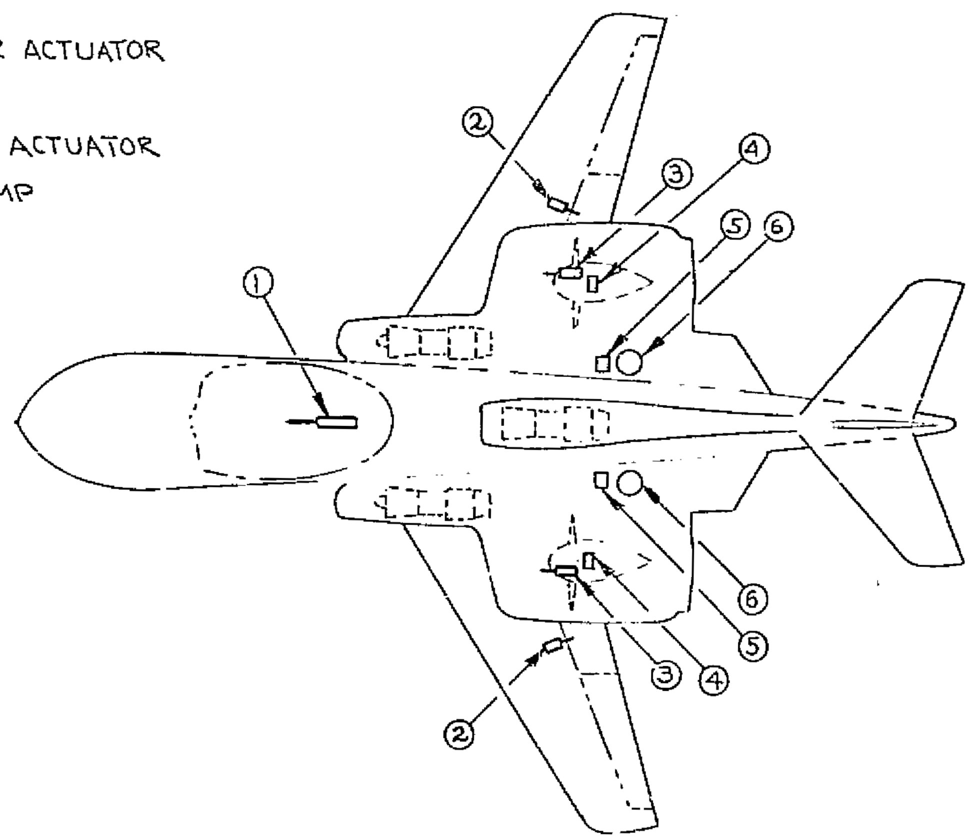
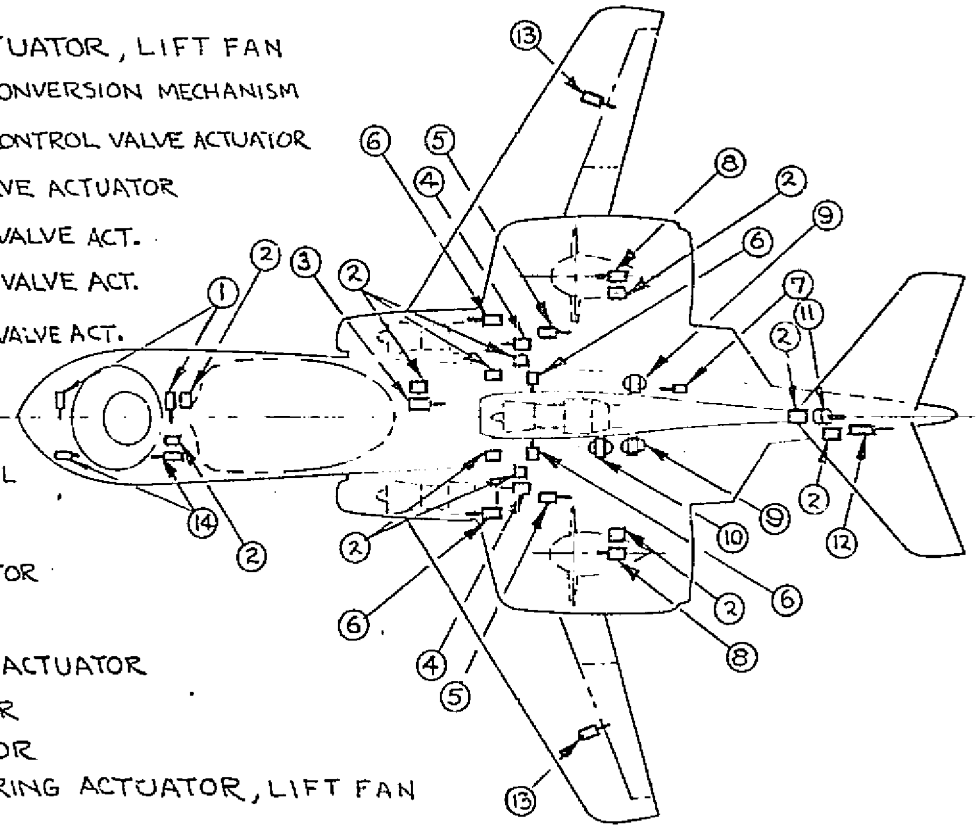


FIGURE 4-11
MODEL 260-RTA-1
MAJOR HYDRAULIC CONTROL COMPONENTS



1. YAW CONTROL ACTUATOR, LIFT FAN
2. TRIPLEX SIGNAL CONVERSION MECHANISM
3. $ET_{\alpha}C$ ISOLATION & CONTROL VALVE ACTUATOR
4. $ET_{\alpha}C$ CONTROL VALVE ACTUATOR
5. HOT GAS SHUT OFF VALVE ACT.
6. HOT GAS ISOLATION VALVE ACT.
7. HOT GAS DIVERTER VALVE ACT.
8. THRUST REDUCTION MODULATION ACT.
9. HOOD YAW CONTROL MOTOR/VALVE
10. HOOD THRUST VECTOR MOTOR/VALVE
11. HORIZONTAL TAIL ACTUATOR
12. RUDDER ACTUATOR
13. AILERON ACTUATOR
14. THRUST VECTORING ACTUATOR, LIFT FAN

pilot and copilot, and individual throttles for engine trimming are provided on the center console. The pilot and copilot controls are mechanically interconnected.

The mechanical controls and actuation systems are shown in Figure 4-12.

4.1.7 ELECTRICAL SYSTEM - Each of the dual electric power systems includes a CSD driven 15 KVA AC generator which provides 115/200 volt, 400 Hz, three phase power. A backup battery is provided for essential power in case both generators are inoperative. Since the aircraft electrical loads exceed a single generator capacity, monitored buses are provided so that selected loads can be removed during single generator operation. The system schematic is shown in Figure 4-13 and the loads in Figure 4-14.

The generators are each mounted on a constant speed drive (CSD). The CSD's are then mounted on and driven by the fan gearboxes. Both the generator and CSD rely on oil cooling and integral oil coolers are provided in each gearbox. Suitable generators and CSD's are available as off-the-shelf equipment

FIGURE 4-12
MODEL 260-RTA-1
MECHANICAL CONTROL AND ACTUATION SYSTEM

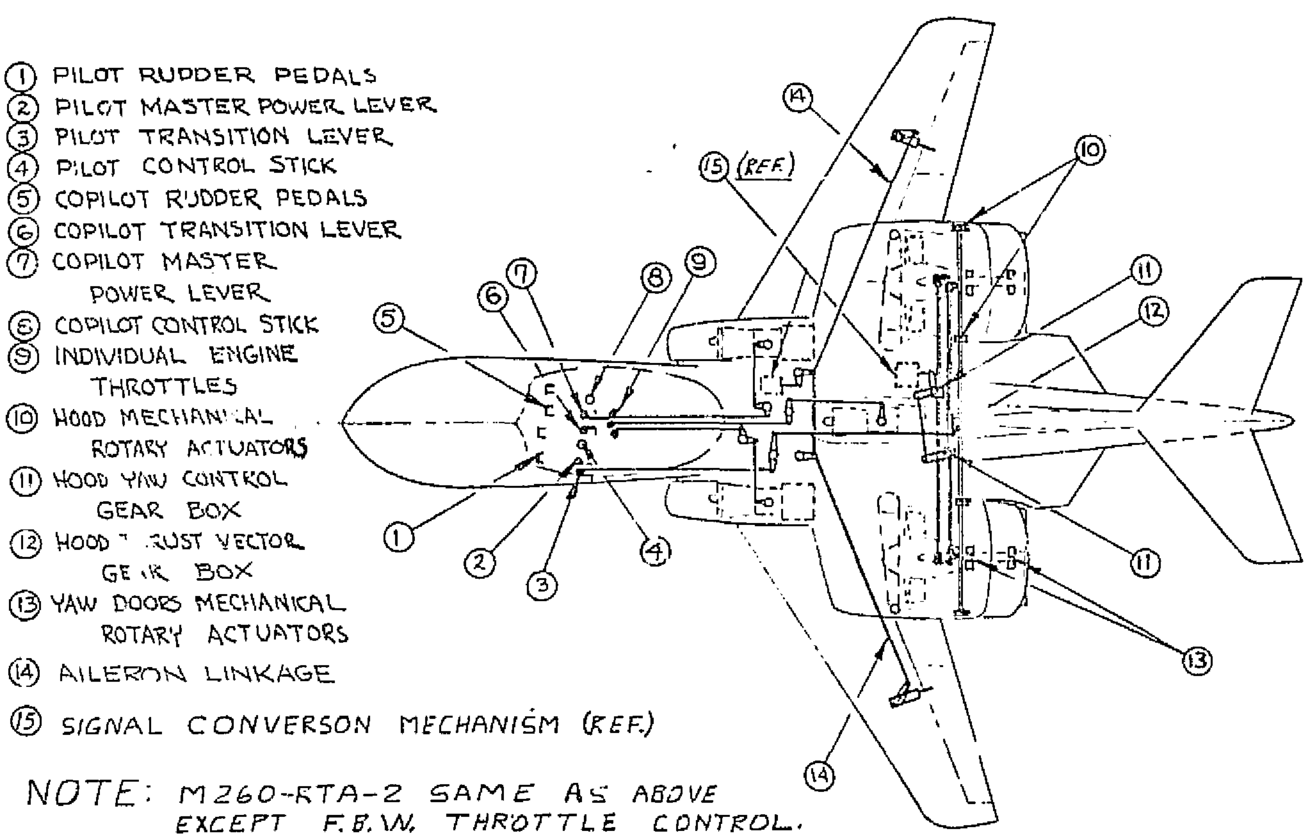


FIGURE 4-13
MODEL 260 RTA
ELECTRICAL SYSTEM

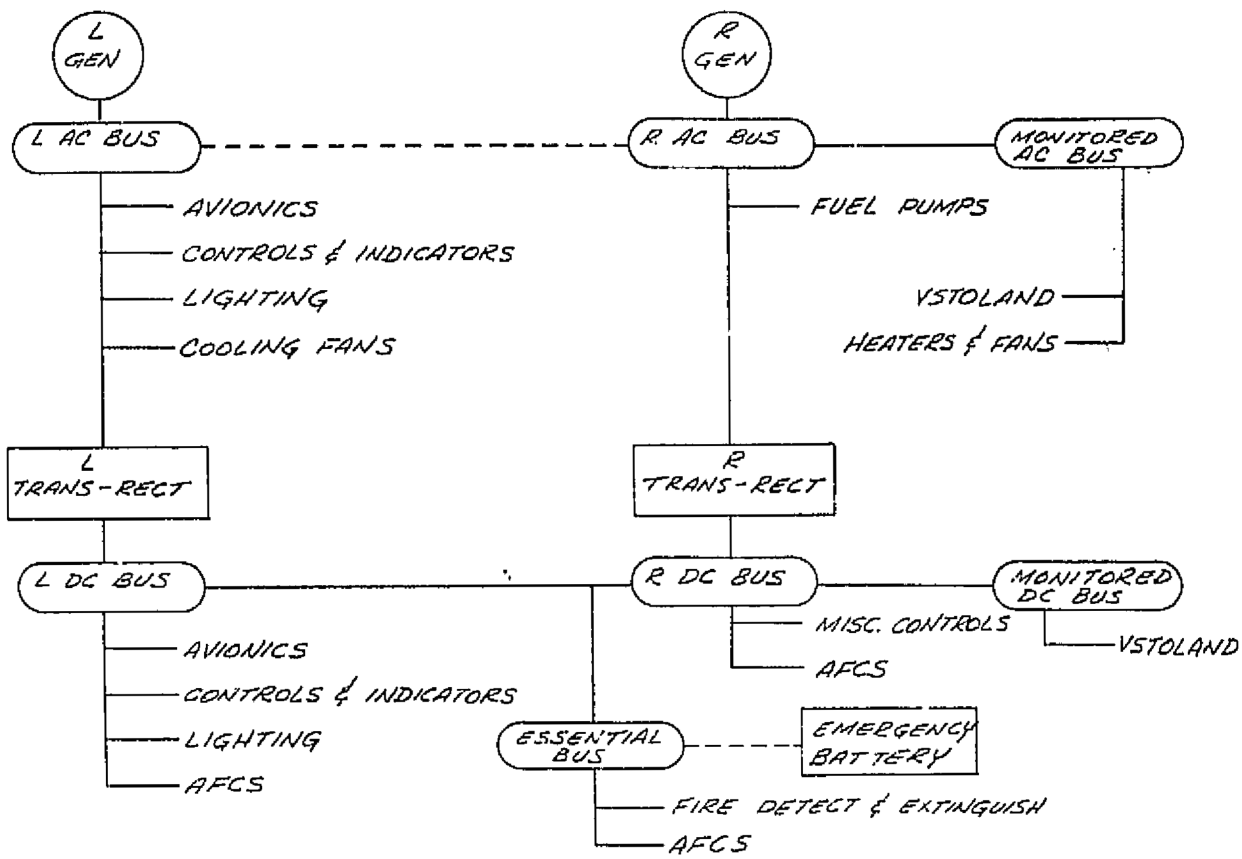


FIGURE 4-14
RTA PRELIMINARY ELECTRIC POWER REQUIREMENTS

EQUIPMENT	NORMAL OPERATION AC (VOLT-AMPS)	NORMAL OPERATION DC (WATTS)	SINGLE GENERATOR OPERATION AC (VOLT-AMPS)	SINGLE GENERATOR OPERATION DC (WATTS)
AVIONICS	2145	118	2145	118
CONTROLS & INDICATORS	249	702	249	702
LIGHTING	811	199	811	199
CABIN/AVIONICS VENT FANS	2131		2131	
FUEL PUMPS	1194		1194	
VSTOLAND	3000	1000		
CABIN HEATERS/FANS	8356			
AC TO DC CONVERSION	2639		1332	
TOTAL (TURBOTIP RTA)	20525 VA	2019 WATTS	7862 VA	1019 WATTS
CGB COOLING FAN	3864		3864	
TOTAL (MECH RTA)	24389 VA	2019 WATTS	11726 VA	1019 WATTS

and require only minor modifications to the CSD because of the wide speed range (approximately 3.4:1) of the fan.

The primary electric power system includes a generator control unit GCU) for each generator. The GCU's will include provisions for voltage regulation and standard protection provisions (over/under voltage, over/under frequency). The GCU's, in conjunction with aircraft mounted current transformers, also provide feeder fault protection.

Secondary 28 volt DC electric power is supplied by two 100 amp transformer-rectifiers (TR's) energized by the aircraft primary AC buses. The TR outputs will be connected in a parallel bus arrangement. TR capacity is such that, in the event of a TR failure, the operative TR can provide total aircraft DC requirements. A monitored DC bus is provided which is automatically deenergized during single generator operation to minimize generator loading. The TR's are fan-cooled units and are readily available off-the-shelf.

Emergency electric power is supplied by a 24 volt nickel cadmium battery. Only essential loads will be energized during emergency operation. It is presently estimated that an 11 amp hour battery capacity will provide adequate emergency capability. No battery charger will be provided in the aircraft. A freshly conditioned battery will be provided for each flight. Suitable batteries should be readily available from government inventories. The location of the major electrical components is shown in Figure 4-15.

4.1.8 AVIONICS AND TEST EQUIPMENT - Minimum avionics to accomplish the communication, navigation, display, and control functions commensurate with the requirements of the flight test demonstration program are installed. The flight controls avionics contains sensors and electronics to properly shape, schedule, amplify, and monitor the input signals supplied for use in driving the appropriate control functions. The flight controls electronics utilize multiple redundancy channel techniques in conjunction with redundant signal conversion mechanisms to provide fail operational flight control. Figure 4-16 shows the location of the major avionics components and the weight/space requirements are shown in Figure 4-17. Approximately 54 cu ft of space, Figure 4-18, is provided in the fuselage for the 2500 lb of NASA supplied test equipment.

4.1.9 FUEL SYSTEM - As discussed in Report MDC A3440, Volume II, the use of fuel in the T-39 wing results in an excessive roll moment of inertia. Therefore, to limit the fuel to the inboard section, a major wing modification was required to seal the rib at Wing Rib 106. The T-39 integral wing tank was not used due

FIGURE 4-15
M260-RTA-1
MAJOR ELECTRICAL SYSTEM COMPONENTS

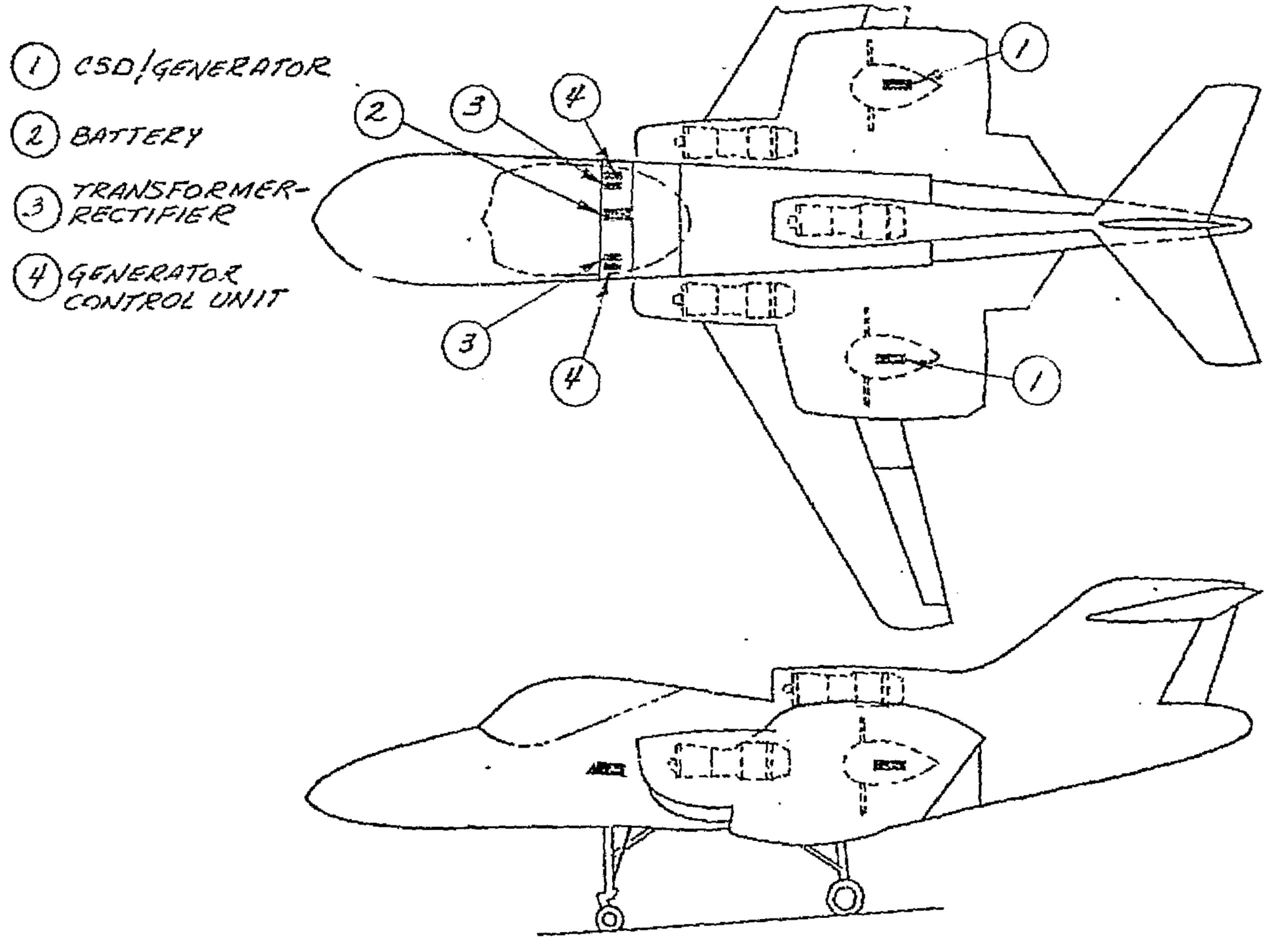
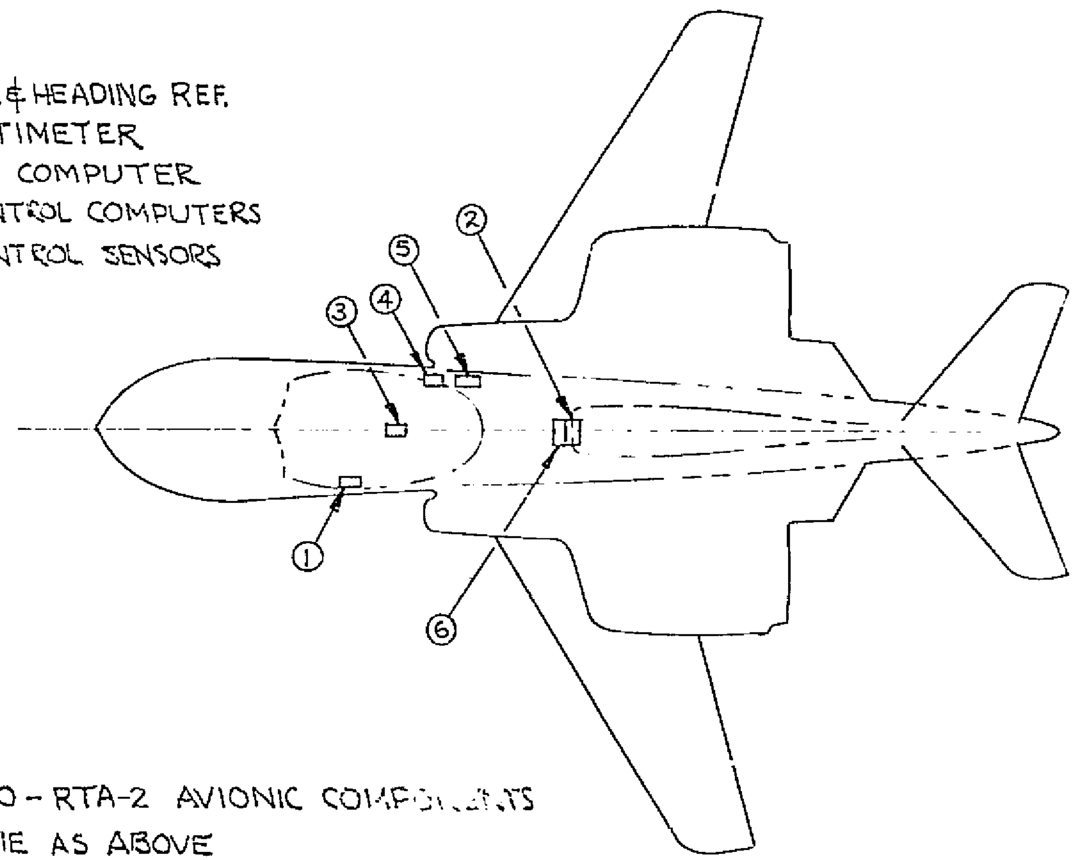


FIGURE 4-16
M260-RTA-1
MAJOR AVIONIC COMPONENTS

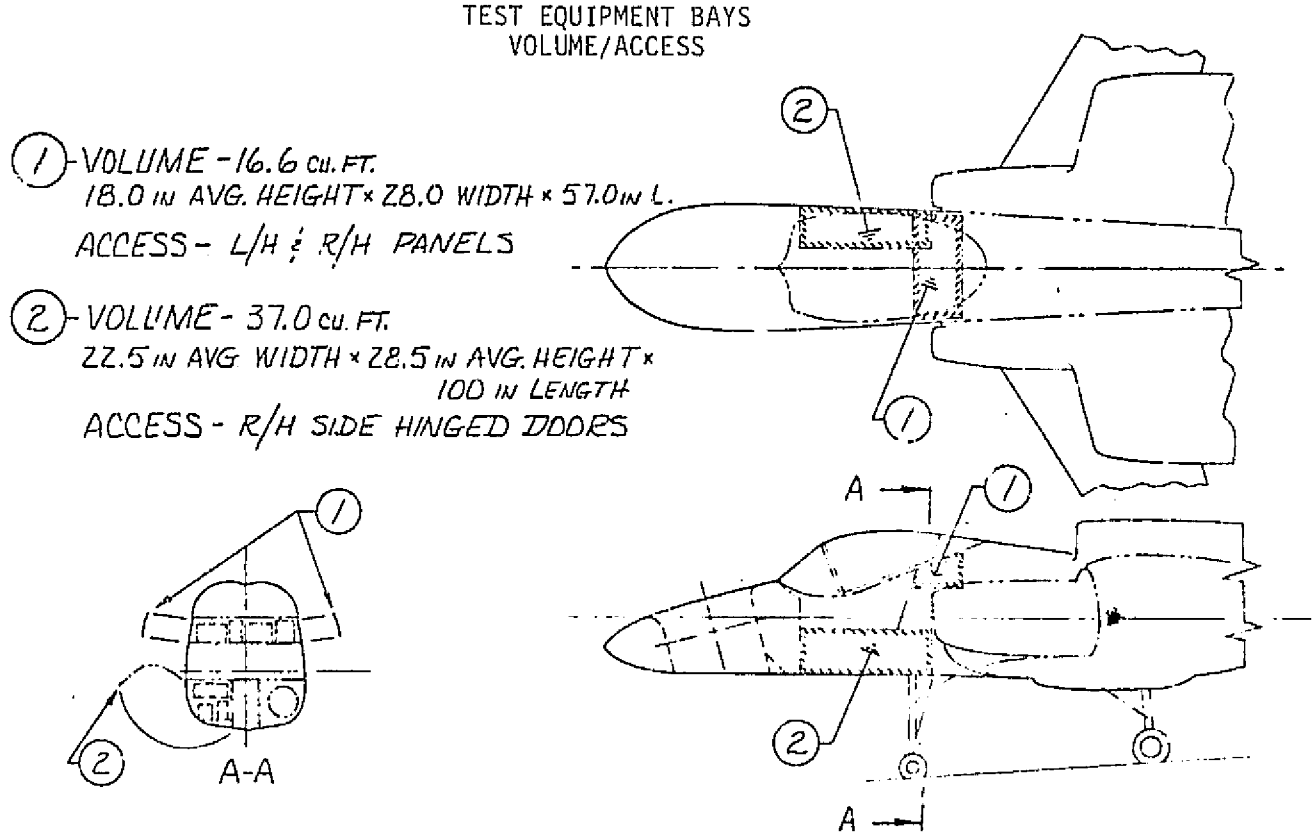


NOTE: M260 - RTA-2 AVIONIC COMPONENTS
SAME AS ABOVE

FIGURE 4-17
RTA AVIONICS EQUIPMENT

	WT (LB)	VOL (CU IN.)
COMMUNICATION, RADIO NAV AND IDENTIFICATION		
UHF AM TRANSCEIVER	9	182
INTERCOMM	7	144
IFF TRANSPONDER	7	173
TACAN	40	925
ANTENNAS	4	137
	(67)	(1561)
NAVIGATION		
ATTITUDE AND HEADING REFERENCE (2)	62	1524
MAGNETIC AZIMUTH DETECTOR (2)	4	35
RADAR ALTIMETER	7	139
AIR DATA SYSTEM	28	800
LOW VELOCITY A/S SYSTEM (2)	10	108
	(111)	(2606)
DISPLAYS		
ATTITUDE DIRECTOR	8	128
HORIZONTAL SITUATION	7	144
ALTIMETER	7	95
STANDBY ATTITUDE	2	28
	(24)	(395)
FLIGHT CONTROL AVIONICS	127	3306
TOTALS	329	7868

FIGURE 4-18
M260-RTA-1
TEST EQUIPMENT BAYS
VOLUME/ACCESS



to the cost associated with these required modifications. The fuel is contained in two fuselage mounted bladder tanks. The lower tank is used as the feed tank and has a capacity of 2400 lb. The upper tank functions as a transfer tank and has a capacity of 3350 lb. A schematic of the fuel system and the equipment locations in the aircraft are shown in Figures 4-19 and 4-20 respectively. Fuselage volume is available if additional tankage is required.

Engine feed is provided by two electrically powered boost pumps located in the feed tank and sized so that either pump can supply fuel to all three engines in the event of a pump failure. Bypass features are provided to allow engine feed even with both pumps failed. Suitable pumps can be found in current service inventories. Fuel transfer to the feed tank is accomplished by gravity flow, utilizing a minimum of components.

The aircraft is refueled by filling the upper transfer tank and gravity filling of the other tank. A single level control valve suffices for this system. The fuel can be removed from the aircraft by suction defueling using the same connect point as is used for single point pressure fueling.

Climb and dive vents are provided through a vent box located in the vertical tail. The vent system consists of fixed tubing, simple check valves, and a fixed geometry scarfed tube vent mast for pressurization. Inflight fuel jettison, aerial refueling, and negative 'g' capabilities are not incorporated in the design.

4.1.10 ECS - A minimal ECS was identified for the RTA to be consistent with the low cost philosophy. A review of the equipment and cockpit heat loads indicated that the cooling requirements could be met with a simple ram air system. Fans are also installed to provide ambient airflow for cooling during ground and low speed operations. Cooling for the electronic test equipment is integral with the package. Electric heaters are also provided for cockpit heating.

4.1.11 FAN/ENGINE MOUNTING AND REMOVAL

Fan/Engine Mount System - The fan/engine combination as shown in Figure 4-21 is a thrust balanced system, whereby the engine mount system does not support any thrust loads.

The forward engine mounts, Figure 4-22, are designed to take vertical and inboard/outboard loads and also double as engine removal pivot points. The aft horizontal mounts are designed to take vertical loads only. The aft, lower center mount is designed to take inboard/outboard loads only. All mounts are

FIGURE 4-19
M260-RTA-1
FUEL SYSTEM SCHEMATIC

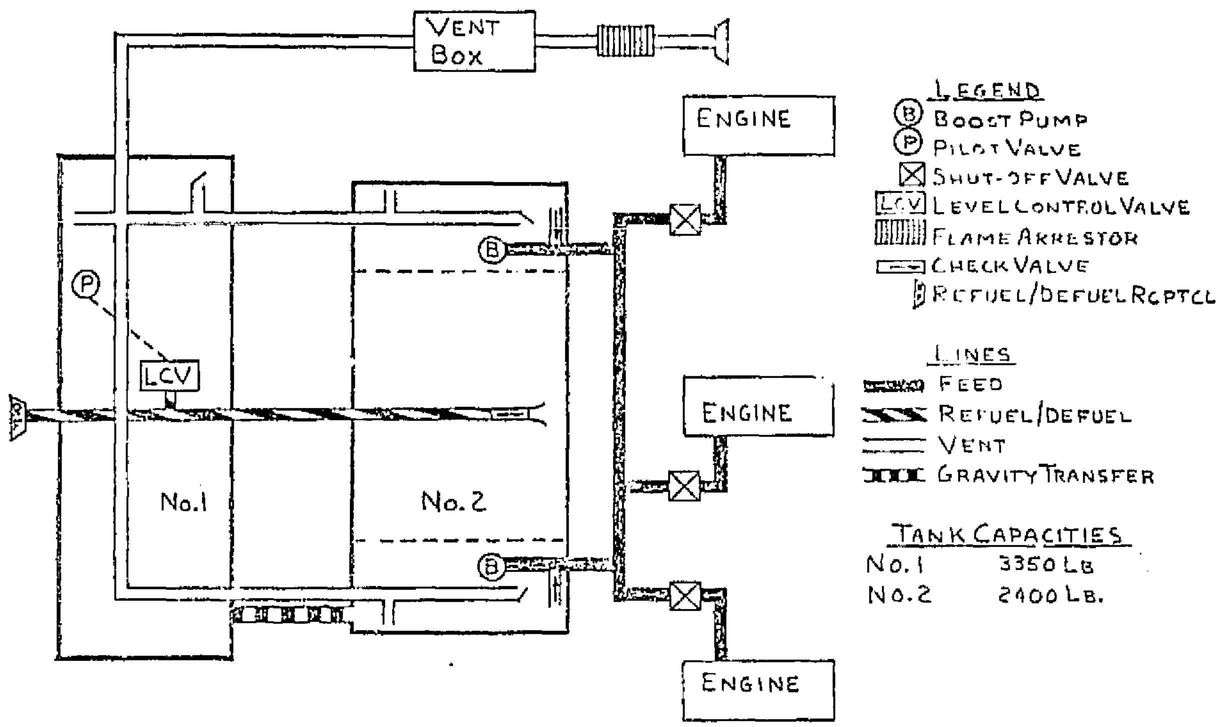
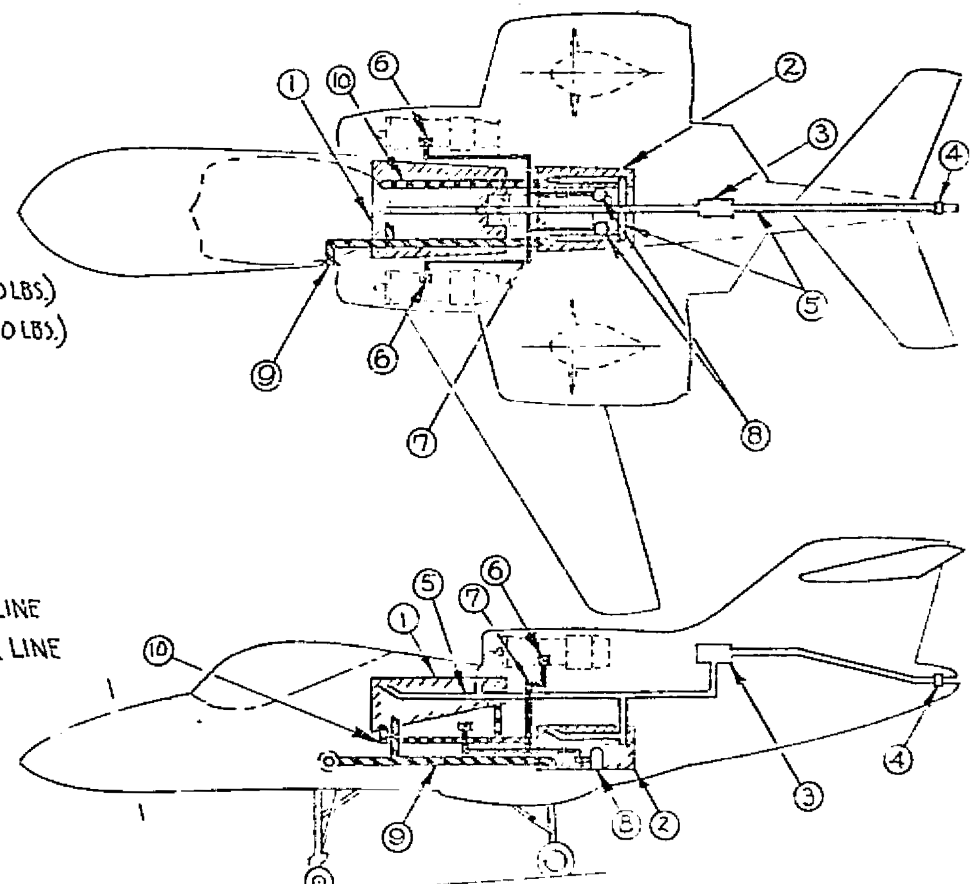


FIGURE 4-20
M260-RT4-1
FUEL SYSTEM



1. No.1 FUEL TANK (3350 LBS.)
2. No.2 FUEL TANK (2400 LBS.)
3. VENT BOX
4. FLAME ARRESTOR
5. VENT LINE
6. SHUT-OFF VALVE
7. ENGINE FEED LINE
8. BOOST PUMP
9. REFUEL/DEFUEL LINE
10. GRAVITY TRANSFER LINE

FIGURE 4-21
M260-RTA-1
MOUNT SYSTEM

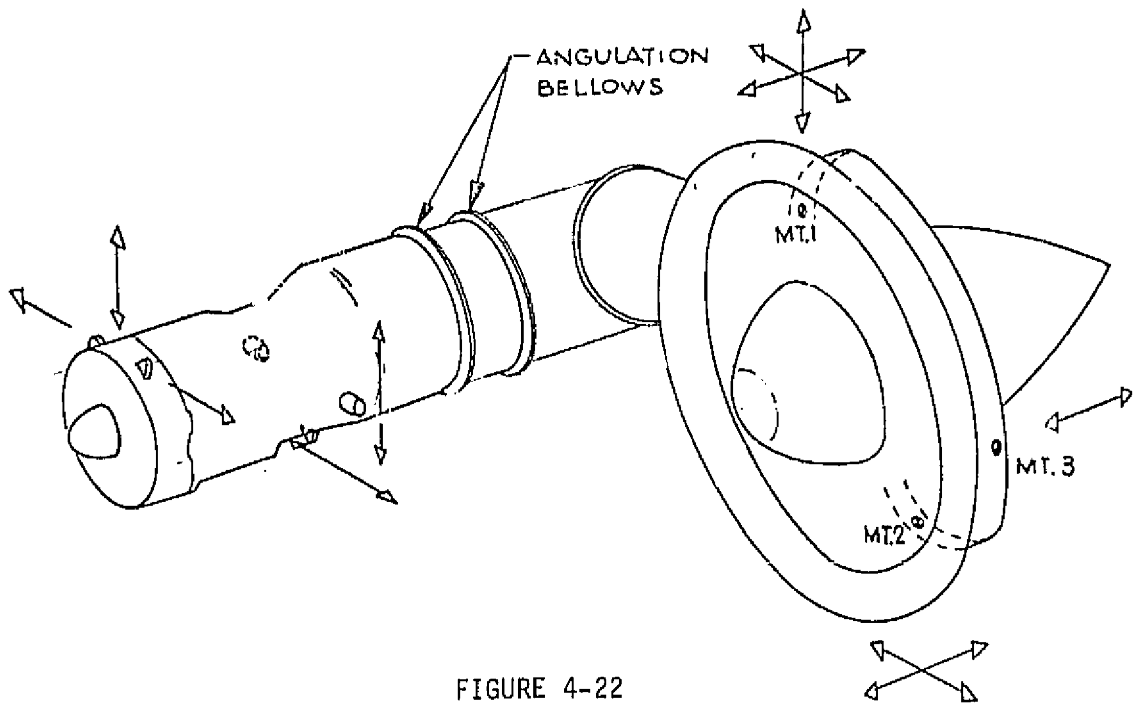
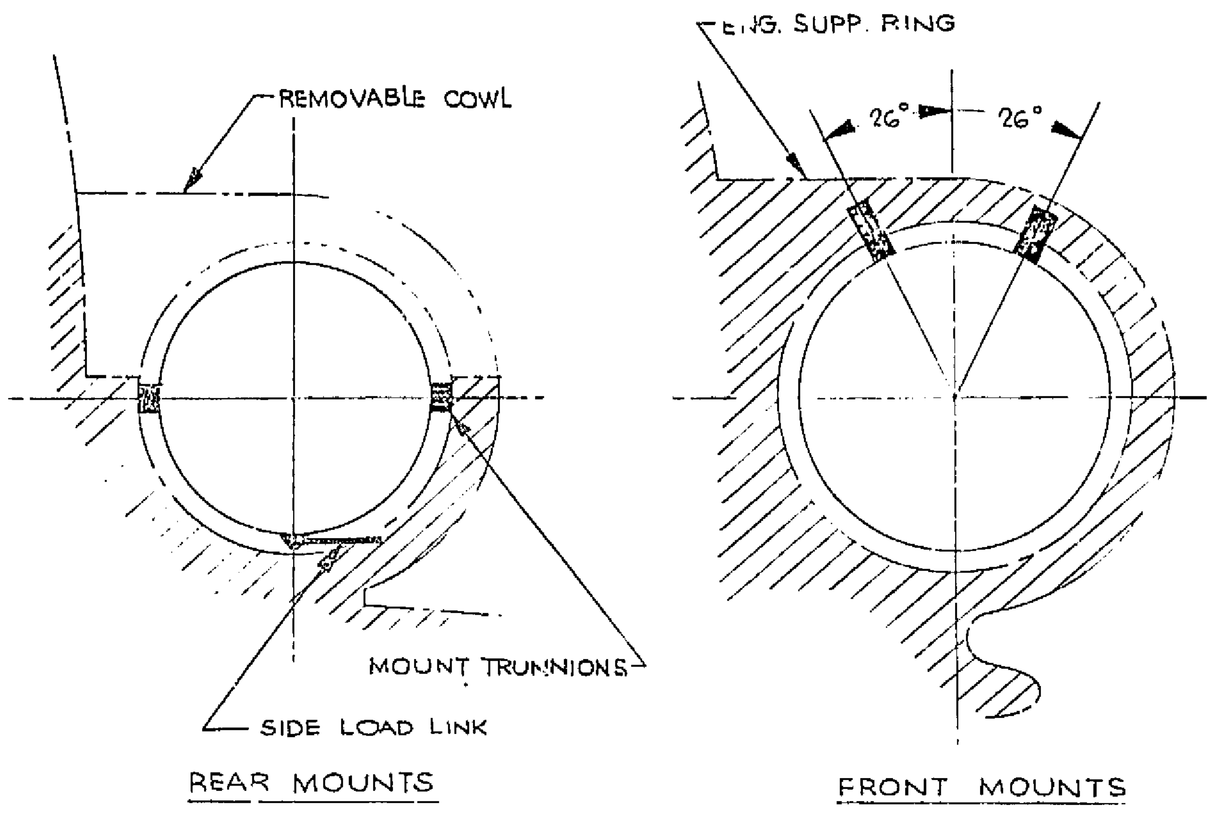


FIGURE 4-22
M260-RTA-1
ENGINE MOUNTS



designed to allow for thermal expansion. There are two in-line angulation bellows aft of the engine exhaust flange to allow for air duct deflections and to isolate these loads from the engine case.

The fan section has a three point equispaced mount system. The upper inboard mount is designed as the main anchor supporting inboard/outboard loads, vertical loads, forward/aft loads and is used for ground handling. The lower inboard mount is designed to support inboard/outboard loads and forward/aft loads. The outboard mount is designed to support forward/aft loads and to double as the fan removal pivot point.

Outboard Engine Removal - The removal of the outboard engine is illustrated in Figure 4-23, and the removal sequence would be as follows:

(a) Remove air turbine starter (ATS) inlet duct and forward nacelle intake. Remove rear mount access doors and rear duct clamp access door.
(b) Install engine removal rails.
(c) Disconnect rear duct clamp.
(d) Disconnect engine mounts.
(e) Remove engine forward along the removal rail.

Center Engine Removal - Three basic removal methods were studied for the center engine.

(a) Removal of engine and diverter valve duct assembly.
(b) Removal of engine and angled rear duct.
(c) Controlled removal of engine and angled rear duct.

Methods (a) and (b) were eliminated for the following reasons: Method (a) - This approach required removal of large sections of cowl, associated structures, plus diverter valve actuation. The engines and duct section must be intricately maneuvered up and aft with the corresponding high risk of damage. Method (b) - This approach required removal of the ATS and inlet duct. The engine must be intricately maneuvered to clear mount structure and engine inlet cone, with corresponding high risk of damage.

Method (c) provided for a completely controlled removal sequence with no intricate engine maneuvering and low risk of damage. The removal is illustrated in Figure 4-24 and requires the following sequence:

(a) Remove required portions of cowl, inlet, and ATS inlet duct.
(b) Disconnect aft angled duct clamp.
(c) Attach hoist to existing ground handling lugs adjacent to rear engine mounts.

C-2

FIGURE 4-23
M260-RTA-1
OUTBOARD ENGINE REMOVAL

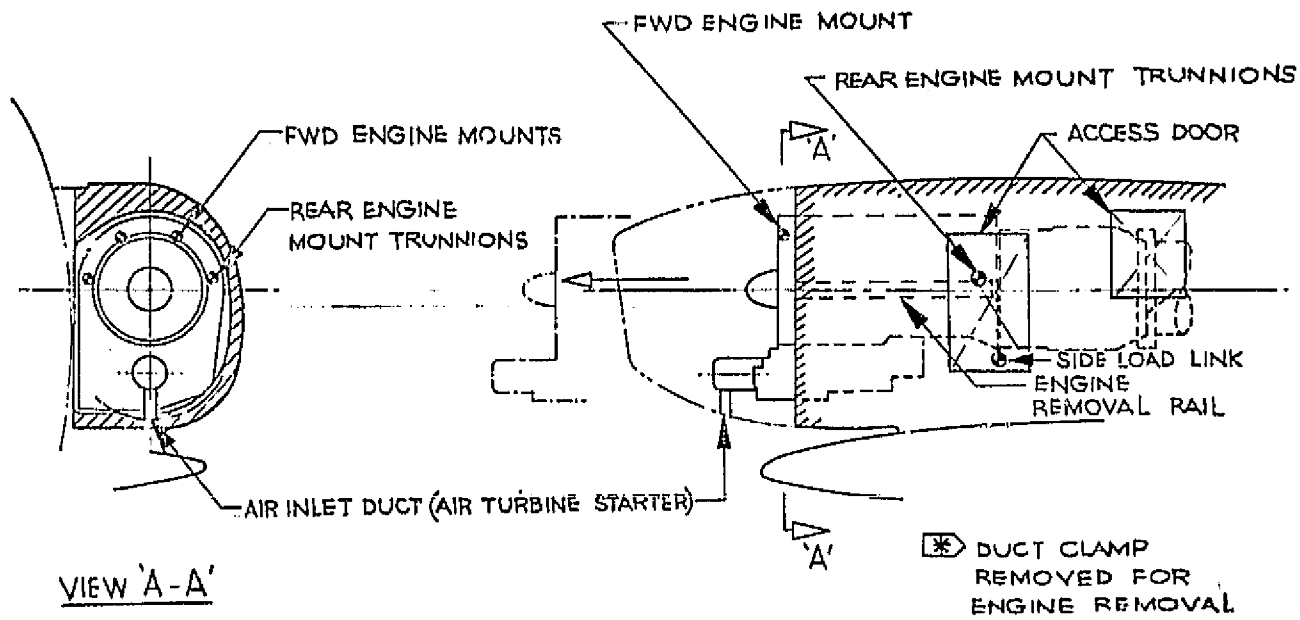
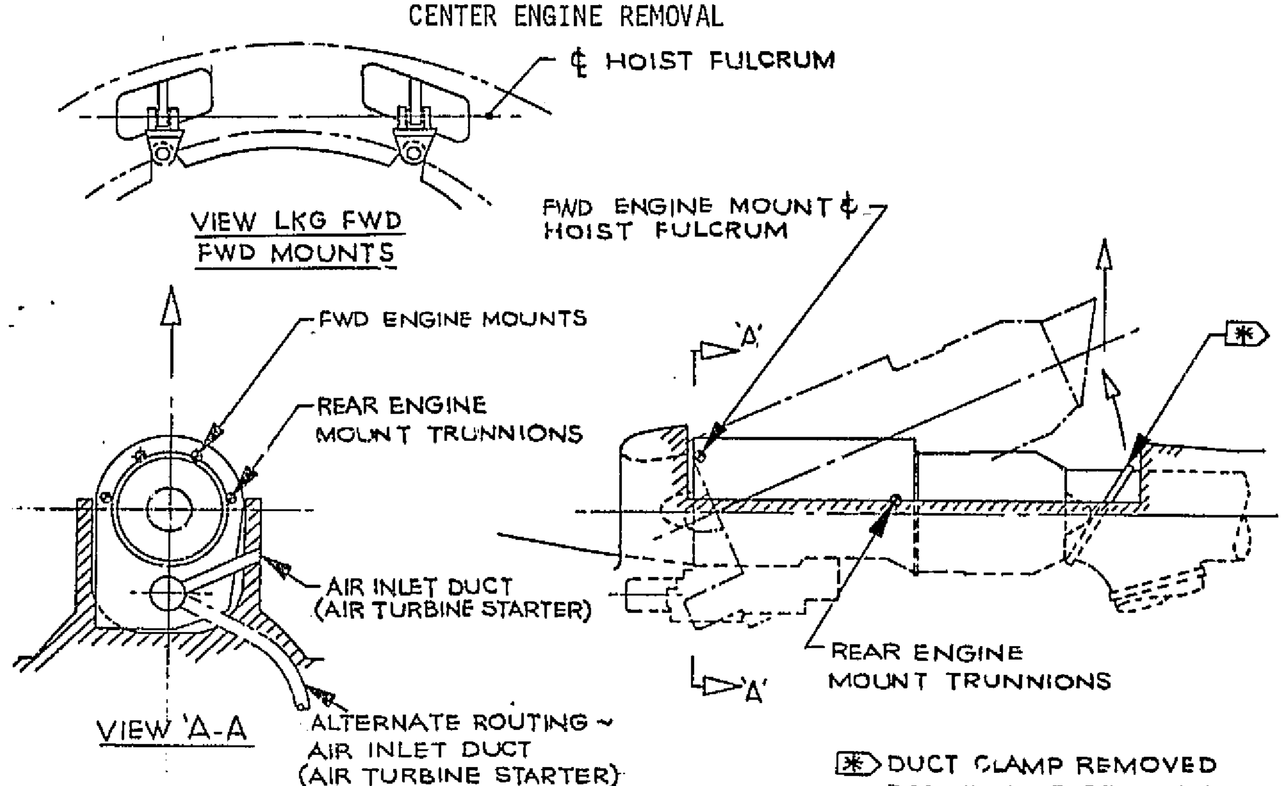


FIGURE 4-24
M260-RTA-1
CENTER ENGINE REMOVAL



(d) Disconnect rear engine mounts and hoist engine using forward engine mounts as hoist pivot points.

(e) Disengage front mount pins and remove engine assembly vertically and over the wing.

Fan Mounted Accessory Installation - The accessory arrangement shown in Figure 4-25 was coordinated with GE to interface with the existing fan gearbox arrangement. The length of the CSD/generator package required a modified aft fan fairing to provide adequate clearance. The service lines, i.e., hydraulic and electrical, are routed through a service strut. The strut position provides a neutral installation and the lines are routed along the wing rear spar to the aircraft fuselage. The aft fan fairing and service strut are easily removable for maintenance.

Air Turbine Starter (ATS) Installation - The ATS is mounted on the forward face of the existing engine gearbox. The ATS inlet is located directly below the engine in the forward nacelle on the outboard engines and on the left side of the forward nacelle on the center engine. An alternate location for the center engine ATS inlet would be on the left side, lower fuselage. The ATS exhausts directly into the engine compartment.

Lift Fan Removal - A controlled removal technique is used for the forward fan assembly to minimize the risk of damage. The forward existing mount point substitutes as the ground handling pivot point. The remaining two existing mount points provide for ground handling attachment. The removal sequence is as follows:

(a) Remove cowl inlet duct to provide access and removal clearance.

(b) Disconnect gimbal, valve actuator, and service lines.

(c) Connect ground hoist to two aft mount ground handling lugs.

(d) Disengage fan assembly from two aft mounts.

(e) Lift fan assembly using remaining forward mount as pivot.

(f) Disengage forward mount pin and remove fan assembly vertically.

The fan removal is illustrated in Figure 4-26.

Lift/Cruise Fan Removal - Controlled removal technique is also used for the lift/cruise fan assembly. The outboard existing mount point also serves as the pivot point during fan removal. The remaining two existing mount points provide for ground handling attachment. The removal sequence is as follows:

(a) Remove forward nacelle and associated structures to gain necessary access and removal clearances.

FIGURE 4-25
M260-RTA-1
FAN MOUNTED
ACCESSORY INSTALLATION

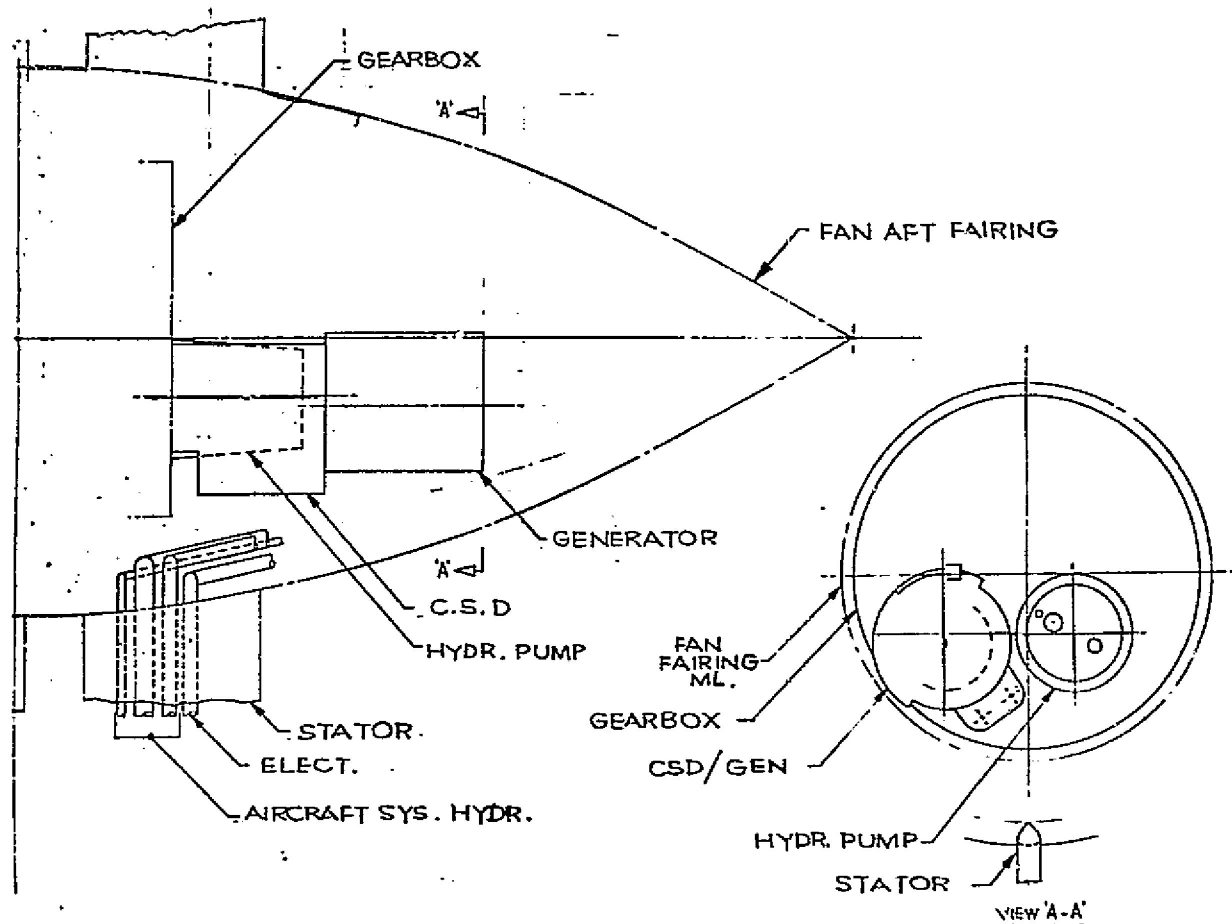
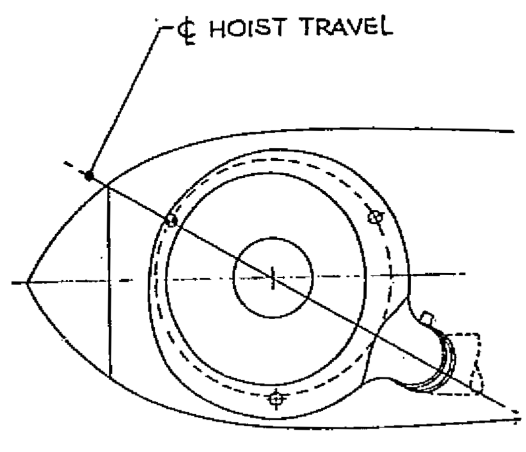
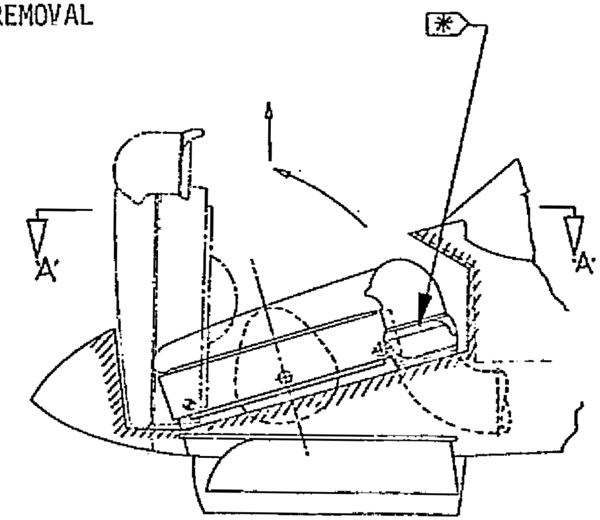


FIGURE 4-26
M260-RTA-1
FORWARD FAN REMOVAL



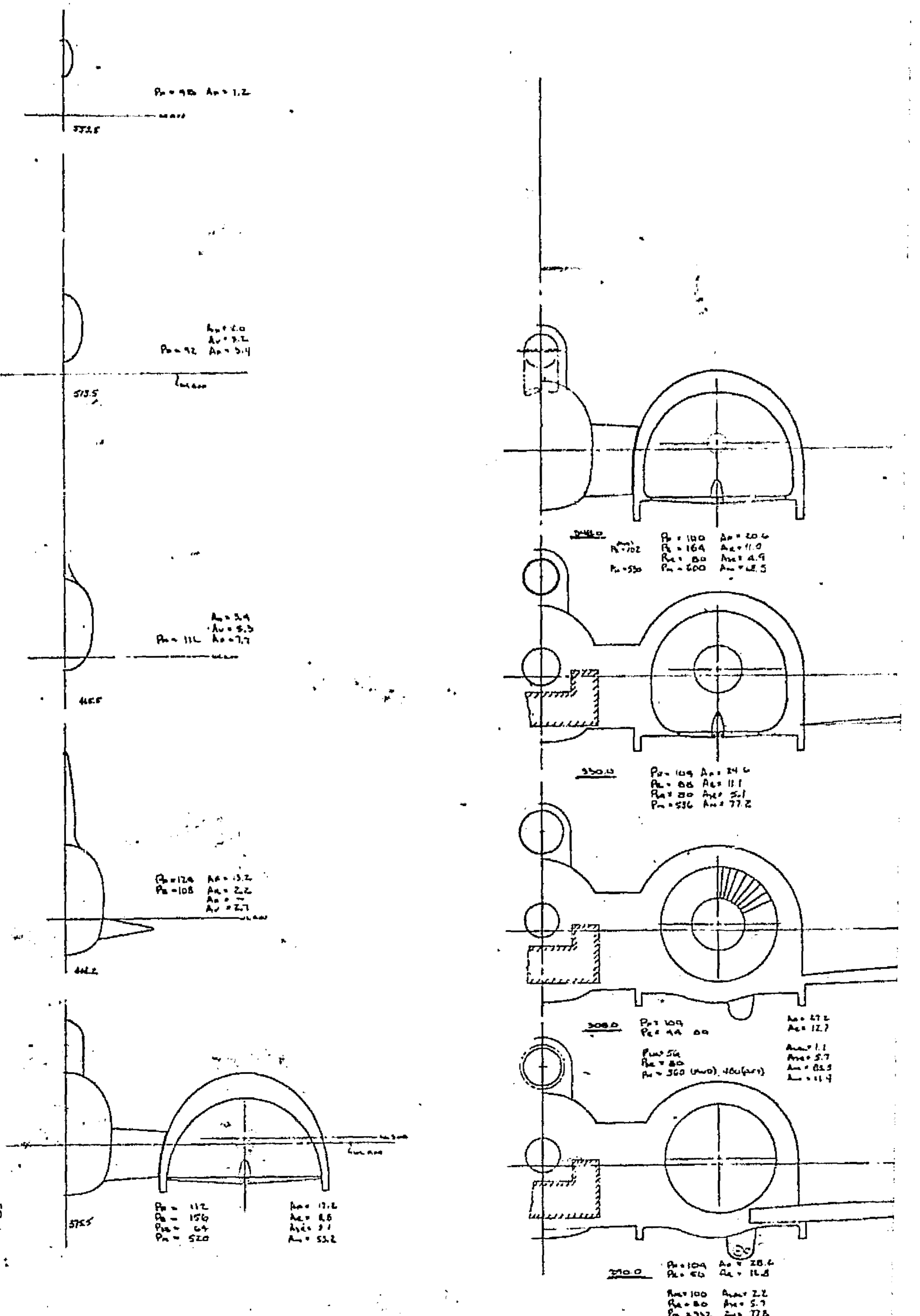
VIEW 'A-A'



● EXISTING MOUNT POINT AND HOIST FULCRUM.

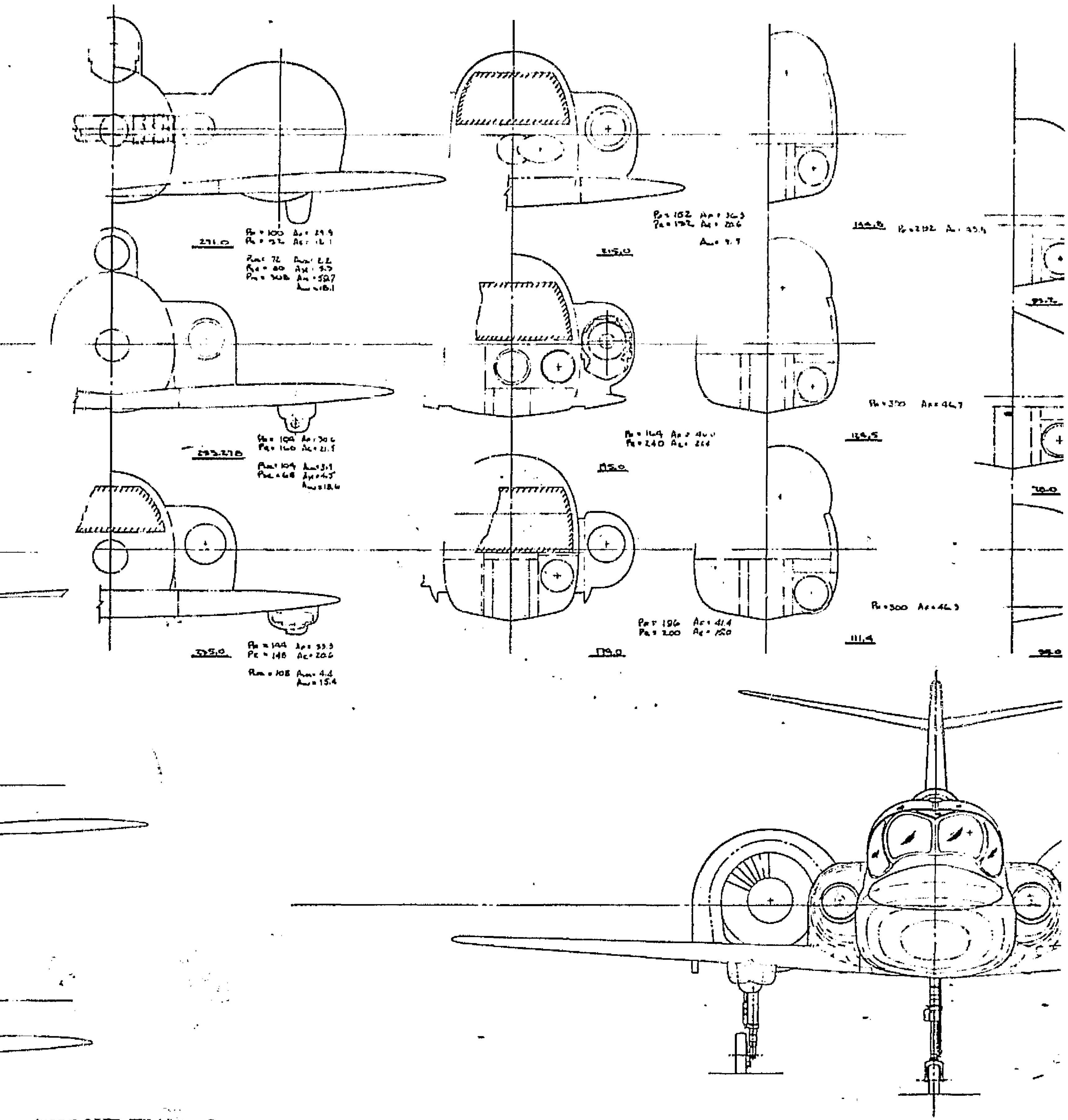
⌖ EXISTING MOUNT POINT AND HOIST POINT

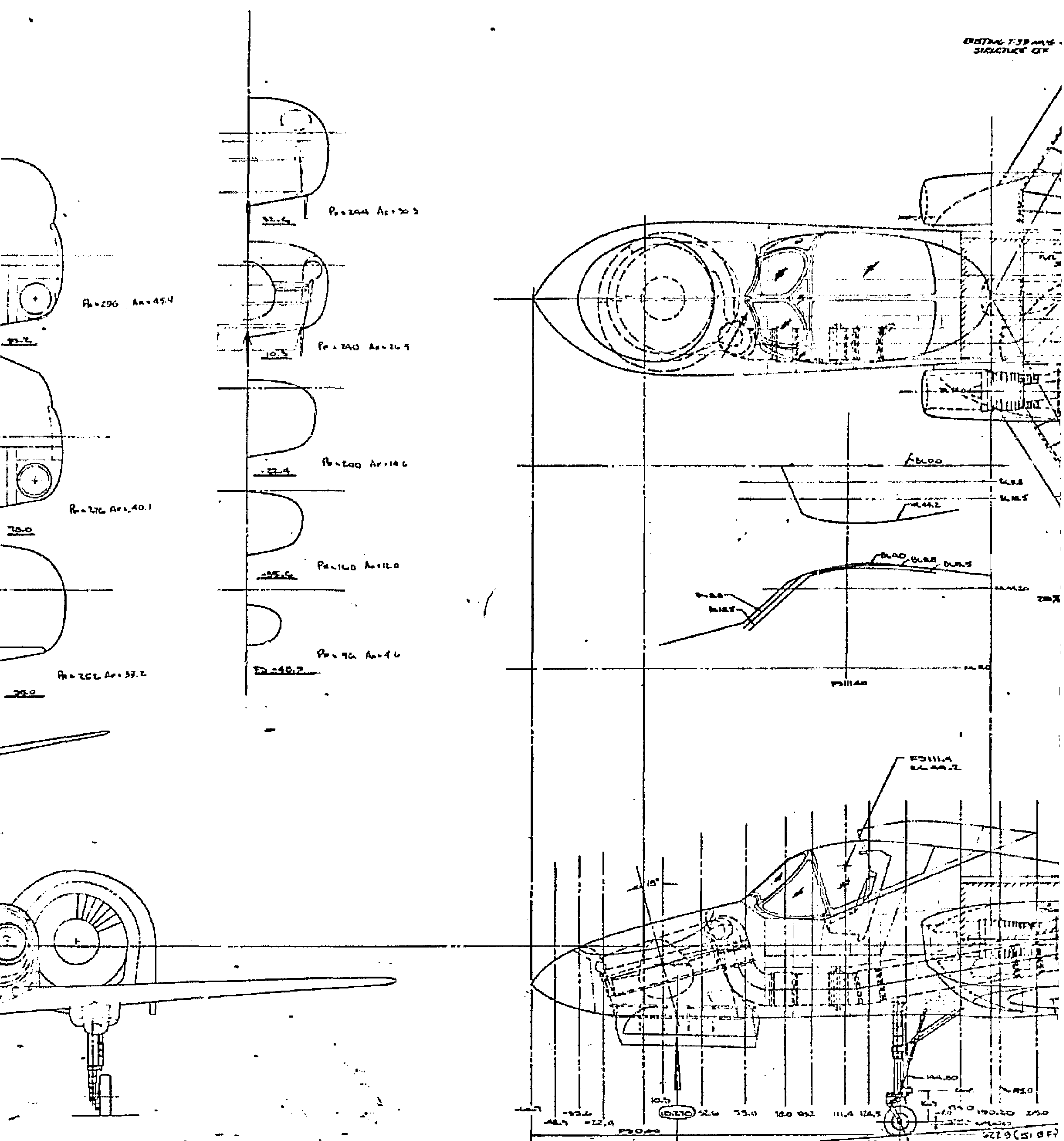
[*] DISCONNECT DUCT & GIMBAL FOR ENGINE REMOVAL



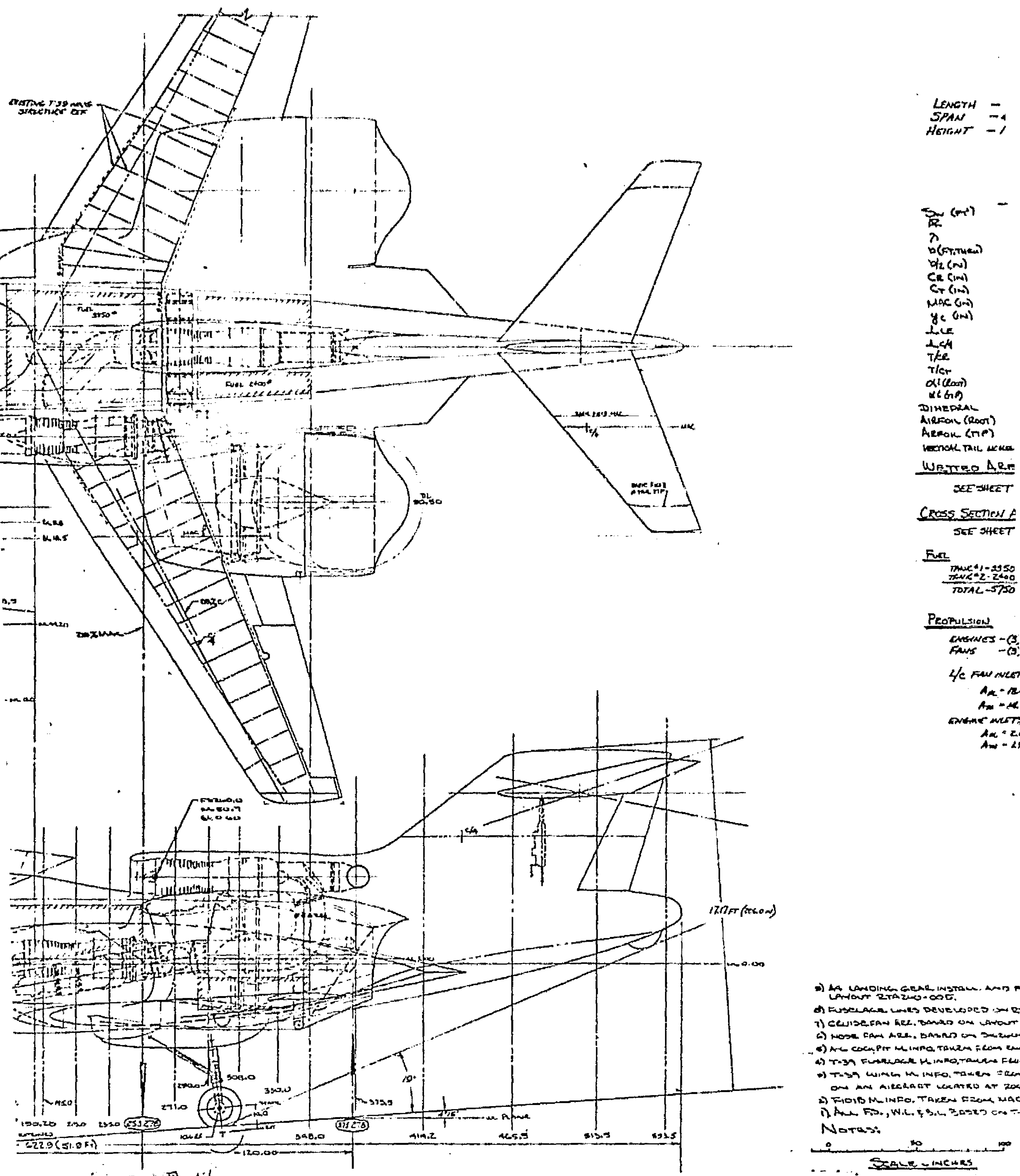
ORIGINAL PAGE IS
OF POOR QUALITY

FOLDOUT FRAME 1

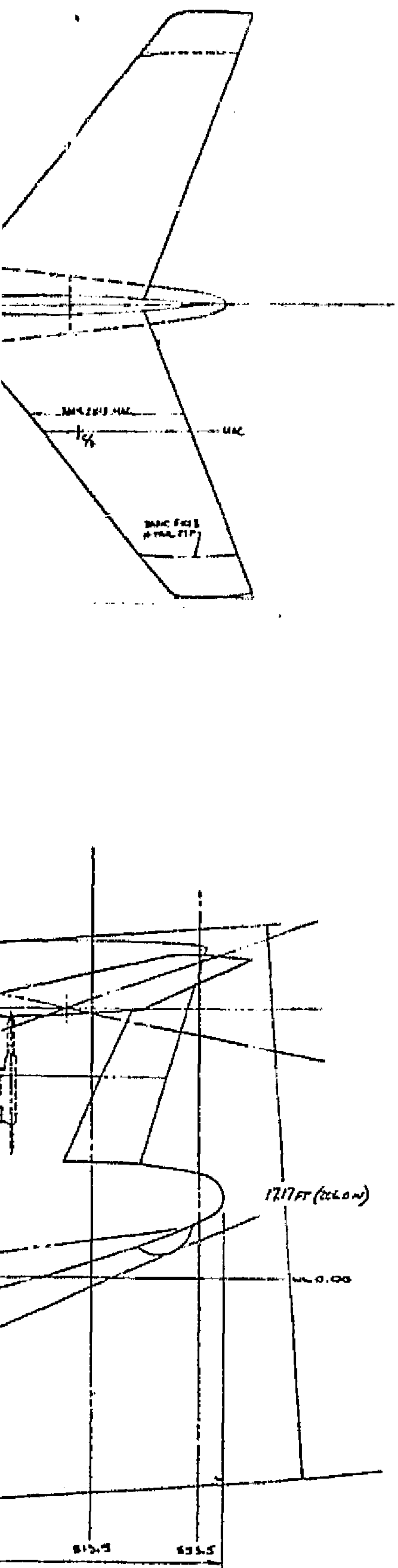




FOLDOUT FRAME 3



FOLDOUT FRAME 4



LENGTH – 51.9 FT
SPAN – 44.43 FT
HEIGHT – 17.17 FT

PHYSICAL CHARACTERISTICS

	T38 WING	F1010 V-TAIL	F1010 H-TAIL (BASIC)	F1018 H-TAIL (MOD)
S_W (FT²)	342.05	89.88	75.41 TRUE	82.02 TRUE
AR	5.77	0.445	3.501	3.959
λ	.325	.508	.4886	.3705
b (FT, TRUE)	44.43	7.50	15.6 (PLAN) (186.0)	18.02 (PLAN) (216.24)
b/2 (IN)	266.56	—	94.4 (TRUE)	109.8 (TRUE)
C_R (IN)	139.86	180.0	78.5	78.5
C_T (IN)	44.91	91.6	30.0	29.07
MAC (IN)	100.6	140.6	58.88	57.57
$\bar{y}_c$ (IN)	110.39	40.12	41.27	46.49
Λ_{LE}	32.35°	35°@20.85%C	35°@29.34%C	35°@29.34%C
$\Lambda_{C/4}$	20.56°	46°	31.25°	31.25°
T/CR	11.30%	7.0%	7.0%	7.0%
T/CT	9.36%	7.0%	6.0%	—
α_i (ROOT)	0°			
α_i (TIP)	-2° 54'			
DIHEDRAL	3.14°		10°	
AIRFOIL (ROOT)	64A SERIES (NASA MOD)	NACA65A007	NACA65A007 MOD	
AIRFOIL (TIP)	64A SERIES (NASA MOD)	NACA65A007	NACA65A006 MOD	

VERTICAL TAIL HEIGHT-AREA RATIO = .85

WETTED AREA

SEE SHEET 2

CROSS SECTION AREA

SEE SHEET 3

FUEL

TANK #1 - 3350
TANK #2 - 2400
TOTAL - 5750#

PROPULSION

ENGINES – (3) G.E. J97 GAS GENERATORS
FANS – (3) LF450 (PR = 1.32)

L/C FAN INLETS

A_{HL} = 18.4 FT²
A_{TH} = 14.5 FT²

ENGINE INLETS

A_{HL} = 2.49 FT² EACH
A_{TH} = 1.92 FT² EACH

9) A4 LANDING GEAR INSTALL. AND FAIRING BASED ON LAYOUT RTA260-005.
8) FUSELAGE LINES DEVELOPED ON RTA260-004 & 3K260Y-31815 (SHT 6)
7) CRUISE FAN ARR. BASED ON LAYOUT RTA260-003.
6) NOSE FAN ARR. BASED ON 3K260Y-31815 DATA.
5) A4 COCKPIT W.L. INFO. TAKEN FROM ENLARGEMENT OF 3VIEW. ALL DWG.
4) T-39 FUSELAGE W.L. INFO. TAKEN FROM ENLARGEMENT OF T.O. DWG.
3) T-39 WING W.L. INFO. TAKEN FROM ACTUAL MEASUREMENTS ON AN AIRCRAFT LOCATED AT ROCKWELL MAINT. CTR. ST. LOUIS
2) F1018 W.L. INFO. TAKEN FROM MAT. LOFT SHEET #33-64
1) ALL F.S., W.L. & B.L. BASED ON T-39 AIRCRAFT COORDINATES.

NOTES:

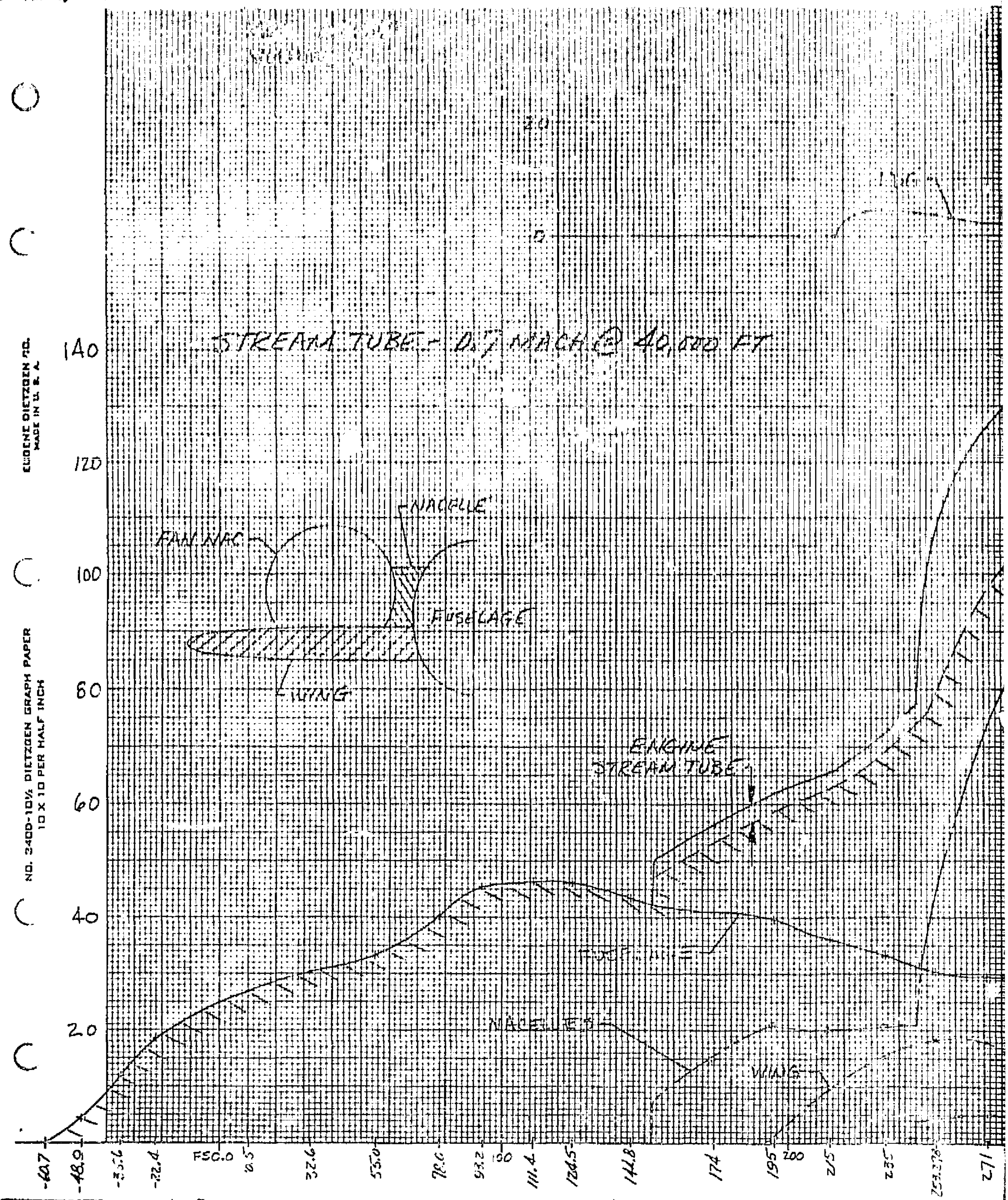
0 50 100

SCALE – INCHES

DWN	MCDONNELL AIRCRAFT COMPANY SAINT LOUIS, MISSOURI
CHK	MCDONNELL DOUGLAS CORPORATION
STR	RESEARCH TECHNOLOGY AIRCRAFT – REDUCED COST STUDY, (3) ENG., (3) FAN
SIZE J	CODE IDENT NO. 76301 — M260-RTA-1 (B)
SCALE 1/20	SHEET 1 OF 3

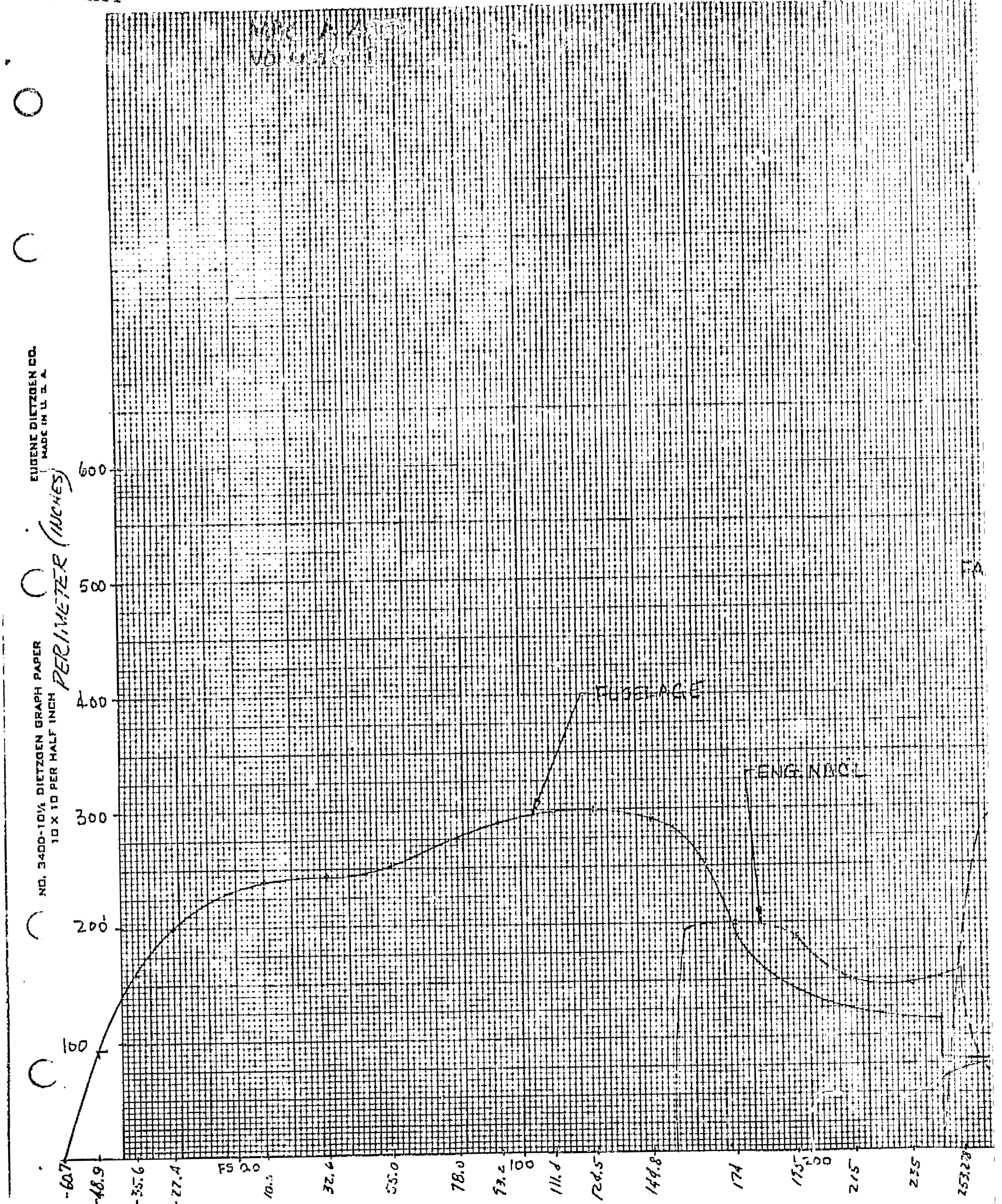
FOLDOUT FRAME 5

ORIGINAL PAGE IS
OF POOR QUALITY



FOLDOUT FRAME

ORIGINAL PAGE IS
OF POOR QUALITY



FOLDOUT FRAME 1

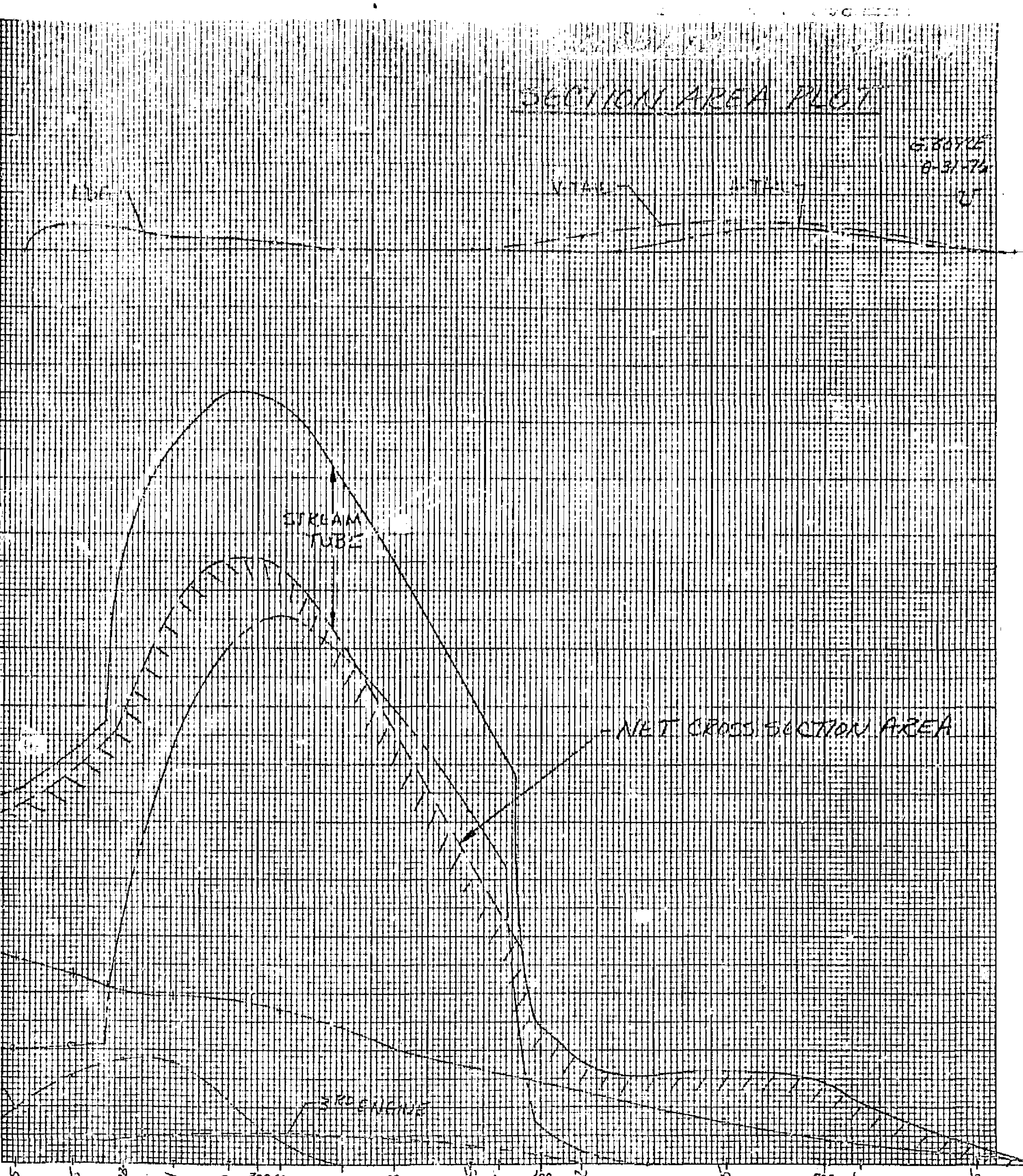
ADVANCED TECHNOLOGY AIRCRAFT

WETTED AREA

G. Boyce
8-30-76

FUSELAGE -	658 FT²
ENG. NACELLES -	247
FAN NACELLES -	430
WING -	400
H-TAIL	158
V-TAIL	149
3RD ENG -	79
LDG -	64
TOTAL	2185 FT²

FOLDOUT FRAME 2



FOLDOUT FRAME

(b) Remove lower strut fairing and disconnect service lines.

(c) Disconnect feeder duct upstream of ETaC valve. Disconnect valve actuator.

(d) Attach ground handling hoist to upper inboard mount point.

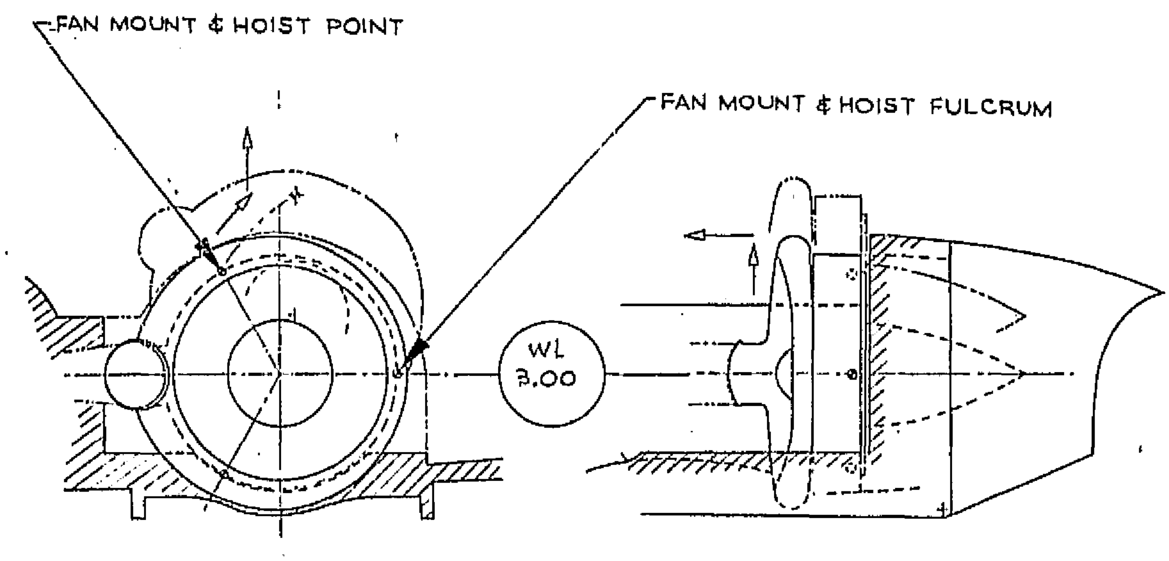
(e) Disconnect upper and lower inboard mounts.

(f) Lift fan assembly about pivot point until fan assembly weight is supported by the hoist.

(g) Disengage pivot pin and remove fan assembly forward.

The lift/cruise fan assembly removal is illustrated in Figure 4-27.

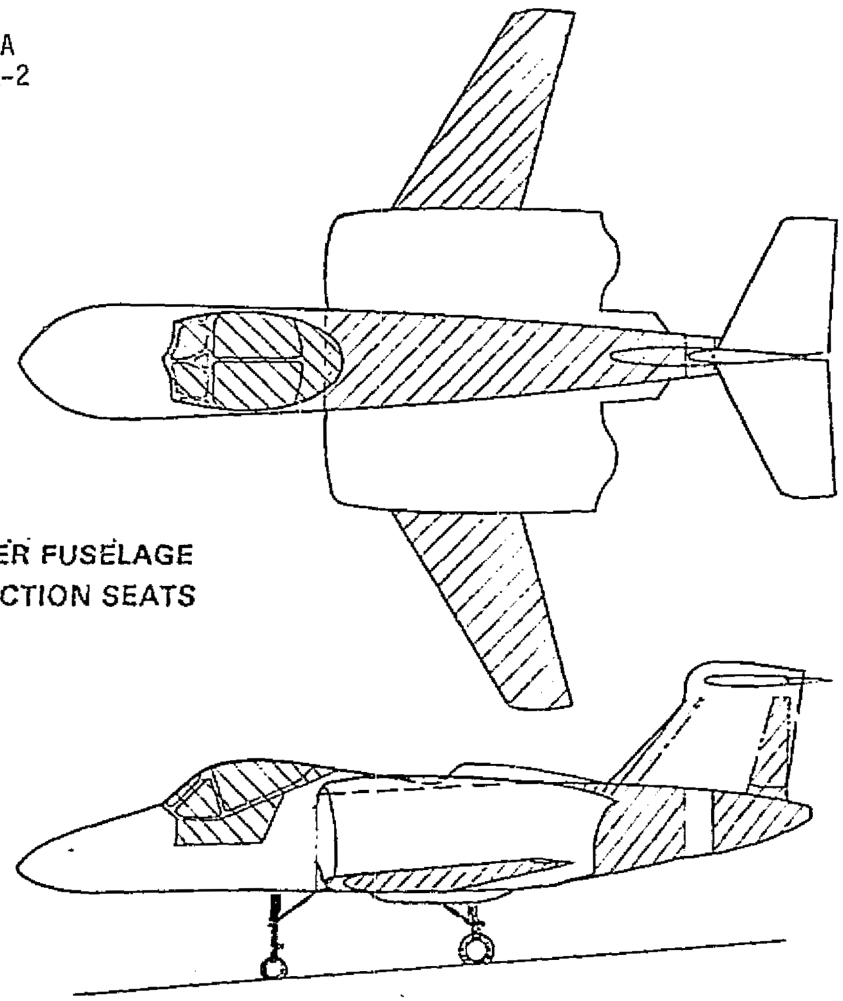
FIGURE 4-27
M260-RTA-1
FAN REMOVAL



4.2 MECHANICAL RTA

4.2.1 GENERAL ARRANGEMENT - The mechanical RTA airframe is virtually identical to the turbotip RTA and reflects a maximum utilization of GFE components as illustrated in Figure 4-28. The primary difference between the two aircraft is that the T-39 aft fuselage and tail are modified rather than using the F-101 empennage. This change was made to reduce weight. The propulsion system consists of three XT701 engines powering three mechanically interconnected 62-inch diameter variable pitch fans (see Volume II). The principal weights and geometric characteristics are shown in Figure 4-29. The general arrangement,

FIGURE 4-28
MECHANICAL RTA
MODEL 260-RTA-2



T-39 WING AND CENTER FUSELAGE
A-6 COCKPIT AND EJECTION SEATS
A-4 LANDING GEAR

FIGURE 4-29
DIMENSIONAL AND DESIGN DATA
MECHANICAL RTA

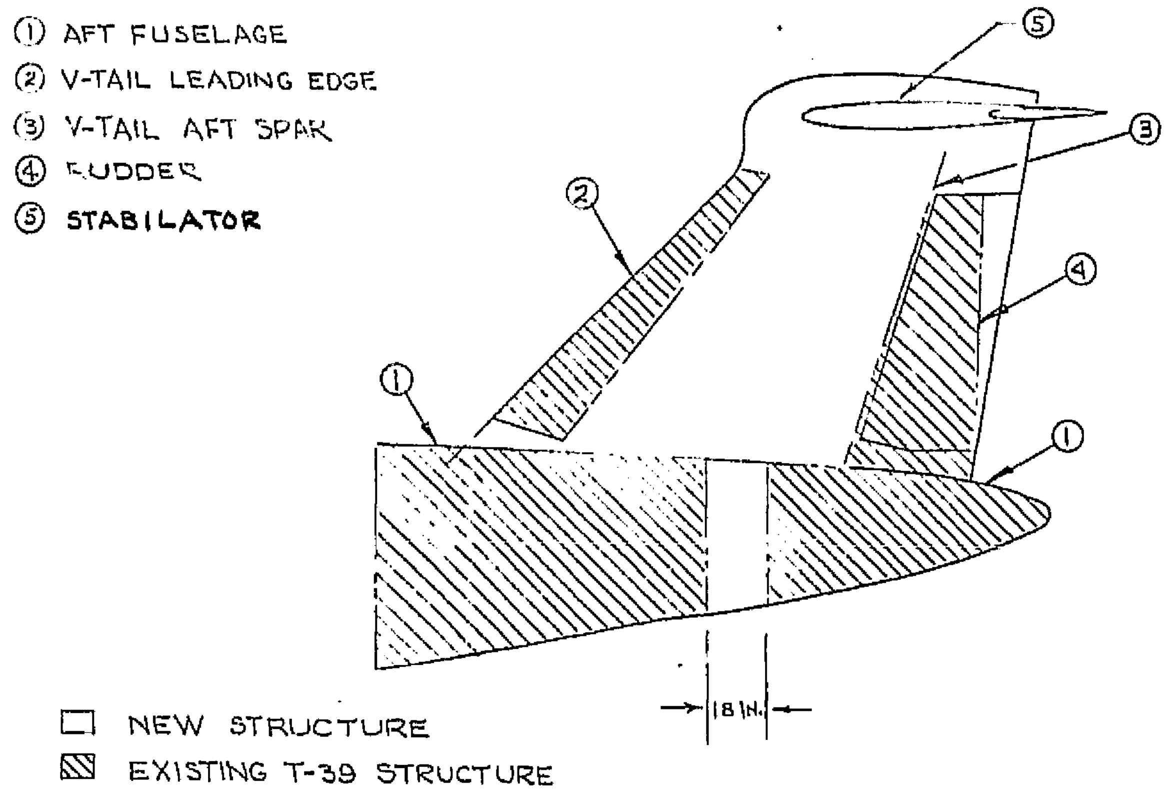
STOGW	(lb)	27,913
	(N)	124,157
VTOGW	(lb)	25,670
	(N)	114,180
OWE	(lb)	20,260
	(N)	90,116
Overall Length	(ft)	52.09
	(m)	15.88
Wing Span	(ft)	44.43
	(m)	13.54
Height	(ft)	16.6
	(m)	5.06
Crew Provisions	(No.)	2
Max. Internal Fuel	(lb)	5,400
	(N)	24,019

		Wing	Horizontal Tail	Vertical Tail
S	(ft^2)	342	90.97	77.0
	(m^2)	31.77	8.45	7.15
AR		5.77	4.095	.914
λ		.323	.400	.454
b	(ft)	44.43	19.3	8.417
	(m)	13.54	5.8	2.57
$\Lambda_{c/4}$		28.56	23.0	38.3
t/c (% Root/Tip)		11.3/9/36	10.0/8.0	10.0/[illegible].0
Airfoil (Root)		64A Series	64A010	63A010
Airfoil (Tip)		Modified	64A008	63A010

wetted areas, and cross sectional areas of the aircraft are shown in Drawing Model 260-RTA-2, Sheets 1, 2 and 3.

4.2.2 AIRFRAME - The fuselage is essentially the same as the turbotip RTA with the exception that the T-39 center and aft fuselage are modified to accommodate the lower fuselage mounted engine and the mechanical transmission. The wing and cockpit are identical to that used for the turbotip RTA. The vertical tail is a modified T-39, Figure 4-30, and the horizontal tail is new construction vice the F-101 empennage used on the turbotip RTA. This change in the tail saved about 544 lb, thereby providing an adequate engine out T/W margin.

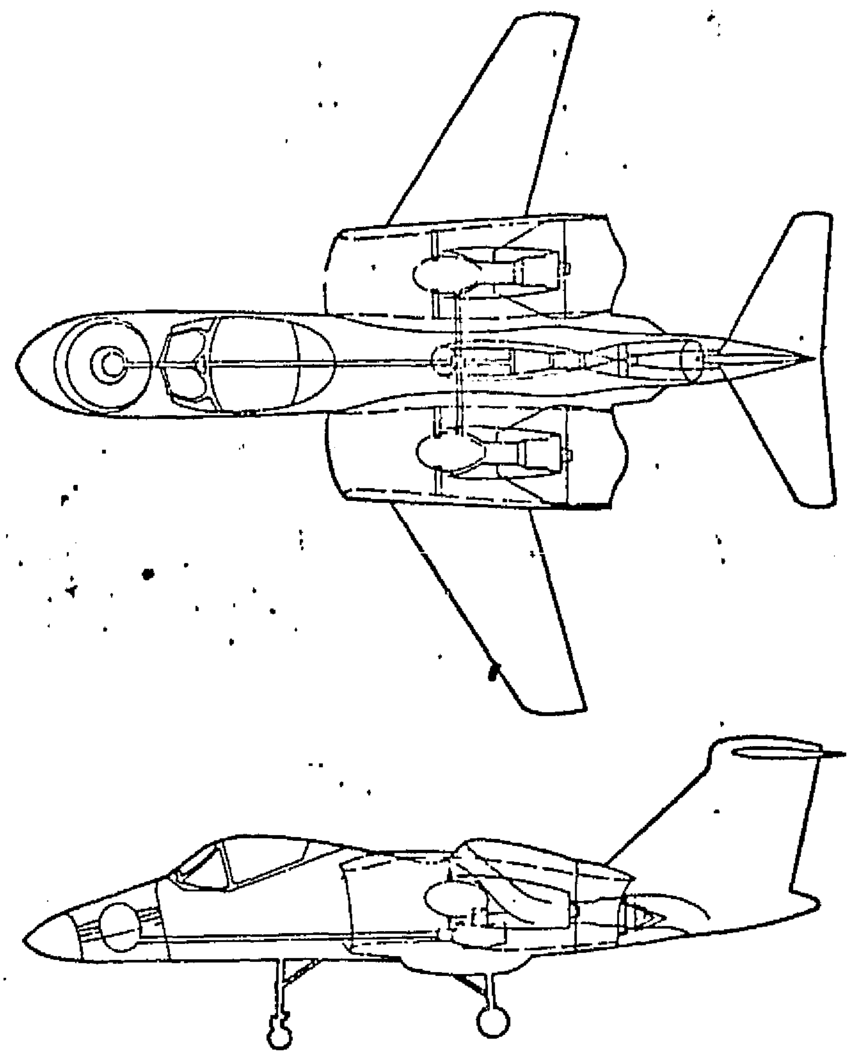
FIGURE 4-30
M260-RTA-2
EMPENNAGE/AFT FUSELAGE



4.2.3 PROPULSION SYSTEM - The propulsion system, illustrated in Figure 4-31, is composed of three Allison XT701 engines driving three Hamilton Standard variable pitch fans. The engines and fans are interconnected by a mechanical transmission system consisting of shafts, combiner gearbox and a lift fan clutch, Figure 3-30.

The lift/cruise fan assemblies (two) are located above the wing and adjacent to the fuselage. The engine, mounted directly behind the fan, is integrated into the lift/cruise nacelle. Fan supercharged air is inducted into the engine and exhausted together with fan by-pass air through the thrust vectoring nozzle. Lift/cruise fan speed is reduced below free turbine speed by a planetary gear set. An overrunning clutch installed ahead of the engine between the engine and the planetary set permits overrunning in the event of an engine failure. The Allison XT701 engine is coupled with a Hamilton Standard variable pitch fan to form the compound turbofan/shaft engine unit identified as the Allison PD370-25A. The fan assembly includes a spiral bevel gear set to transmit power to the other two fans for control power or power transmission during an engine out condition.

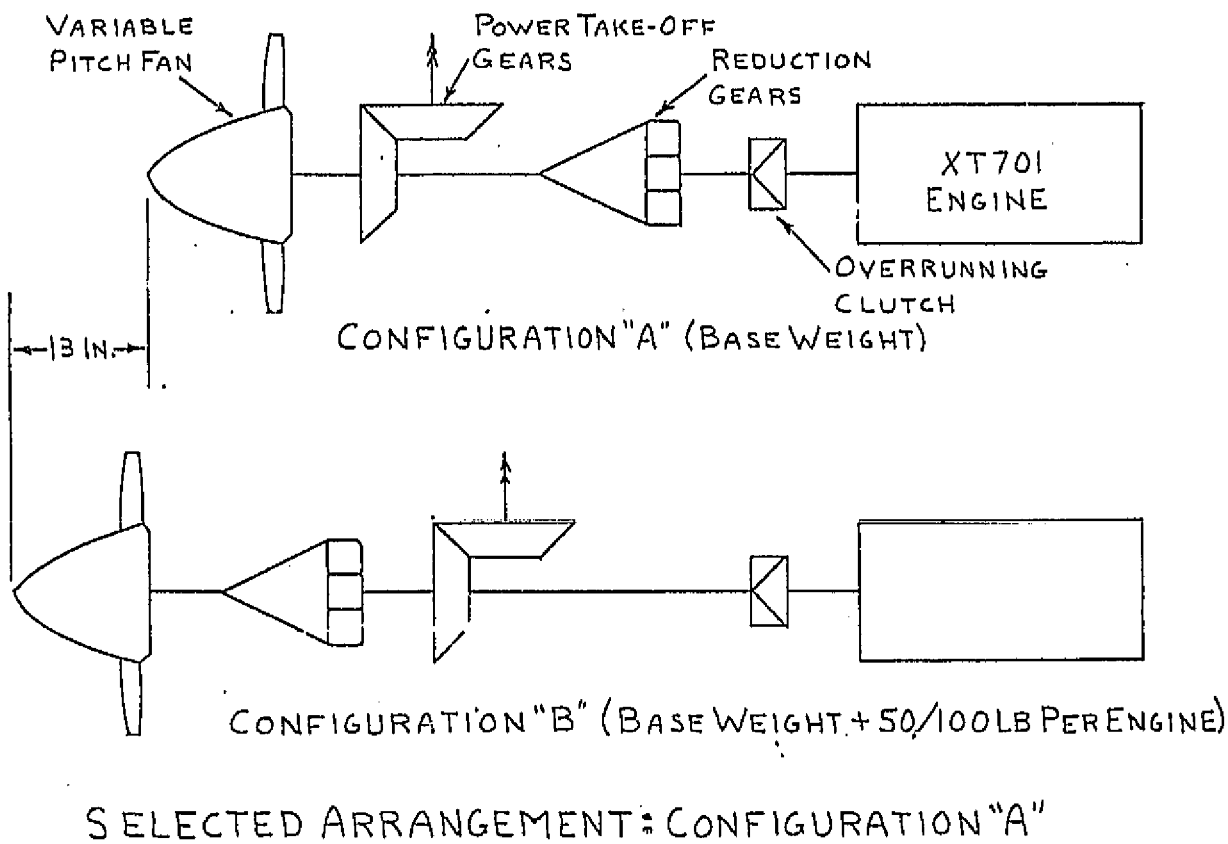
FIGURE 4-31
MODEL 260-RTA-1
PROPULSION SYSTEM



The lift/cruise fan gear arrangement, shown in Figure 4-32, was selected after discussions with DDA. The selected arrangement resulted in a 13 inch shorter assembly and a weight savings of 50 to 100 lb per engine. Another benefit of this arrangement is that if a reduction gear set fails, the affected fan would still be operable for both thrust and control.

The Hamilton Standard variable pitch lift fan is located forward of the cockpit. The fan assembly includes a spiral bevel gear set to transmit power from the forward drive shaft. Power is transmitted from the combiner gearbox to the lift fan through a wet disk clutch which provides for disengagement of the fan during ground operation, if desired, or in the cruise mode.

FIGURE 4-32
M260-RTA-2
LIFT/CRUISE FAN GEAR ARRANGEMENT SELECTION



The third (center) engine drives into the combiner gearbox through a spur gear set into the transmission system. The combiner box also accepts power from the two lift/cruise engines for distribution during control excursions and engine out operation.

The thrust vectoring and yaw controls are the same as discussed for the turbotip RTA.

A design and development study for the mechanical transmission system was subcontracted to Detroit Diesel Allison (DDA) through MCAIR Work Statement WS-SDPS-860. Study ground rules and results of the DDA effort are presented in Volume II of this report.

4.2.4 ACCESSORY INSTALLATION - A study was conducted of alternate ways to drive the hydraulic pumps and CSD/generators. One approach used a power takeoff from the combiner box to drive a separate gearbox for the two pumps and two CSD/generators. This approach resulted in additional development costs, a weight increase, and a potential single point failure for all hydraulic and electrical power. Since DDA was in the process of redesigning the engine accessories, it was decided to mount the hydraulic pumps and CSD/generator directly on the engine. This selected approach was cost effective and provided a dual power source.

The accessory configuration shown in Figure 4-33 was selected for the following reasons:

- The CSD and generator was readily accessible commensurate with the higher maintenance requirements.
- The oil pump was located near the bottom of the engine gearbox to provide optimum scavenging.
- The ATS was located so that the inlet duct (which is flush with the engine shroud) is accessible from the lower vector doors. A starter cart hose extension would preclude the necessity of having the ground crew place their arms through the actuated vector doors. The ATS exhaust is directed into the shrouded engine compartment.
- The main fuel pump and control are located adjacent to an existing stator thereby shortening fuel lines.
- The hydraulic pump was located above a drip shielded CSD/generator.

All fuel and water lines were routed through an existing major stator which was separate from the electrical and hydraulic lines. The aircraft hydraulic and electrical lines were routed through another existing major stator and were in turn separated in this stator. The fan variable pitch hydraulics were located in the forward section of this same stator. These stators would also house the forward/lower engine support struts. Due to the neutral accessory installation, the service lines change routing in the airframe section but not in the engine compartment thereby resulting in minimum engine buildup requirements. All interfaces were coordinated with DDA.

4.2.5 SUBSYSTEMS - The design approach for the aircraft subsystems was identical to that used in the turbotip RTA. The component differences, which are primarily associated with the mechanical propulsion system, are described in the following.

The hydraulic system differs in that thrust modulation is controlled by blade pitch actuators versus ETaC/TRM and the pumps are engine mounted. The hydraulic power generation and flight control component locations are shown in Figures 4-34 and 4-35, respectively. The landing gear and brake subsystems are identical.

FIGURE 4-33
M260-RTA-1
ENGINE MOUNTED ACCESSORY INSTALLATION

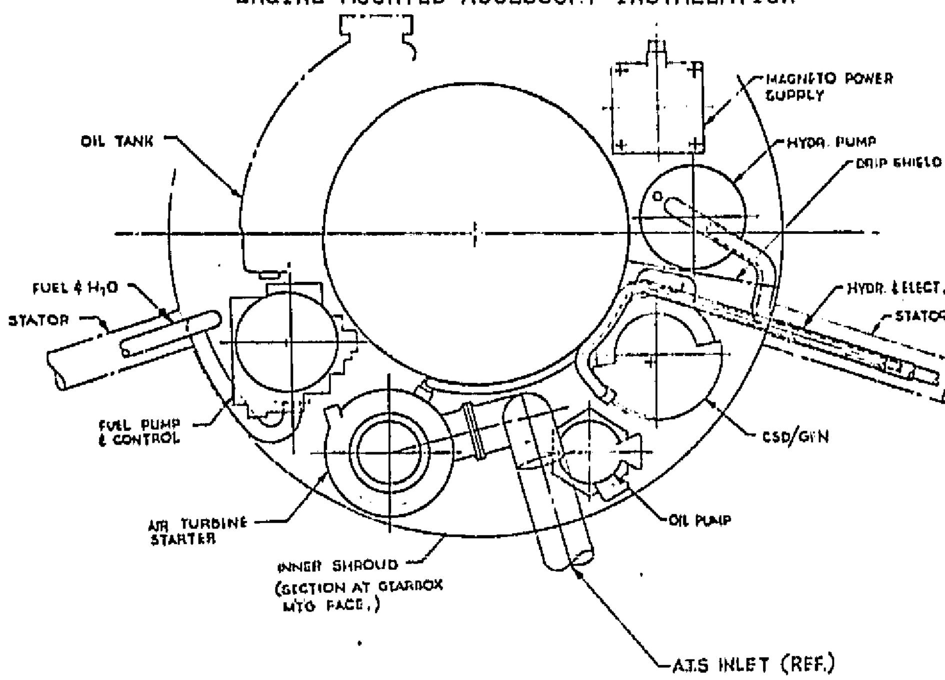


FIGURE 4-34
M260-RTA-2
MAJOR HYDRAULIC POWER GENERATION AND UTILITY COMPONENTS

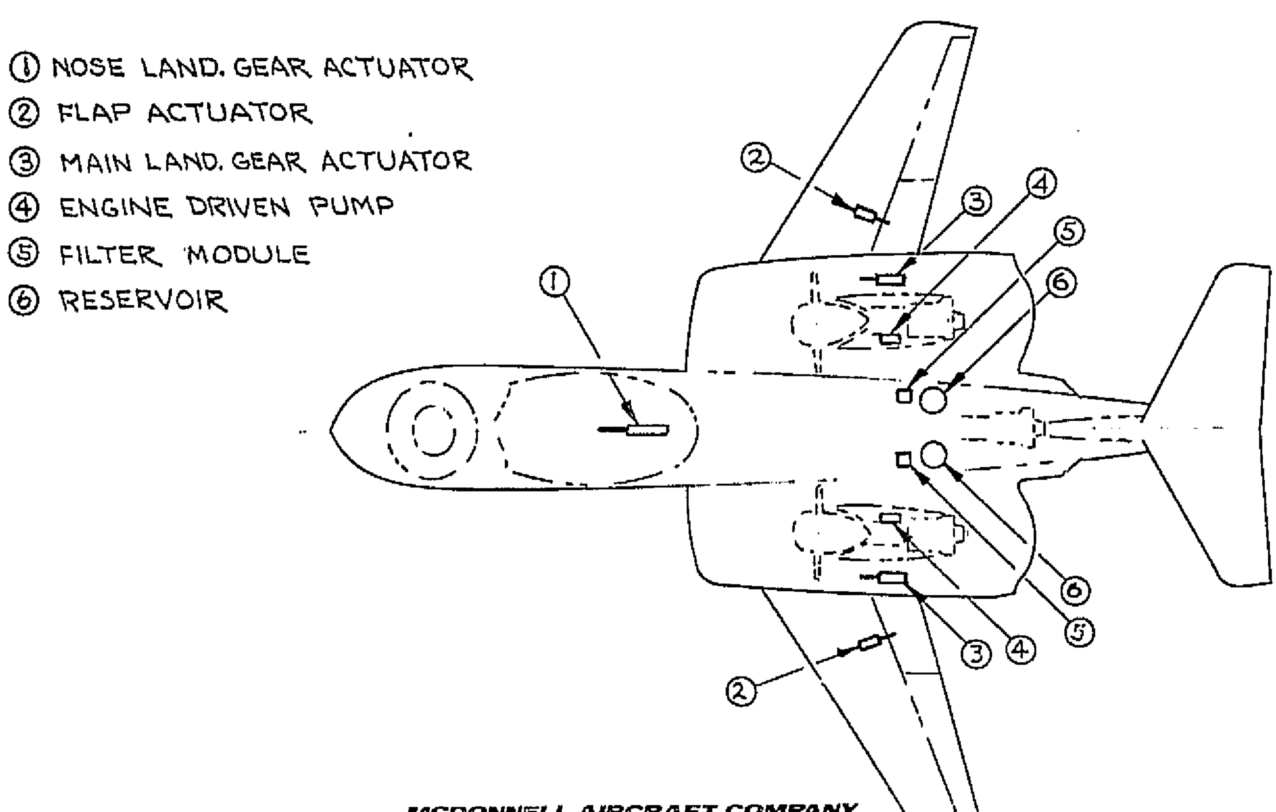
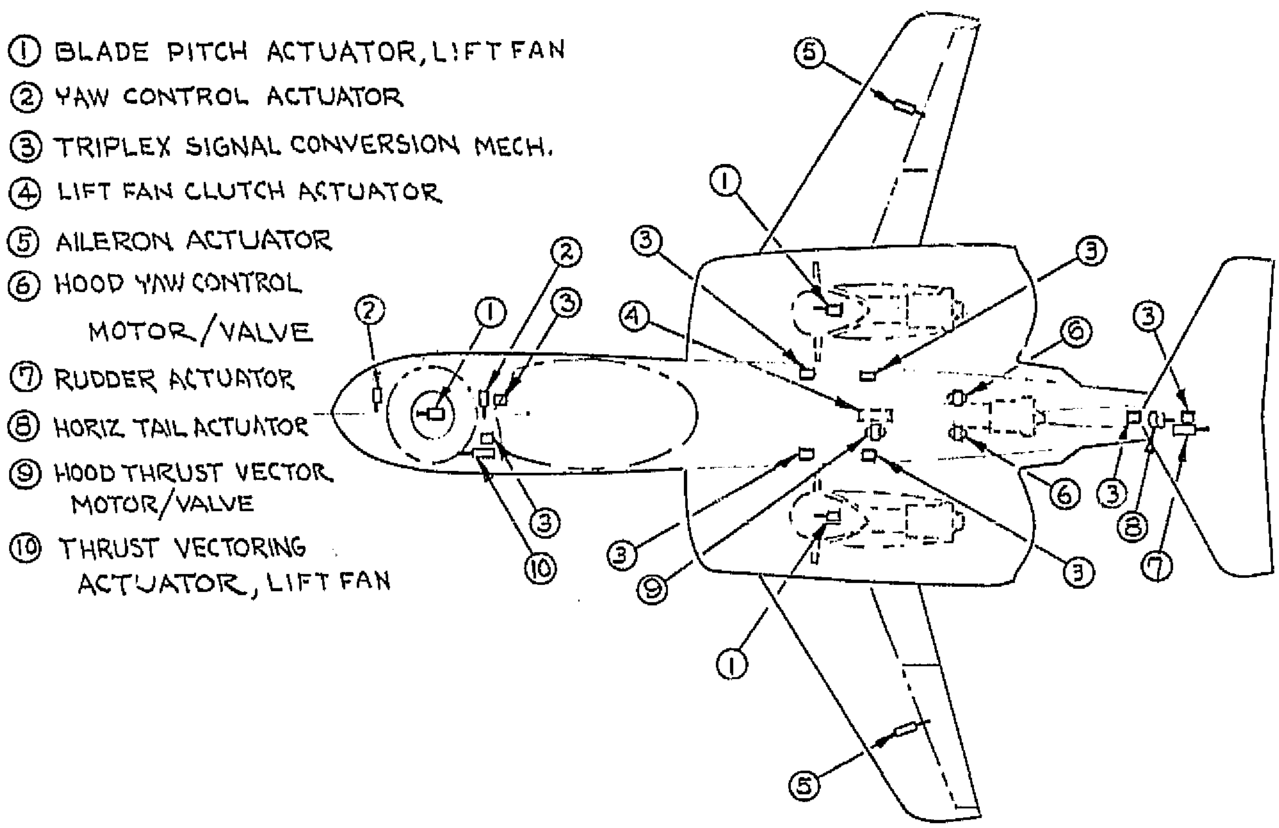


FIGURE 4-35
M260-RTA-2
MAJOR HYDRAULIC CONTROL COMPONENTS



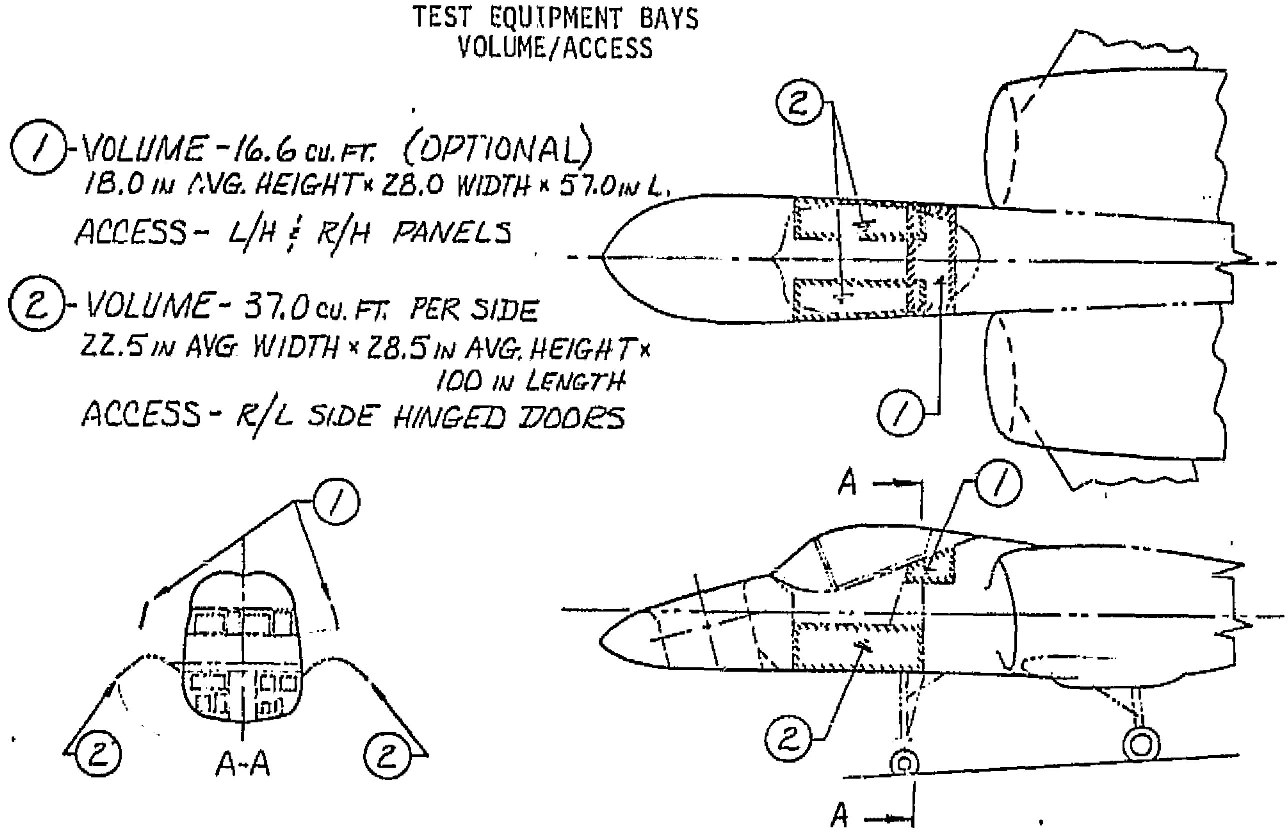
The electrical system is essentially the same except that the CSD/generators are mounted on the engines. Each CSD/generator is cooled through a fuel/oil heat exchanger which is integral with the engine fuel system. Since the engine speed range is lower than the turbotip fan speed range, the selected CSD can be used without modification.

The avionics are identical to the turbotip RTA system and additional space for test equipment is provided as shown in Figure 4-36. The fuel system is the same except that smaller fuselage fuel tanks are used. The tank capacities are 4700 lb in the feed tank and 700 lb in the transfer tank.

The primary flight controls are also the same except for the use of blade pitch control instead of ETaC. The power management controls are the same except the throttles are control-by-wire instead of mechanical.

The ECS is identical. An air cooled heat exchanger is provided to cool the combiner gearbox lube oil system.

FIGURE 4-36
M260-RTA-2
TEST EQUIPMENT BAYS
VOLUME/ACCESS

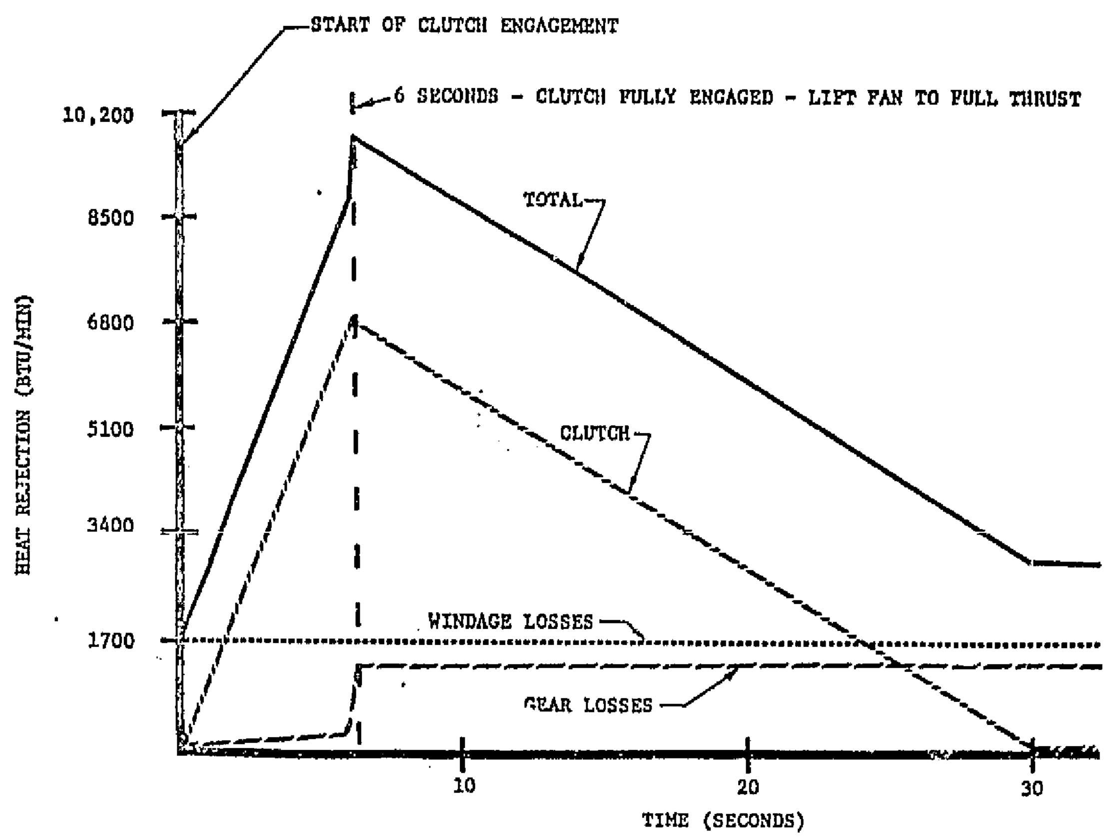


4.2.6 TRANSMISSION GEARBOX COOLING - The combiner gearbox/clutch assembly for the transmission system, discussed in Volume II of this report, has a significant heat rejection rate and requires cooling during all flight modes, Figure 4-37. This heat rejection consists of windage losses, gear losses, and clutch engagement heat dissipation. Since a large part of the aircraft operating life will be spent in the powered lift mode, it was decided to provide cooling capacity for only the windage and gear losses (3100 BTU/min) and consider the clutch engagement heat to be a transient condition.

Various heat dissipation methods, i.e., water, water boiler, fuel/oil and air/oil, were analyzed. The weights of the water and water/ethyl alcohol heat sinks were found to be prohibitive. A water boiler augmented by a fuel/oil heat exchanger provided a significant weight decrease; however, the complexity and development were determined to be excessive for an RTA.

The fuel/oil heat exchanger is normally the light weight, low volume approach and was analyzed. It was found that a fuel flow rate of approximately

FIGURE 4-37
MODEL 260-RTA-2
COMBINER GEAR BOX HEAT REJECTION



16,000 lb/hr was required to cool the 3100 BTU/min heat load and still provide an acceptable fuel temperature (135°F) to the engine. This fuel flow rate is about twice that available during powered lift operations. Therefore, the fuel/oil heat exchanger approach was dismissed.

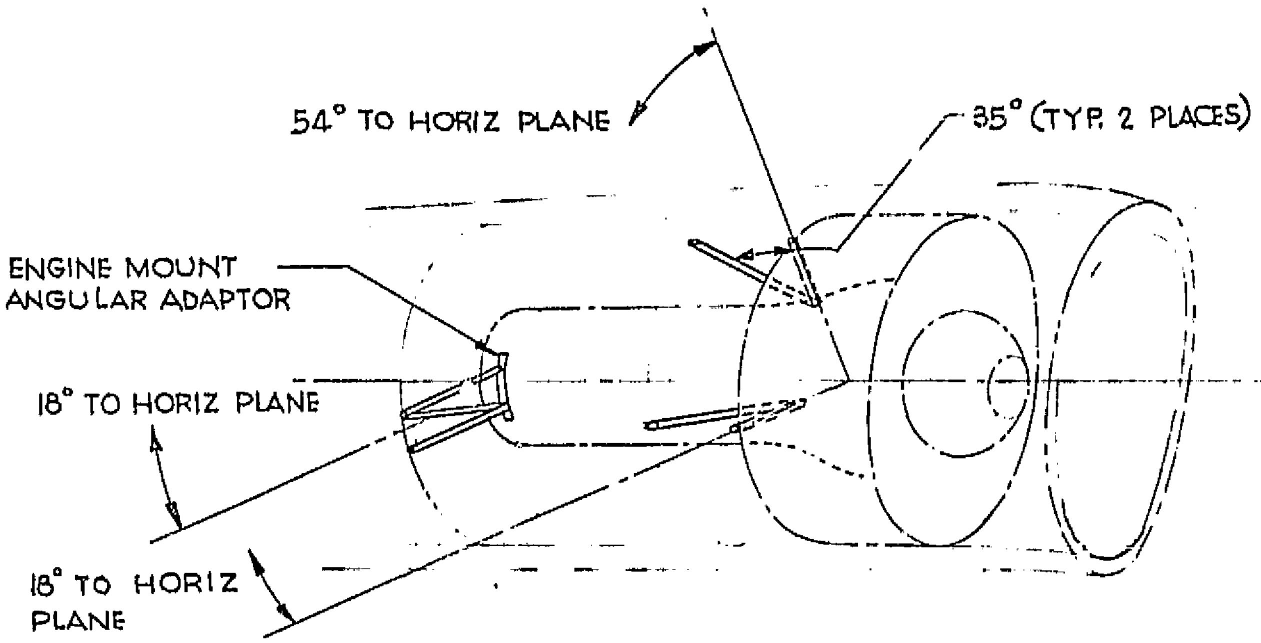
It was found that the simplest and lightest weight system was an air/oil heat exchanger using an electric fan for air flow during low speed operation. The selected heat exchanger is compatible with the 3100 BTU/min cooling requirements. The large thermal mass of the combiner gearbox and lube oil system will restrict the temperature to reasonable levels during the clutch engagement cycle.

The cooling for the lift/cruise engine gearboxes and the lift fan gearbox is provided integral with the units.

4.2.7 FAN/ENGINE MOUNTING AND REMOVAL

Integrated Fan/Engine Mount System - The integrated fan/engine mount system, shown in Figure 4-38, is comprised of two forward truss sets and three rear struts.

FIGURE 4-38
M260-RTA-2
ENGINE MOUNTS



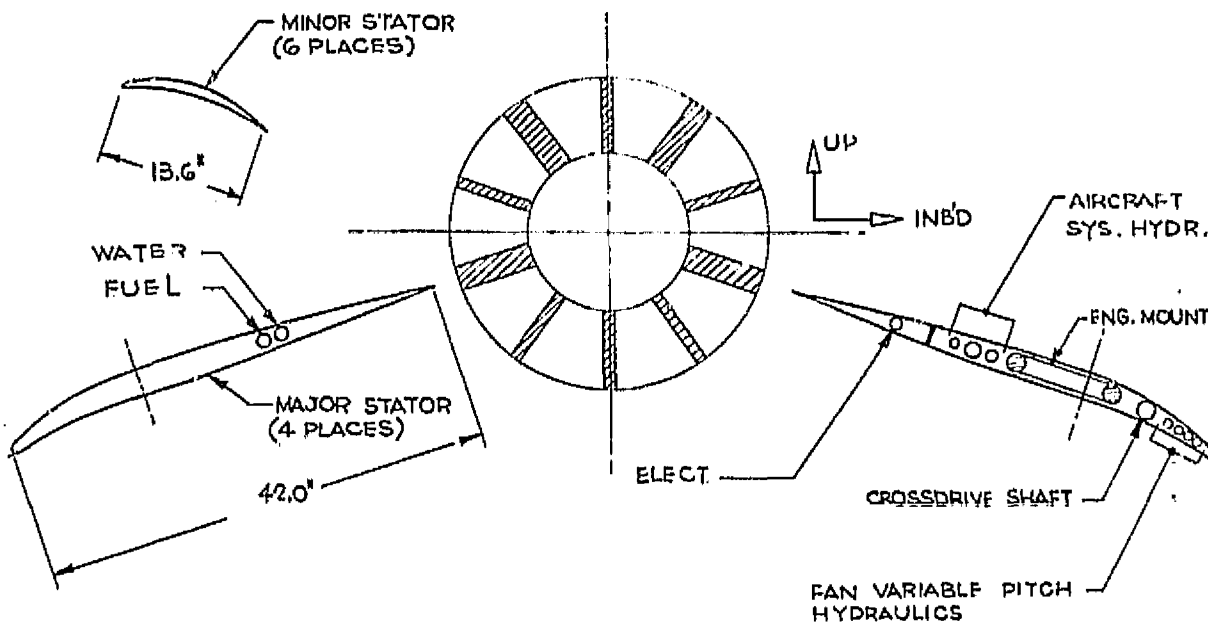
The forward truss sets each consist of two struts inclined at 40° to each other in the forward/aft plane. The upper inboard truss set is inclined at 54° above the horizontal and the lower inboard truss set is inclined at 18° below the horizontal. These angles coincide with the major stator angles. Aerodynamic fairings are provided.

The three rear struts are arranged as two parallel struts inclined at 18° below the horizontal with the third strut as a diagonal brace. These struts are faired so as to provide minimum impact on airflow. An angular adapter ring is provided since the existing aft engine mount pad is inclined at 30° below the horizontal.

The lower forward truss set is designed to carry the forward/aft loads, inboard/outboard loads and thrust loads into the fuselage. The upper forward truss set is designed to carry the vertical and yaw loads and the rear struts carry the vertical loads, inboard/outboard loads and torque.

<u>Fan Stator Arrangement</u> - There are two types of stators used - major and minor. The stator arrangement consists of 10 stators; 4 major stators and 6 minor as shown in Figure 4-39. The major stator chord length is approximately

FIGURE 4-39
M260-RTA-2
STATOR ARRANGEMENT



42.0" with a maximum section depth of 2.6". The major stator space envelope is utilized as follows:

<u>Left Engine</u>

<u>Upper inboard</u> stator contains main fan/engine support struts.

<u>Lower inboard</u> stator contains main fan/engine support structs, cross drive shaft, hydraulic and electrical lines and fan variable pitch hydraulic lines.

<u>Lower outboard</u> stator contains fuel and water lines.

<u>Upper outboard</u> stator is unused.

<u>Right Engine</u>

<u>Upper inboard</u> stator contains main fan/engine support struts.

<u>Lower inboard</u> stator contains fuel and water lines, cross drive shaft and main fan/engine support struts.

<u>Lower outboard</u> stator contains hydraulic and electrical lines and fan variable pitch hydraulic lines.

<u>Upper outboard</u> stator is unused.

The 6 minor struts have a chord length of approximately 13.6" and maximum section depth of approximately 1.25". The minor stators are equispaced angularly at 36° and contain no service lines.

Outboard Engine Removal - Both the engine and fan assembly must be removed simultaneously since they are an integrated assembly. The integrated assembly approach was accepted for the RTA since it was cost effective and resulted in a weight saving of approximately 120 lb/engine assembly. This approach was also consistent with the RTA philosophy of not paying a penalty for maintainability.

The removal sequence is dictated by existing mount features, airframe mount support, accessibility and integrated fan/engine removal. Three removal concepts were studied:

(a) Removal of integrated fan/engine aft via vector doors

(b) Removal of integrated fan/engine outboard and sideways over the wing

(c) Controlled removal of integrated fan/engine up and over the wing

Methods (a) and (b) were eliminated for the following reasons:

Method (a): Fan/engine package must be carefully maneuvered in a very restricted area to clear mount support ring. Rear vector doors, actuation and support structure must be removed. Fan/engine package must be carefully maneuvered aft and down through a very restricted area with resultant high risk of damage.

Method (b): Fan/engine package must be maneuvered forward and sideways to clear engine support ring. The turbine section must also clear the exhaust cowl structure. This would require careful maneuvering with corresponding high risk of damage.

Method (c) was selected and provides for a completely controlled removal sequence with no intricate fan/engine package maneuvering.

The removal sequence shown in Figure 4-40 is as follows:

(a) Remove required portions of cowl, inlet, and stator fairings to gain necessary access.

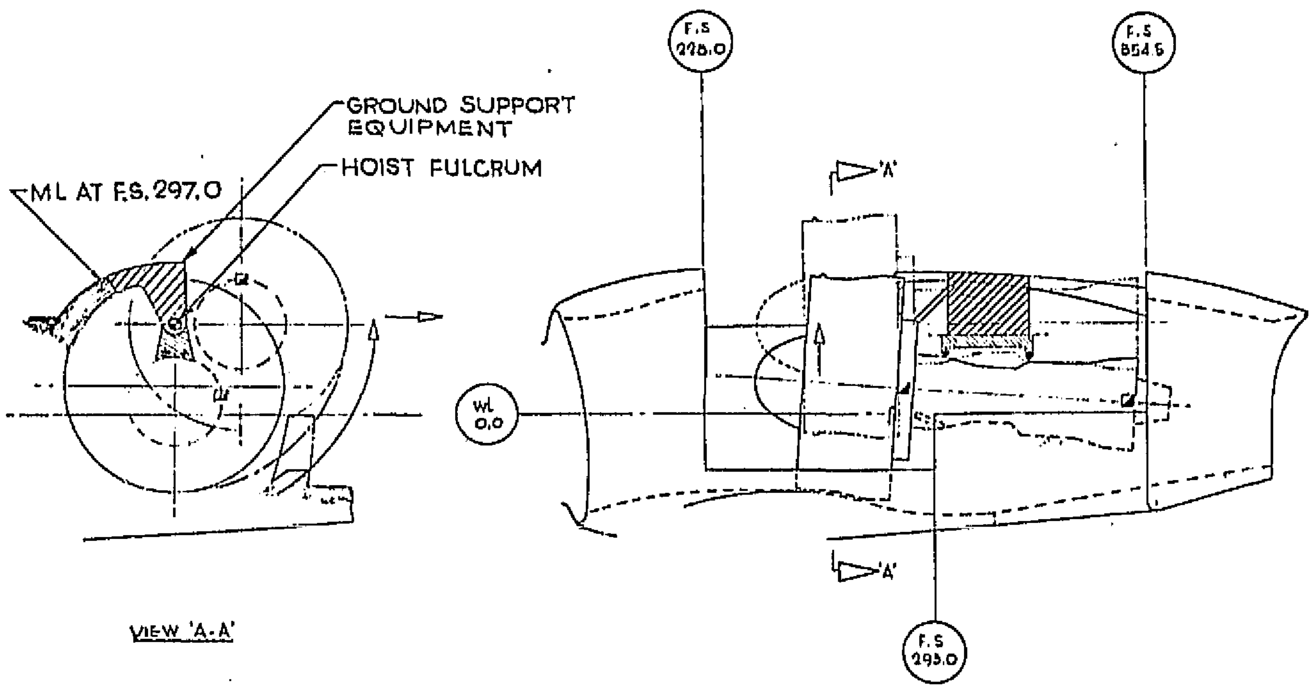
(b) Disconnect service lines and remove engine exhaust tail cone.

(c) Attach ground handling support bracket to existing ground handling lugs. Attach main ground handling support arm to structure and to above support bracket providing a pivot line parallel to engine centerline.

(d) Attach ground hoist lines to existing forward and aft outboard engine mount pads.

(e) Disconnect forward and rear engine support struts.

FIGURE 4-40
M260-RTA-2
OUTBOARD ENGINE REMOVAL



NOTE; TAIL CONE REMOVED FOR ENGINE REMOVAL.
INTEGRATED FAN/ENGINE REMOVAL SAVES 240 LBS.
⊗ HOIST FULCRUM.
◪ HOIST POINTS USING EXISTING.
UNUSED MOUNT POINTS.
⊕ EXISTING GROUND HANDLING POINTS.

(f) Swing engine up and outboard using pivot line provided, until the hoist points are vertical and support the fan/engine package.

(g) Disengage pivot pins and remove fan/engine package outboard and over the wing.

Center Engine Removal - The removal sequence is dictated by existing engine mounting features, mounting support to airframe and accessibility. Access to the engine is gained via "bomb bay" type access doors in the lower fuselage.

The removal sequence would be as follows and is shown in Figure 4-41.

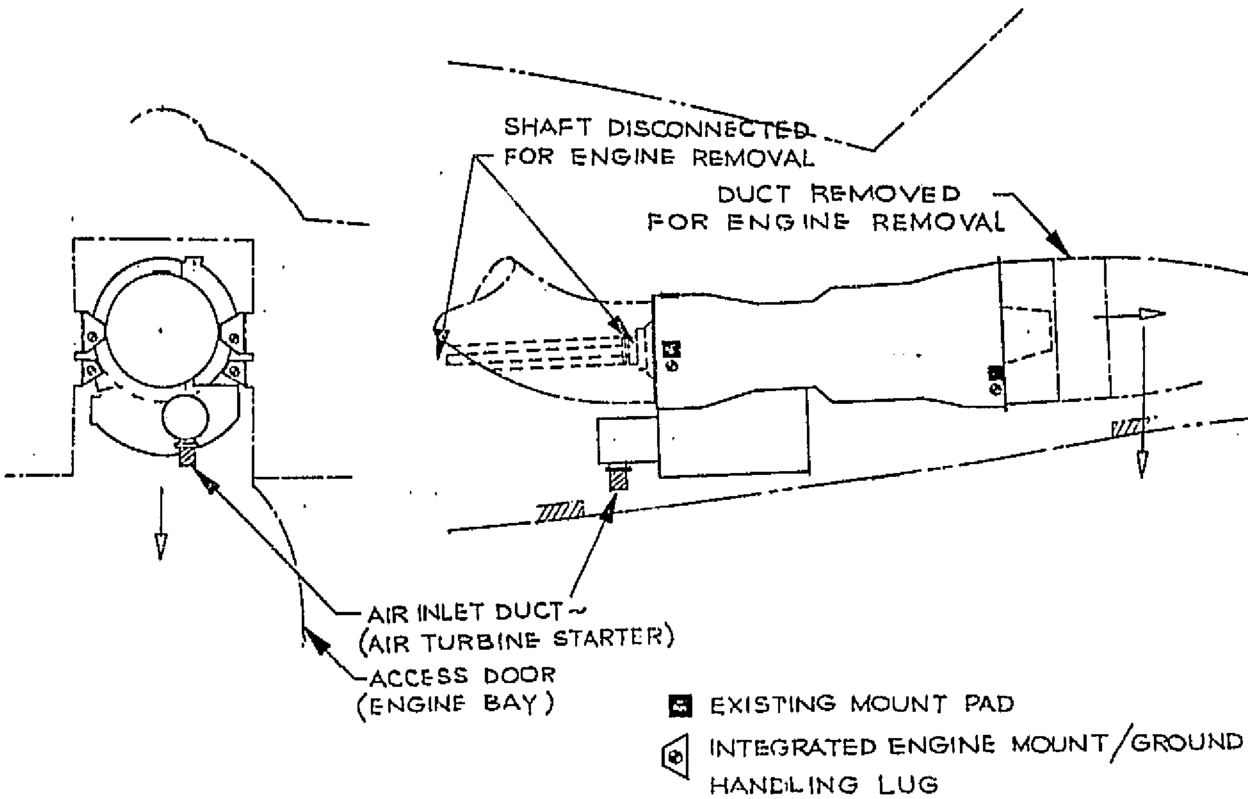
(a) Disconnect drive shaft at aft face of combiner gearbox.

(b) Remove section of duct aft of engine exhaust tail cone.

(c) Disconnect service lines to engine.

(d) Attach ground support cart to engine ground handling points provided on engine mount brackets.

FIGURE 4-41
M260-RTA-2
CENTER ENGINE REMOVAL



(e) Disconnect engine mount brackets from airframe and translate engine aft sufficient to gain access to shaft coupling.

(f) Disconnect shaft coupling at engine and lower engine from airframe.

Forward Fan Removal - A controlled removal technique is used to remove the forward fan assembly. The existing fan assembly has a bolt hole mounting flange. Since the 60 holes are not necessary for mounting purposes a three position equispaced mount feature was used, Figure 4-42. Two of the mounts would pick up a series of existing holes in the fan assembly and then be secured to the airframe. The third mount would provide a pivot point and ground handling lug.

The removal sequence would be as follows:

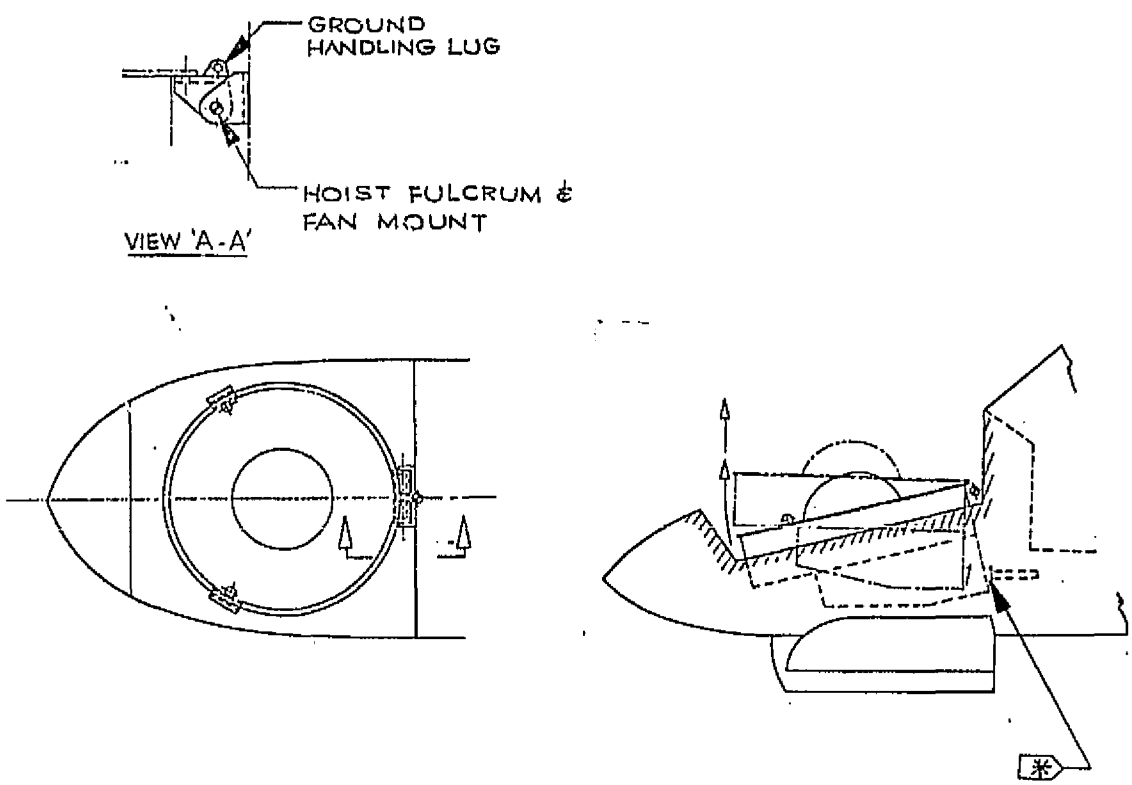
(a) Remove cowl and associated structures to gain necessary access and removal clearances.

(b) Disconnect drive shaft at fan assembly coupling.

(c) Attach ground handling aids adjacent to two forward mount points.

(d) Attach hoist to these two aids and disconnect fan assembly from two forward mounts.

FIGURE 4-42
M260-RTA-2
FORWARD FAN REMOVAL



HOIST POINT ¢ FAN MOUNT
HOIST FULCRUM ¢ FAN MOUNT
DISCONNECT DRIVE SHAFT
FOR ENGINE REMOVAL

(e) Hoist fan assembly to balanced horizontal position.

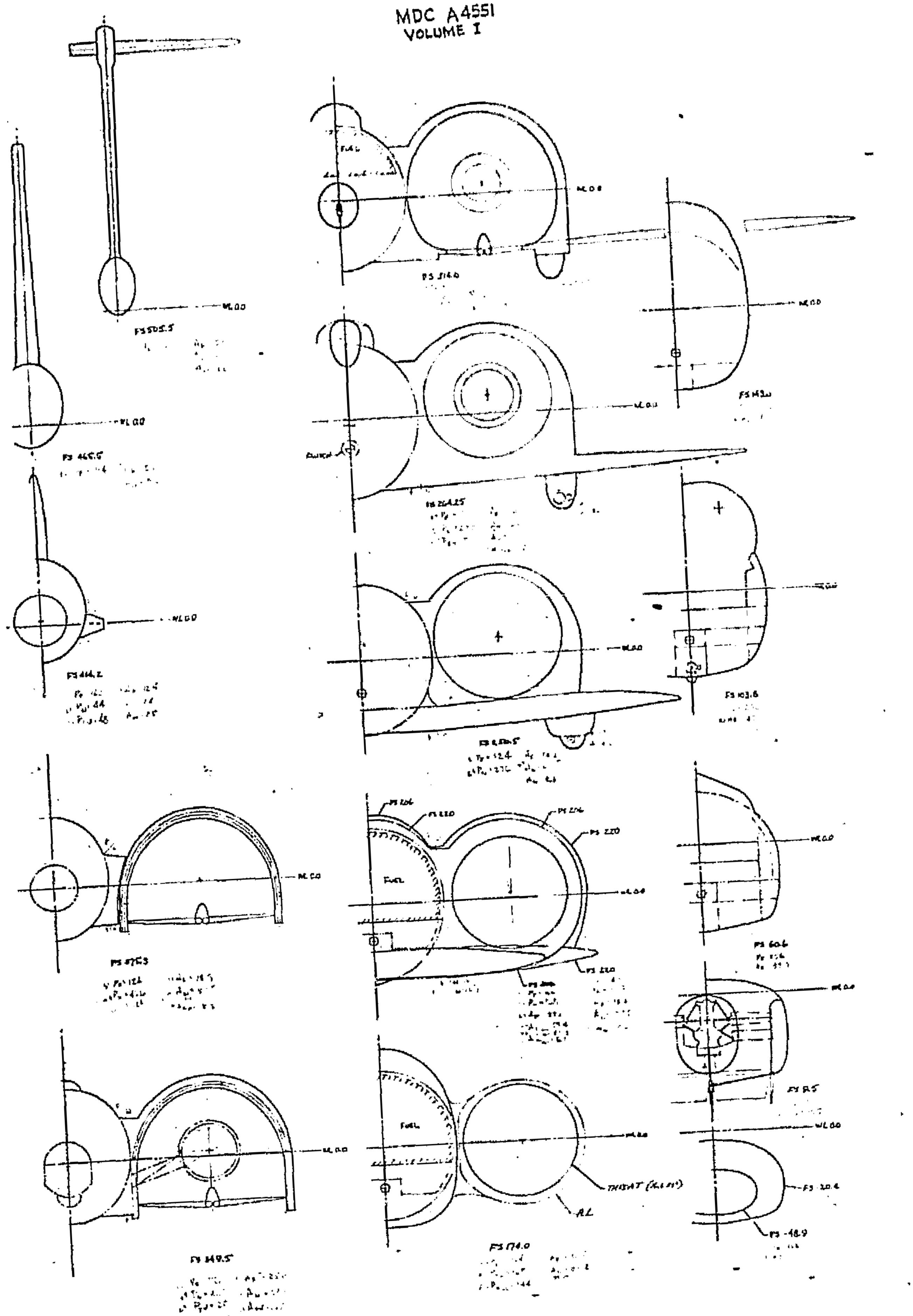
(f) Attach further hoist to ground handling lug provided on rear pivot/mount point.

(g) Disengage pivot pins and hoist fan vertically out of airframe.

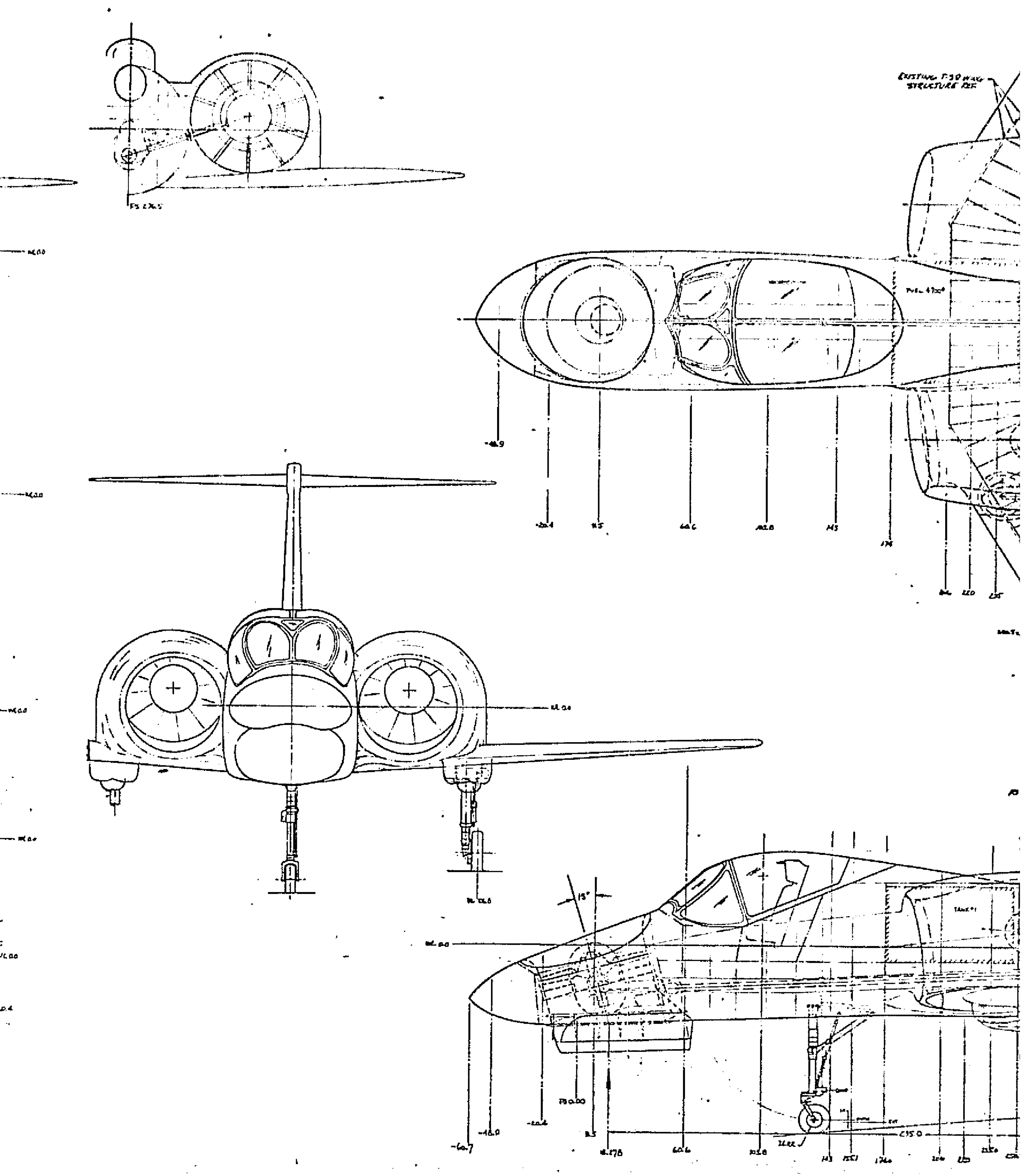
4.3 WEIGHTS

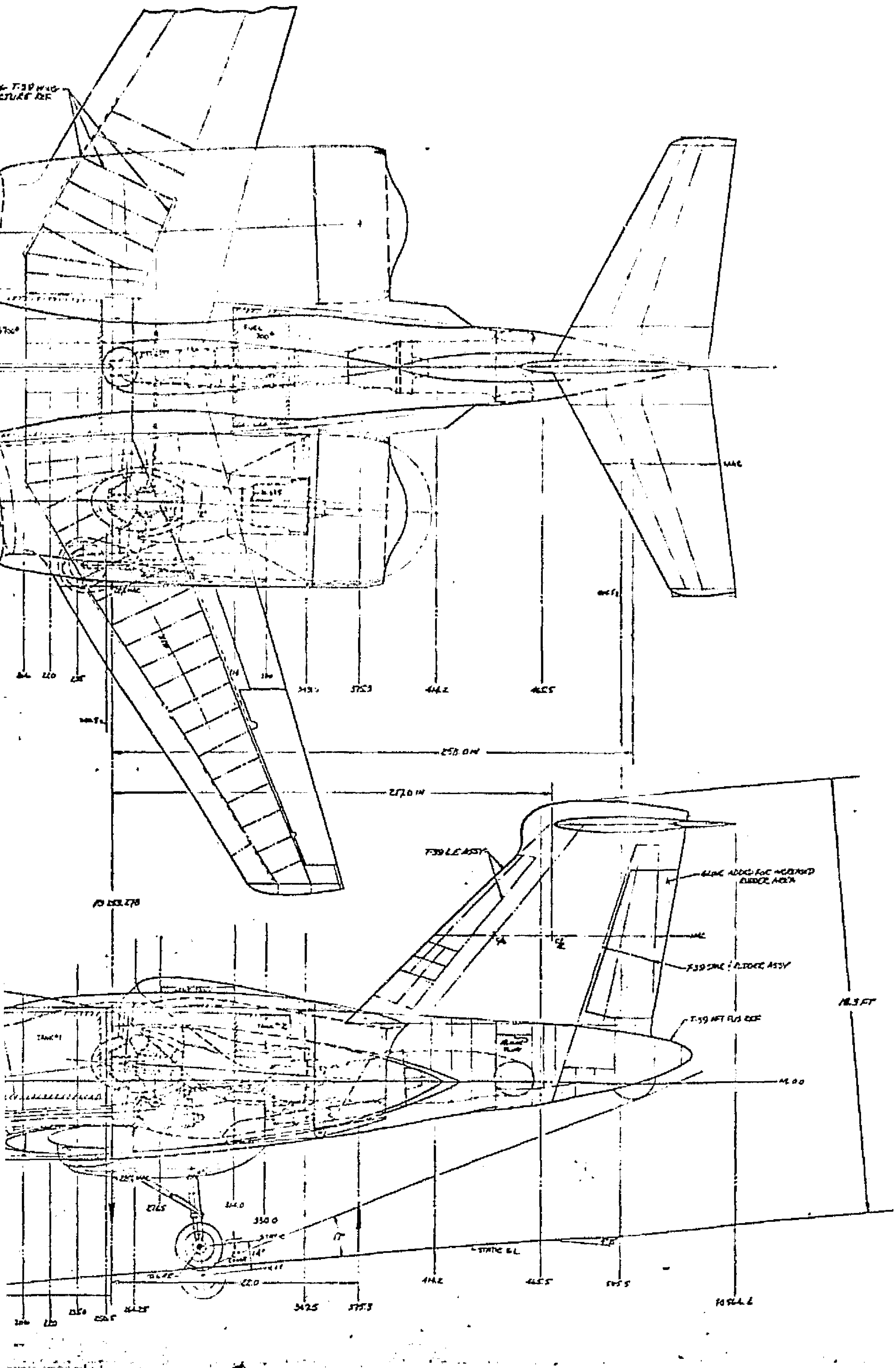
The weight, balance, and inertia summaries for the turbotip and mechanical configurations are presented in Figures 4-43, 4-44, and 4-45 respectively. Both aircraft are capable of a limit load factor of 2.5 g at a 25,000 lb BFDGW.

The two configurations are derivatives of the T-39 Sabreliner with the following exceptions. Model 260-RTA-1 (turbotip) utilizes the aft fuselage and empennage of the F-101 with a pneumatically interconnected propulsion



FOLDOUT FRAME /





PHYSICAL
SPAN
HEIGHT
LENGTH

	7 W
S FT²	3
AR	
λ	..
b (FT)	4
b/2 (IN)	26
CR (IN)	15
CT (IN)	4
MAC (IN)	1
$\bar{y}$c (IN)	1
Λ LE	5
Λ c/4	2
T/CR	1
T/CT	
DIHEDRAL	
AIRFOIL (ROOT)	
AIRFOIL (TIP)	

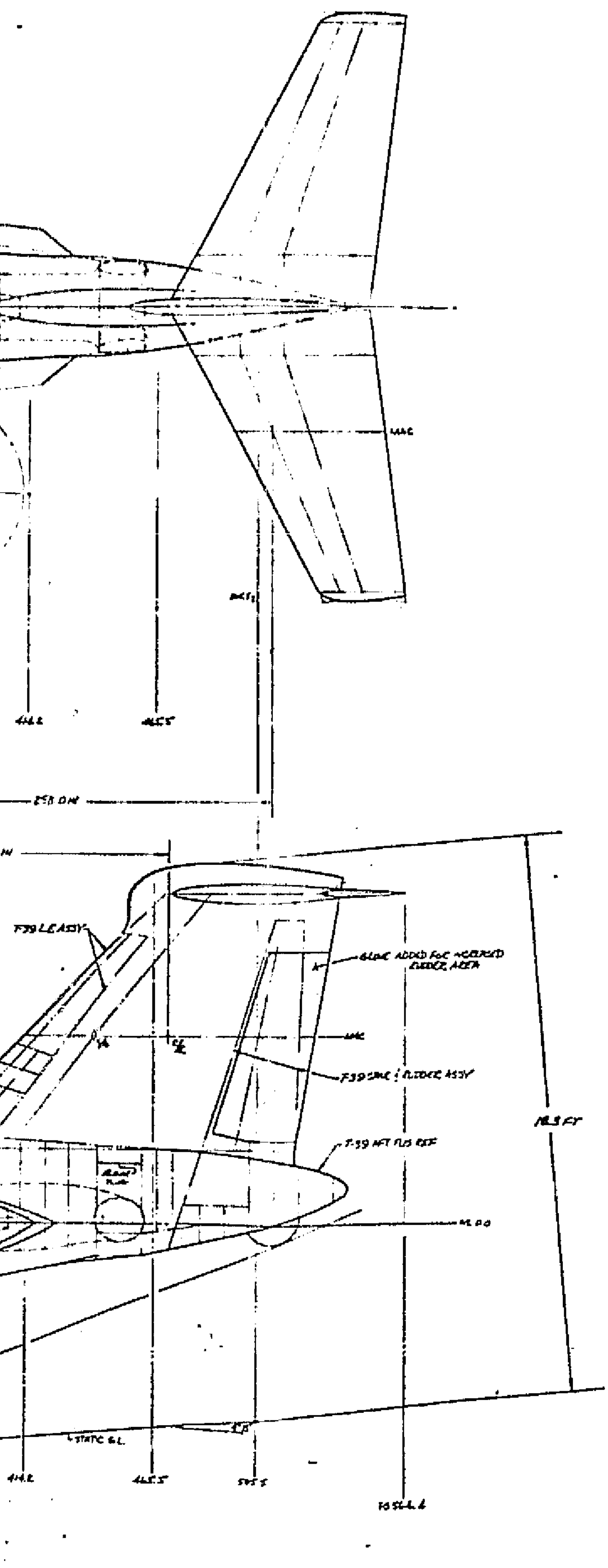
PROPULSION
(2) - P
(1) - T
(1) -
4c FAN

FUEL AVAIL

WETTED AR

CROSS SECTI

3) T-3
2) AL
1) ALL
NOTE:



PHYSICAL CHARACTERISTICS

SPAN	44.43 FT
HEIGHT	16.6 FT
LENGTH	52.09 FT

	T-39 WING	MODIFIED T-39 V-TAIL	H-TAIL
S FT^2	342	77	90.97
AR	5.77	.914	4.095
λ	.323	.454	.400
b (FT)	44.43	8.417	19.3
b/2 (IN)	266.56	—	115.8
C_R (IN)	159.86	152.0	80.8
C_T (IN)	44.91	69.0	32.32
MAC (IN)	100.6	115.7	60.02
$\bar{y}$ (IN)	110.6	44.18	49.65
Λ_{LE}	32.55°	45°	28.2°
$\Lambda_{c/4}$	28.56°	38.3°	23.0°
T/C_R	11.5%	10%	10%
T/C_T	9.36%	10%	8%
DIHEDRAL	3.14°	—	—
AIRFOIL (ROOT)	64A SERIES (NAA MOD)	NACA 63A010	NACA 64A010
AIRFOIL (TIP)		"	" 64A008

PROPULSION

(2) - PD 370-25A COMPOUND L/C TURBOFAN/TURBOSHAFT ENGINES, ALLISON
(1) - T701-16 ENGINE, ALLISON
(1) - HAMILTON STANDARD 62 IN DIA V/STOL LIFT FAN

L/C FAN INLETS
A_{HL} = 21.9 FT^2 EA
A_{TH} = 17.5 FT^2 EA
CENTER ENGINE INLET
A_{HL} = 340 IN^2
A_{TH} = 272 IN^2

FUEL AVAILABLE
FUSELAGE - 5400 LB (BLADDER TANKS)
TANK #1 - 4700#
TANK #2 - 700#

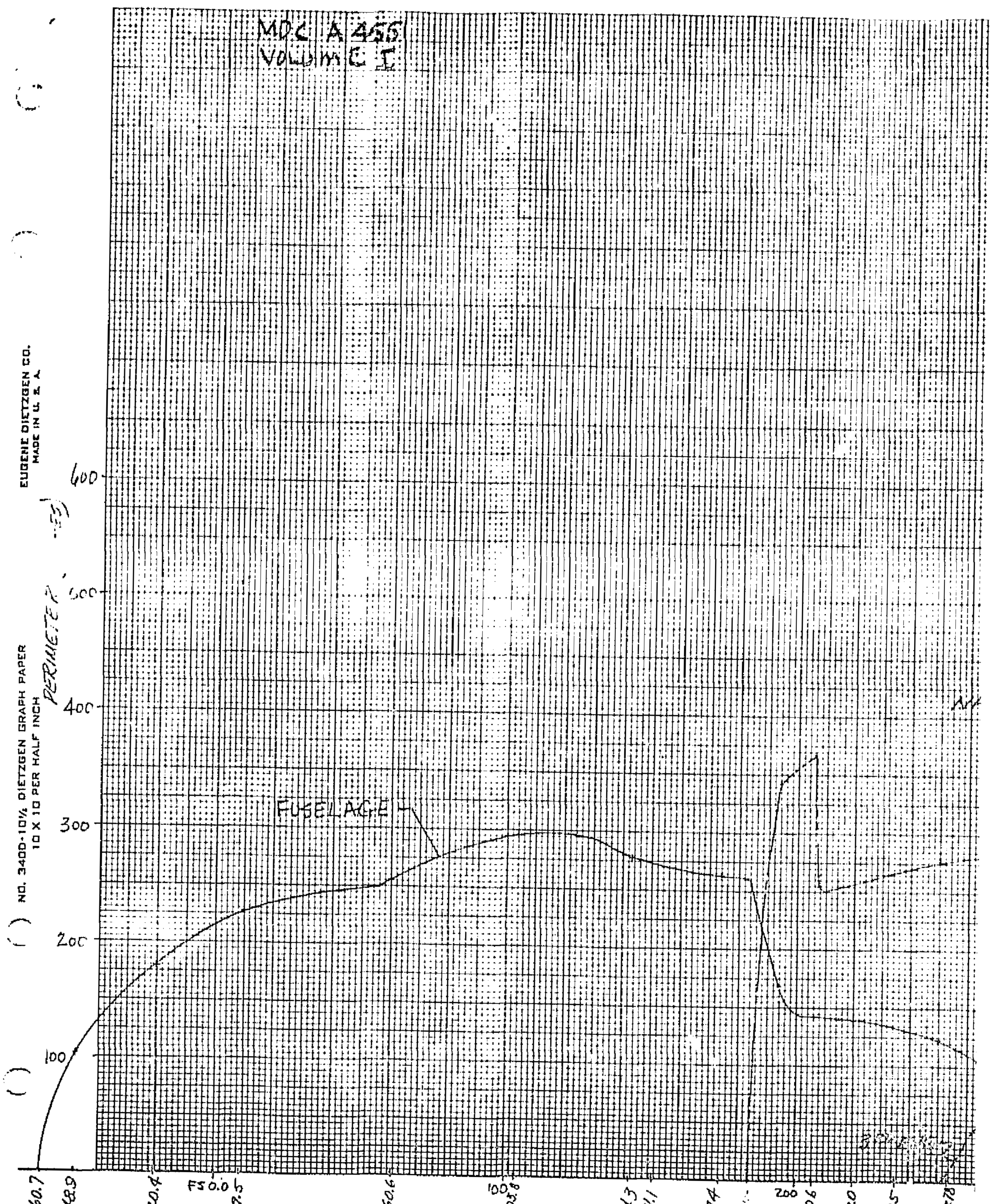
WETTED AREA
SEE SHEET #2

CROSS SECTIONAL AREA
SEE SHEET #3

3) T-39 WING & FUS DATA BASED ON T-39 ARRANGEMENT DWG.
2) A-4 LANDING GEAR INSTALL BASED ON LAYOUT RTA-260-005
1) ALL FS., WL & BL BASED ON T-39 AIRCRAFT COORDINATES.
NOTES:

DRN ZOYCE	MCDONNELL AIRCRAFT COMPANY / MCDONNELL DOUGLAS CORPORATION
	RESEARCH TECHNOLOGY AIRCRAFT - MECHANICAL SYSTEM (B)
SIZE J	CODE IDENT NO. 76301	M-260-RTA-2
SCALE 1/20	WT	SHEET

ORIGINAL PAGE IS
OF POOR QUALITY



FOLDOUT FRAME

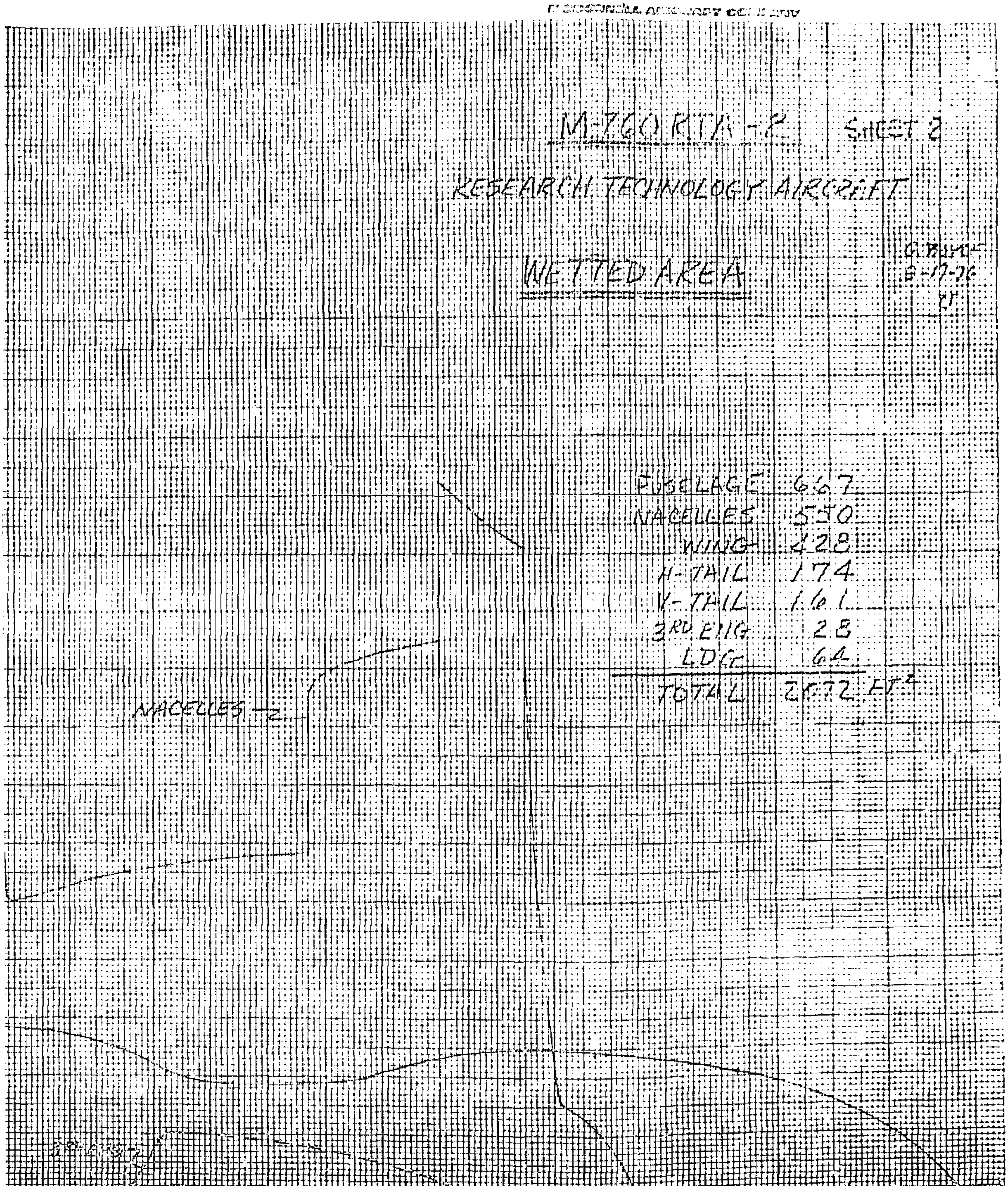
M-760 RTA-2 SHEET 2

RESEARCH TECHNOLOGY AIRCRAFT

WETTED AREA

8-17-76

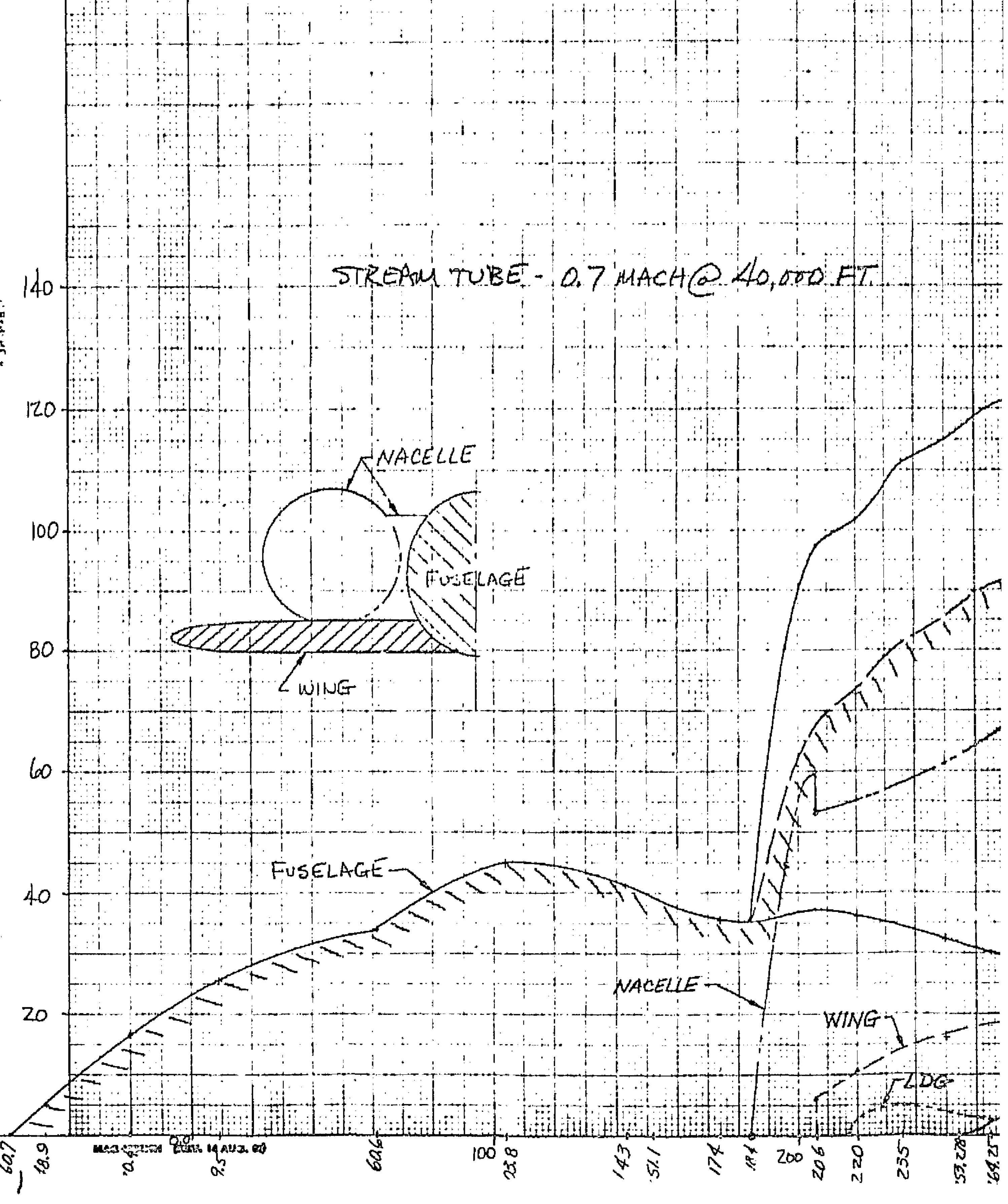
FUSELAGE	667
NACELLES	550
WING	428
H-TAIL	174
V-TAIL	161
3RD ENG	28
LDG	64
TOTAL	2072 FT²



DATE

MODEL

MDC A 4551
VOLUME I



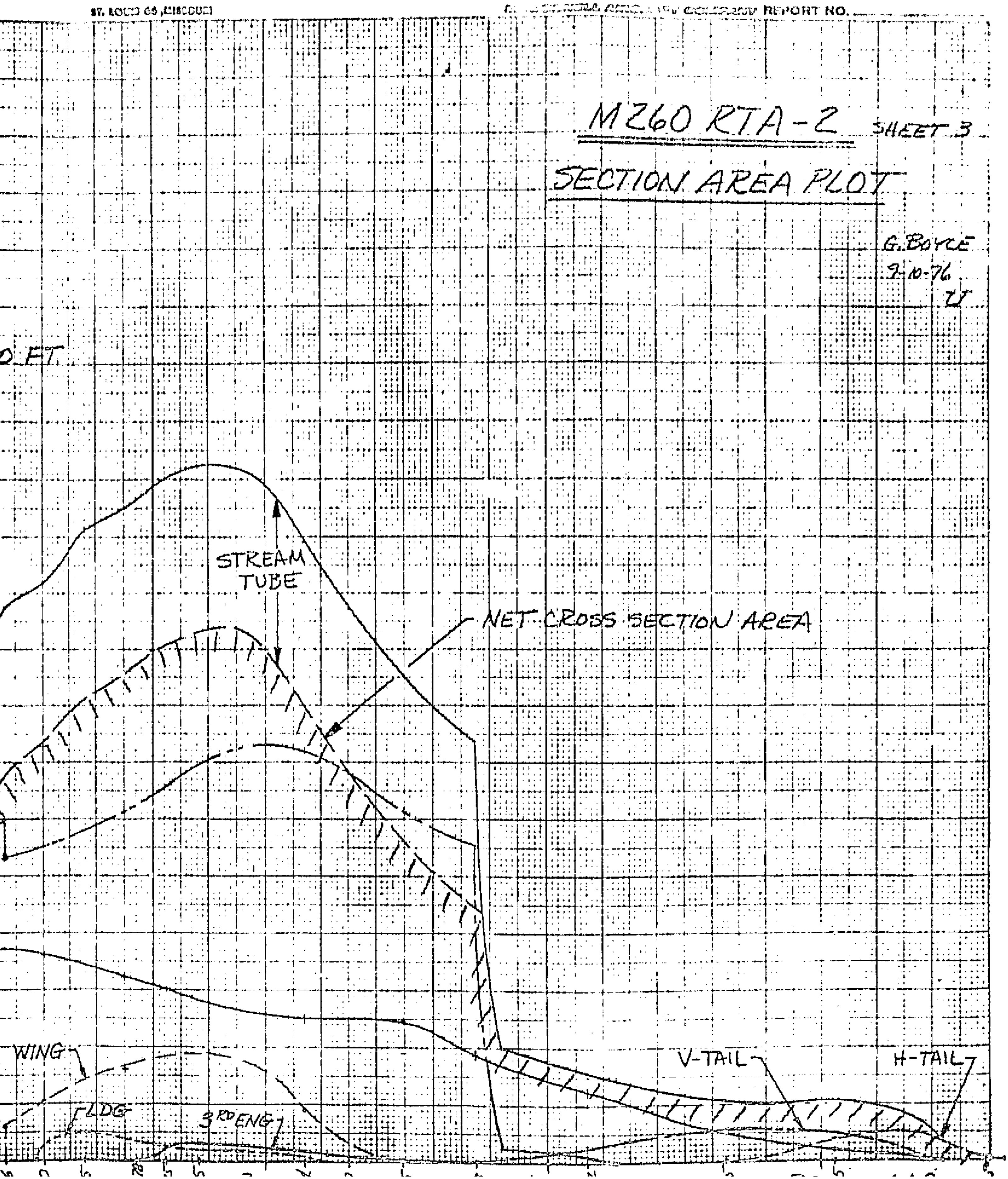

MZ60 RTA-2
SHEET 3
SECTION AREA PLOT
G. BOYCE
9-10-76
O FT.
STREAM TUBE
NET CROSS SECTION AREA
WING
LDG
3RD ENG
V-TAIL
H-TAIL

FIGURE 4-43
RTA AIRCRAFT GROUP WEIGHT STATEMENT

ITEM	Model 260-RTA-1 (TURBOTIP)	Model 260-RTA-2 (MECHANICAL)
WING	1900	1900
VERTICAL TAIL	471	210
HORIZONTAL TAIL	416	270
FUSELAGE	3526	3476
NOSE LANDING GEAR	220	220
MAIN LANDING GEAR	670	670
ENGINE SECTION	180	404
PROPULSION		
GAS GENERATORS	2217	3337
WATER INJECTION SYS.		125
FUEL SYSTEM	407	407
CONTROLS	60	60
LIFT FAN/& L/C FANS(-2)	855	2267
LIFT FAN LOUVERS	200	160
LIFT/CRUISE FANS	1770	
L/C FAN DEFLECTORS	1380	1300
FAN GEARBOXES		1300
TRANSMISSION - DUCTS & VALVES	1897	
GEARBOX & SHAFT		546
START	100	100
FLIGHT CONTROLS	835	835
INSTRUMENTS	257	257
HYDRAULICS	175	175
ELECTRICAL	325	325
ELECTRONICS	230	230
ARMAMENT		
FURNISHINGS	568	568
AIR CONDITIONING	150	150
ANTI-ICE		
AUXILIARY GEAR	7	7
WEIGHT EMPTY	18736	19289
CREW	360	360
TRAPPED FUEL	80	80
OIL	135	135
Oz & MISC.	140	140
H_2O		180
OPERATING WEIGHT EMPTY	19451	20260
PAYLOAD	2500	2500
FUEL	(MISSION DEPENDENT)	
TAKE-OFF GROSS WEIGHT		

FIGURE 4-44
BALANCE AND INERTIA DATA

	M260-RTA-1 (Turbotip)	M260-RTA-2 (Mechanical)
STOGW - lb	26320	26531
Ix - (slug - ft^2)	19400	23000
Iy - (slug - ft^2)	52400	54000
Iz - (slug - ft^2)	67500	68500
F.S.	253.9	252.5
W.L.	2.0	-1.5
OWE + Payload - lb	21951	22760
Ix - (slug - ft^2)	14500	17250
Iy - (slug - ft^2)	50900	53000
Iz - (slug - ft^2)	62000	63000
F.S.	254.0	256.5
W.L.	2.0	-3.0

FIGURE 4-45
RTA SUMMARY WEIGHT DATA

	M260-RTA-1 (Turbotip)	M260-RTA-2 (Mechanical)
Structure	7383	7150
Propulsion	8886	9602
Subsystems	2467	2537
Weight Empty	18736	19289
Non-expendable V.L.	715	971
O.W.E.	19451	20260

system, while the Model 260-RTA-2 (mechanical) incorporates the T-39 aft fuselage with a modified Sabreliner empennage and a mechanically interconnected propulsion system.

The components both aircraft have in common are the modified Sabreliner wing, A-6 cockpit (including canopy and windshield), and A-4 landing gear. New, all metal structure is required to integrate the propulsion system and the above mentioned components into the proposed configurations.

Propulsion weights are based on vendor data with General Electric providing turbotip system information, and Allison/Hamilton Standard providing the mechanical system data.

4.4 DATA BASE SUMMARY

Figure 4-46 shows the various data elements used as a basis for analysis and budgetary cost estimates of the selected program and aircraft approaches. The major airframe components required for each aircraft and their required modifications/integrations are discussed in Section 4.1 for the turbotip RTA, and in Section 4.2 for the mechanical RTA. Aerodynamic, propulsion, control characteristics, and weight are summarized in Sections 2, 3, 4, and 5 for the RTA vehicles. The technical development program, including both ground and flight test programs, is summarized in Volume III. Systems requiring development are identified together with test objectives, instrumentation, and facilities required. A milestone schedule shows the integration of development tests for systems and aircraft and dock dates of major contractor and government furnished items.

4.4.1 MAJOR EQUIPMENT LIST - The major government furnished airframe components, tooling, and test equipment are shown in Figure 4-47, and the major subsystem components are shown in Figure 4-48. Items are identified as GFE and CFE, and are common for both airplane configurations except where noted.

4.4.2 GROUND SUPPORT EQUIPMENT - It is assumed on-site GSE at contractor or government test facilities will suffice for the major support needs of the technology aircraft. This includes support for the various subsystems such as hydraulics, electrical, fuel, landing gear and brakes, and CNI.

Special support equipment, such as required for the Automatic Flight Control Set, will be furnished by the contractor. Certain government furnished special GSE will be required for the propulsion system components, including checkout and handling equipment. The latter includes slings and transport adapters for the gas generators and fans. A preflight console for the instrumentation data system will also be required as GFE.

FIGURE 4-46
DATA BASE-TECHNOLOGY AIRCRAFT PROGRAM

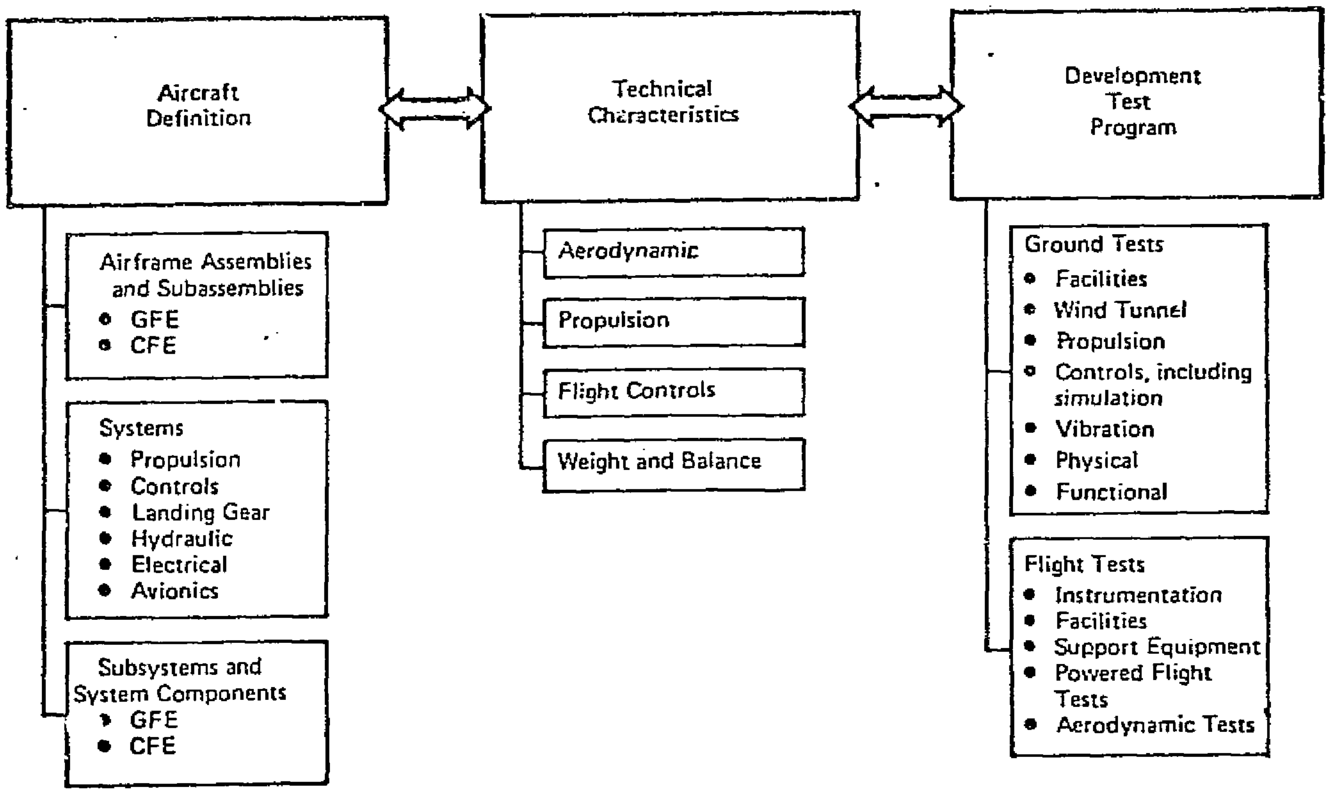


FIGURE 4-47
MAJOR GFE AIRFRAME COMPONENTS, TOOLING,
AND TEST EQUIPMENT

	per A/C
T-39 Center Fuselage	1
T-39 Wing Structural Assembly	1
Master Tooling at T-39 Center Fuselage Splice Bulkheads	1 Set
Full Set of T-39 Wing Tooling	1 Set
(a) F-101 Aft Fuselage and Empennage	1
(b) T-39 Aft Fuselage and Vertical Tail	1
A-6 Cockpit Assembly	1
A-6 Canopy and Actuation System	1
A-6 Oxygen System	1
A-4 Landing Gear	1
Ejection Seat (Zero-Zero)	2
Three Degree of Freedom Test Rig	1

(a) Turbotip Airplane Only

(b) Mechanical Airplane Only

FIGURE 4-48
MAJOR SUBSYSTEM EQUIPMENT

	per A/C	CFE	GFE
PROPULSION SYSTEM			
(a) YJ97-100 Gas Generator	3		X
(a) LF459 Lift Fan	3		X
(a) Air Turbine Starter	3		X
(a) L/C Vectoring Nozzle and Louvers	2	X	
(a) Nose Fan Vectoring Nozzle and Louvers	1	X	
(a) Interconnect Ducts, Bellows	1 Set	X	
(a) Gas Generator Isolation Valves	3	X	
(a) System Isolation Valves	2	X	
(a) Fan Isolation Valve	3	X	
(a) FTaC Modulation Valves	3	X	
(a) Diverter Valve	1	X	
(a) Lift/Cruise Fan Gearbox	2		X
(b) PD370-274 Lift/Cruise Fan Assembly (includes engine, fan, gearbox, and clutch)	2		X
(b) XT701 Turboshaft Engine	1		X
(b) Lift Fan	1		X
(b) Combiner Gearbox	1	X	
(b) Lift Fan Clutch	1	X	
(b) Interconnecting Drive Shaft	1 Set	X	
(b) Drive Shaft Support Bearings	1 Set	X	
FUEL SYSTEM			
Boost Pump	2		X
Gaging System (includes probes)	1		X
Miscellaneous Shutoff and Vent Valves, Regulators	1 Set		X
Fuel Tanks (Fuselage)	1 Set		X

(a) Turbotip Airplane Only

(b) Mechanical Airplane Only

FIGURE 4-48 (CONTINUED)
MAJOR SUBSYSTEM/EQUIPMENT

	per A/C	CFE	GFE
FLIGHT CONTROLS			
Automatic Flight Control Set	1	X	
Flight Control Actuators	1 Set	X	
Signal Conversion Mechanisms	1 Set	X	
Pilot Cockpit Controls	1 Set	X	
Mechanical Transition Control System	1	X	
(a) Mechanical Throttle Control System	1	X	
UNDERCARRIAGE			
A-4 Main Landing Gear and Mechanism	2		X
A-4 Main Wheel, Brake, Tire	2		X
A-4 Nose Landing Gear and Mechanism	1		X
A-4 Nose Wheel, Tire	1		X
A-4 Brake Control Valve	2		X
HYDRAULIC SYSTEMS			
Variable Displacement Pressure Compensated Pump	2		X
Reservoir (Bootstrap with RLS)	2		X
Shutoff Valve in the RPS	1		X
Miscellaneous Components for 2 Power Control Hydraulic Systems (3000 psi)	2	X	
ELECTRICAL SYSTEMS			
AC Generator, 15 KVA	2	X	
Constant Speed Drive 15 KW	2	X	
Transformer-Rectifier, 100 Amp	2		X
Battery, 11 Amp Hour	1		X
Inverter, 250 VA	1		X
Generator Control Unit (compatible with generator listed above)	2	X	

(a) Turbotip Airplane Only

FIGURE 4-48 (CONTINUED)
MAJOR SUBSYSTEM/EQUIPMENT

	per A/C	CFE	GFE
ENVIRONMENTAL CONTROL SYSTEM			
Electric Driven Fan (Cockpit and Equipment Cooling)	4	X	
(b) Air/Oil Heat Exchanger (Combiner Gearbox)	1	X	
(b) Electric Driven Cooling Fan (Combiner Gearbox)	1	X	
Miscellaneous Valves and Ducts	1 Set	X	
AVIONICS			
Communication, Radio Nav and Identification			
UHF AM Transceiver	1		X
Intercomm	1		X
IFF Transponder	1		X
TACAN	1		X
Antennas			
UHF/L Band	1	X	
Transponder	1	X	
Navigation			
Attitude and Heading Reference	2		X
Magnetic Azimuth Detector	2		X
Radar Altimeter	1		X
Air Data System			
Air Data Computer	1	X	
Pitot Static Probe	2		X
Alpha/Beta Sensors	2		X
Total Temp Sensor	2		X
Low Velocity A/S System	2	X	
Displays			
Attitude Director	1		X
Horizontal Situation Indicator	1		X

(b) Mechanical Airplane Only

FIGURE 4-48 (CONTINUED)
MAJOR SUBSYSTEM/EQUIPMENT

	per A/C	CFE	GFE
AVIONICS (Continued)			
Displays (Continued)			
Altimeter	1		X
Standby Attitude Indicator	1		X
MISCELLANEOUS			
Fire Detection and Extinguishing System	1 Set	X	
Miscellaneous Flight Instruments	1 Set		X
100 Channel PCM Tape and Telemetry	1 System		X
Measureands	80	X	

5. AIRCRAFT CONTROL

Aircraft control of the Research Technology Aircraft (RTA) and the specific control techniques are described in this section. The control capabilities have been determined and are compared to the minimum recommended control performance by the NASA study guidelines.

5.1 BA. C CONTROL CONCEPTS

Control of the RTA aircraft throughout the aerodynamic and powered lift flight envelope is provided by conventional aerodynamic control surfaces in combination with a three fan/three engine propulsion system. Control in the powered lift portion of the flight envelope is provided by an airspeed dependent blend of the aerodynamic control surfaces with vectored thrust of two lift/cruise fans and a forward fuselage mounted lift fan. During transition from hover to conventional aerodynamic flight, the powered lift controls are phased out as the effectiveness of the aerodynamic surfaces increases with airspeed. After conversion to conventional aerodynamic flight, the lift fan is shut down and the powered lift controls are disabled. General descriptions of the aerodynamic controls and the powered lift controls are given below.

5.1.1 AERODYNAMIC CONTROLS - Stabilator, aileron, and rudder control surfaces provide aircraft pitch, roll, and yaw control within the conventional aerodynamic flight envelope, and airspeed dependent partial control within the powered lift flight envelope. All control surfaces are driven by irreversible, hydraulically powered actuators which are operational throughout the entire composite flight envelope. Thrust is generated in conventional aerodynamic flight by the two lift/cruise fans powered by two gas generators.

5.1.2 POWERED LIFT CONTROLS - Although aircraft control in portions of the powered lift flight envelope is provided in part by the aerodynamic control surfaces, the powered lift controls as discussed here refer to the means by which the thrust of the two lift/cruise fans and the forward fuselage mounted lift fan is modulated and vectored for STOL and VTOL operation.

Powered lift aircraft control for VTOL operation is provided by vertically vectored lift/cruise fan thrust and lift fan thrust. Attitude control is accomplished by modulating and/or vectoring the thrust of the three fans so as to produce the required roll, pitch, and yaw control moments. Height control in VTOL operation is functionally identical to total lift control and is accomplished by modulation of gas generator power.

Thrust modulation is utilized to produce roll and pitch attitude control moments and is accomplished by controlling the transfer of energy between the three fans. A roll control moment is produced by increasing the thrust of one lift/cruise fan and decreasing the other while maintaining constant lift fan thrust. A nose up pitch control moment is simularly produced by increasing the lift fan thrust while decreasing the thrust of each lift/cruise fan. A nose down pitch control moment is produced by reversing the sense of the thrust modulation at each fan. Constant total lift is maintained by coordinating the thrust modulation of the three fans so as to produce no change in composite vertical thrust.

Yaw control is produced by differential lateral vectoring of the lift fan thrust and lift/cruise fan thrust. Deflecting the lift fan thrust to the right or left while simultaneously deflecting the thrust of both lift/cruise fans in the opposite direction respectively produces a left or right yaw control moment. Because the effective thrust deflection angles required to produce necessary yaw control capability are small, the coupled effect on total lift is negligible. Thrust deflection is produced by lift fan yaw vectoring louvers and lift/cruise fan yaw control vanes.

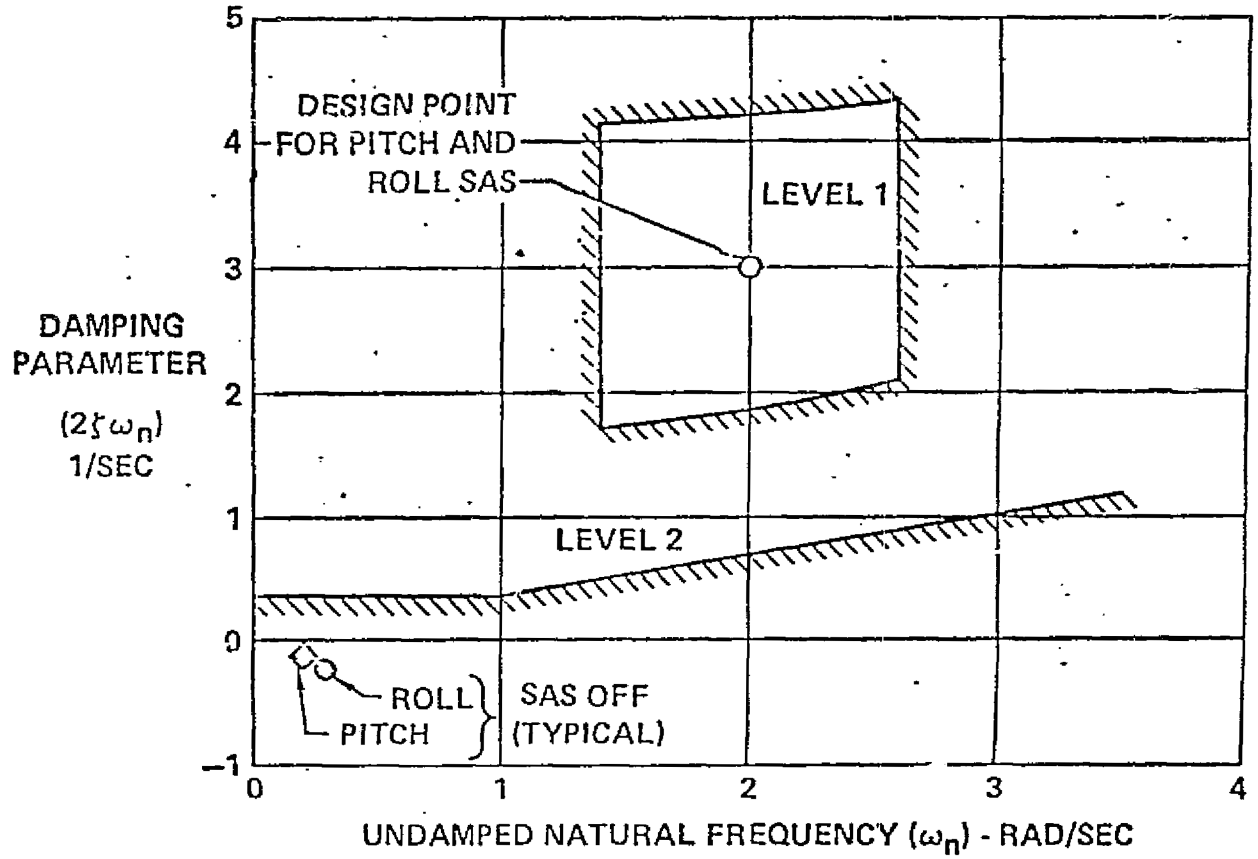
Thrust vectoring louvers underneath the lift fan and vectoring nozzles on the lift cruise fans provide thrust vector control in transition. The lift fan and lift/cruise fan thrust vectors are scheduled to maintain zero pitching moment due to direct thrust forces. The current design vector rate is 50 degrees per second. The higher vectoring rate is, of course, desirable to enhance the research capabilities of the RTA.

5.1.3 CONTROL REQUIREMENTS - The control design requirements were established to insure good maneuvering capability and, also, to provide sufficient forces and moments to stabilize the aircraft and to control disturbance and cross-coupling effects. The primary control design guidelines for maneuver control are summarized in Figure 5-1. Design control power is interrelated with aircraft stability requirements in that the characteristics of the stability augmentation system affect the installed control power requirements. The inherent aerodynamic stability of a V/STOL aircraft decreases with reduction of airspeed until at a point approaching hover the aircraft becomes unstable. The inherent dynamic characteristics of the aircraft were evaluated and found to possess low frequency divergent osicllatory pitch and roll modes as shown in Figure 5-2. To achieve the specified hover stability, aircraft attitude

FIGURE 5-1
PRIMARY VTOL CONTROL GUIDELINES
RESEARCH TECHNOLOGY AIRCRAFT

		LEVEL 1	LEVEL 2
ATTITUDE CONTROL			
ACCELERATION (RAD/SEC^2)	ROLL	± 0.90	± 0.40
	PITCH	± 0.50	± 0.30
	YAW	± 0.30	± 0.20
ANGLE IN 1 SEC (DEGREES)	ROLL	± 15	± 7
	PITCH	± 8	± 5
	YAW	± 5	± 3
COMBINED CONTROL		100% + 30% + 30%	
HEIGHT CONTROL			
WITH 50% ATTITUDE CONTROL (g)		±0.1	–0.1, +0.05
TRANSIENT RESPONSE (TIME CONSTANT)			
ATTITUDE CONTROL (SECONDS)		0.2	0.3
HEIGHT CONTROL (SECONDS)		0.3	0.5

FIGURE 5-2
COMPLIANCE WITH STABILITY CRITERIA



and rate feedback loops are provided by a stability and control augmentation system which brings the aircraft pitch and roll natural frequency and damping response characteristics into compliance with the Level 1 guideline stability criteria as indicated by the design point shown in Figure 5-2.

The composite VTOL control power design guidelines are indicated in Figure 5-3. The closed loop attitude control powers are dictated by requirements for hover dynamic stability and attitude change in one second per inch of control stick displacement. These requirements may be translated into specified moment/inertia (M/I) ratios as indicated in Figure 5-3.

5.2 TURBOTIP RTA CONTROL

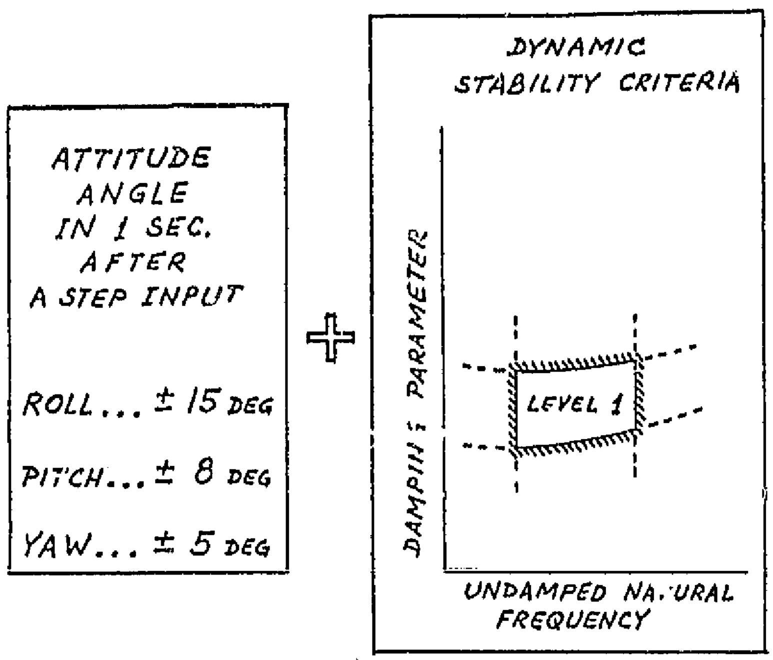
The specific methods employed by the Turbotip RTA to implement transfer of energy between fans for thrust modulation are discussed in this section. This is followed by a comparison of the Turbotip RTA control capabilities with the NASA study guidelines.

5.2.1 THRUST MODULATION - Transfer of energy between the three fans and coordination of thrust modulation so as to maintain constant total lift is provided by the combined action of the Energy Transfer and Control (ETaC) system and the fan Thrust Reduction Modulation (TRM) system. ETaC valves are located in the interconnecting ducts between fans at the inlets to the tip turbines. Partial closure of the ETaC valve at one fan produces an increase in thrust at all other fans without changing the thrust of its associated fan substantially. The TRM system provides complimentary control of the thrust of each individual fan and is coordinated with the ETaC system to negate the increase in total lift produced by this effect, as well as to provide greater control moment thrust differentials and better control response characteristics than are obtainable by the ETaC system operating alone.

ETaC and TRM provide two paths for control force and moment signals as depicted in Figure 5-4, resulting in increased flight safety and survivability in the event of a major failure during VTOL. Loss of an ETaC or TRM function at a fan affects only a portion of the control capability, usually less than 50%, which results in only a minor degradation of handling qualities.

5.2.2 CONTROL DURING NORMAL OPERATION - The Turbotip RTA was analyzed to determine the thrust modulation requirements. Attitude control in hover was determined to be more demanding of thrust modulation than control in transition or STOL. The thrust modulation levels required to satisfy the VTOL design control power requirements were determined and are shown in Figure 5-5. The

FIGURE 5-3
VTOL DESIGN CONTROL POWER



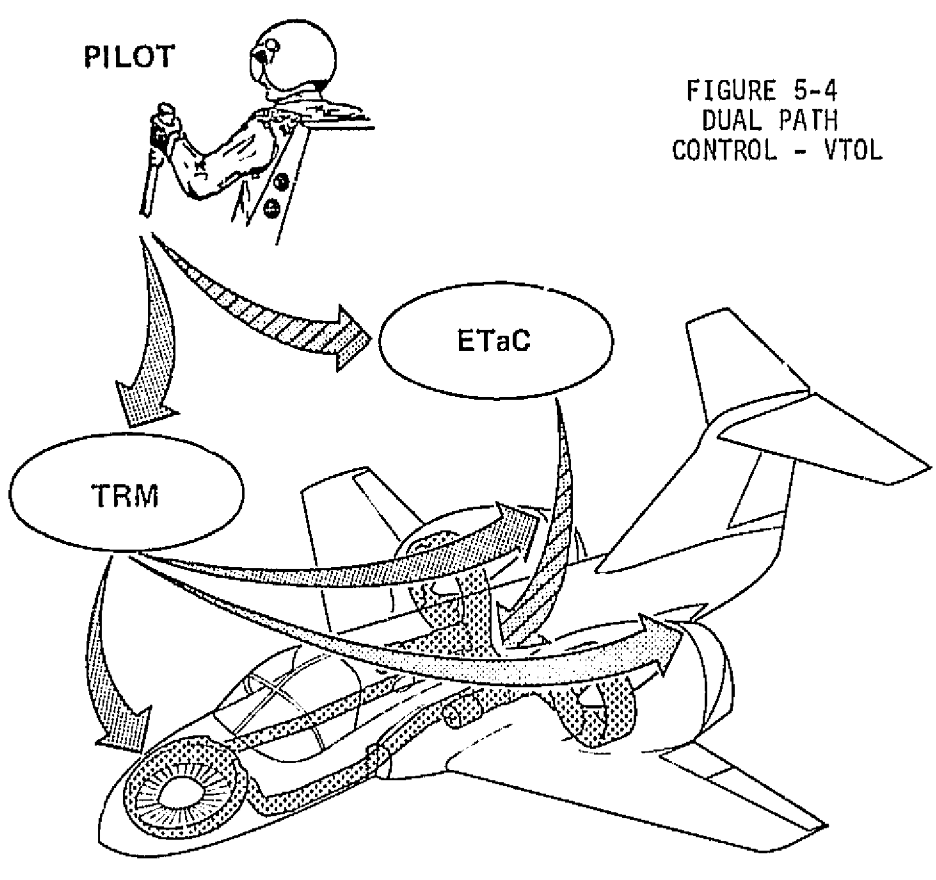
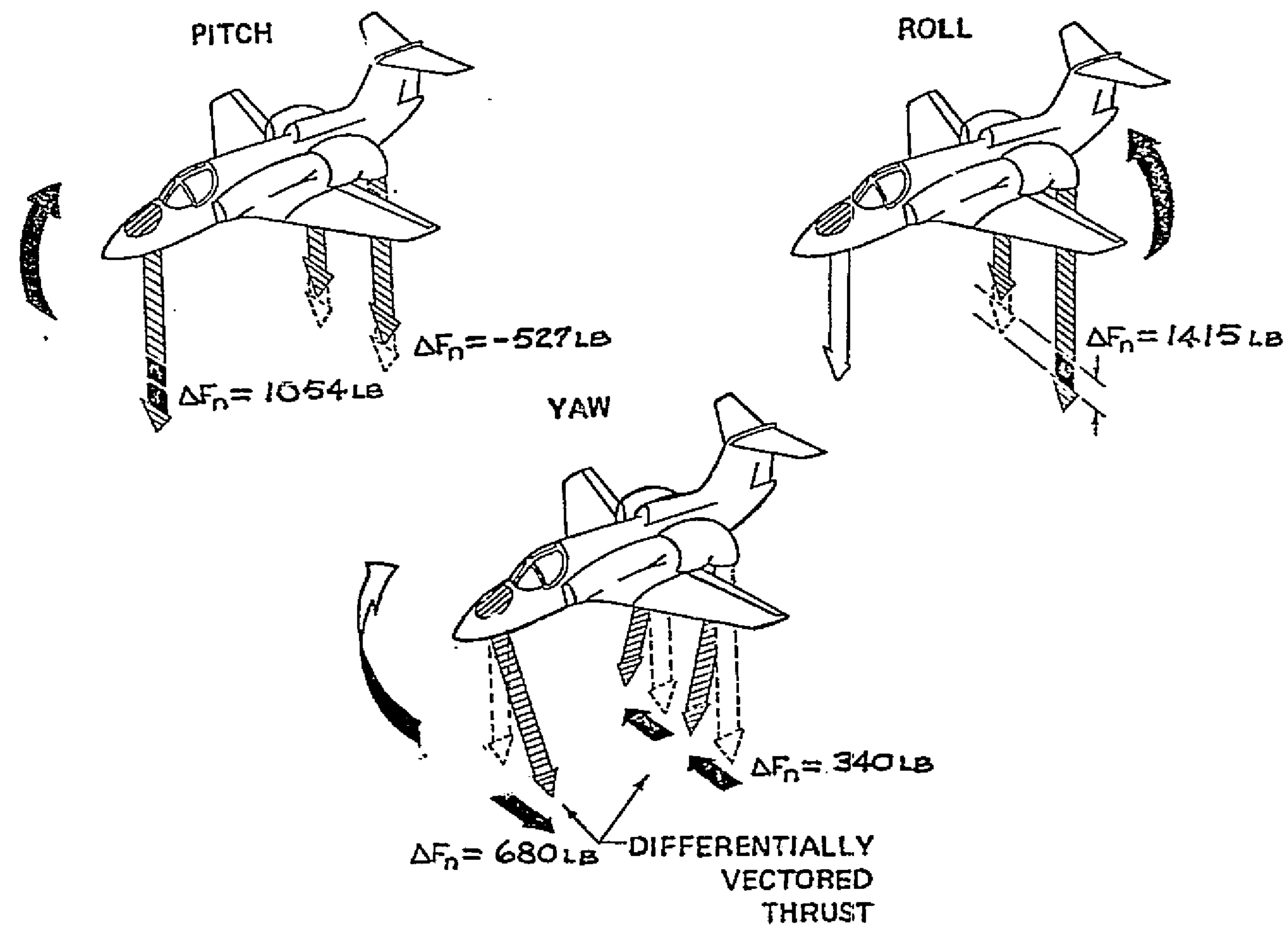


FIGURE 5-4
DUAL PATH
CONTROL - VTOL

FIGURE 5-5
TURBOTIP RTA
THRUST MODULATION REQUIREMENTS



indicated thrust increments (decrements) for pitch, roll and yaw control corresponds to the vertical takeoff gross weight.

Available thrust modulation levels are defined by a temperature limit and other practical considerations. The VTOL control requirements are shown superimposed on a graph of available control power in Figure 5-6. The 3 second 1600°F EGT rating shown in the figure is an upper bound on available modulation capability and is well in excess of the design guideline requirements for combined attitude control and height control with 50% attitude control. Thrust requirements in both cases, as well as the T/W = 1.05 sustained thrust capability, are all well below the intermediate power rating.

The 25% thrust modulation curve shown in Figure 5-6 represents a practical design goal which insures better control characteristics with respect to cross-coupling effects and control response. Combined attitude and height with 50% attitude control requirements are shown in Figure 5-6 to be well within this design goal. The relationship of the 25% thrust modulation roll and pitch attitude control capability to Level 1 guideline control criteria of Figure 5-1 is shown in Figure 5-7. Also, the yaw attitude control capability based upon maximum effective thrust deflection angles of 4° and 8° at the lift fan and the lift/cruise fans, respectively, is shown in Figure 5-7. The yaw attitude control capability design point at the OWE + Payload weight is also indicated. Yaw attitude control capability is usually most critical at the landing gross weight as the ratio of thrust to be deflected to aircraft yaw inertia is generally smallest at this gross weight. In all three cases, the attitude control capability is shown in Figure 5-7 to be excess of the dual Level 1 guidelines of required attitude angle change in one second and M/I ratio.

5.2.3 CONTROL POWER CAPABILITY FOR RESEARCH - The Turbotip RTA aircraft has excess control margins for future research programs in the area of control power requirements. The margins based upon a 25% thrust modulation and on the 1600°F EGT temperature limit for pitch and roll, and 4° and 8° of thrust deflection for yaw are shown in Figure 5-8. The 4° and 8° yaw thrust deflection limits are somewhat arbitrary and may be raised to provide additional control capability.

5.2.4 CONTROL WITH ONE ENGINE INOPERATIVE - Level 2 guideline control power requirements for operation in the one engine out emergency flight condition are substantially reduced. The relationship of Level 2 emergency control requirements to the one gas generator out propulsion system capabilities is shown in

FIGURE 5-6
TURBOTIP RTA
LEVEL 1 VTOL CONTROL DEFINITION

VTOGW = 25,286 LB

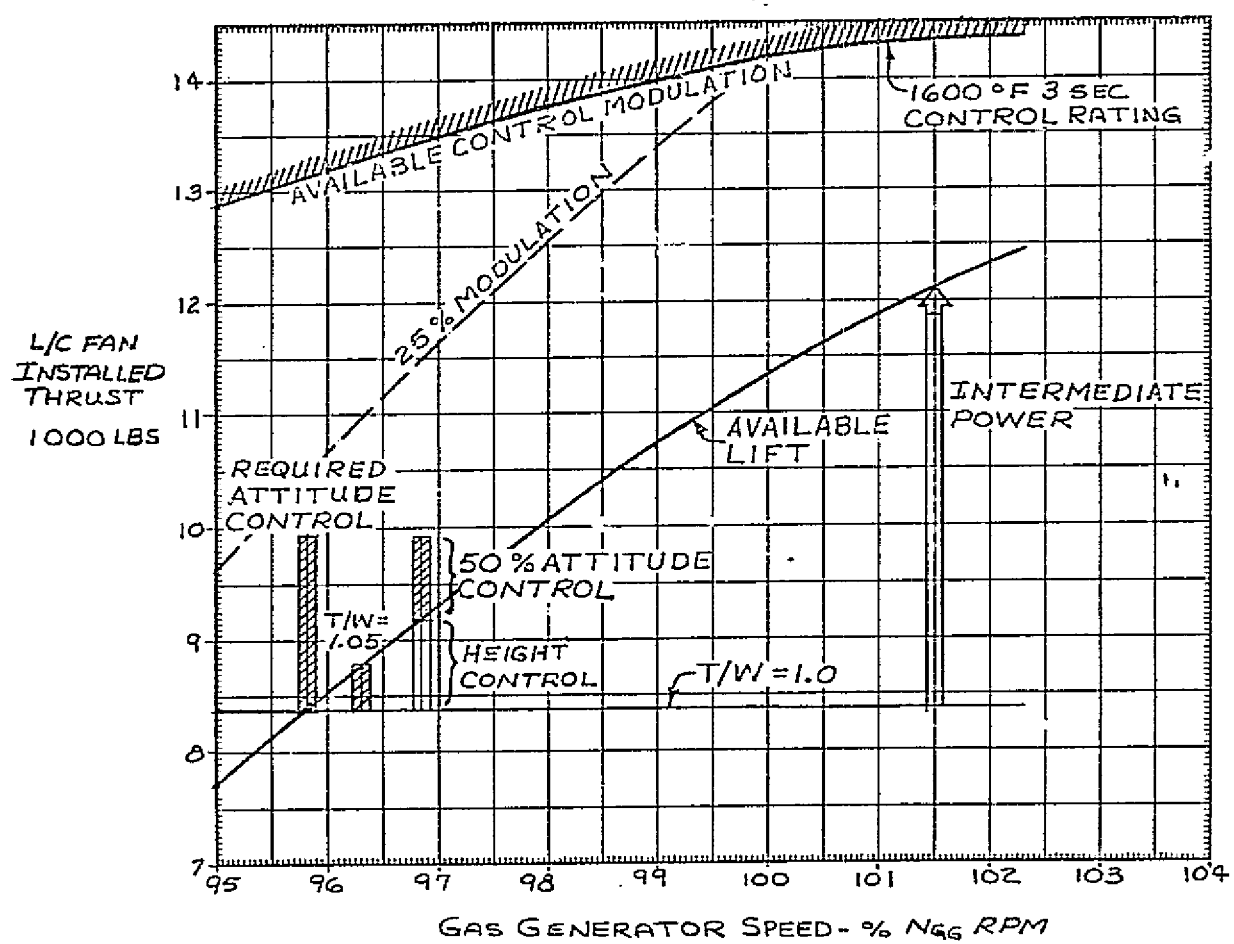


FIGURE 5-7
TURBOTIP RTA
COMPLIANCE WITH LEVEL 1
CONTROL CRITERIA

VTOGW = 25,286

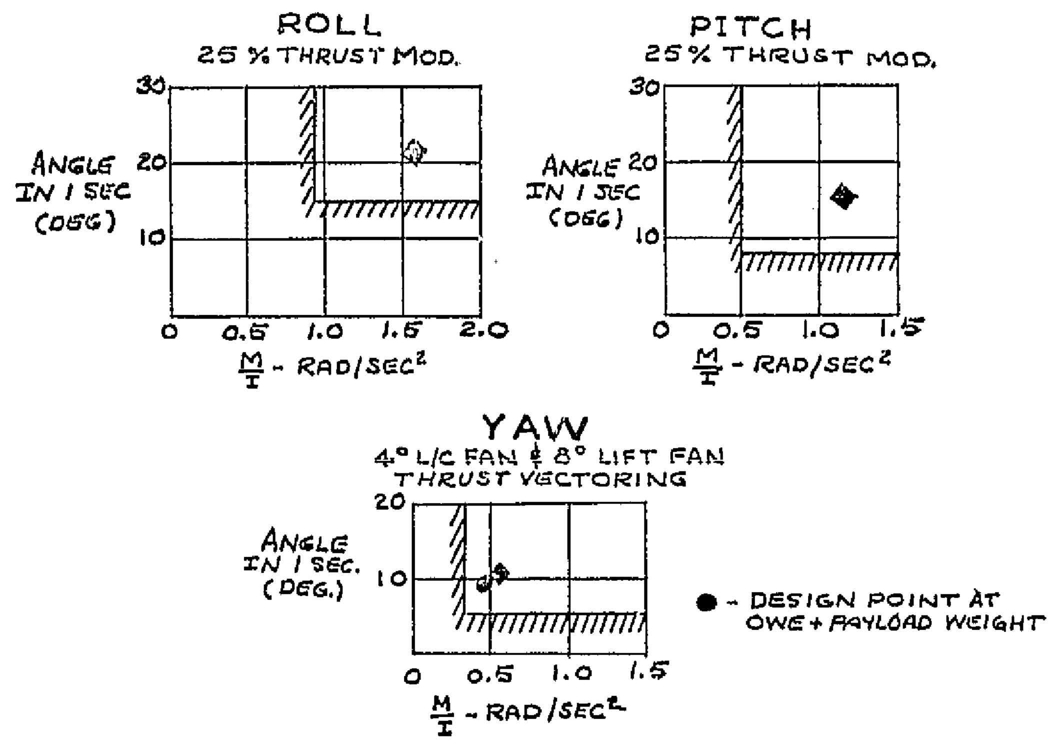


FIGURE 5-8
TURBOTIP RTA
CONTROL CAPABILITY FOR RESEARCH

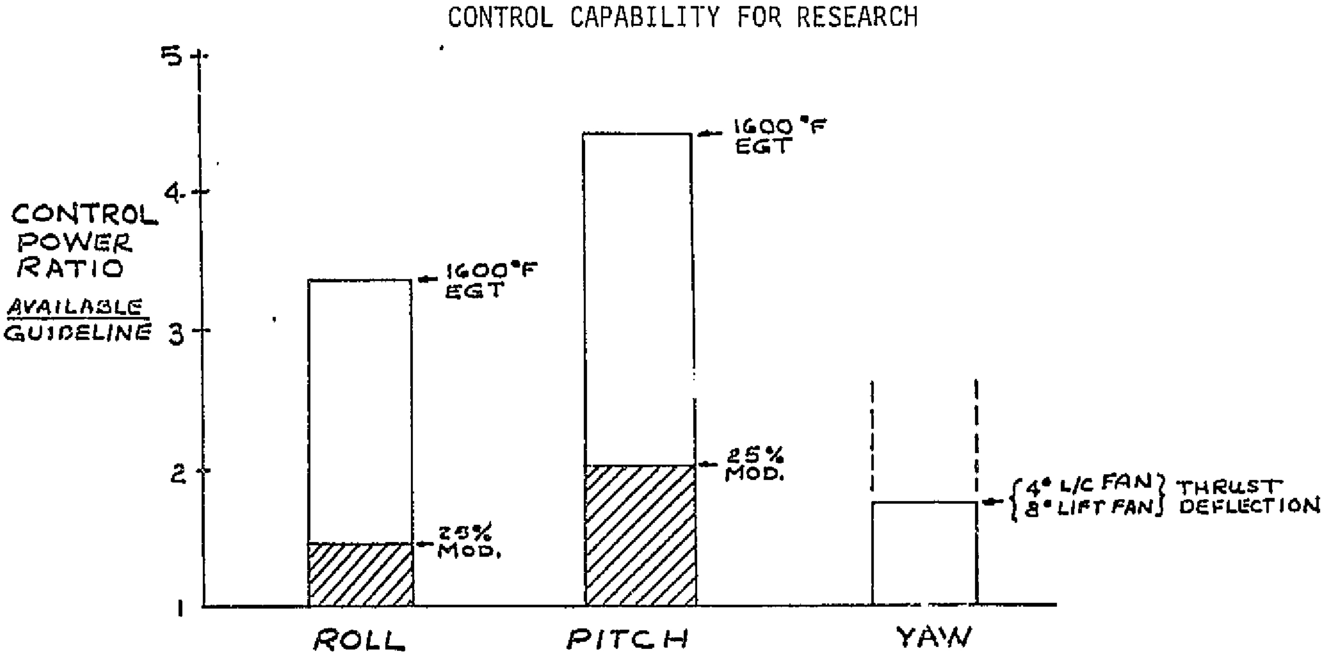


Figure 5-9. It is evident from the figure that control capability as determined by the 3 sec. 1600°F EGT temperature limit is well in excess of the control modulation requirements which are all below the intermediate power rating.

5.2.5 GYROSCOPIC COUPLING - Gyroscopic coupling is present between the pitch, roll and yaw axes due to the combined angular momentum of the fans and gas generators. Because of gyroscope coupling, a control input to the pitch axis induces angular accelerations about the roll and yaw axes. Similarly, control inputs to the roll and yaw axes each induce angular acceleration about the pitch axis. The pertinent control power guideline requirements states that at least 90% of the specified normal control power capability shall remain after compensation for gyroscopically induced accelerations of magnitude corresponding to that produced by maneuvers employing guideline level control power.

The gyroscopic coupling evaluations were performed at a power setting corresponding to T/W = 1.0 at VTO gross weight. The magnitudes of the gyroscopically induced accelerations were determined by employing the guideline maneuver for each axis which required the greatest amount of control power modulation with the attitude stabilization systems engaged. For the pitch and roll axes, maneuvers corresponding to achieving the guideline attitude angular change in one second were employed and the maximum resultant angular rates were used to compute the maximum coupled gyroscopic accelerations. For the yaw axis, a maneuver producing the guideline angular acceleration (M/I ratio) was employed and the coupled gyroscopic acceleration at 1 second was utilized in the analysis. These maneuvers and the magnitudes of the coupled gyroscopic accelerations are shown in Figure 5-10. Reductions in available control power based upon 25% thrust modulation for the pitch and roll axes, and 4° and 8° thrust deflection limits for yaw control are shown in Figure 5-11. The remaining control power in each case is significantly greater than the 90% guideline.

5.2.6 CENTER OF GRAVITY TRIM - Coincident center of thrust and center of gravity (cg) at VTO gross weight is a design requirement which is accomplished by appropriate distribution of installed lift between the lift fan and the two lift/cruise fans. In addition, the turbotip aircraft design is such that there is no shift in the cg location over the weight range from the STO gross weight to the OWE + Payload weight. Therefore, cg trim requirements were not considered in determining the thrust modulation control margins.

FIGURE 5-9
TURBOTIP RTA
LEVEL 2 VTOL CONTROL DEFINITION

VLGW = 23660 LB

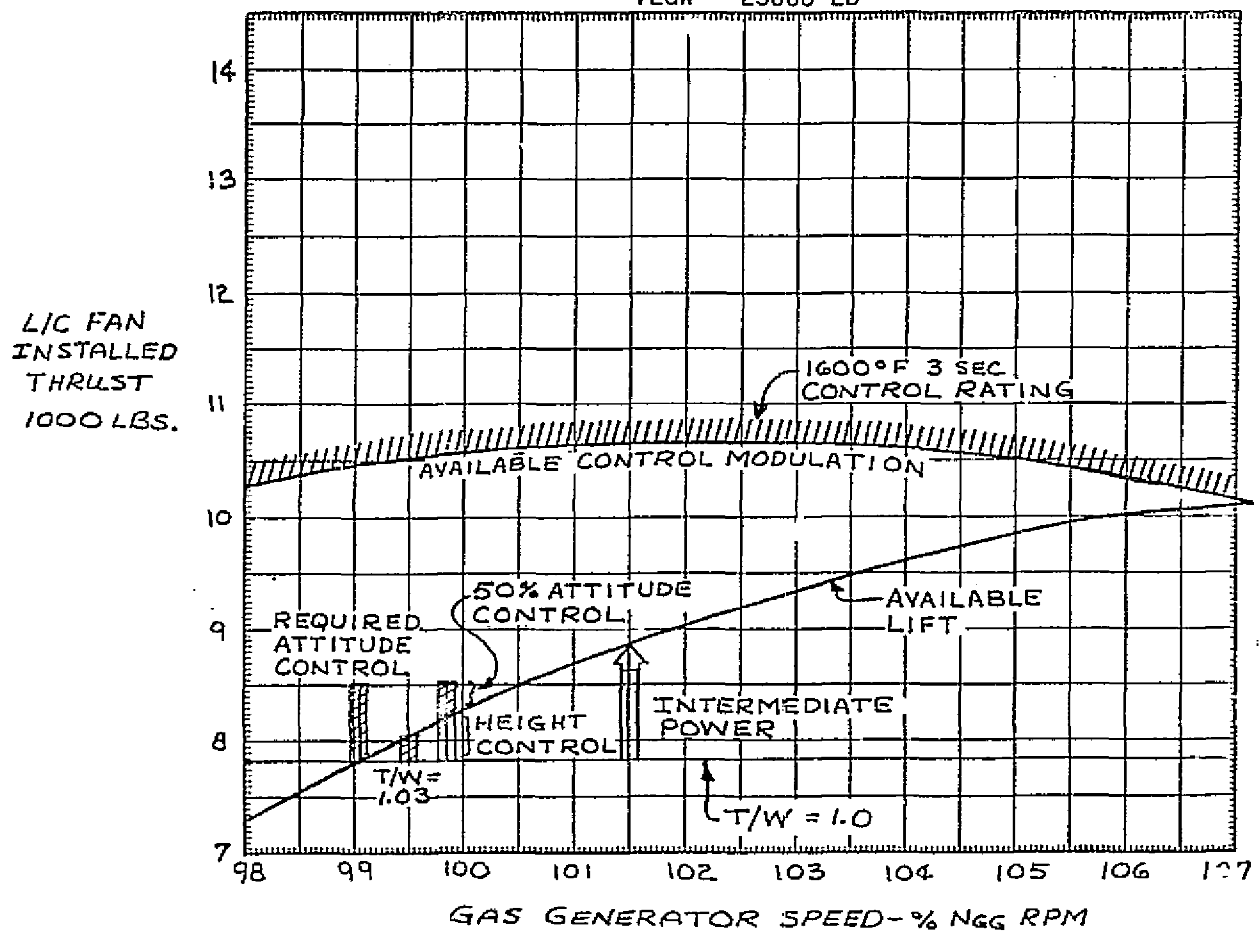


FIGURE 5-10
TURBOTIP RTA
CONTROL POWER REQUIREMENTS
TO ARREST GYROSCOPIC MOMENTS

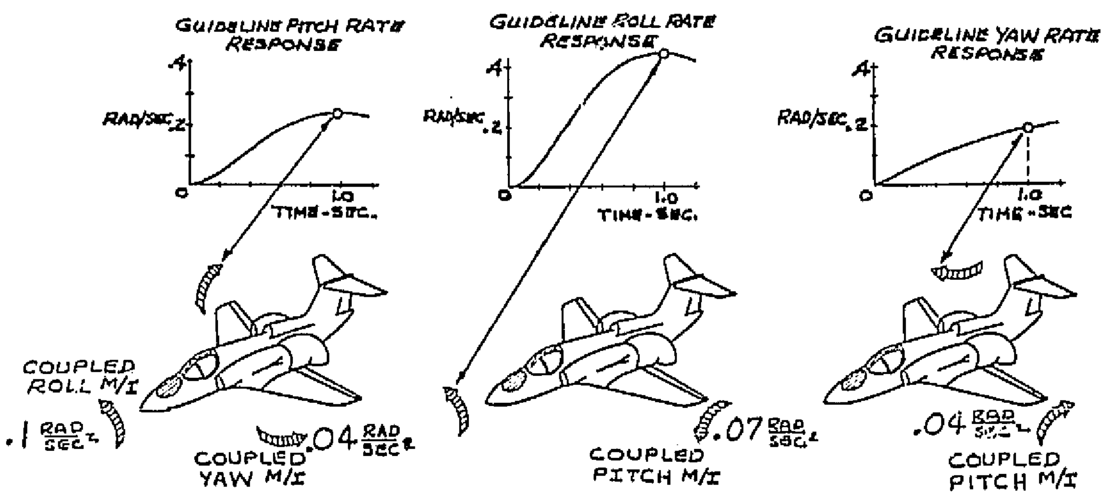
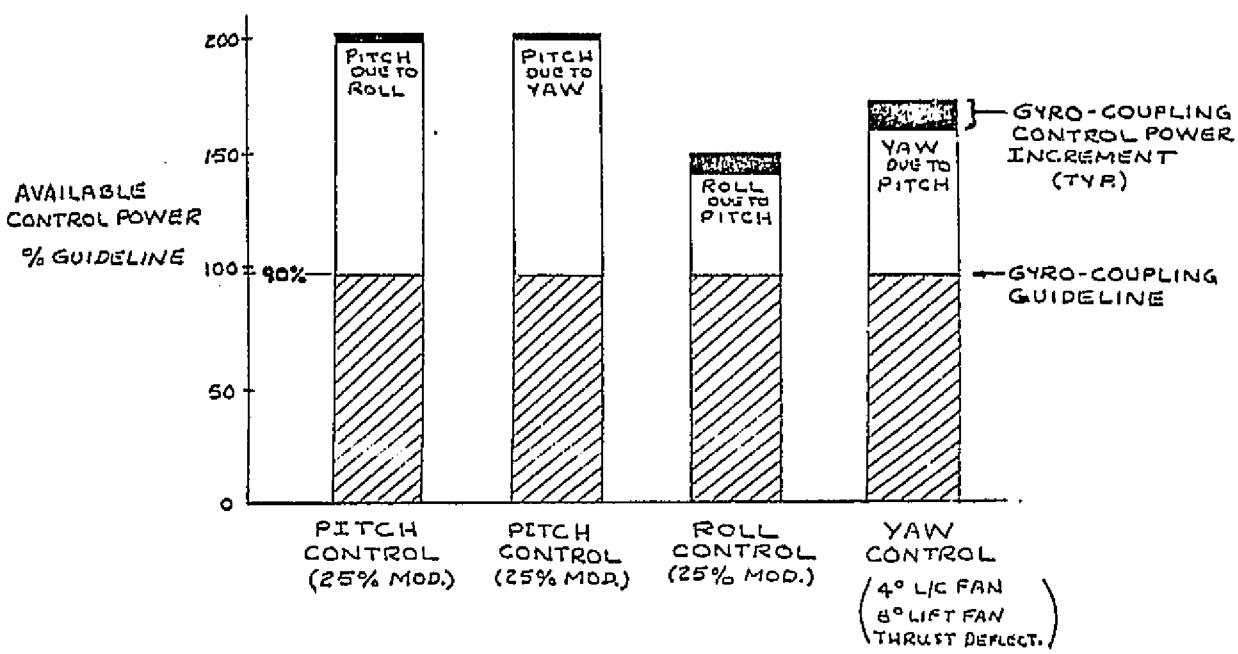


FIGURE 5-11
TURBOTIP RTA
GYROSCOPIC COUPLING EFFECTS
ON CONTROL POWER



5.2.7 CONTROL SYSTEM RESPONSE - Control response consists of two components: (1) fan thrust response from an increase of gas energy, and (2) thrust reduction modulation. The uncompensated response characteristics of the first of these components is relatively slower than the second. The excess control margin in both normal and engine out operation permits effective use of lead compensation to significantly improve the fan thrust increase response time. Lead compensation magnifies the initial value of control commands which causes a more rapid initial rate of increase in fan thrust. The initial magnification is washed out exponentially with the net result being a more rapid rise in fan thrust to the commanded level.

Estimated values of each of the two response components and their combination based upon previous LF460 fan studies and ETaC test results are shown in Figure 5-12. The response time corresponding to the normal operation VTO gross weight is indicated on the Level 1 graph, that at the one engine out VL gross weight is indicated on the Level 2 graph, and that at the OWE + Payload weight on both graphs. With lead compensation, the response effective time constants are well below the guideline limits. The response time in the one engine out condition is at a lower power setting and is therefore only slightly greater than that for normal operation indicating an insignificant degradation in the control response time.

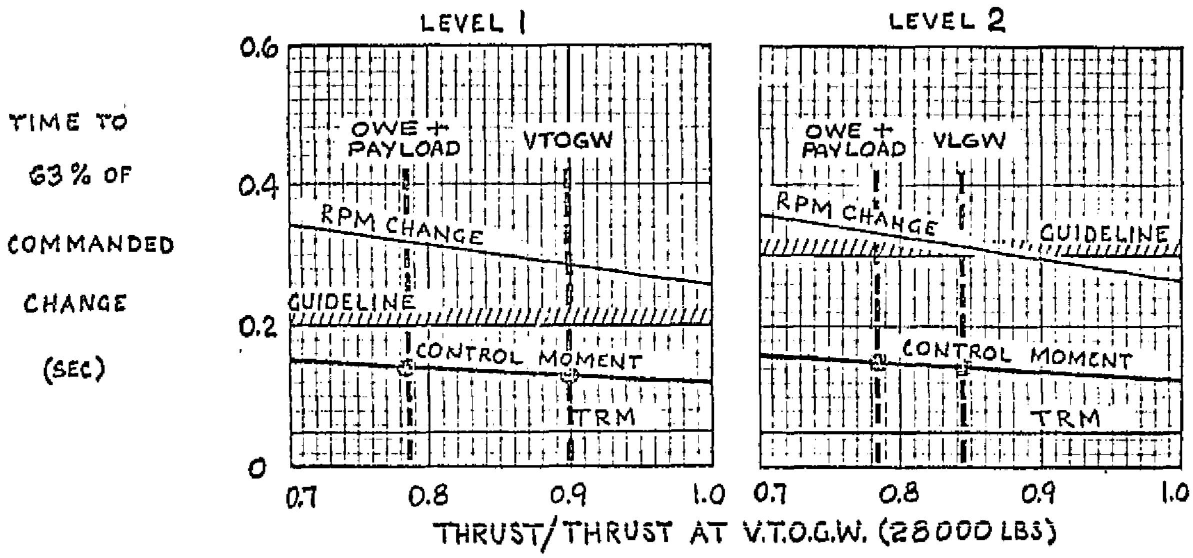
5.3 MECHANICAL RTA CONTROL

The control concept of the shaft coupled system is based on fan blade pitch (β) changes for aircraft pitch and roll control, and a combination of β, fan rpm, and engine power for height control.

5.3.1 ATTITUDE CONTROL - Control signals cause a differential β change between the lift cruise fans for roll control, and between lift cruise fans and the lift fan for pitch control. Power transfer associated with these differential β changes occurs inherently via interconnecting shafts. Yaw control is provided by yaw vanes located at the fan flow exits. The yaw system and the transition thrust vectoring system are common to both, mechanical and turbotip RTA. The attitude control concept is illustrated in Figure 5-13.

5.3.2 THRUST MANAGEMENT CONTROL - The thrust management control concept, providing the height and total power control functions, is illustrated in Figure 5-14. This concept is based on direct command of engine power via fuel controls and an automatic compensation system operating through β control. Because fan thrust response to β control in the compensation scheme improves

FIGURE 5-12
ESTIMATED CONTROL RESPONSE
(LEAD COMPENSATION WITH ETaC)



TRM = THRUST REDUCTION MODULATION

FIGURE 5-13
MECHANICAL RTA
POWERED LIFT ATTITUDE CONTROL

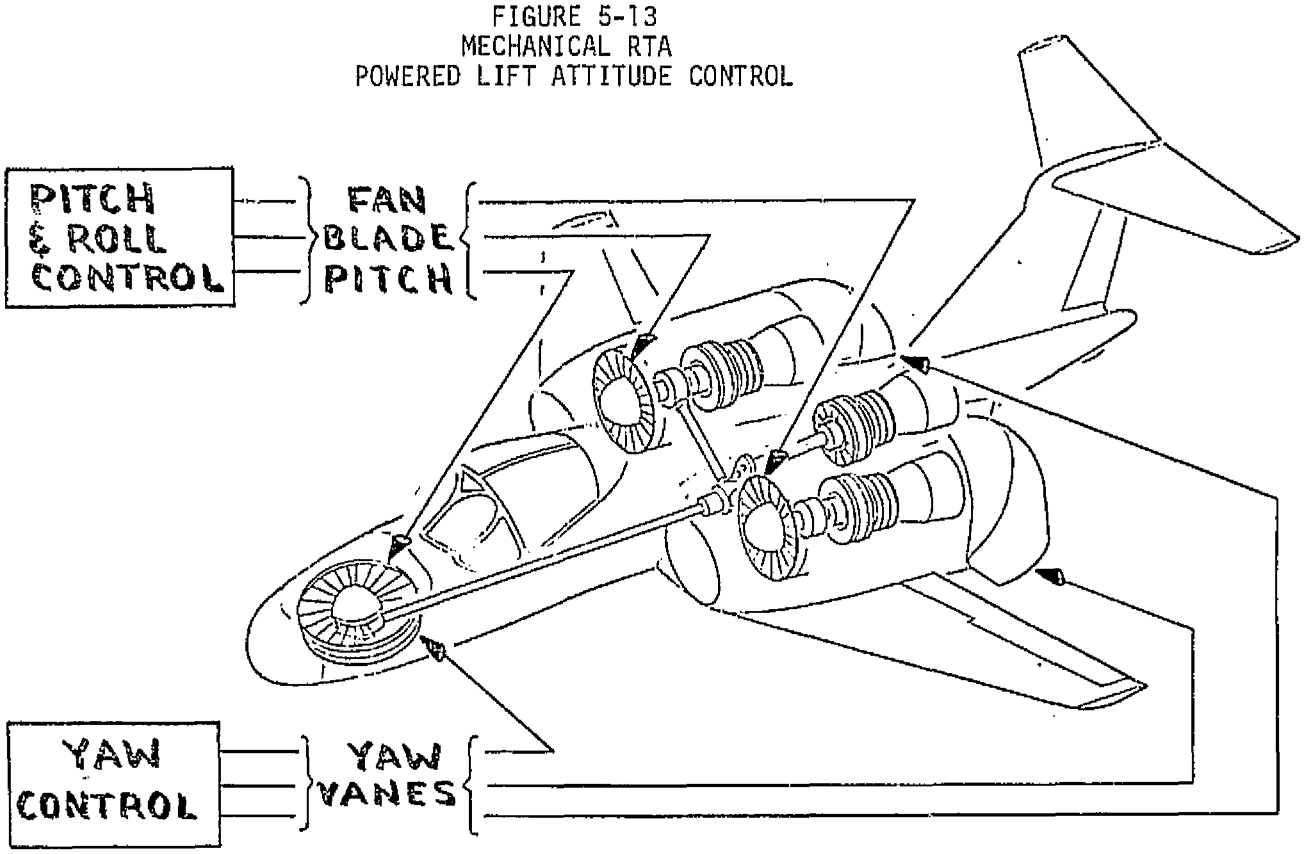
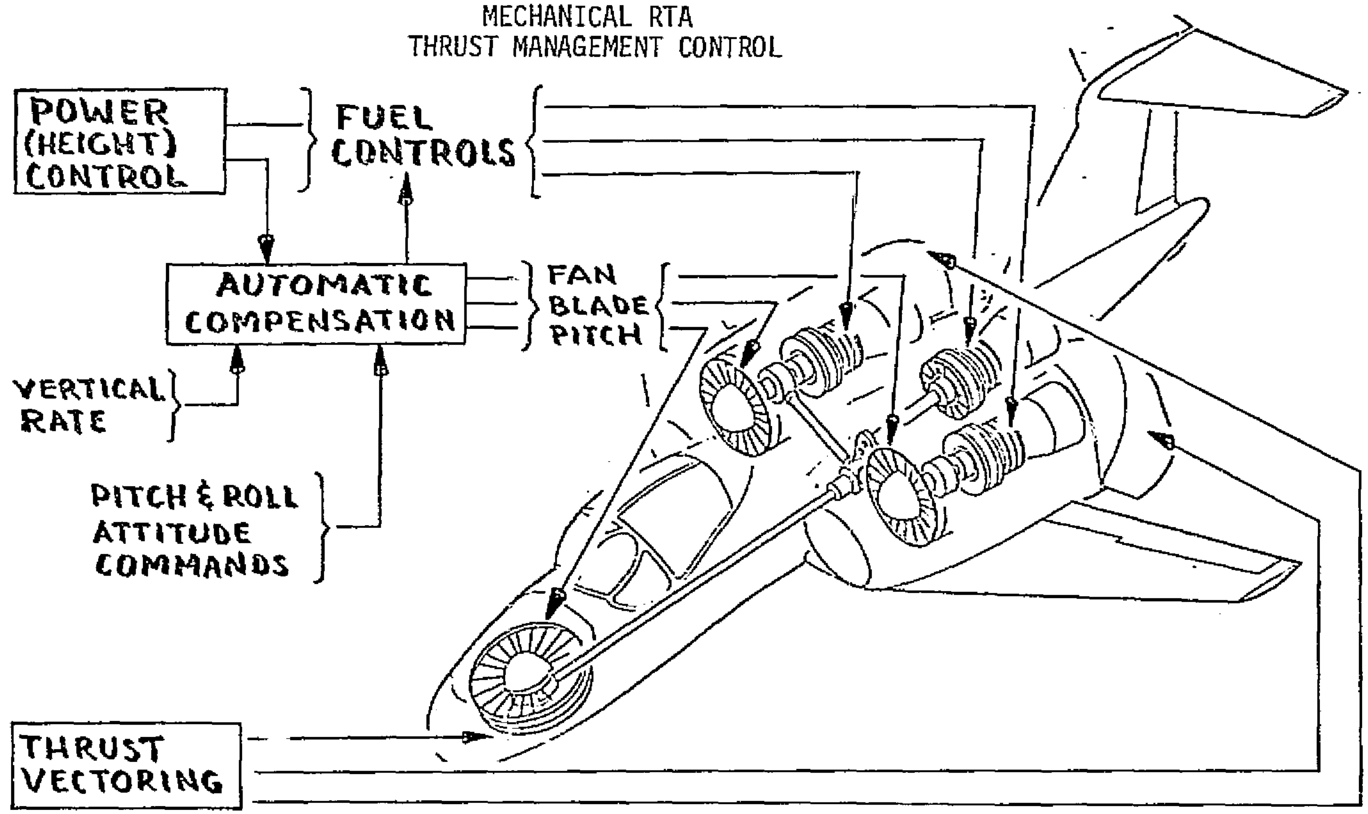


FIGURE 5-14
MECHANICAL RTA
THRUST MANAGEMENT CONTROL



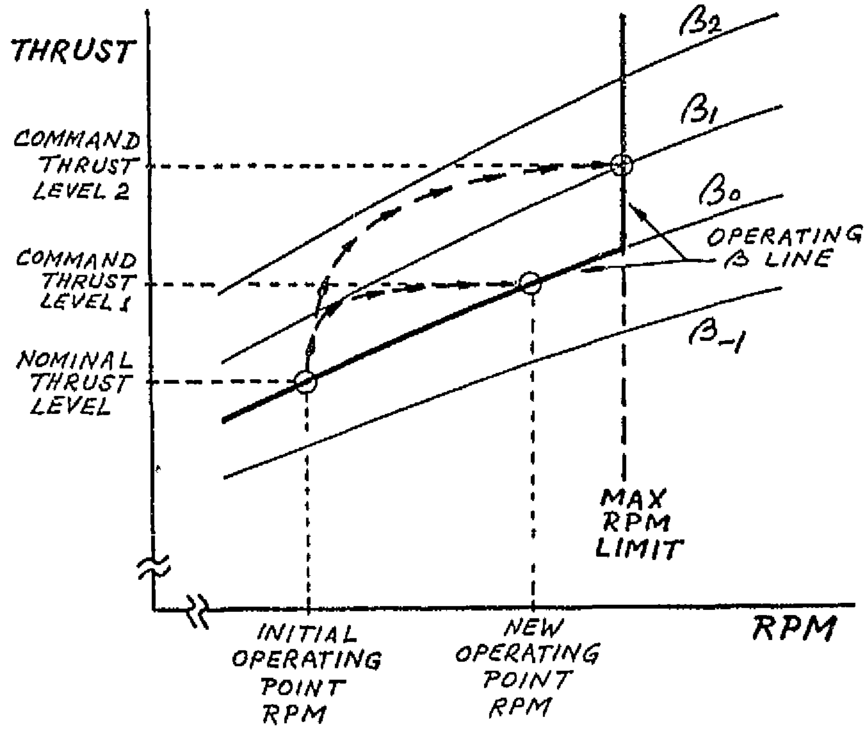
effectively total thrust response for height control. The automatic compensation also provides vertical rate damping to improve height control precision and reduce external disturbances effects on aircraft control through a low authority closed loop control of total thrust. The automatic control further compensates for any lift changes resulting from power transfer during pitch and roll control inputs. Thus the automatic compensation system provides important interface between pitch/roll and thrust management control systems and serves to decouple their respective control axes. It also provides, in conjunction with the aircraft's stability augmentation system, control priority management to insure that the more critical pitch and roll control functions are satisfied before the height (total thrust) control demands.

The basic principle of operation of this thrust management control concept is briefly described as follows.

A control signal from the power lever calling for total power increase is transmitted to the engine fuel controls. A signal difference is created immediately between the command signal and the fan rpm feedback signal. This difference is applied to the fan β controls causing an increase in blade angle of attack at all fans, increasing total thrust. However, because the blade angle change cuts into the β margin reserved for aircraft attitude control, a steady state change in β is not permitted. Therefore, fan rpm is allowed to increase which reduces the difference between the rpm feedback and the command signals. Since β change is proportional to the signal difference, β is reduced and settles out at its original steady state level as the rpm feedback signal approaches the command signal. Thus, attitude control margin is preserved. The principle of operation of this system is illustrated in Figure 5-15.

The above described control laws are altered, however, at the limit fan rpm (free turbine speed). At this time a steady state β change is permitted with a correspondingly reduced attitude control margin. A full attitude control input at this time commands at least one fan above its maximum β capability, while no restrictions are encountered on the fans commanded to lower β. This combination of differential β settings requires lower total horsepower at the fans than the engines are delivering, with a resultant tendency to overspeed the fnas (free turbine). The overspeed control at the engines automatically reduces fuel flow which reduces engine power, thereby causing a reduction of total thrust output. Thus, a simultaneous command of full aircraft attitude control and thrust to weight (T/W) capability at maximum

FIGURE 5-15
MECHANICAL RTA
CONTROL SYSTEM CONCEPT



VTO gross weight is resolved by giving priority to the attitude control moment over the T/W command. The minimum design T/W during full control application is 1.0.

In the event of loss of automatic compensation, this thrust management concept reverts to an open loop engine power control system. Without the benefit of improved thrust response for height control, vertical rate damping, control interface, and priority management functions, aircraft control is degraded but still adequate to satisfy emergency level requirements. This design approach is consistent with the RTA philosophy of design considering the RTA's current state-of-the-art capabilities. This system was also modeled for the September 1976 FSAA flight simulation of the RTA. The results show that it provided satisfactory control characteristics. It is important to point out, however, that this control concept was developed based on a set of typical propulsion characteristics provided by the engine manufacturer solely for use in the flight simulation program. Propulsion performance data provided specifically for this RTA definition study was insufficient to either confirm or revise this control system design.

5.3.3 CONTROL THRUST REQUIREMENTS - To achieve compliance with the control design guidelines, the required mechanical RTA thrust modulation levels are as shown in Figure 5-16. The magnitude of thrust change needed for pitch is approximately the same as for the turbotip RTA. Roll thrust modulation, however, is considerably higher in this configuration than in the Turbotip RTA because gas generators are located coaxially with the lift cruise fans resulting in higher roll inertia of the aircraft and a shorter distance between the lift/cruise nozzles. Thrust deflection for yaw control compares very closely with the Turbotip RTA yaw requirements.

5.3.4 CONTROL THRUST CAPABILITY - The engine company data provided for this study was plotted in terms of fan thrust versus fan speed for three fan blade pitch angles as shown in Figure 5-17. While the amount of data is not adequate to define the simulated control system as discussed earlier in this section, it was sufficient to determine the available control margins with respect to the RTA's operational gross weights at the constant maximum permissible fan RPM. As shown in Figure 5-17, the available fan thrust margin for attitude control at VTOGW T/W = 1 is nearly twice the guideline requirement if a reduction of stall margin is permitted as discussed in Section 3.2.

FIGURE 5-16
MECHANICAL RTA
THRUST MODULATION REQUIREMENTS

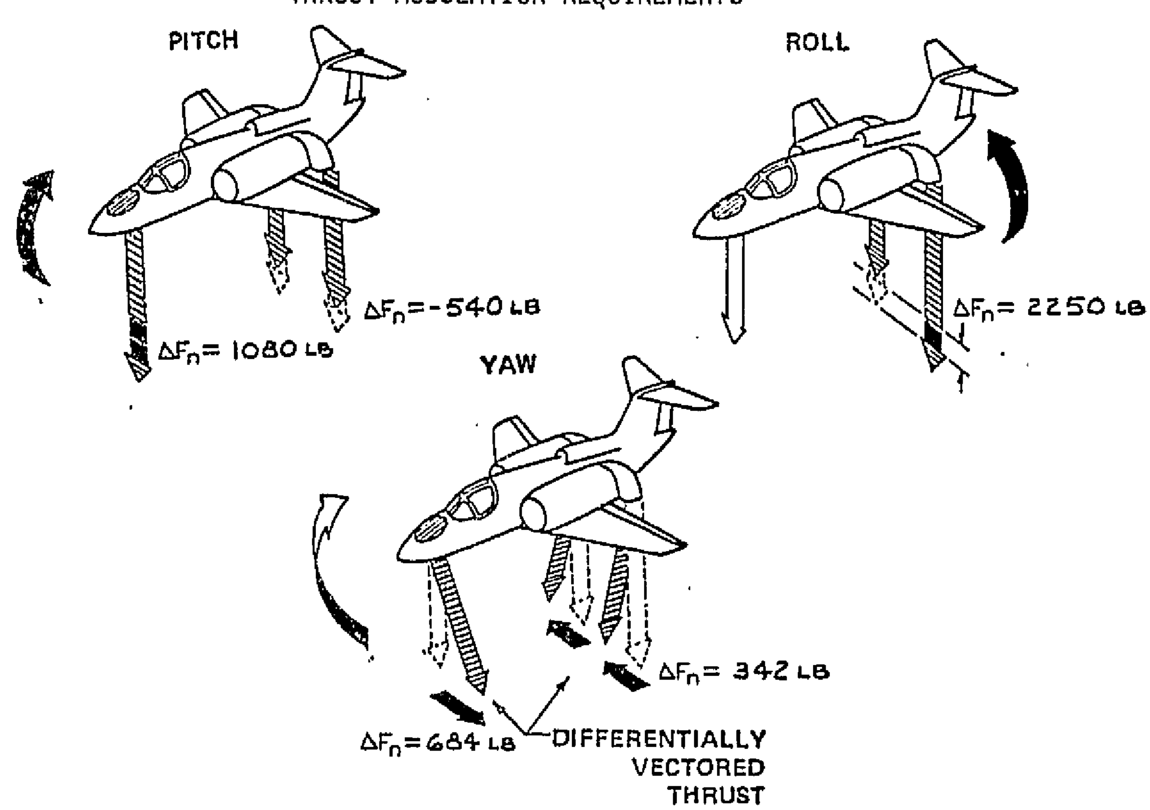
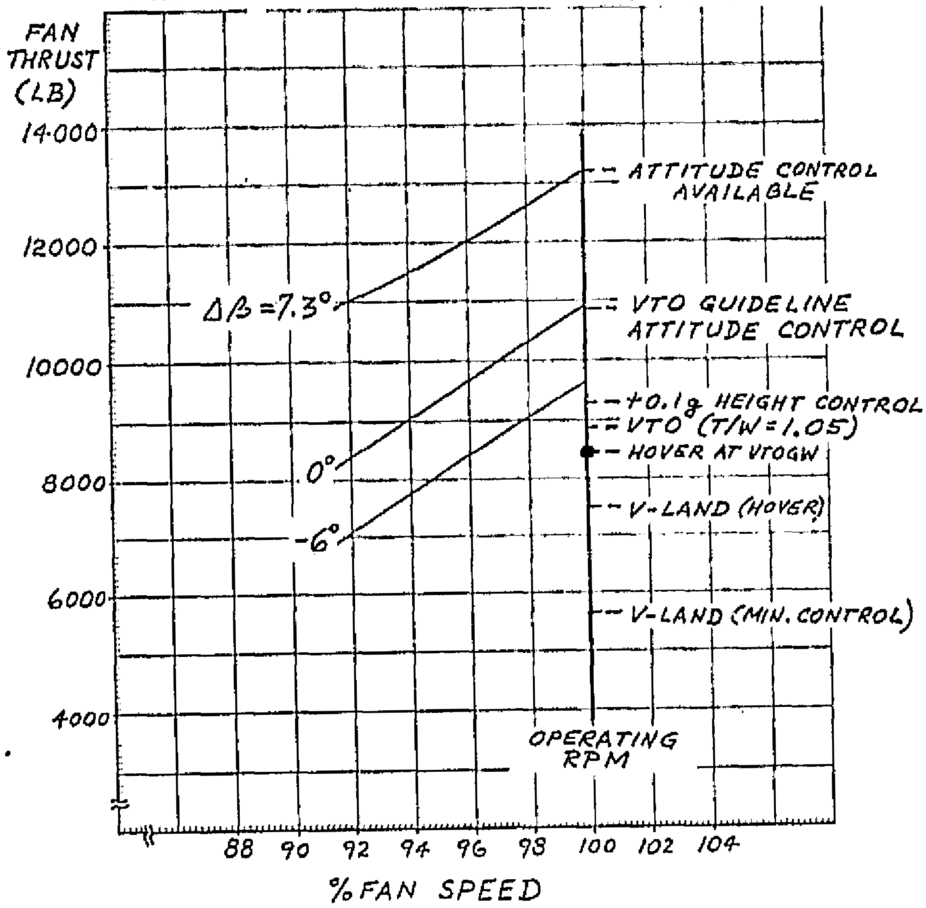


FIGURE 5-17
CONTROL THRUST DEFINITION
BASED ON CONSTANT RPM CONTROL SYSTEM CONCEPT



5.3.5 GYROSCOPIC COUPLING - The gyroscopic coupling analysis performed for the mechanical RTA was identical to that previously described in Section 5.2 for the Turbotip RTA. The results of this analysis are shown in Figure 5-18. Comparison of these values with those for the Turbotip RTA shown in Figure 5-10 in Section 5.2 indicates the gyroscopic coupling to be less in all cases. This is due primary to the smaller mechanical fan inertias.

FIGURE 5-18
MECHANICAL RTA GYROSCOPIC COUPLING

Coupled Axis/Input Axis	Coupled M/I (rad/sec^2)
Roll/Pitch	.06
Pitch/Roll	.05
Pitch/Yaw	.02
Yaw/Pitch	.02

5.4 FLIGHT CONTROL SYSTEM

The basic aircraft configurations of the turbotip and mechanical versions of the RTA and their respective control concepts are very similar. Only the methods of thrust modulation for aircraft attitude and height control, stemming from the means of energy distribution and transfer, are different between the two systems. This allows a common definition of the flight control system outside of the specific thrust modulation techniques.

5.4.1 CONTROL SYSTEM FUNCTIONAL REQUIREMENTS - General flight simulation and V/STOL flight experience very strongly reinforce the VTOL requirement for attitude stabilization. This was recently confirmed by RTA simulation tests conducted by MCAIR on the FSAA under contract to NASA/Navy. The results, confirm most of the previously established desirable functions and modes for the RTA. Pitch and roll attitude command is desired in hover. Above 30-40 knots, the pilots prefer rate command for greater maneuver capability but with attitude stabilization between maneuvers to alleviate workload. The directional axis command of yaw rate with a heading hold mode to keep the aircraft from weathervaning provides the desired characteristics in hover. In transition, feedback of lateral acceleration, roll rate, bank angle, yaw rate, and side velocity is used to insure lateral-directional stability and good turn coordination performance.

The heavy pilot workload on approach to a vertical landing can be reduced by an automatic vectoring mode operating as a function of range-to-go or altitude information. Automatic trim and manual takeoff trim setting selection eliminate pilot concern about maintaining the aircraft in trim, while the vertical rate damper reduces effects of disturbances and enhances precision of height control.

A list of the desired flight control system functions and modes for the RTA is contained in Figure 5-19.

5.4.2 FLIGHT CONTROL SYSTEM DESCRIPTION - The selection of a suitable flight control system for the technology demonstration aircraft is based on the premise that the aircraft should demonstrate all aspects of the technology which it represents. The elements and functions of the powered lift control system and the characteristics of V/STOL operation in general require a control system which is highly flexible. To satisfy this requirement, the Active Control System (ACS) approach was selected. The ACS is defined as a control-by-wire through a dedicated flight control system computer as shown in Figure 5-20.

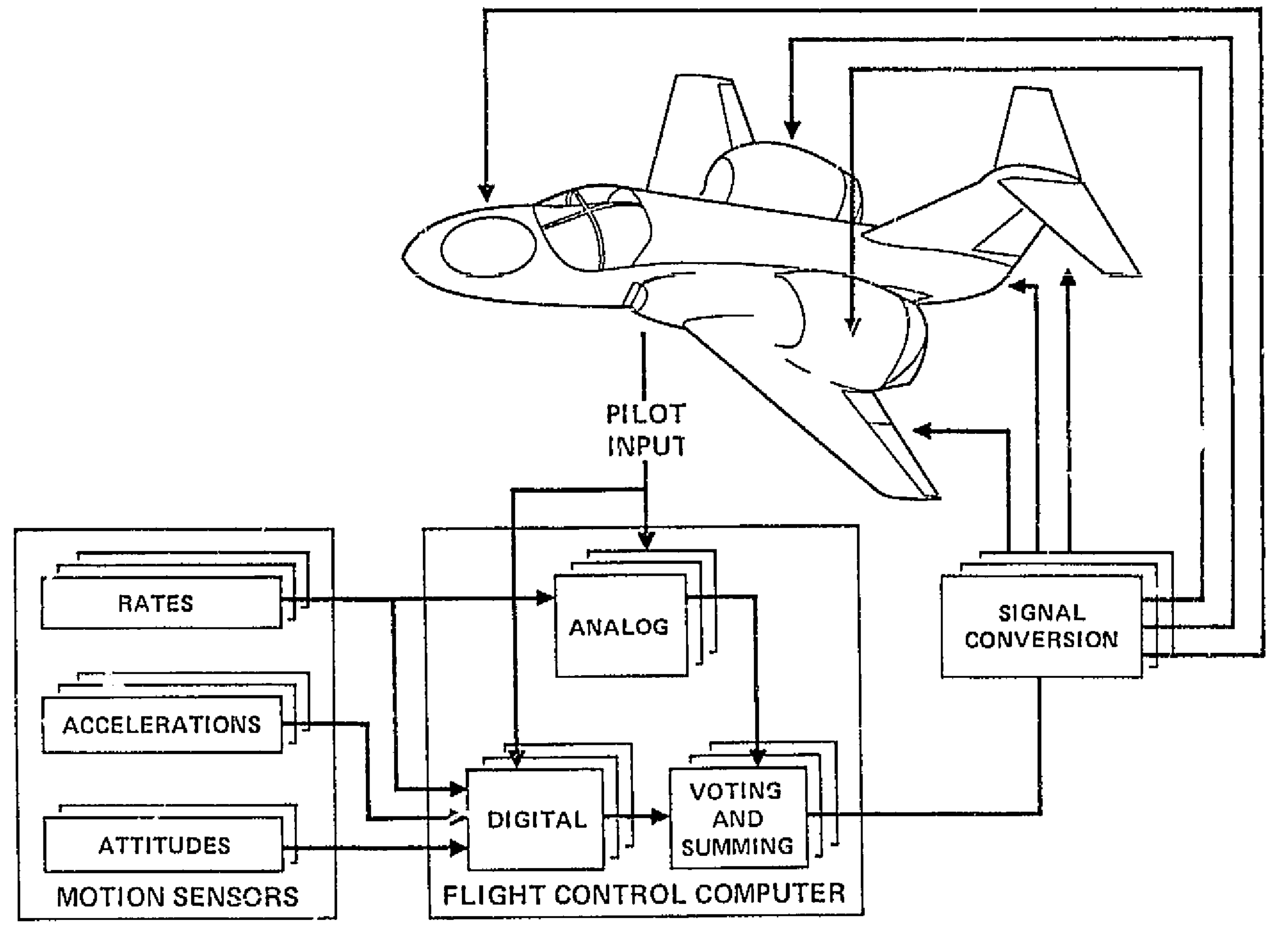
The ACS is a triplex hybrid implementation of digital and analog functions combined to achieve desired high reliability and flight safety goals. The analog computer provides the minimum flight control functions deemed necessary for safe flight, particularly during approach and landing following a complete digital system failure. The digital computer provides the capacity and flexibility required to incorporate many different primary flight modes, which are desirable for widening the scope of research applications. Both the anlog and digital computations are performed simultaneously. The redundancy provides capability to land safely in the event of either complete digital or complete analog control function computation failure with reduced performance in either case.

The digital control-by-wire flight control system in the RTA is nearly ideal for applications to research. The high flexibility of the digital computer to accept changes and modifications not only permits optimization of the basic aircraft handling qualities throughout the flight envelope but also allows a wide range of aircraft configuration variations, flight conditions, and mission tasks for research.

FIGURE 5-19
FLIGHT CONTROL SYSTEM FUNCTIONS/MODES

- CONVENTIONAL FLIGHT REGIME
  - MOTION DAMPING AND COMMAND SHAPING FUNCTIONS
- TRANSITION FLIGHT REGIME
  - PITCH RATE COMMAND/ATTITUDE HOLD
  - ROLL RATE COMMAND/ATTITUDE HOLD/HEADING HOLD
  - YAW RATE DAMPING/TURN COORDINATION
- HOVER FLIGHT REGIME
  - PITCH ATTITUDE COMMAND
  - ROLL ATTITUDE COMMAND
  - YAW RATE COMMAND/HEADING HOLD
- FLIGHT PATH CONTROL MODE
  - AUTOVECTOR
- AUXILIARY FUNCTIONS
  - AUTOMATIC TRIM
  - TAKEOFF TRIM SELECT
  - VERTICAL RATE DAMPER
  - EXHAUST SPLAY (FLOW FIELD EVALUATION)

FIGURE 5-20
RTA BASELINE FLIGHT CONTROL SYSTEM



6. CONCLUSIONS

Based on the results of this study it is concluded that the subject lift/cruise V/STOL technology demonstrator aircraft:

- o meet or exceed all of the design guideline requirements
- o make maximum utilization of existing components
- o provide excess control power for research purposes
- o offer significant research capability at low cost
- o provide for demonstrating aero/propulsion efficiencies and cruise fan integrity at cruise and loiter conditions
- o utilize a low cost RTA approach for subsystem design
- o provide a low risk approach for demonstrating the lift/cruise fan V/STOL aircraft concept.

APPENDIX A

DESIGN GUIDELINES AND CRITERIA
FOR
DESIGN DEFINITION STUDY OF A
LIFT CRUISE FAN TECHNOLOGY V/STOL AIRCRAFT

ATTACHMENT 1

DATED DECEMBER 1975

Attachment 1 to National Aeronautics and Space Administration, Ames Research Center, Moffett Field, California Letter RFP 2-26146(RFL) dated April 9, 1976.

ATTACHMENT 1

DESIGN GUIDELINES AND CRITERIA

FOR

DESIGN DEFINITION STUDY OF A

LIFT CRUISE FAN TECHNOLOGY V/STOL AIRCRAFT

DECEMBER 1975

The purpose of these guidelines is to provide a basis for comparing the conceptual designs of V/STOL Technology aircraft using the lift/cruise fan propulsion system. These guidelines will provide direction for only those items required for conceptual design considerations. This is not an attempt to provide criteria for either the preliminary or detail design of military aircraft.

Except where specific criteria are given, handling qualities shall be consistent with the intent of AGARD-R-577-70 and MIL-F-83300. Under MIL-F-83300, the aircraft will be considered in the class II category. Two levels of operation will be considered. Level I is normal operation with no failures. Level 2 is operation with a single reasonable failure of the propulsion or control system.

Upon any reasonable failure of a power plant or in the control system, the aircraft shall be capable of completing a STOL flight mode takeoff and continuing sustained flight. With failure of the most critical power plant, Level 2 performance shall be achieved at sea level and at 90°F under the following conditions: (a) STOL Mode - capability for continuing flight on a flight path 1 1/2° above the horizontal at a weight which shall include 2500 lbs. payload and fuel sufficient for 11 STOL test missions; (b) VTOL Mode - capability for a thrust to weight ratio of 1.03 without altitude control at a weight which shall include 2500 lbs. payload and fuel sufficient for 2 VTOL test missions. Fan failure during low speed flight is not a design requirement (as similarly the case for rotor type or propeller-driven concepts), althought consideration of a turbo-engine failure is a design requirement.

1.0 Flight Safety and Operating Criteria

1.1 Handling Qualities Criteria (low speed powered lift mode)

Definitions of the two levels are as follows:

Level 1: Flying qualities are satisfactory for research and technology demonstration missions when flown by an engineering test pilot.

Level 2: Flying qualities are adequate to continue flight and land. The pilot work load is increased but is still within the capabilities of an engineering test pilot.

1.1.1 Attitude Control Power (S.L., 90°F).

Applicable for all aircraft weights and at any speed up to V_{con}. For purposes of this study, the VTOL values will apply near hover (0 to 40 kts); where the STOL values will apply when operating above 40 knots. The Tables list minimum values, higher levels are desirable for research purposes.

Level 1: The low speed control power shall be sufficient to satisfy the most critical of the three following sets of conditions:

Conditions (a) -- to be satisfied simultaneously,

(1) Trim with the most critical CG position.

(2) In each control channel provide control power, for maneuver only, equal to the most critical of the requirements given in the following table.

Axis	Maximum Control Moment / Inertia		Attitude Angle in 1 sec after a Step Input	
	VTOL	STOL	VTOL	STOL
Roll	$\pm$ 0.9 rad/sec^2	$\pm$ 0.6 rad/sec^2	$\pm$ 15 deg	$\pm$ 10 deg
Pitch	$\pm$ 0.5 rad/sec^2	$\pm$ 0.4 rad/sec^2	$\pm$ 8 deg	$\pm$ 6 deg
Yaw	$\pm$ 0.3 rad/sec^2	$\pm$ 0.2 rad/sec^2	$\pm$ 5 deg	$\pm$ 3 deg

These maneuver control powers are applied so that 100% of the most critical and 30% of each of the remaining two need occur simultaneously.

Condition (b) -- At least 50% of the above control power shall be available for maneuvering, after the aircraft is trimmed in a 25 knot crosswind.

Condition (c) -- At least 90% of the control power specified in condition (a) shall be available after compensation of the gyroscopic moments due to the maneuvers specified in condition (a). This condition includes trim with the most critical CG positon.

Level 2: The low speed control power shall be sufficient to satisfy, simultaneously, the following:

(1) With the most critical CG position trim after any reasonable single failure of power plant or control system.

(2) In each control channel, provide control power, for maneuver only, equal to at least the following:

Axis	Maximum Control Moment / Inertia		Attitude Angle in 1 sec after a Step Input	
	VTOL	STOL	VTOL	STOL
Roll	$\pm$ 0.4 rad/sec^2	$\pm$ 0.3 rad/sec^2	$\pm$ 7 deg	$\pm$ 5 deg
Pitch	$\pm$ 0.3 rad/sec^2	$\pm$ 0.3 rad/sec^2	$\pm$ 5 deg	$\pm$ 5 deg
Yaw	$\pm$ 0.2 rad/sec^2	$\pm$ 0.15 rad/sec^2	$\pm$ 3 deg	$\pm$ 2 deg

Simultaneous maneuver control power need not be greater than 100% - 30% - 30%.

1.1.2 Flight Path Control Power (SL to 1000 ft., 90°F).

1.1.2.1 VTOL (0-40 kt TAS and zero rate of descent)

At applicable aircraft weights and at the conditions for 50% of the maximum attitude control power of critical axis specified in para. 1.1.1 it shall be possible to produce the following incremental accelerations for height control:

Level 1: (a) In free air $\pm$ 0.1g

(b) With wheels just clear of the ground

+0.10g, ±0.05g

Level 2: (a) In free air -0.1g, +0.05g

(b) With wheels just clear of the ground

-0.10g, +0.00g

It shall also be possible to produce the following horizontal incremental acceleration, but not simultaneously with height control.

Level 1: $\pm$ 0.15g

Level 2: $\pm$ 0.10g

At applicable aircraft weights it shall be possible to produce the following stabilized thrust-weight ratios without attitude control inputs.

Level 1: $\frac{F}{W}$ = 1.05 in free air (Takeoff power rating)

Level 2: $\frac{F}{W}$ = 1.03 in free air (Emergency power rating)

With the most critical engine failed, Level 2 performance shall be achieved at a weight which shall include 2500 lbs. payload and fuel sufficient for 2 VTOL test missions (figure 1a).

1.1.2.2 VTOL and STOL Approach (40 kts. to V_{CON})

At the applicable landing weight the aircraft shall be capable of making an approach at 1000 FPM rate of descent while simultaneously decelerating at 0.08g along the flight path.

It shall be possible to produce the following incremental normal accelerations by rotation alone (angle of attach change and constant thrust) in less than 1.5 seconds at the STOL landing approach airspeed where reasonable rotation (angle of attack changes) will produce at least 0.15g's.

Level 1: $\pm$ 0.1g

Level 2: $\pm$ 0.05g

It shall be possible to produce the following normal accelerations in at least 0.5 seconds for flight path, flare, or touchdown control by either thrust changes or combined thrust changes and rotation at STOL landing approach speeds below which 0.15g's can be produced by reasonable rotation alone.

Level 1: $\pm$ 0.1g

Level 2: $\pm$ 0.05g

1.1.3 VTOL and STOL Low Speed Control System Lags (S.L. to 1000 ft. 90°).

The effective time constant (time to 63% of the final value) for attitude control moments and for flight path control forces shall not exceed the levels given in the following table.

	Level 1	Level 2
Attitude Control Moments	0.2 sec	0.3 sec
Flight Path Control Forces	0.3 sec	0.5 sec

With a step-type input at the pilot's control the commanded control moment or force shall be applied within the following:

Level 1: 0.3 seconds for 0.5 inches of pilot's control

0.5 seconds for full pilot's control

Level 2: 0.5 seconds for full pilot's control

1.1.4 Stability (S.L. to 1000 ft., 90°F)

1.1.4.1 Hovering

The frequency and damping of the airframe/control system dynamics, in the hovering condition, shall be within the following limits for the three rotary axes:

Level 1: Optimum damping and frequency zone established from the Ames six-degree-of-freedom moving base simulator (figure 2).

Level 2: The zone given in figure 2. The boundary of this zone corresponds to a damping factor of 0.166 for values of ω_n above 1 rad sec.

1.1.4.2 Low Speed

Level 1: The dominant oscillatory modes shall be maintained as close as possible to the optimum zone specified in section 1.1.4.1 while maintaining other oscillatory modes damped. Aperiodic modes, if unstable, shall have a time to double amplitude of greater than 20 sec.

Level 2: The dominant oscillatory modes shall be maintained within the Level 2 zone given in figure 2. Other oscillatory modes may be unstable provided their frequency is less than 0.84 rad/sec and their time to double amplitude greater than 12 sec. Aperiodic modes, if unstable, shall have a time to double amplitude of greater than 12 sec.

1.1.4.3 Cruise

The aircraft as configured for cruise flight shall be statically stable at all gross weights with a stability margin of 0.05 at the critical center of gravity without stability augmentation.

1.2 STOL Takeoff Performance

The climbout gradient in the takeoff configuration, at takeoff gross weight, with gear down and most critical power plant failed at lift off shall be positive and the aircraft will continue to accelerate.

During takeoff wing lift shall not exceed 0.8 CL_{MAX}. No catapults or arresting gear will be utilized. The rolling coefficient of friction will be 0.03. (for calculations)

1.3 Conversion Requirements (STOL and VTOL)

It must be possible to stop and reverse the conversion procedure quickly and safely without undue complicated operation of the powered lift controls.

The maximum speed in the powered-lift configuration shall be at least 20% greater than the power-off stall speed in the converted configuration for level 1 operation and the speed in the powered lift configuration shall be at least 10% greater than the power off stall speed for the level 2 operation.

2.0 Mission

2.1 Mission Summary

2.1.1 Land Operation -- The VTOL and STOL test missions are described in figure 1.

- Minimum Mission Time - Level 1

VTOL Missions	1/2 hour
STOL Missions	1 hour
Cruise/Endurance Mission	2 hours

- Payload (not including crew) 2500 lbs (minimum)
  50 cu. ft.

2.1.2 Shipboard Operation -- The aircraft shall be capable of operating from the deck of a naval aircraft carrier.

2.2 Minimum Cruise Speed

- 300 KEAS at sea level and 0.7 at 25,000 ft.

3.0 General Design Guidelines

3.1 Austerity is to be stressed but not by compromising safety.

3.2 The limit load factor will be no less than +2.5g, -0.5g at design gross weight.

3.3 Sufficient attitude control power will be available to perform research on control requirements. The contractor shall indicate those axes where greater control power than required in section 1.0 would be made available for research purposes.

3.4 New aircraft components will be designed for approximately 500 flight hours.

3.5 Additional Information

- Crew	2 pilots (flyable by one pilot only, or by either pilot)
- Sink rate at touchdown	12 fps at max landing weight, 15 fps desired

- Pressurized cockpit is desired but not required
- Oxygen required
- Cockpit Environmental System — Minimum
- Pilot's Primary Flight Controls — Stick and Pedal
- Ejection System for both pilots
- Maximum possible visibility

3.6 The contractor shall furnish as a minimum:

a. Conceptual design aircraft layout drawings.

b. Mil Std. 1374 Part 1 shall be used to show the empty weight breakdown into the usual structural and system group including additions and deletions to the original aircraft.

c. Low speed performance envelope at design gross weight.

d. Conceptual definition of proposed aircraft low speed control and stabilization system.

e. Control moment coefficients and control power about each axis with all gas generators operating and with most critical gas generator failed.

f. Engine and fan data which were used to calculate mission performance in all flight modes.

4.0 Summary of Costing Information required for the Research and Technology Aircraft

The Cost Breakdown is for a two airplane buy. The Cost Breakdown shall be stated in five pricing elements; engineering labor, manufacturing labor, materials and purchased items, other direct costs, and spares (if any). A listing of Government Furnished Equipment (GFE) assumed in the costing shall be included. It is intended that the costing information shall be complete in that the total costs of the subitems listed in paragraphs 4.1 thru 4.8 shall equal the total costs of the aircraft excluding the GFE items.

4.1 Airframe Design and Modification including:

- Landing Gear
- Subsystem and conventional controls
- Cockpit
- Ejection seats
- Wings
- Fuselage
- Empennage
- Miscellaneous

4.2 Propulsion system including:

- Components in 5.0
- Transmission components
- Transmission subsystem
- Thrust vectoring
- Miscellaneous

4.3 Control System including:

- Fly-by-wire controls
- Augmentation systems
- Miscellaneous

4.4 Propulsion System Testing including:

- Components in 5.0
- Transmission components
- Thrust vectoring

- ° Qualification tests
- ° Aircraft ground tests
- ° Miscellaneous

4.5 Control System Aircraft Testing including:

- ° Component tests
- ° System integration
- ° Aircraft ground tests

4.6 Aircraft Ground Tests

- ° Excluding aircraft ground tests in sections 4.4 and 4.5

4.7 Ejection Seat Tests

4.8 Flight Tests

- ° Contractor Flight Test

4.9 Government Furnished Equipment including:

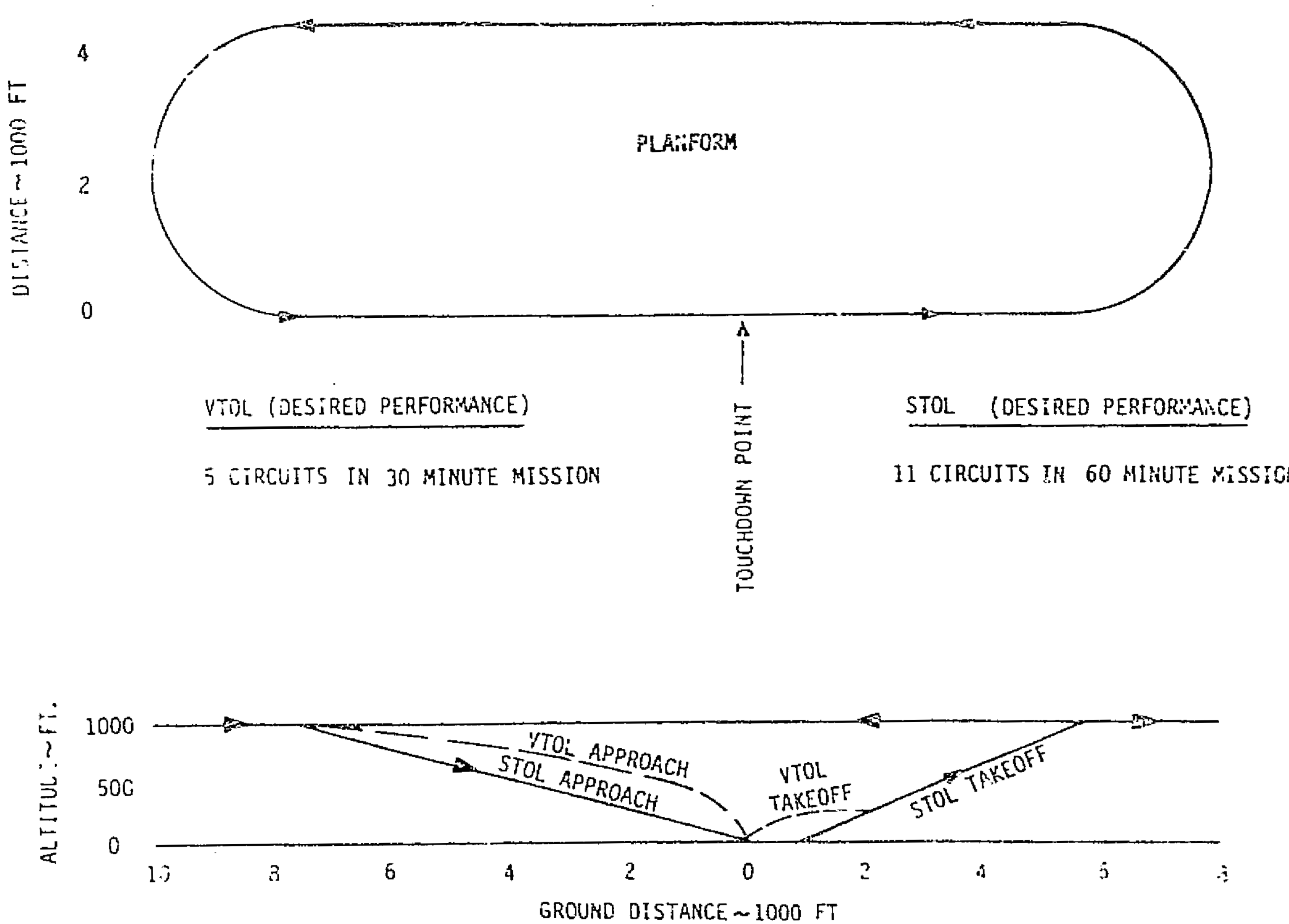
- ° NA265-40 basic airframe
- ° Airframe components
- ° Fans
- ° Engines
- ° Research instrumentation
- ° Miscellaneous

5.0 Summary of the Costing Information required for the high risk propulsion components

The costs for each component shall be stated in four pricing elements; engineering labor, manufacturing labor, material and purchased items, and other direct costs. For each of the pricing elements, the component costs shall be stated for the following categories: data base requirements (effort required to accumulate required data before detail design including data search, analysis, tests, etc.), design and manufacture, component testing, and unit qualification testing. Thus each component costs shall be stated in a four by four matrix.

ORIGINAL PAGE IS OF POOR QUALITY

FIGURE 1 - TYPICAL VTOL AND STOL TEST MISSIONS



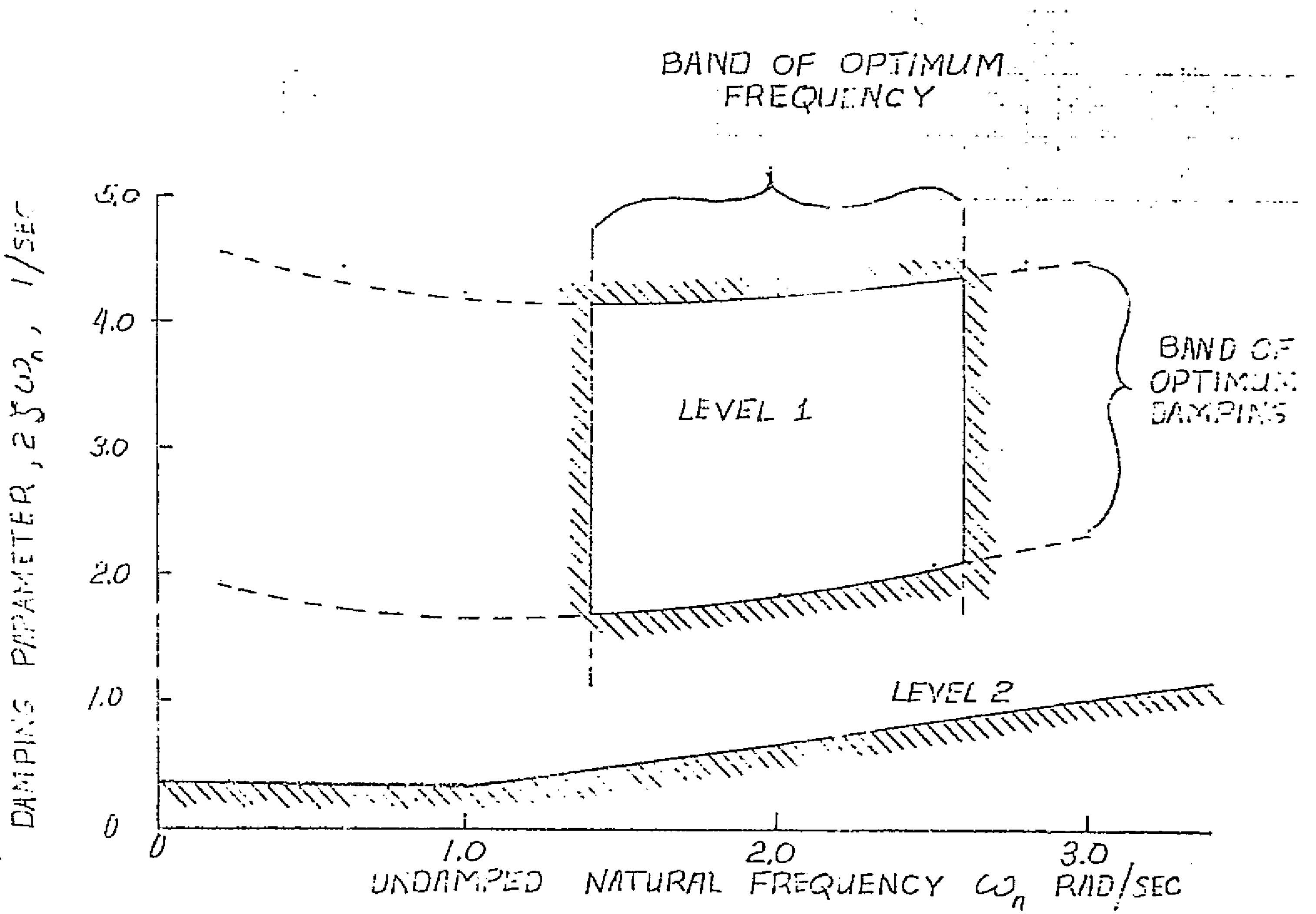


FIGURE 2. DYNAMIC STABILITY CRITERIA

APPENDIX B

TURBOTIP RTA PROPULSION SYSTEM DATA PACKAGE

(3) YJ97/(3) LCF459 FANS

I. PERFORMANCE DATA

(3) YJ97-GE-100/(3) LCF 459 FANS
NORMAL POWERED LIFT
INSTALLED VTOL PROPULSION SYSTEM PERFORMANCE
SLS, 89.8°F

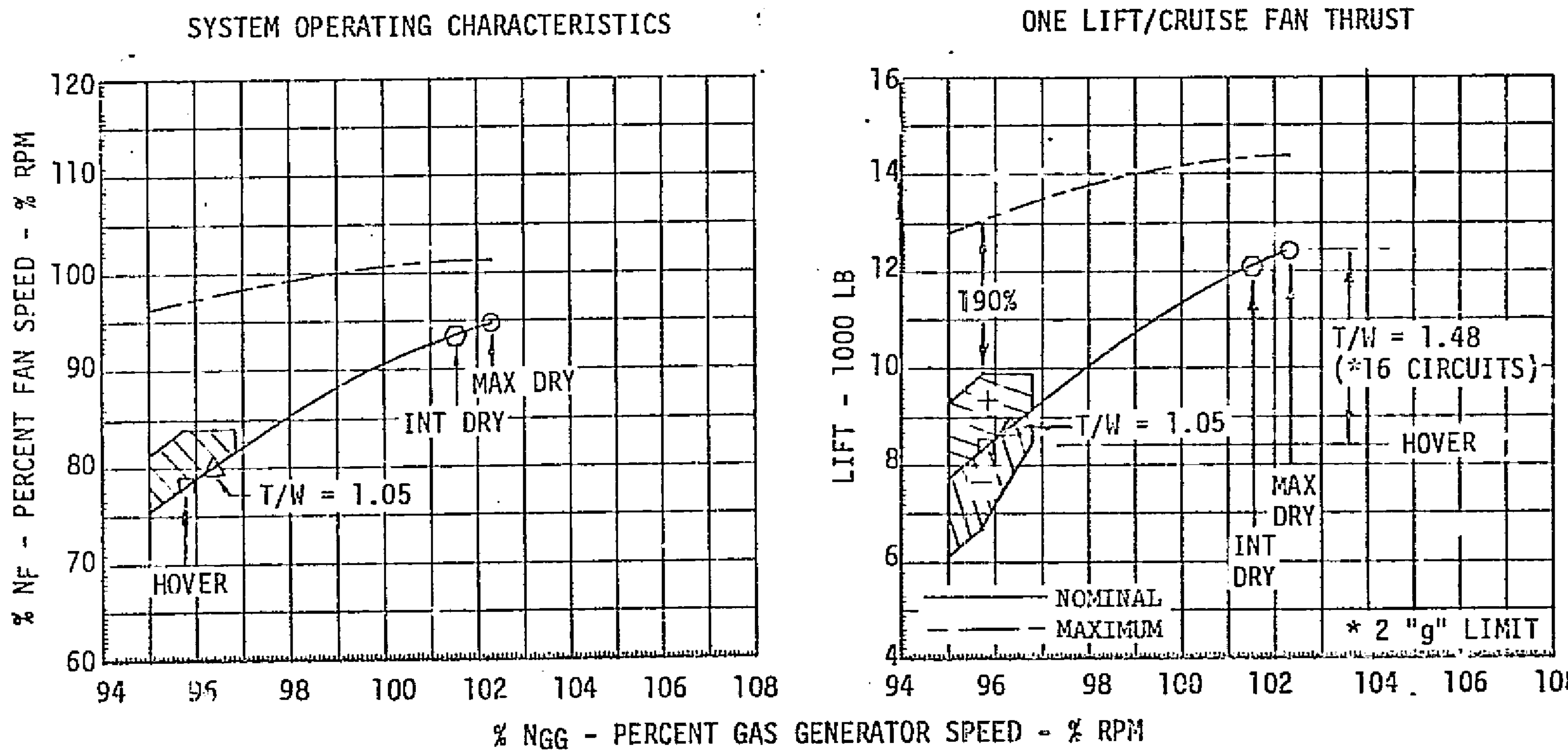
Parameters		Max Dry	Int. Dry	Reduced Power	
		POWER RATINGS			
Gas Generator					
RPM	% N_{GG}	102.3	101.5	98	95
TIT	°F				
EGT	°F	1452	1431	1284	1153
EGP	psia	51.19	50.42	44.39	38.25
W_g	lb/sec	66.41	65.77	60.31	54.04
W_f	lb/hr	4754	4627	3735	2948
W_a	lb/sec	65.42	64.81	59.57	53.49
H_2O/W_a	%	0.0	0.0	0.0	0.0
H_2O	LB/SEC	0.0	0.0	0.0	0.0
Fans					
RPM (Fwd)	% N_F	94.7	93.7	85.4	75.4
RPM (L/C)	% N_F	94.7	93.7	85.4	75.4
W_a (Fwd)	lb/sec	571.84	565.39	516.56	451.41
W_a (L/C)	lb/sec	571.84	565.39	516.56	451.41
F_N (Fwd)	lb	12560	12276	10121	7855
F_N (L/C)	lb	12436	12154	10021	7777
Total Propulsion System					
Lift	lb	37432	36584	30163	23409
W_f	lb/hr	14262	13881	11205	8844
Lift SFC	lb/hr-lb	.381	.379	.372	.378
Airflow	lb/sec	1911.8	1890.6	1728.4	1514.7

(3) YJ97-GE-100/(3) LCF459 FANS
ENGINE OUT LIFT RATINGS
INSTALLED VTOL PROPULSION SYSTEM PERFORMANCE
SLS, 89.8°F

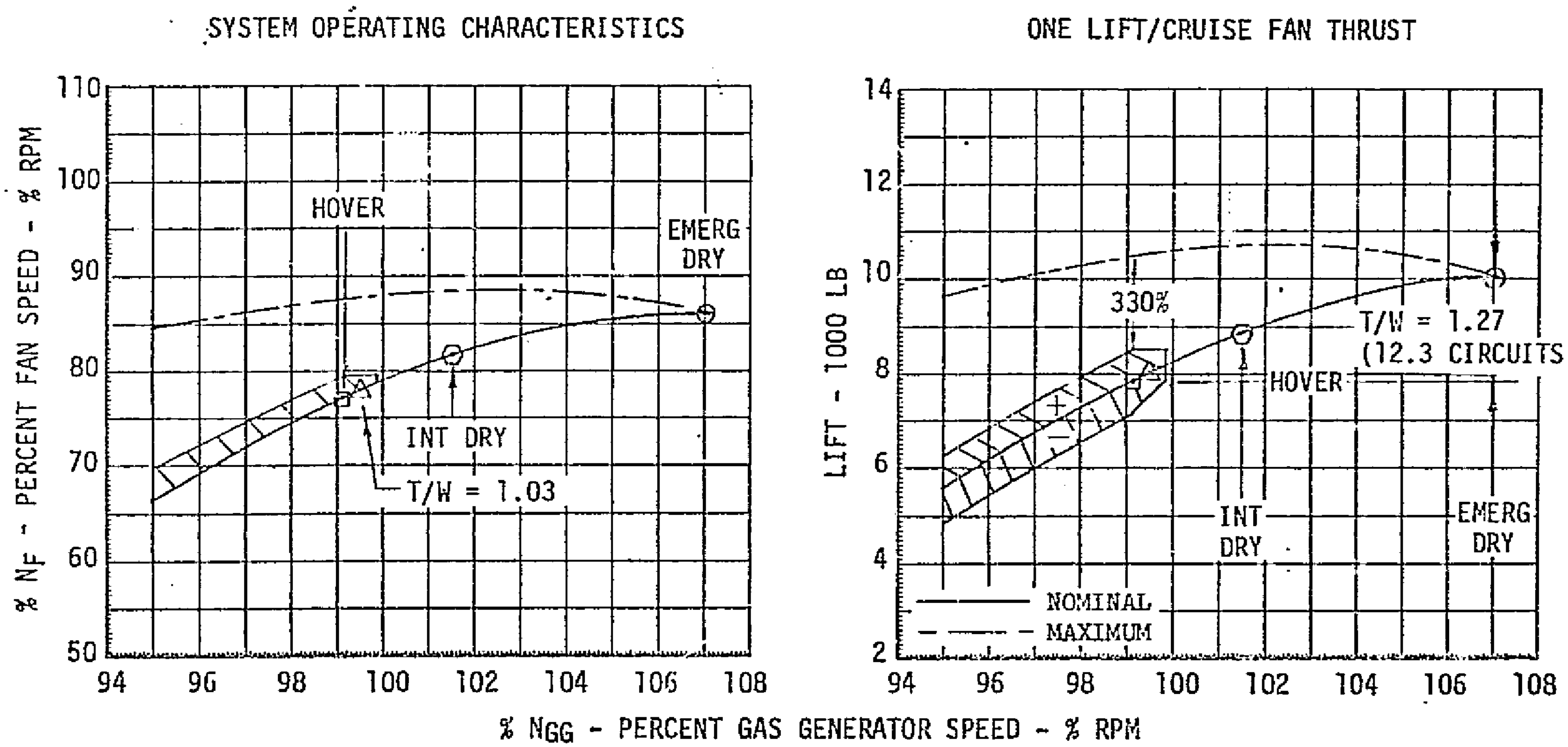
Parameters		Power Ratings			
		Emergency Dry	Intermediate Dry	Reduced Power	
Gas Generator					
RPM	% N_{GG}	107.1	101.5	98	95
TIT	°F				
EGT	°F	1600	1418	1274	1144
EGP	psia	54.47	49.79	43.82	37.71
W_g	lb/sec	68.08	65.70	60.16	53.83
W_f	lb/hr	5468	4570	3690	2905
W_a	lb/sec	66.90	64.76	59.44	53.29
H_2O/W_a	%	0.0	0.0	0.0	0.0
H_2O	lb/sec	0.0	0.0	0.0	0.0
Fans					
RPM (Fwd)	% N_F	86.1	81.6	74.5	66.5
RPM (L/C)	% N_F	86.1	81.6	74.5	66.5
W_a (Fwd)	lb/sec	520.82	496.2	453.2	401.8
W_a (L/C)	lb/sec	520.82	496.2	453.2	401.8
F_N (Fwd)	lb	10,156	8955	7336	5632
F_N (L/C)	lb	10,055	8866	7263	5576
Total Propulsion System					
Lift	lb	30,266	26,687	21,862	16,784
W_f	lb/hr	10,936	9140	7380	5810
Lift SFC	lb/hr-lb	.361	.342	.338	.346
Airflow	lb/sec	1696.3	1682.9	1537.9	1365.3

TURBOTIP RTA
INSTALLED CONTROL PERFORMANCE FOR THREE ENGINE VTO OPERATION
(3) YJ97-GE-100/(3) LCF459 FANS
SEA LEVEL, 89.8°F

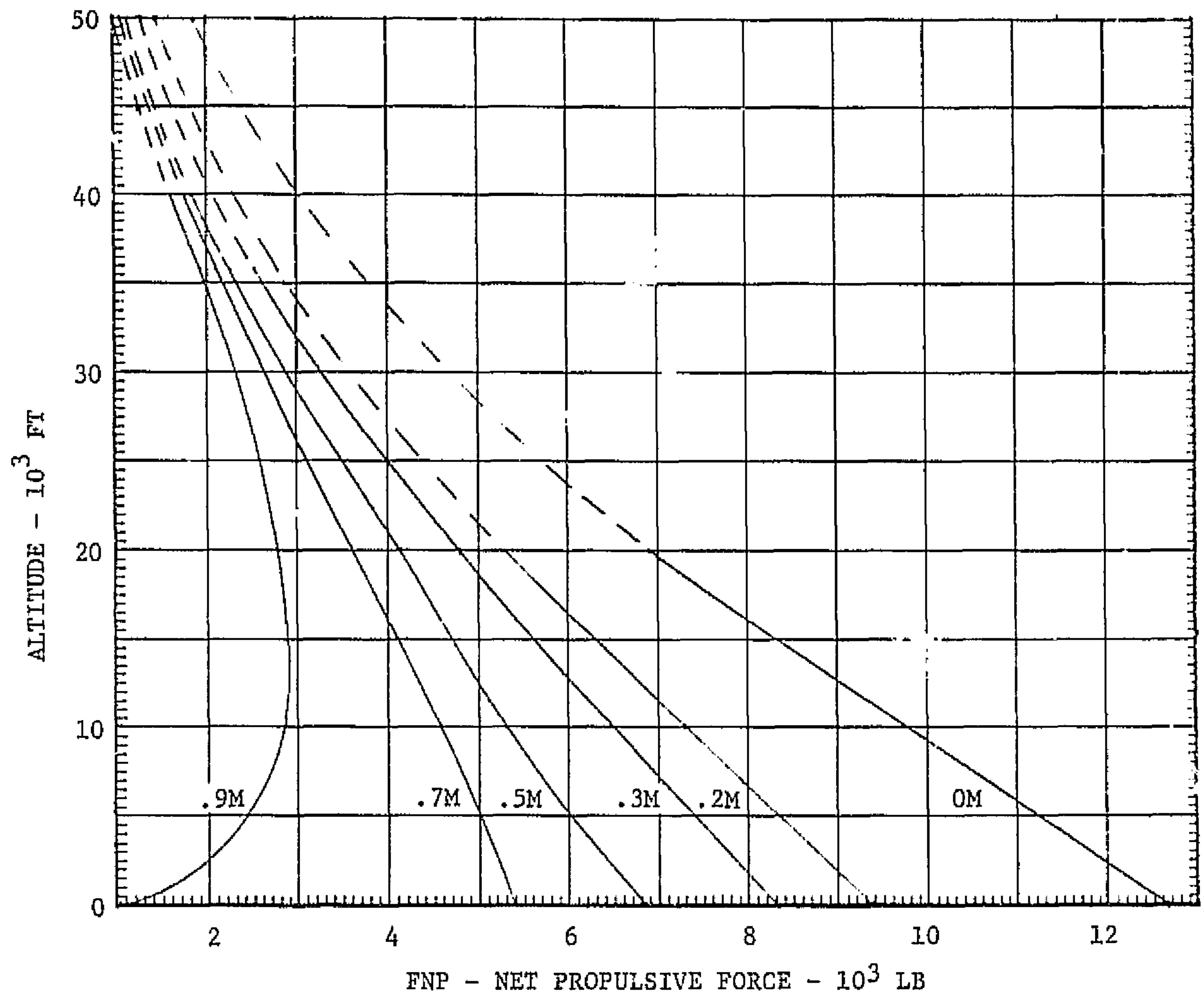
GROSS WEIGHT = 25286 LB (5 CIRCUITS FUEL)



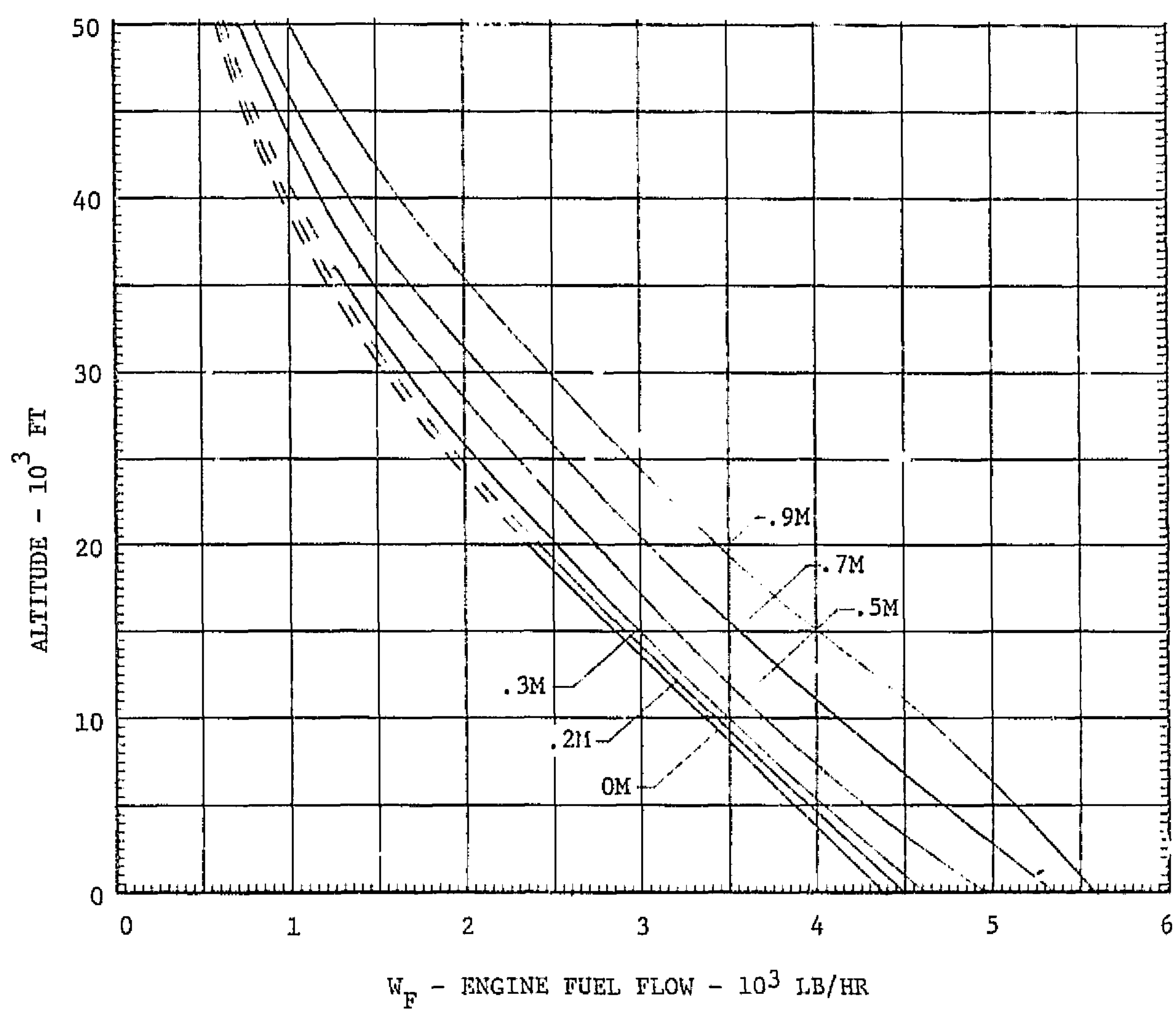
TURBOTIP RTA
INSTALLED CONTROL PERFORMANCE FOR ENGINE OUT VTO OPERATION
(3) YJ97-GE-100/(3) LCF459 FAN SYSTEM
SEA LEVEL, 89.8°F
GROSS WEIGHT = 23660 LB (2 CIRCUITS FUEL)



NET PROPULSIVE FORCE PER L/C FAN
(3) YJ97/(3) LCF459 FAN SYSTEM
INTERMEDIATE POWER, STD DAY
(ONE ENG/ONE FAN)



FUEL FLOW PER GAS GENERATOR
(3) YJ97/(3) LCF 459 FAN SYSTEM
INTERMEDIATE POWER, STD DAY
(ONE ENG/ONE FAN)

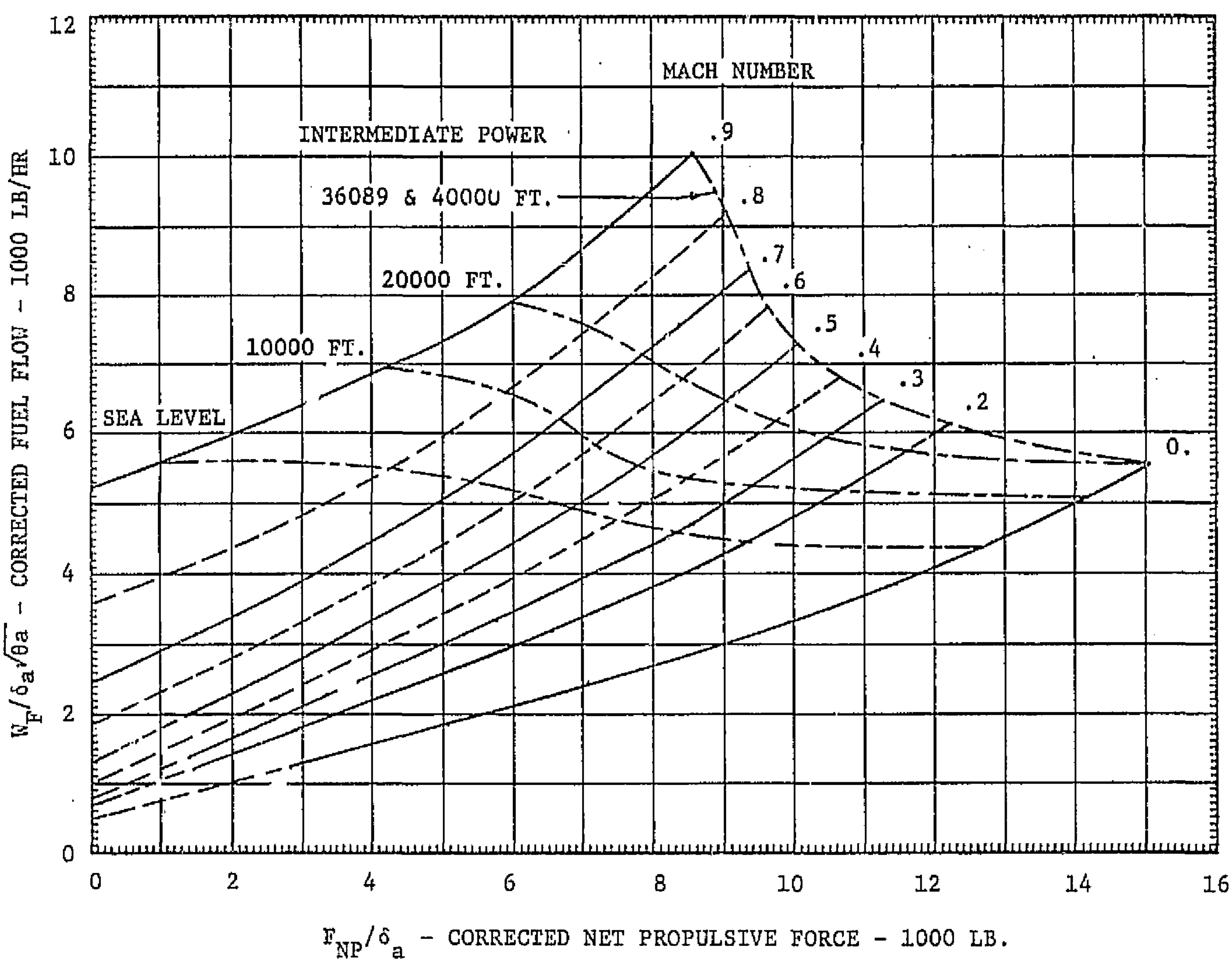


REDUCED POWER PERFORMANCE

(3) YJ97-GE-100/(3) LCF 459 FAN SYSTEM

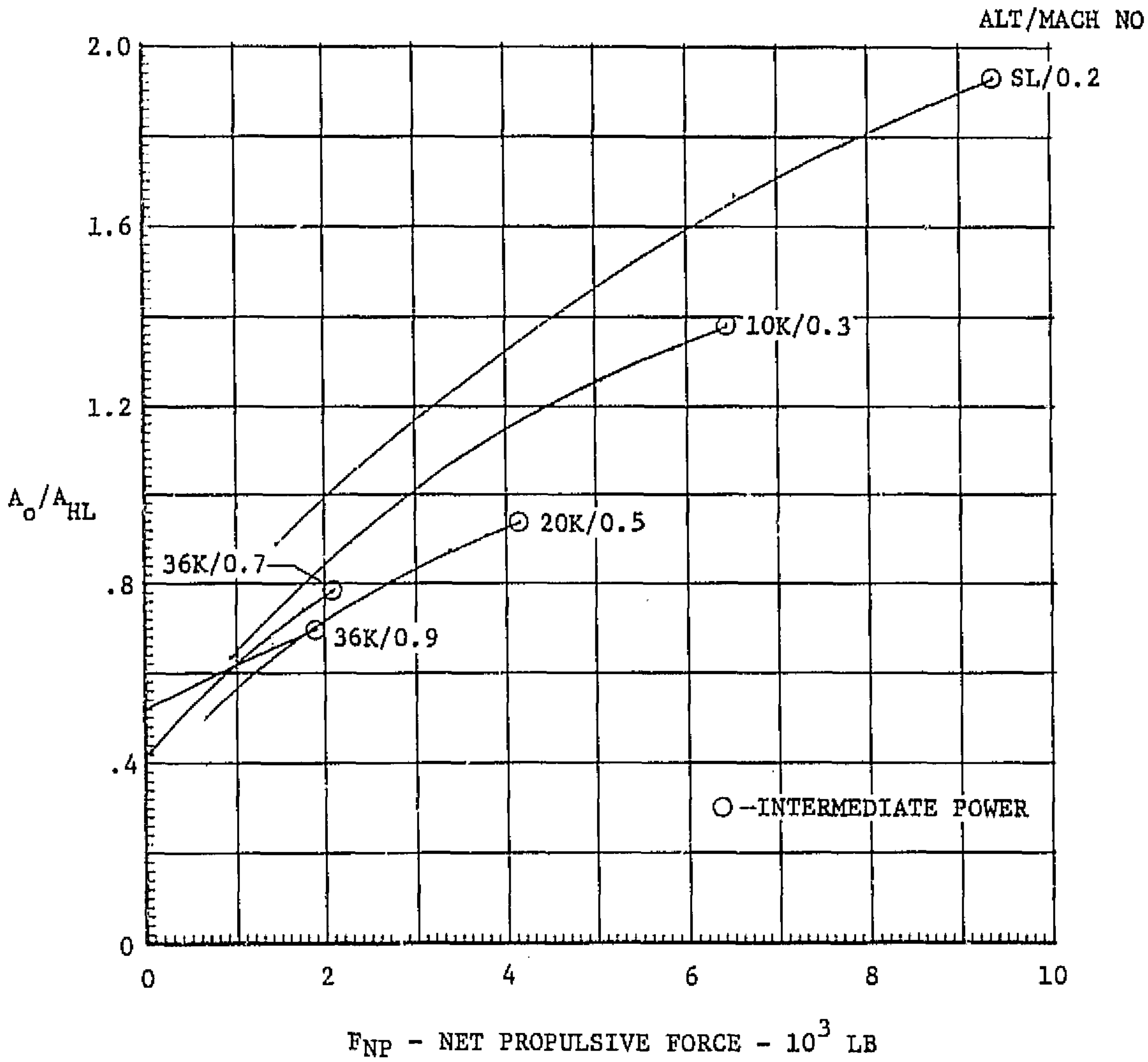
(ONE ENGINE/ONE FAN)

STANDARD DAY



C.3

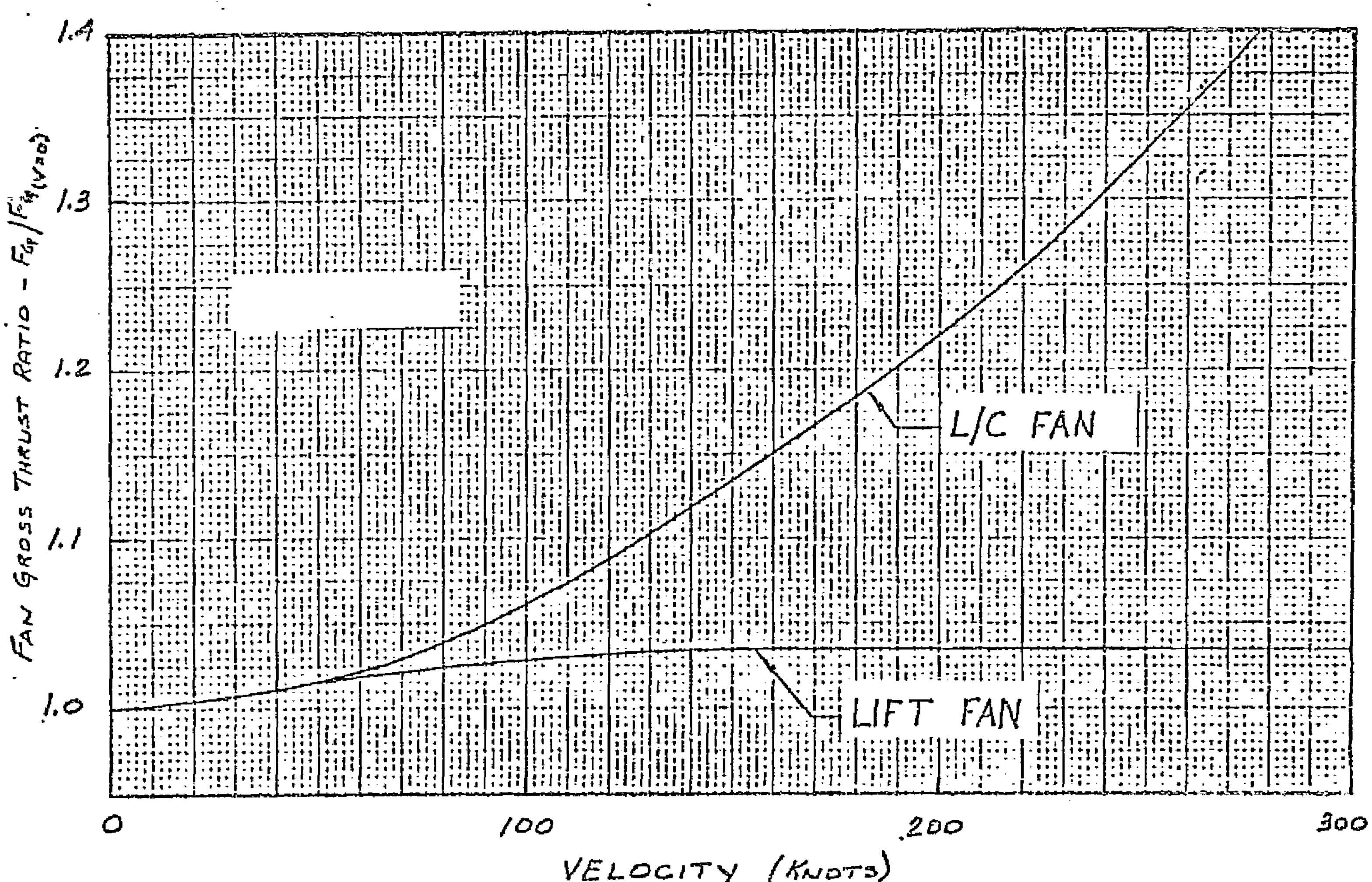
LIFT/CRUISE FAN INLET MASS FLOW RATIO
(3) YJ97/(3) LCF459 FANS



GAS RTA

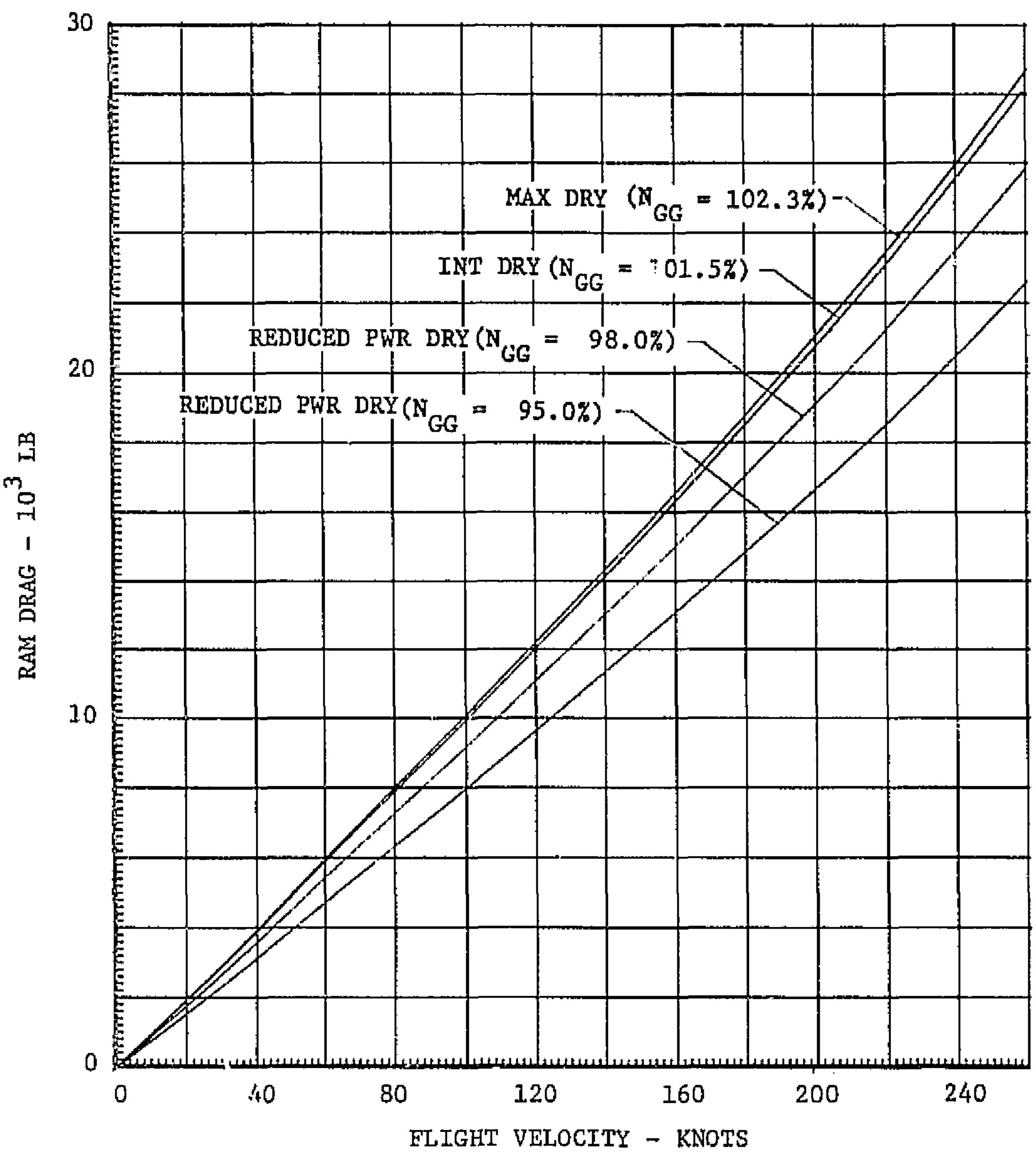
GROSS THRUST PERFORMANCE DURING POWERED LIFT OPERATION

SEA LEVEL, 89.8 °F



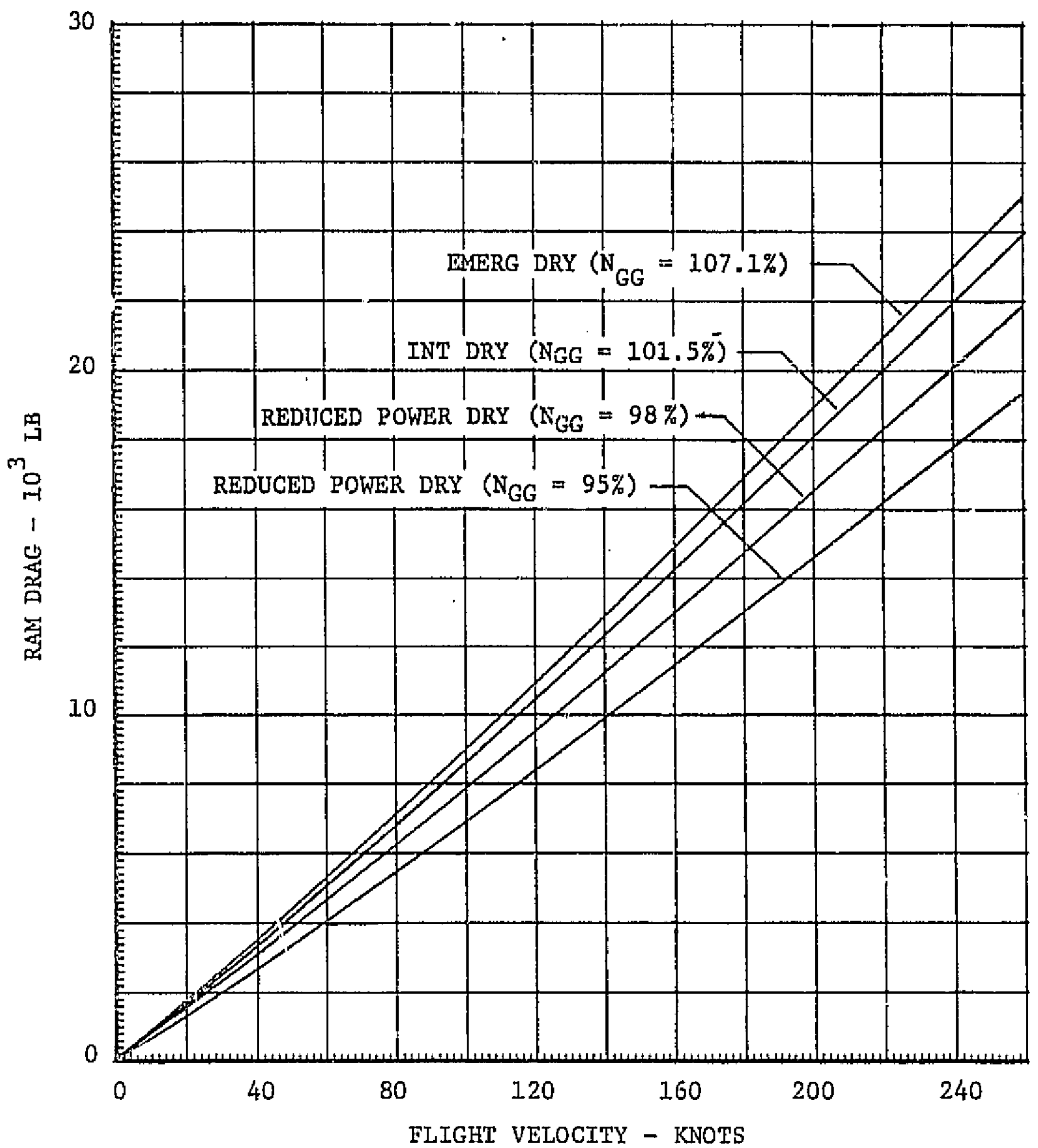
TOTAL RAM DRAG DURING POWERED LIFT FLIGHT
(3) YJ97/(3) LCF 459 FANS
S.L., 89.8°F

NORMAL OPERATION



TOTAL RAM DRAG DURING POWERED LIFT FLIGHT
(3) YJ97/(3) LCF 459 FANS
S.L., 89.8°F

ENGINE OUT OPERATION

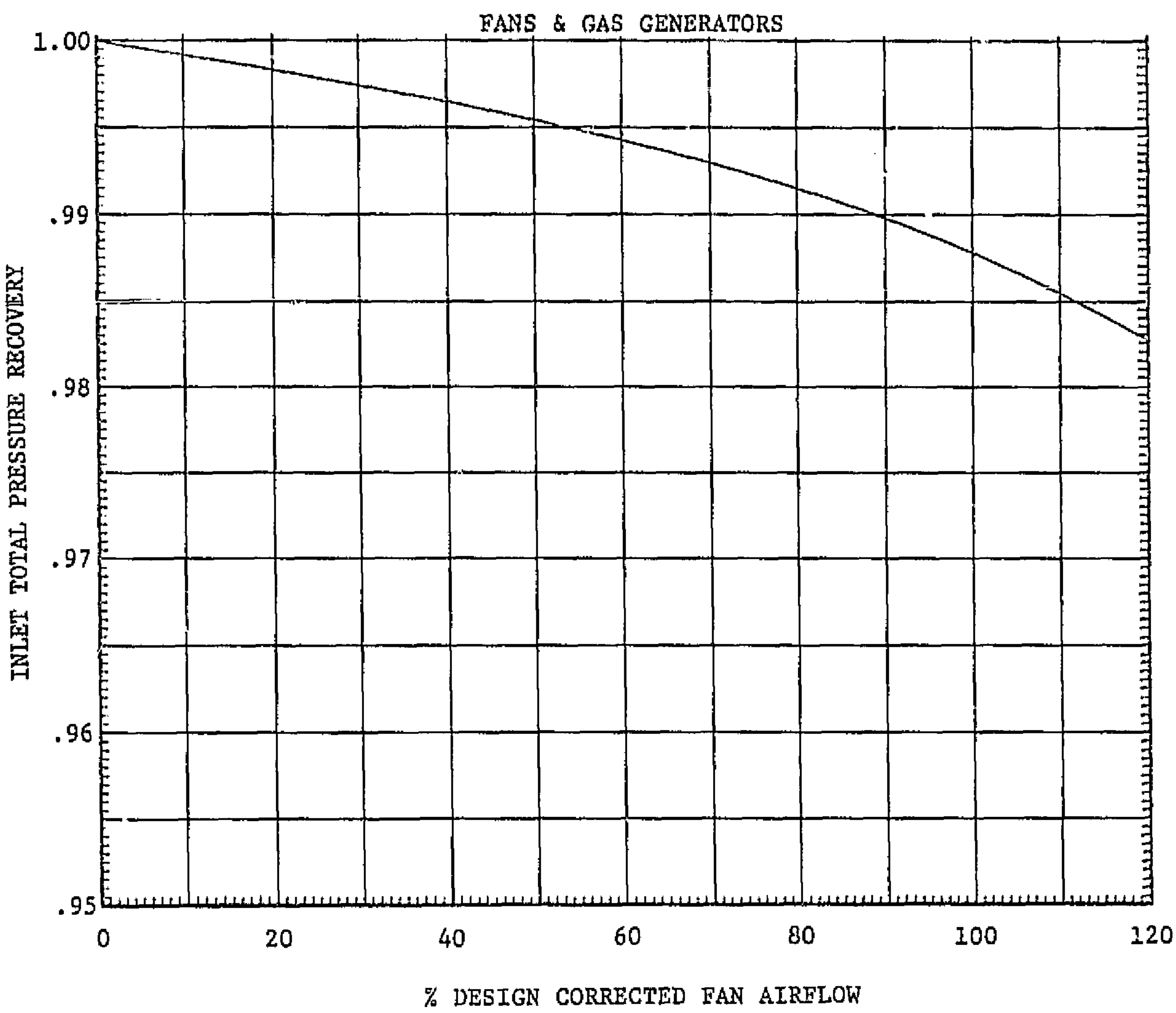


II. INSTALLATION FACTORS

GAS RTA
PROPULSION SYSTEM INSTALLATION FACTORS
VTO OPERATION

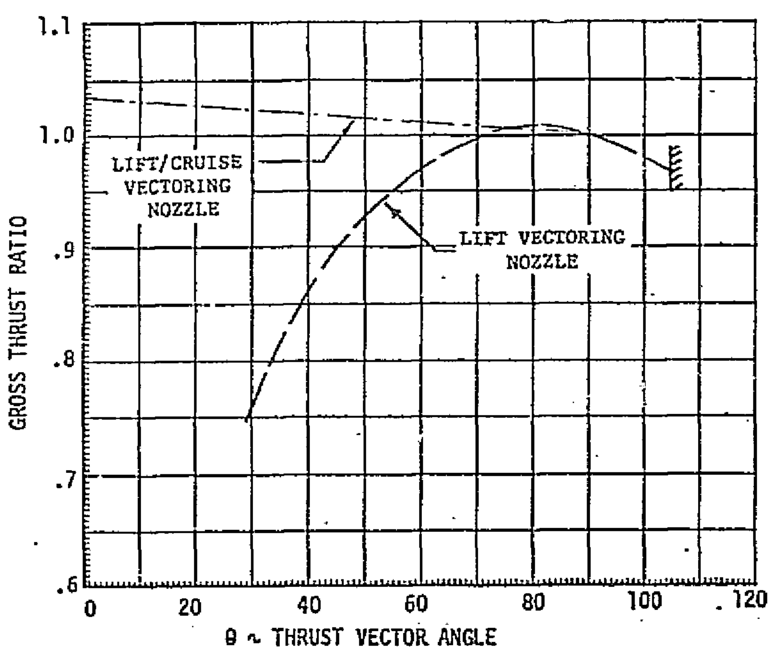
Component Installation Loss	Installation Factor
GAS GENERATORS	
Inlet Pressure Recovery	Page 13
Compressor Bleed	1/2%
Horsepower Extraction	25 HP/Eng.
L/C FANS	
Inlet Pressure Recovery	Page 13
Horsepower Extraction	50 HP/Fan
Nozzle Thrust Coefficient	.940
NOSE FAN	
Inlet Pressure Recovery	.988
Nozzle Thrust Coefficient	.950
INTERCONNECTING DUCTING	
L/C Fan Duct Pressure Loss	10.3%
Nose Fan Duct Pressure Loss	10.3%
Scroll Pressure Loss	5%
ADDITIONAL PERFORMANCE ALLOWANCES	
Thrust Derate	3%

GAS RTA
ESTIMATED TOTAL PRESSURE RECOVERY
S.L.S., 89.8°F

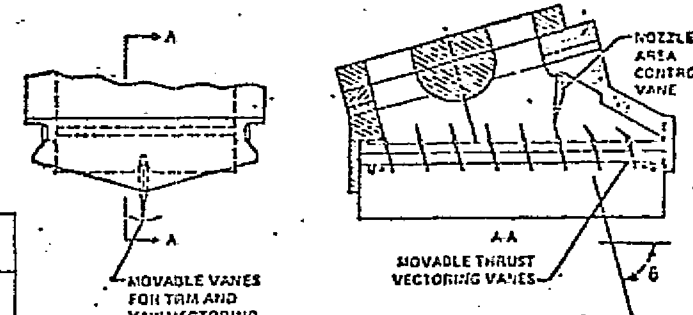


VECTORING SYSTEM PERFORMANCE

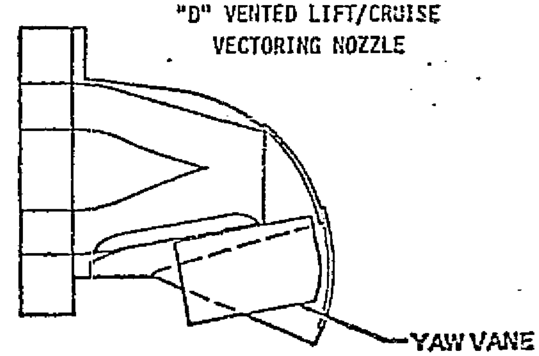
GROSS THRUST VARIATION
DURING VECTORING



INTERNAL CASCADE LIFT VECTORING NOZZLE



"D" VENTED LIFT/CRUISE
VECTORING NOZZLE



90° VECTOR POSITION

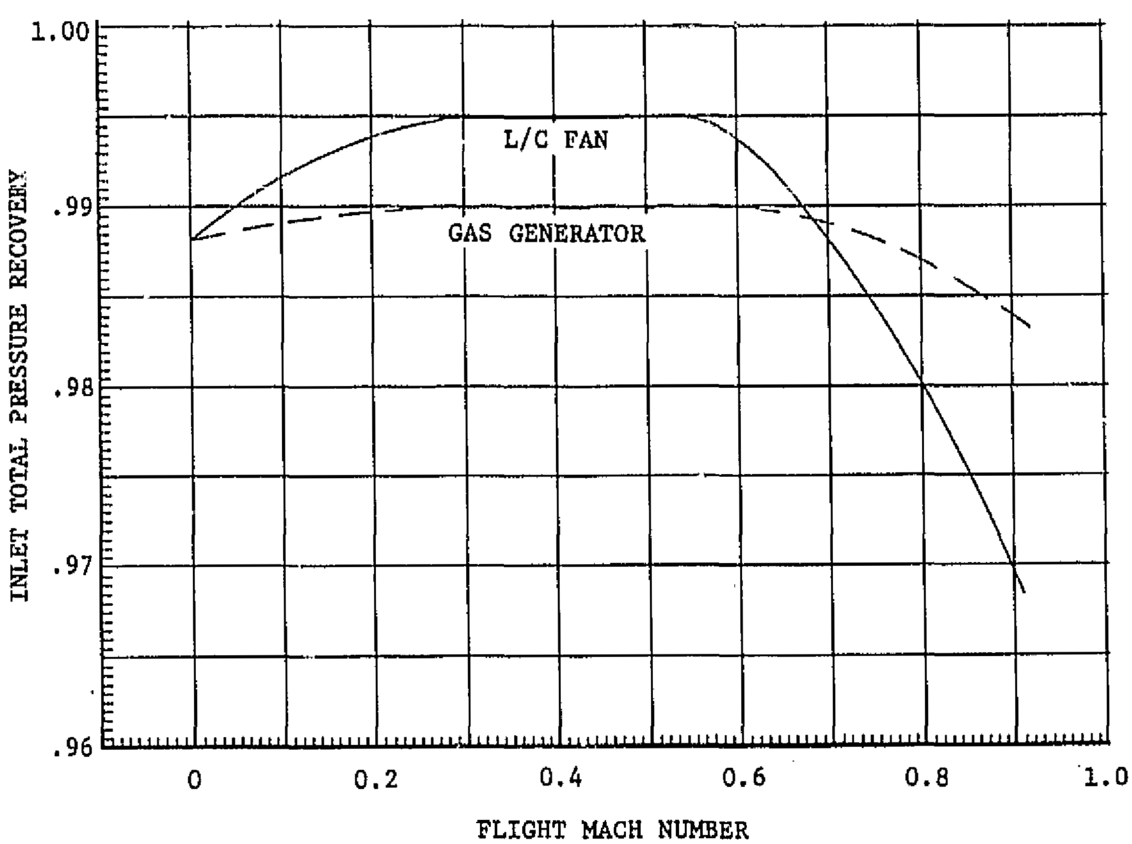
GAS RTA
PROPULSION SYSTEM INSTALLATION FACTORS
CONVENTIONAL FLIGHT

Component Installation Loss	Nominal Installation Factor
GAS GENERATORS	
Inlet Pressure Recovery	(a)
Compressor Bleed	1%
Horsepower Extraction	25 HP/Eng.
L/C FANS	
Inlet Pressure Recovery	(a)
Horsepower Extraction	25 HP/Fan
Nozzle Thrust Coefficient	.98
INTERCONNECTING DUCTING	
L/C Fan Duct Pressure Loss	4.2%
Nose Fan Duct Pressure Loss	—
Scroll Pressure Loss	5%
ADDITIONAL PERFORMANCE ALLOWANCES	
Thrust Derate	3%
Engine Bay Ventilation and ECS Drags	(b)

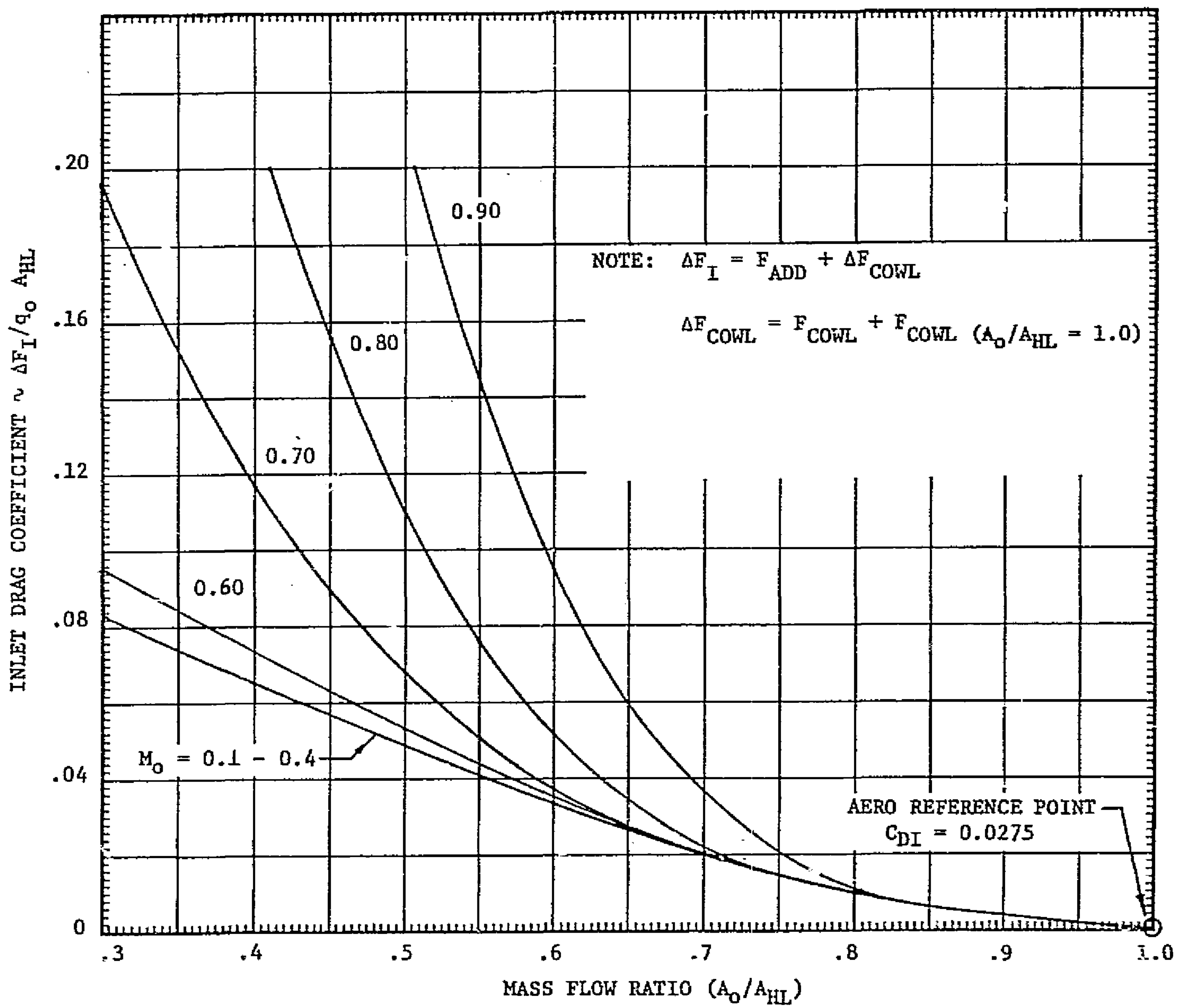
Notes: (a) During cruise, fan and gas generator recoveries vary per Page 16

(b) Ten percent of intermediate power gas generator ram drag at all Mach/altitude/power settings.

GAS RTA
ESTIMATED INLET TOTAL PRESSURE RECOVERY
CONVENTIONAL FLIGHT



ESTIMATED INLET DRAG CHARACTERISTICS

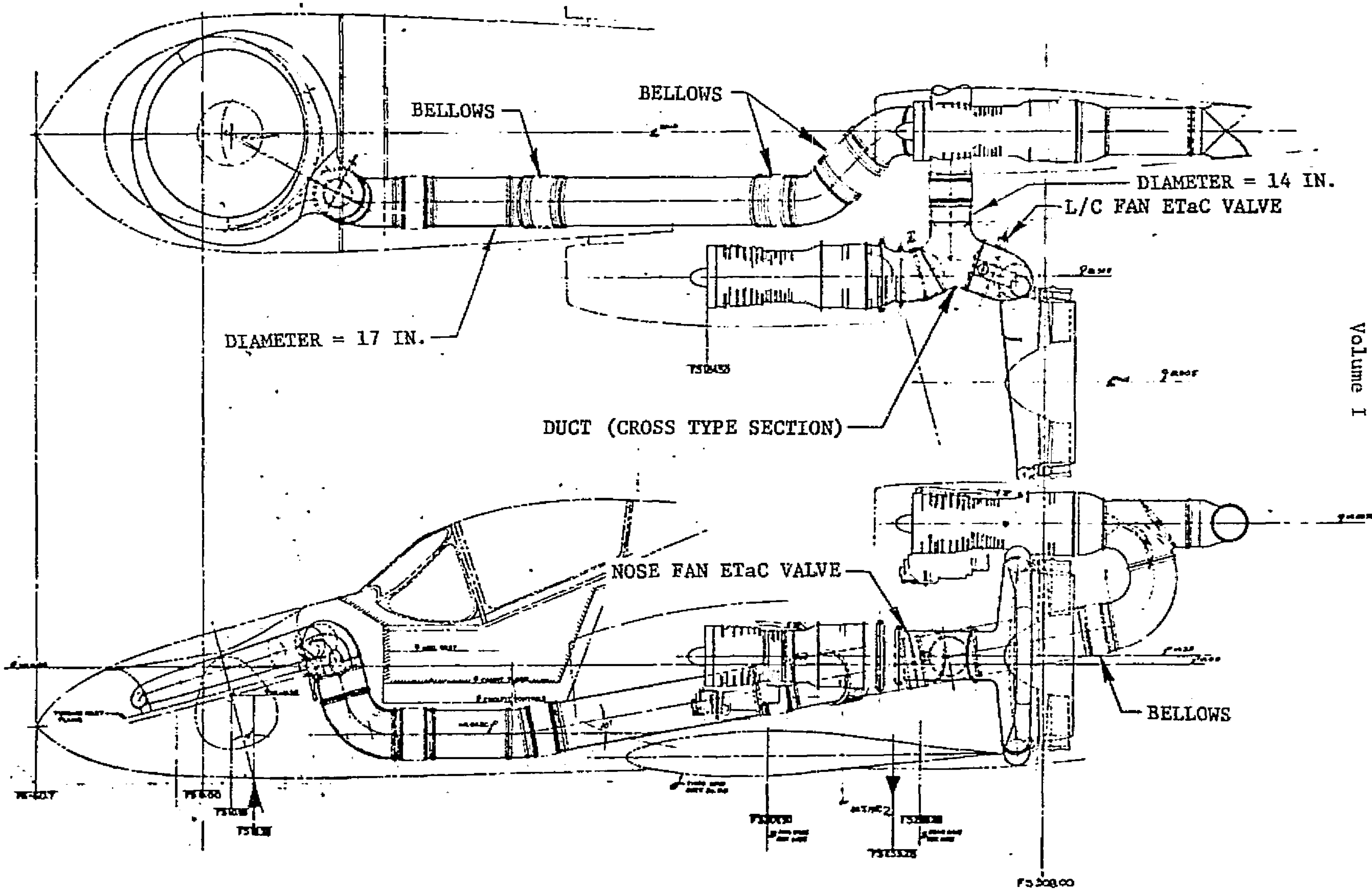


NOZZLE DRAG

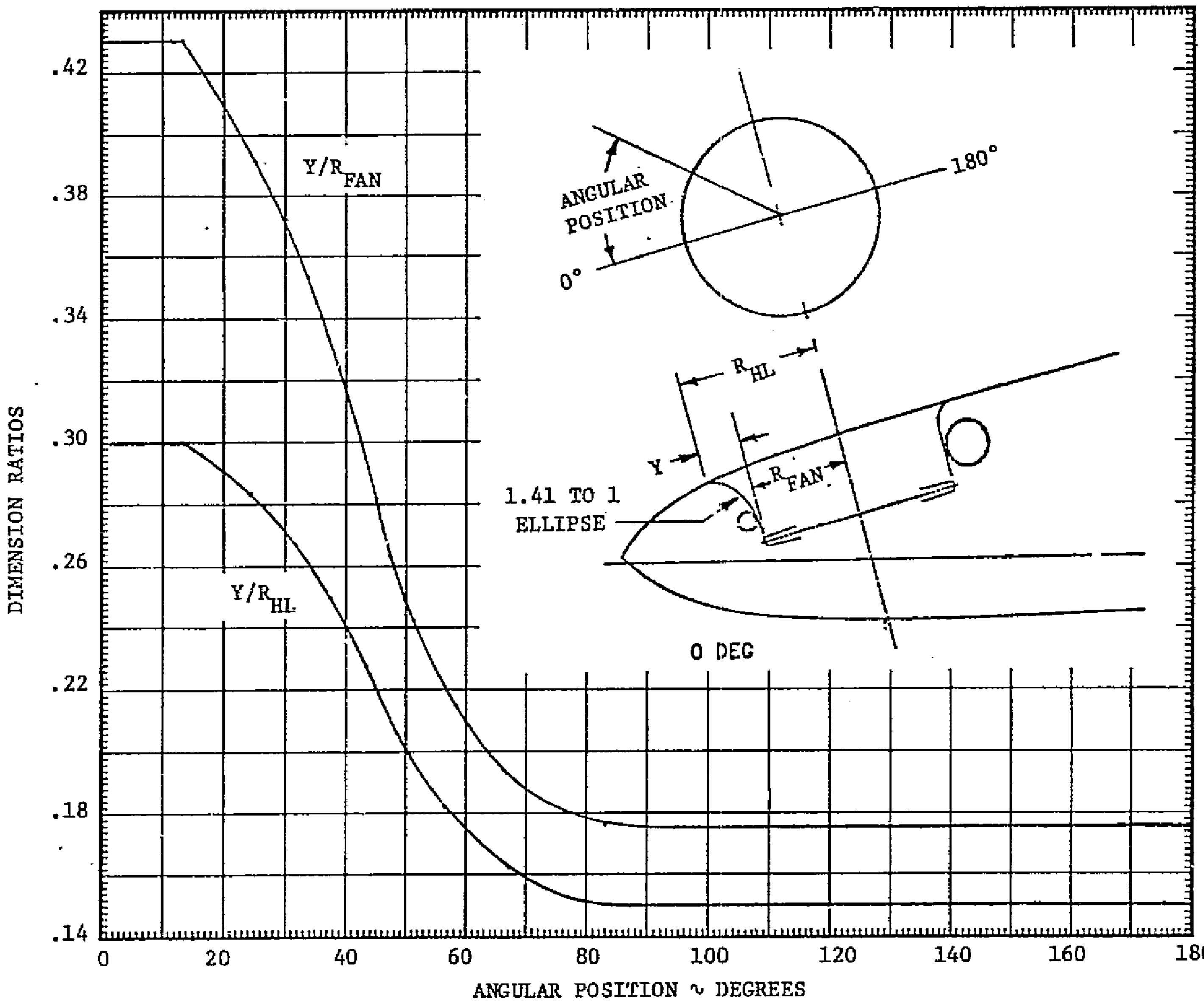
NO NOZZLE DRAG INCLUDED IN NPF

III. PHYSICAL DIMENSIONS AND WEIGHTS

PROPULSION SYSTEM INSTALLATION
M-260 RTA

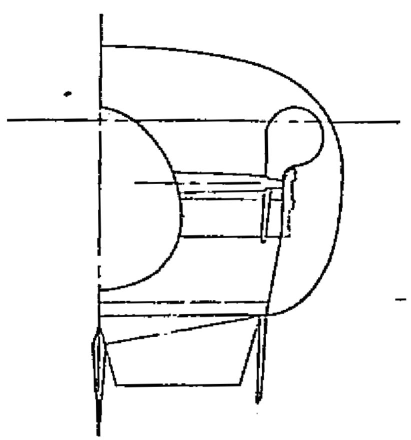


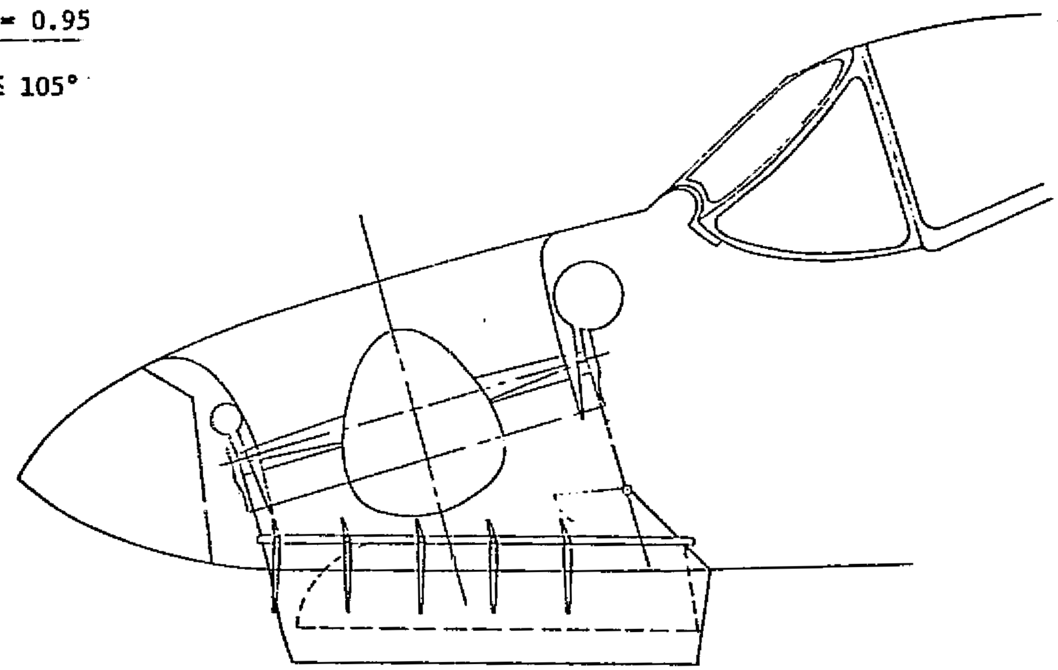
LIFT FAN INLET DEFINITION



TURBOTIP RTA
NOSE FAN UNIT DESIGN GUIDELINES

- <u>INTERNAL GEOMETRY</u>
  - INLET LIP CONTOUR = 1.41:1 ELLIPSE
  - CONTRACTION RATIO = 1.84
- <u>INLET AREAS</u>
  - A_{HL} = 4185 IN.^{2}
  - A_{TH} = 2278 IN.^{2}
- <u>INSTALLED PERFORMANCE</u>
  - INLET RECOVERY $\geq$ .988
  - NOZZLE VELOCITY COEFFICIENT = 0.95
- <u>VECTORING REQUIREMENTS</u>
  - ARTICULATING VANES: $40° \leq \theta \leq 105°$
  - YAW VANES: $\pm 16°$





LIFT CRUISE INLET DESIGN GUIDELINES

o INTERNAL GEOMETRY

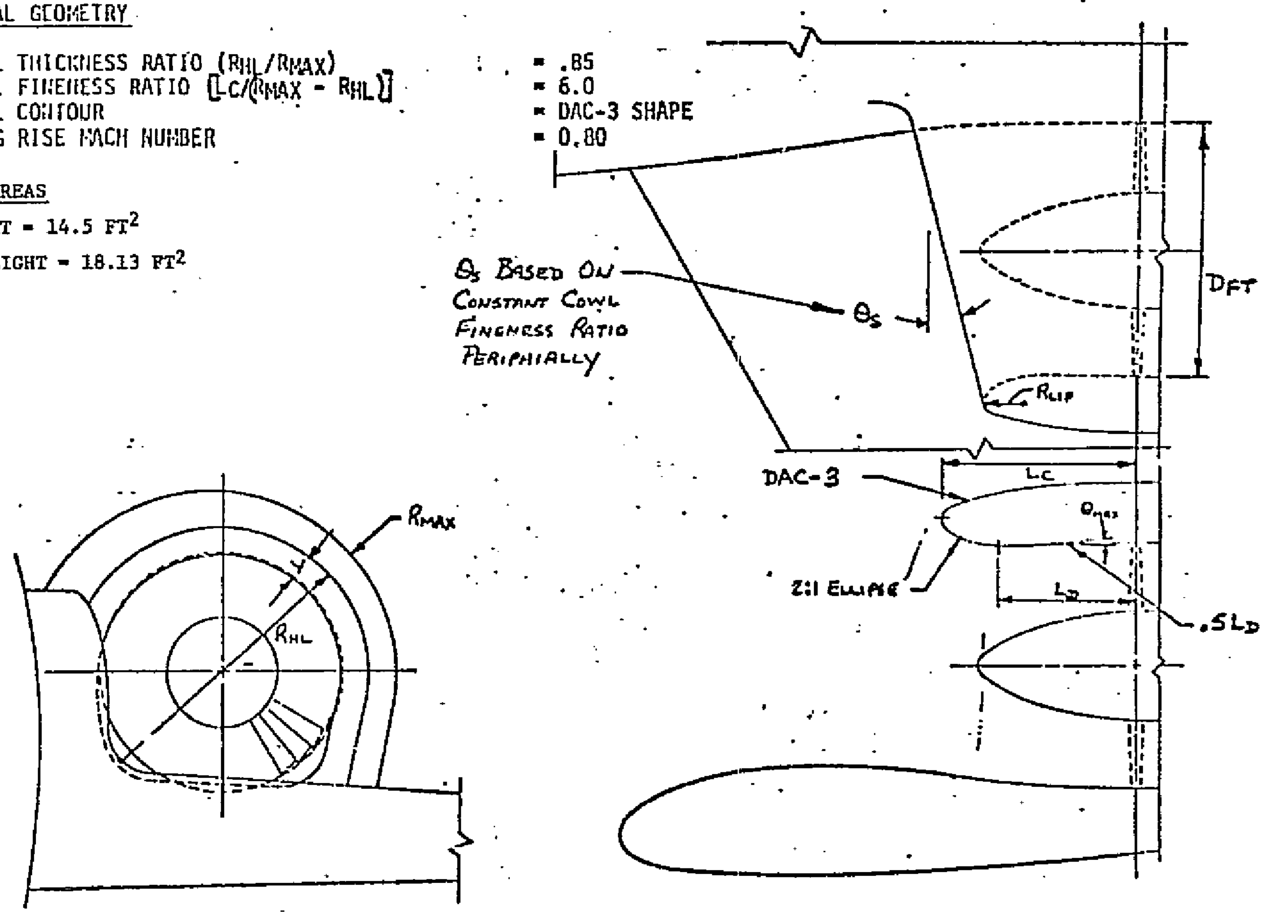
- MAX INTERNAL WALL ANGLE (θ_{MAX}) = 7° @ .5 L_D
- INLET LIP THICKNESS RATIO (Y/R_{HL}) = .10
- INLET LIP CONTOUR = 2:1 ELLIPSE
- LIP LEADING EDGE RADIUS (R_{LIP}) = .05 RHL
- INTERNAL DUCT CONTOUR = CUBIC CONTOUR
- DIFFUSER LENGTH RATIO (L_D/D_{FT}) = 0.5

o EXTERNAL GEOMETRY

- COWL THICKNESS RATIO (R_{HL}/R_{MAX}) = .85
- COWL FINENESS RATIO $[L_C/(R_{MAX} - R_{HL})]$ = 6.0
- COWL CONTOUR = DAC-3 SHAPE
- DRAG RISE MACH NUMBER = 0.80

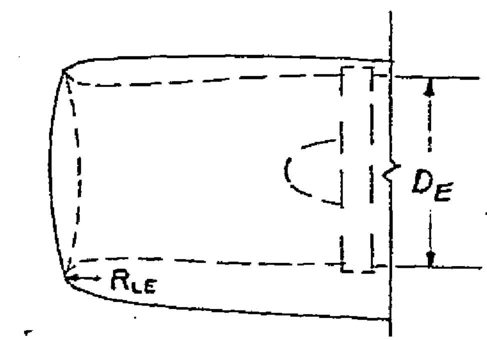
o INLET AREAS

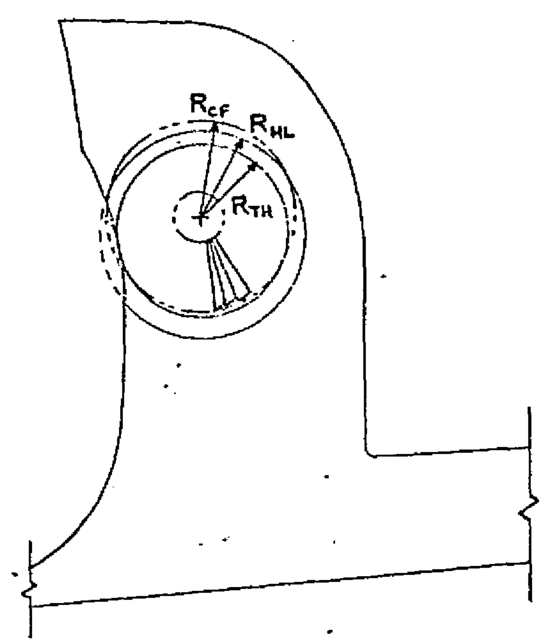
- THROAT = 14.5 FT^2
- HIGHLIGHT = 18.13 FT^2

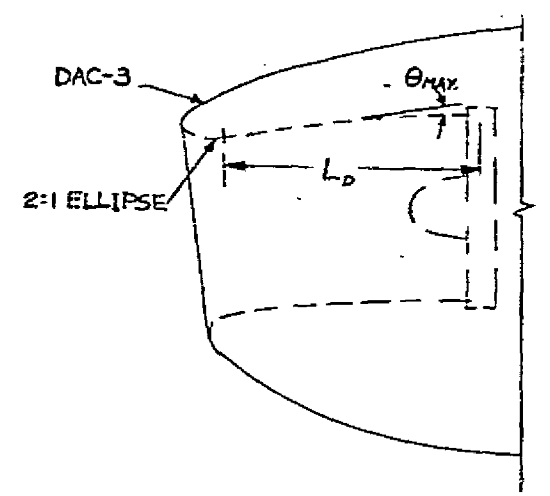


GAS GENERATOR SIDE INLET DESIGN CRITERIA

- INTERNAL GEOMETRY
  - MAX INTERNAL WALL ANGLE (θ_{MAX}) = 5° at .5 L_D
  - CONTRACTION RATIO $(R_{HL}/R_{TH})^2$ = 1.40
  - INLET LIP CONTOUR = 2:1 ELLIPSE
  - INTERNAL DUCT CONTOUR = CUBIC CONTOUR
  - DIFFUSER LENGTH RATIO (L_D/D_E) = 1.39
  - ENGINE FACE DIAMETER (D_E) = 20.15 IN.
- EXTERNAL GEOMETRY
  - LIP LEADING EDGE RADIUS (R_{LE}) = b^2/a
  - COWL CONTOUR = DAC-3 SHAPE
- INLET AREAS
  - A_{HL} = 2.688 FT^2
  - A_{TH} = 1.920 FT^2





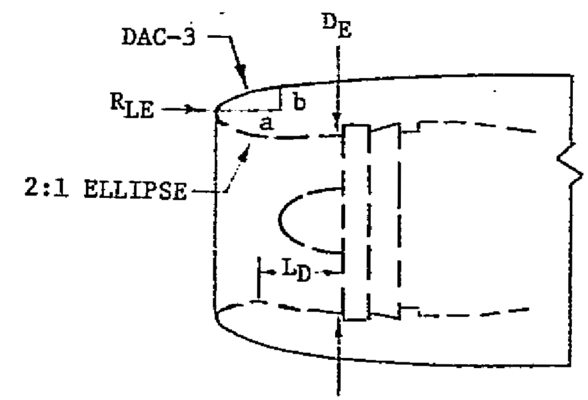


GAS GENERATOR TOP INLET DESIGN CRITERIA

(FOR 3rd ENGINE)

o INTERNAL GEOMETRY

- MAX INTERNAL WALL ANGLE (θ_{MAX}) = 9° at .5 L_D
- CONTRACTION RATIO $(R_{HL}/R_{TH})^2$ = 1.40
- INLET LIP CONTOUR = 2:1 ELLIPSE
- INTERNAL DUCT CONTOUR = CUBIC CONTOUR
- DIFFUSER LENGTH RATIO (L_D/D_E) = .50
- ENGINE FACE DIAMETER (D_E) = 20.15 IN.

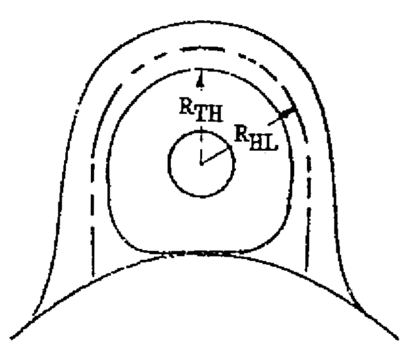


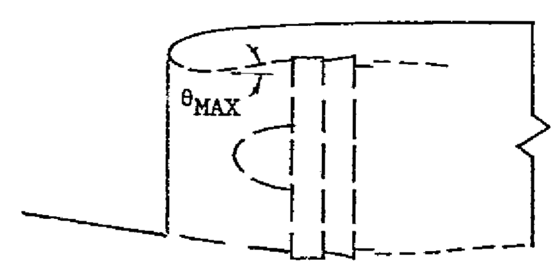
o EXTERNAL GEOMETRY

- LIP LEADING EDGE RADIUS (R_{LE}) = b^2/a
- COWL CONTOUR = DAC-3 SHAPE

o INLET AREAS

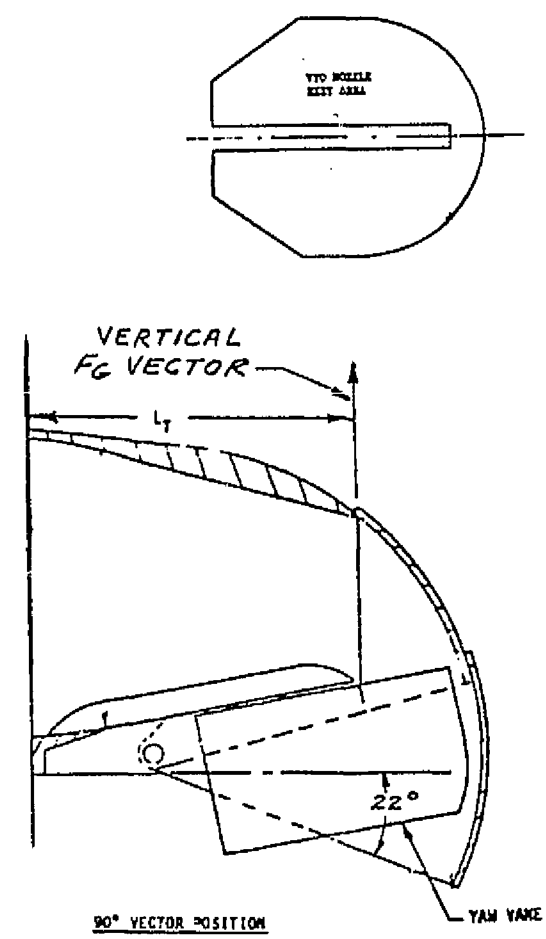
- A_{HL} = 2.710 FT^2
- A_{TH} = 1.920 FT^2





L/C NOZZLE SIZING CHARACTERISTICS

(3) YJ97-GE-100/(3) LCF 459 FANS

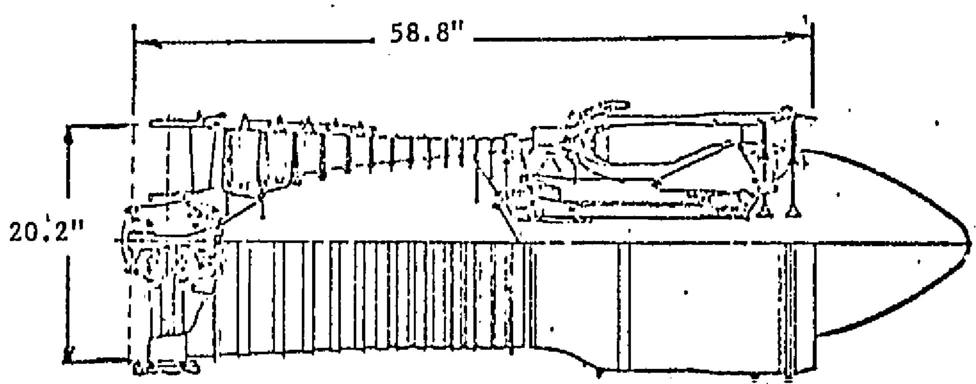


$A_{EX_{VTO}}$ ∿ VTO NOZZLE EXIT AREA ≈ 4487 IN.^{2}

$A_{EX_{cruise}}$ ∿ CRUISE NOZZLE EXIT AREA = 1900 IN.^{2}

L_T = 58.85 IN.

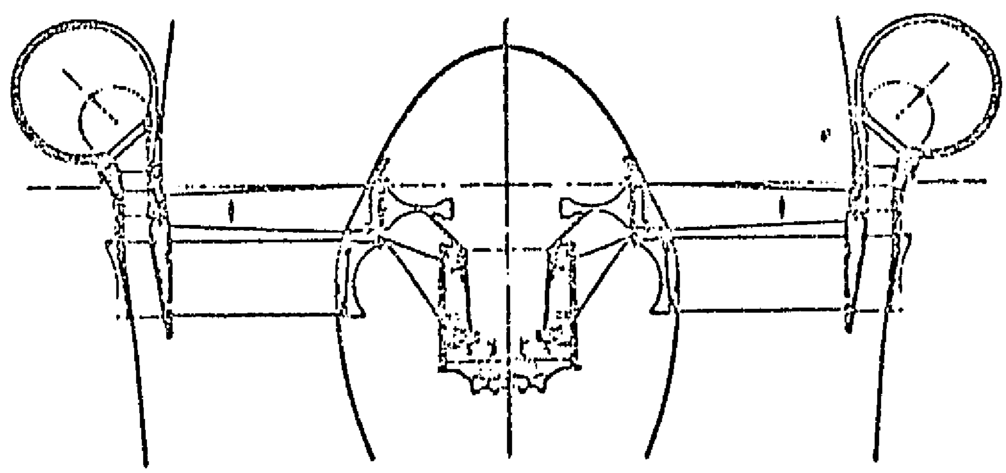
YJ97 CYCLE DESIGN CHARACTERISTICS



UNINSTALLED CHARACTERISTICS AT INTERMEDIATE
S.L. STATIC, STANDARD DAY

% N_{GG}, % RPM	101.5
CPR	14.07
WA_{GG}, LB/SEC	69.2
TIT, °R	2487.1
EGT, °R	1834.6
EGP, PSIA	52.967
W_{GAS}, LB/SEC	70.54
WFM, LB/HR	4822
WEIGHT, LB	739
HPI	13700
HPI/WEIGHT	18.54
HPI SFC	.352

LCF459 TURBOTIP FAN DESIGN CHARACTERISTICS



UNINSTALLED CHARACTERISTICS AT INTERMEDIATE
S.L. STATIC, STANDARD DAY

Aero Design FPR	1.319
TDPR	1.19
Fan Airflow, lb/sec	646
Turbine Gas Flow, lb/sec	70.54
Thrust, lb	14,152
SFC	.341
Fan Diameter, in.	59
Fan Weight, lb	850

APPENDIX C

MECHANICAL RTA PROPULSION SYSTEM DATA PACKAGE
(3) PD370-25A/(3) 62 INCH VARIABLE PITCH FANS

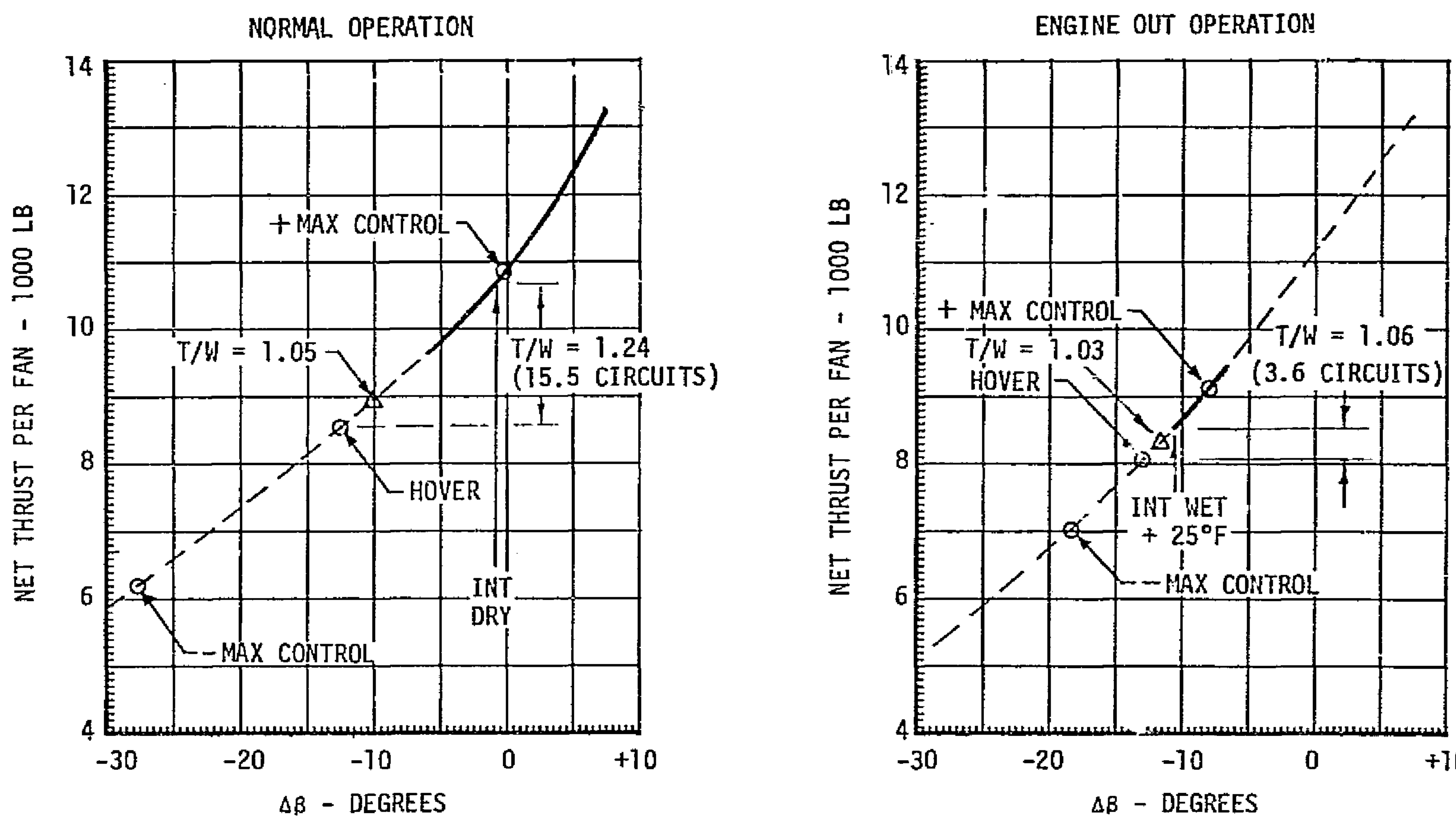
I. PERFORMANCE DATA

MECHANICAL RTA
INSTALLED VTOL PROPULSION SYSTEM PERFORMANCE
SLS, 89.8°F

PARAMETERS		POWER RATINGS			
		NORMAL INT. DRY (T/W=1.25)(1)	NORMAL (MCAIR EST) (T/W=1.05)(1)	ENGINE OUT INT. WET + 25° (T/W=1.06)(2)	ENGINE OUT (MCAIR EST) (T/W=1.03)(2)
SUPERCHARGED GAS GENERATOR (CORE)					
RPM	% N_{GG}	$\boxed{104.6}$	$\boxed{101.1}$	$\boxed{108.1}$	$\boxed{106.5}$
BOT	°F	$\boxed{2290}$	$\boxed{2077}$	$\boxed{2375}$	$\boxed{2297}$
W_A	LB/SEC	44.9	40.1	45.9	45.2
W_F	LB/HR	3618	2846	3887	3739
W_{GAS}	LB/SEC	45.9	40.9	47.0	46.2
W_{H_2O}	LB/SEC	0	0	1.4	1.36
UNSUPERCHARGED GAS GENERATOR (CORE)					
RPM	% N_{GG}	$\boxed{100.8}$	$\boxed{97.2}$	$\boxed{108.0}$	$\boxed{106.4}$
BOT	°F	$\boxed{2290}$	20[illegible]	$\boxed{2375}$	$\boxed{2297}$
W_A	LB/SEC	41.3	38.5	41.7	41.1
W_F	LB/HR	3346	2741	3579	3430
W_{GAS}	LB/SEC	42.2	39.3	42.7	42.1
W_{H_2O}	LB/SEC	0	0	1.3	1.23
FANS					
RPM (LIFT & L/C FANS)	% N_F	$\boxed{100.0}$	$\boxed{100.0}$	$\boxed{100.0}$	$\boxed{100.0}$
W_A (L/C FAN)	LB/SEC	638.1	581.8	563.5	554.2
W_A (LIFT FAN)	LB/SEC	641.8	583.3	571.5	562.1
$\beta_{L/C}/\beta_{LIFT}$	DEGREES	-0.40/-0.10	-10.0/-9.85	-10.60/-9.30	-13.20/-11.90
TOTAL PROPULSION SYSTEM					
LIFT	LB	$\boxed{32,138}$	$\boxed{26,953}$	$\boxed{25,756}$	$\boxed{24,978}$
W_F	LB/HR	10,582	8433	7466	7169
LIFT SFC	LB/HR/LB	.329	.313	.290	.287

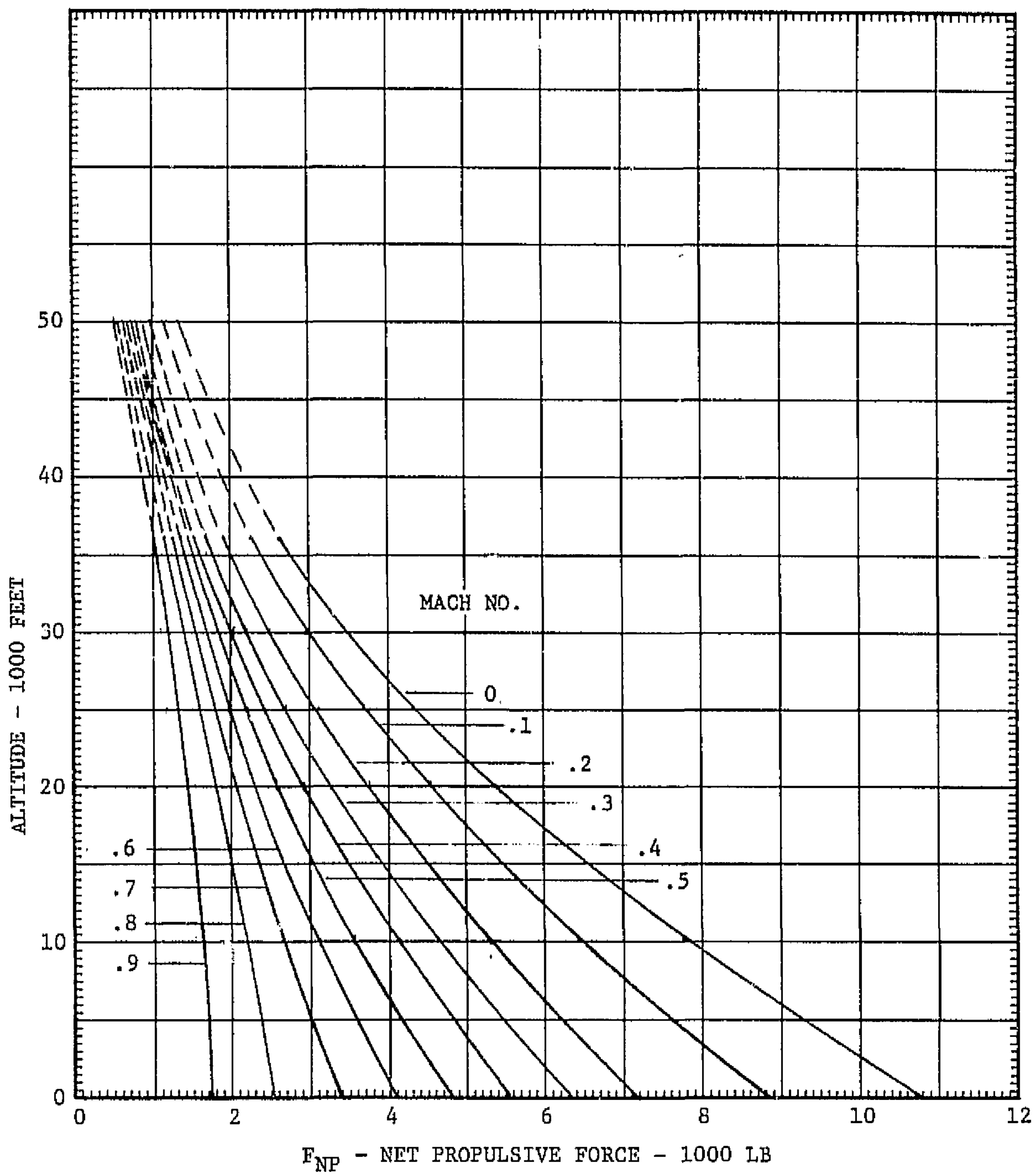
(1) 5 CIRCUITS FUEL
(2) 2 CIRCUITS FUEL

MECHANICAL RTA CONTROL PERFORMANCE
LIFT/CRUISE FAN
SLS, 89.8°F
100% N_F

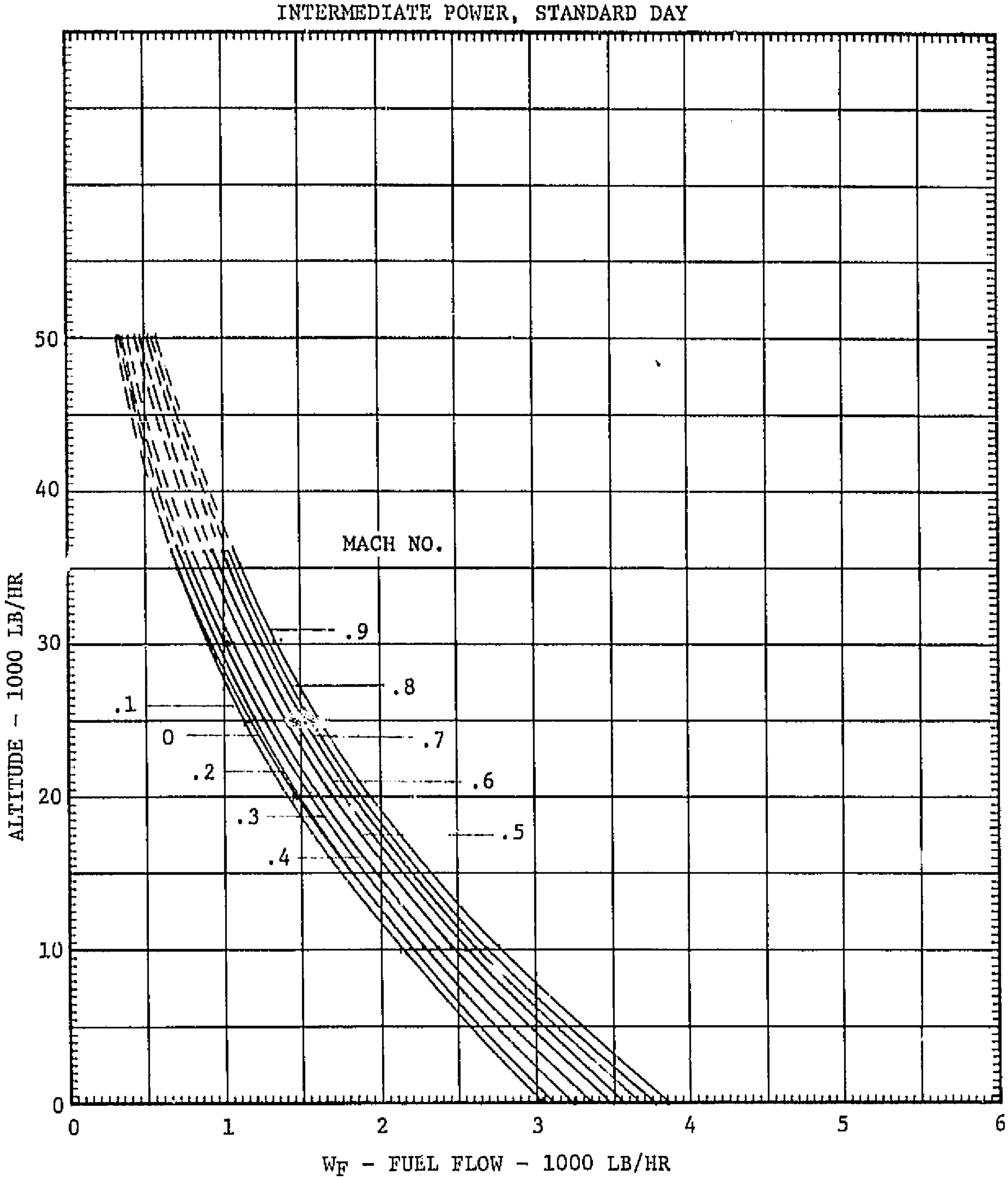


——— DDA DATA
- - - - - MCAIR EXTRAPOLATION

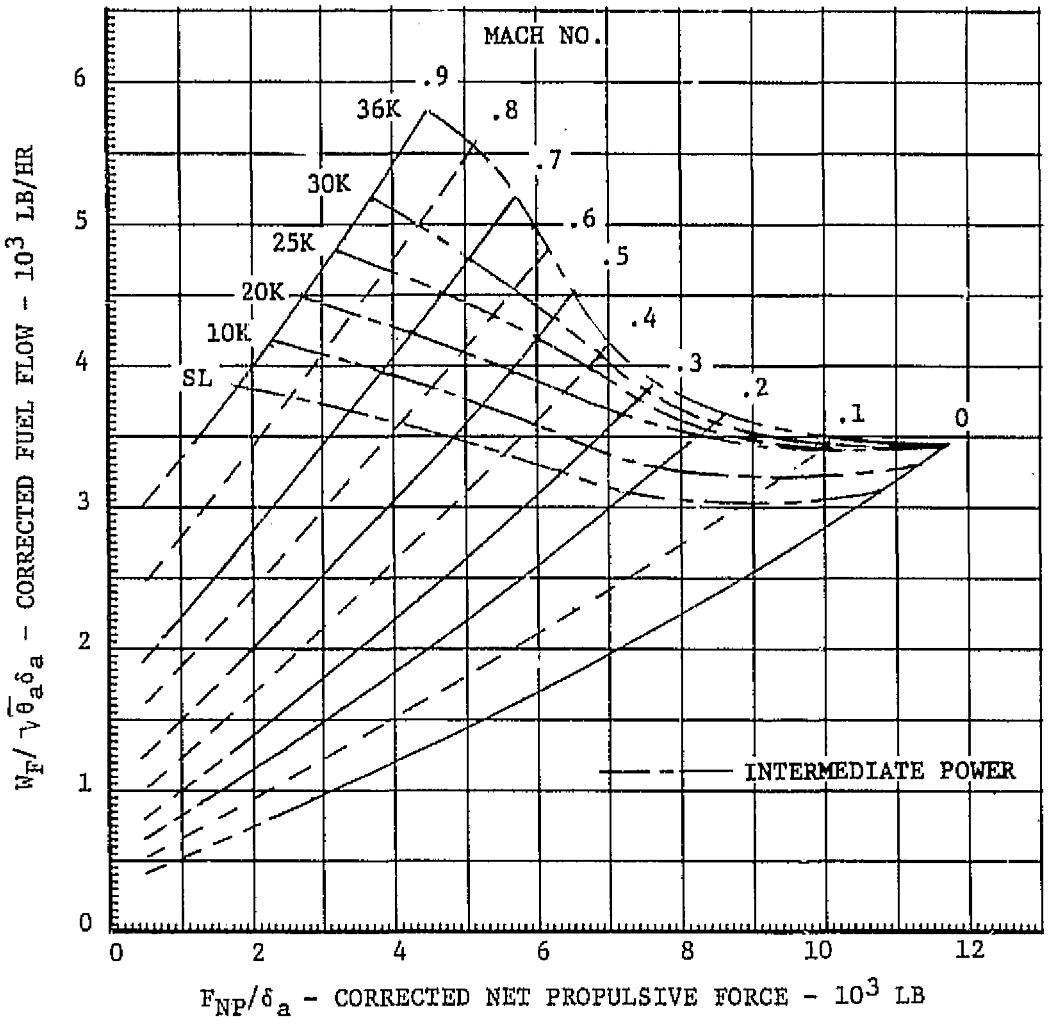
MECHANICAL RTA
NET PROPULSIVE FORCE PER L/C FAN
(3) PD370-25A/(3) FAN SYSTEM
INTERMEDIATE POWER
STANDARD DAY



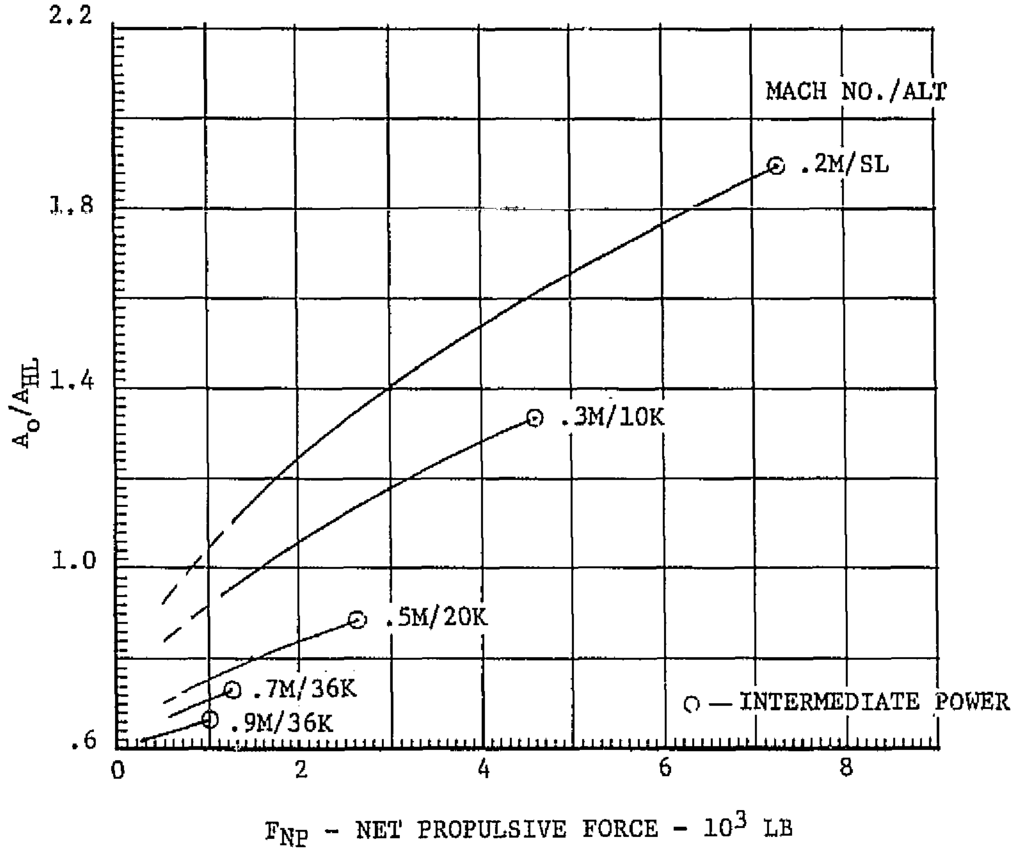
MECHANICAL RTA
FUEL FLOW PER GAS GENERATOR
(3) PD370-25A/(3) FAN SYSTEM
INTERMEDIATE POWER, STANDARD DAY



REDUCED POWER PERFORMANCE
PD370-25A MECHANCIAL FAN RTA
STANDARD DAY



LIFT/CRUISE FAN INLET MASS FLOW RATIO
PD370-25A MECHANICAL FAN RTA
AERO DESIGN FPR = 1.2, FAN DIA. = 62 IN.

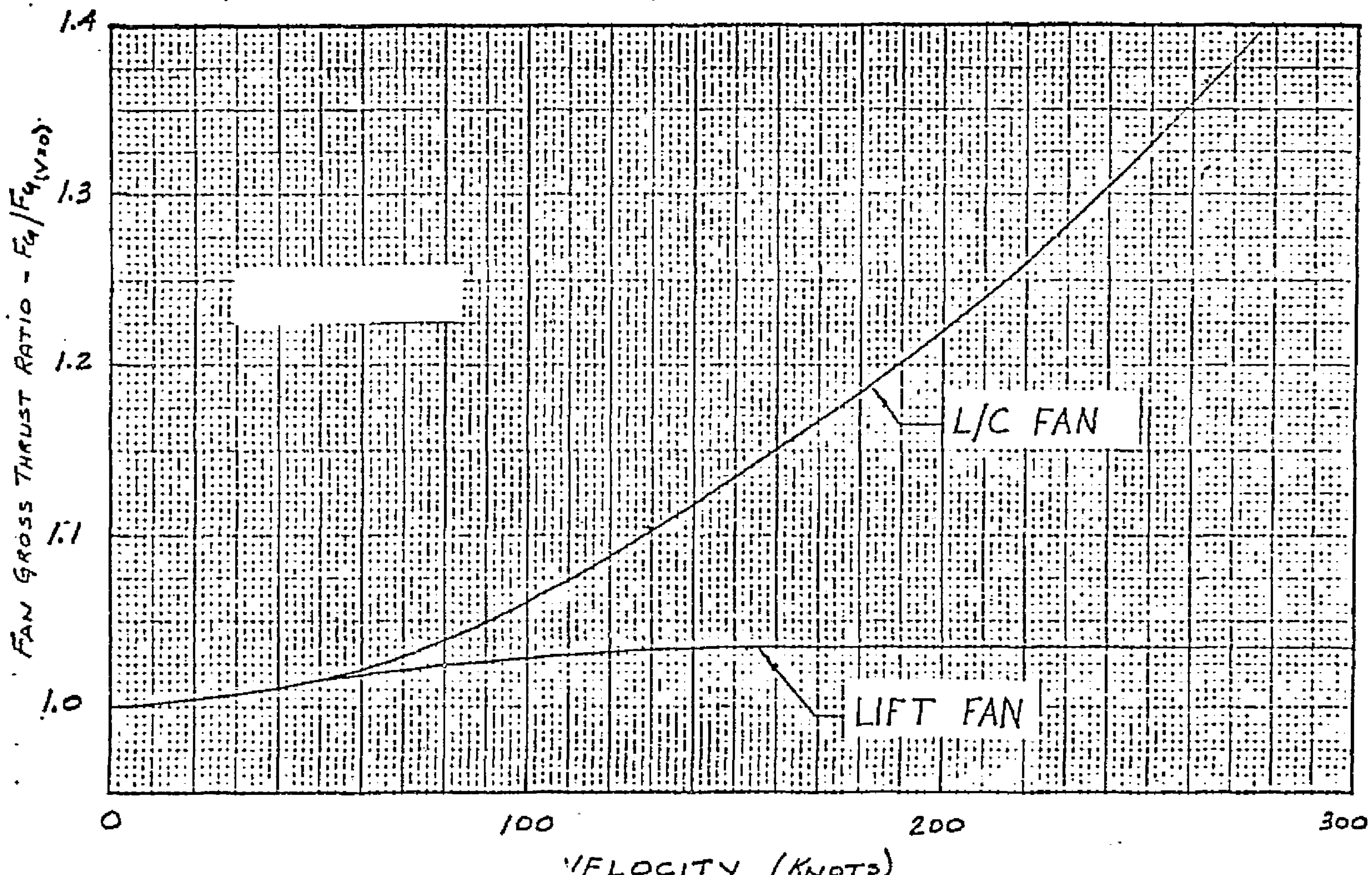


NOTES: 1. $A_{HL} = 1.25\ A_t$
2. A_t BASED ON FOLLOWING:
- THROAT MACH NO. = 0.77
- FLIGHT MACH NO. = 0.3
- INTERMEDIATE POWER AT 36,089 FT WITH 3% AIRFLOW MARGIN

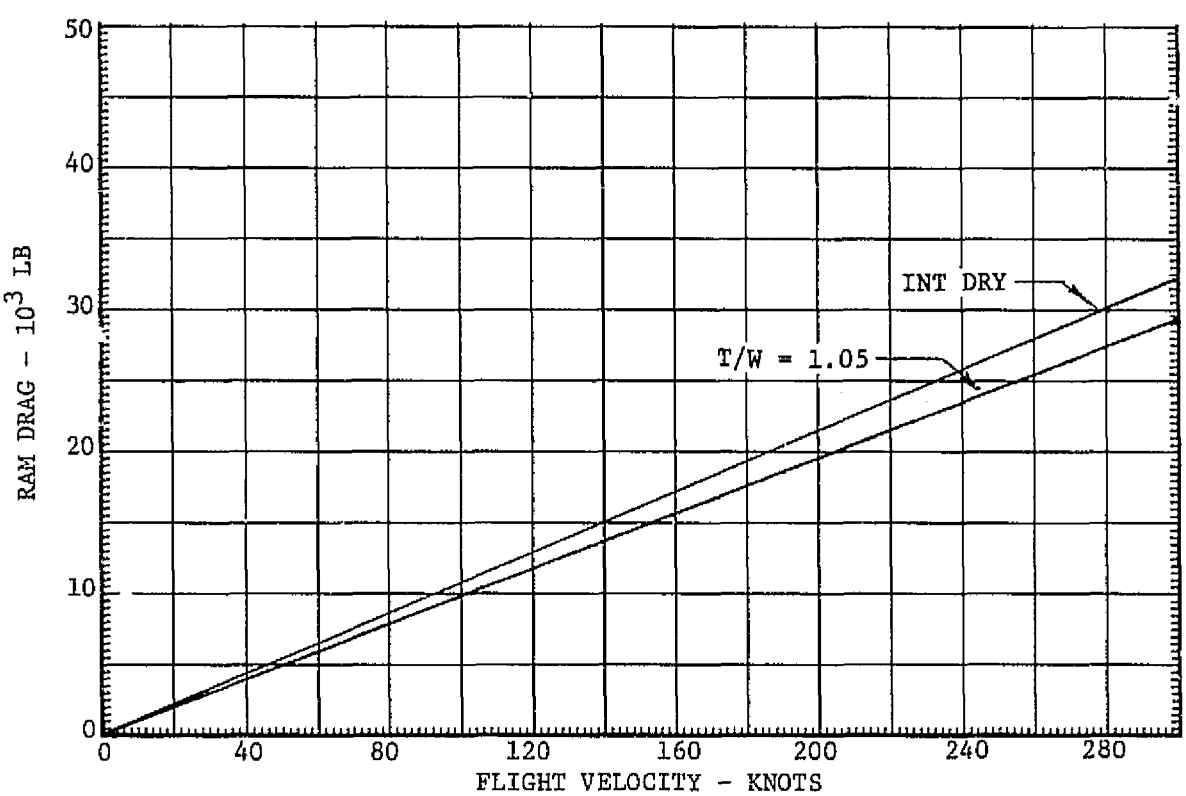
MODEL 260

GROSS THRUST PERFORMANCE DURING POWERED LIFT OPERATION

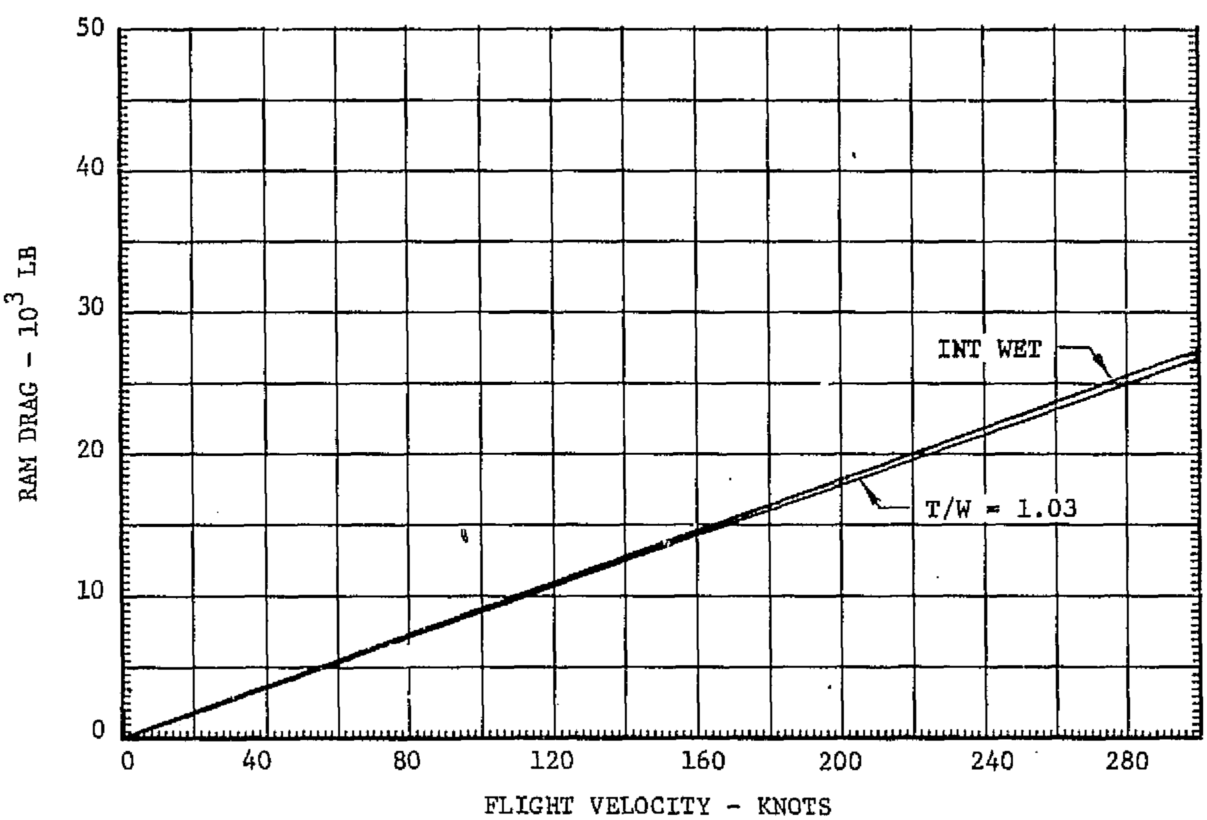
SEA LEVEL, 89.8°F



RAM DRAG DURING POWERED LIFT FLIGHT
PD370-25A MECHANICAL FAN SYSTEM
S.L.S., 89.8°F
NORMAL TAKEOFF DRY



RAM DRAG DURING POWERED LIFT FLIGHT
PD370-25A MECHANICAL FAN SYSTEM
S.L.S., 89.8°F
ENGINE OUT
INT WET



II. INSTALLATION FACTORS

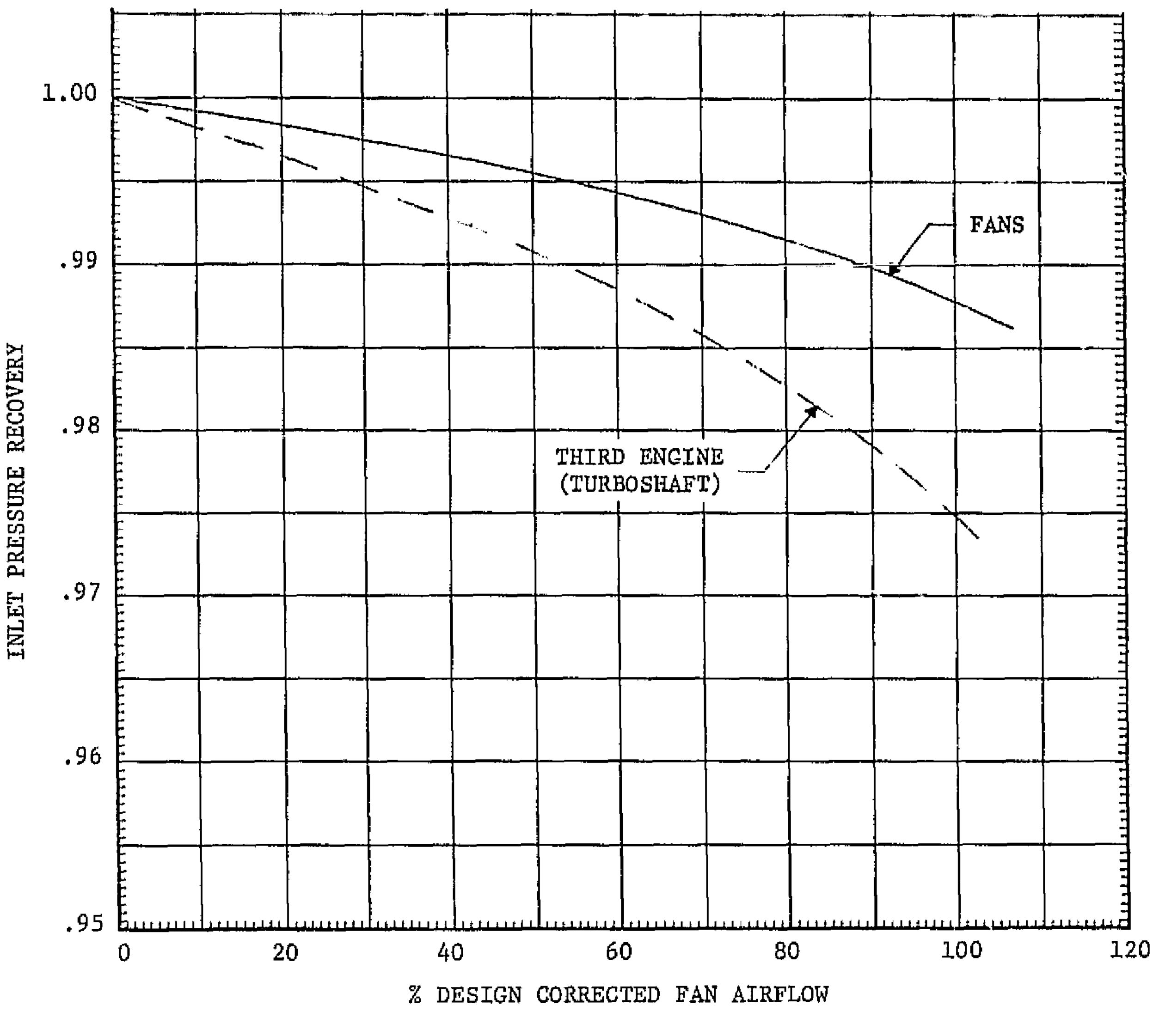
MECAHNICAL RTA
PROPULSION SYSTEM INSTALLATION FACTORS
VTO OPERATION

COMPONENT	NOMINAL INSTALLATION FACTOR
TURBOSHAFT ENGINES	
PRESSURE RECOVERY[d]	.975
COMPRESSOR AIRBLEED[a] (LB/SEC)	0.10
HORSEPOWER EXTRACTION[a]	125[c] 0[d]
L/C FAN AND NACELLE	
PRESSURE RECOVERY	.988
DUCT PRESSURE LOSS (FAN COLD STREAM)	(b)
NOZZLE THRUST COEFFICIENT	.94
LIFT FAN SYSTEM	
PRESSURE RECOVERY	.988
NOZZLE THRUST COEFFICIENT	.95
CENTER GEARBOX	
HORSEPOWER EXTRACTION	0
HORSEPOWER LOSS	(b)
ADDITIONAL PERFORMANCE ALLOWANCES	
NET THRUST DERATE	0
SUPERCHARGING	100%
FAN HP ALLOWANCE DUE TO DEAD ENGINE	2%
GROUND EFFECTS/REINGESTION	0

NOTES:
- (a) PER ENGINE
- (b) FURNISHED BY ENGINE CO.
- (c) SUPERCHARGED ENGINE
- (d) NON-SUPERCHARGED ENGINE

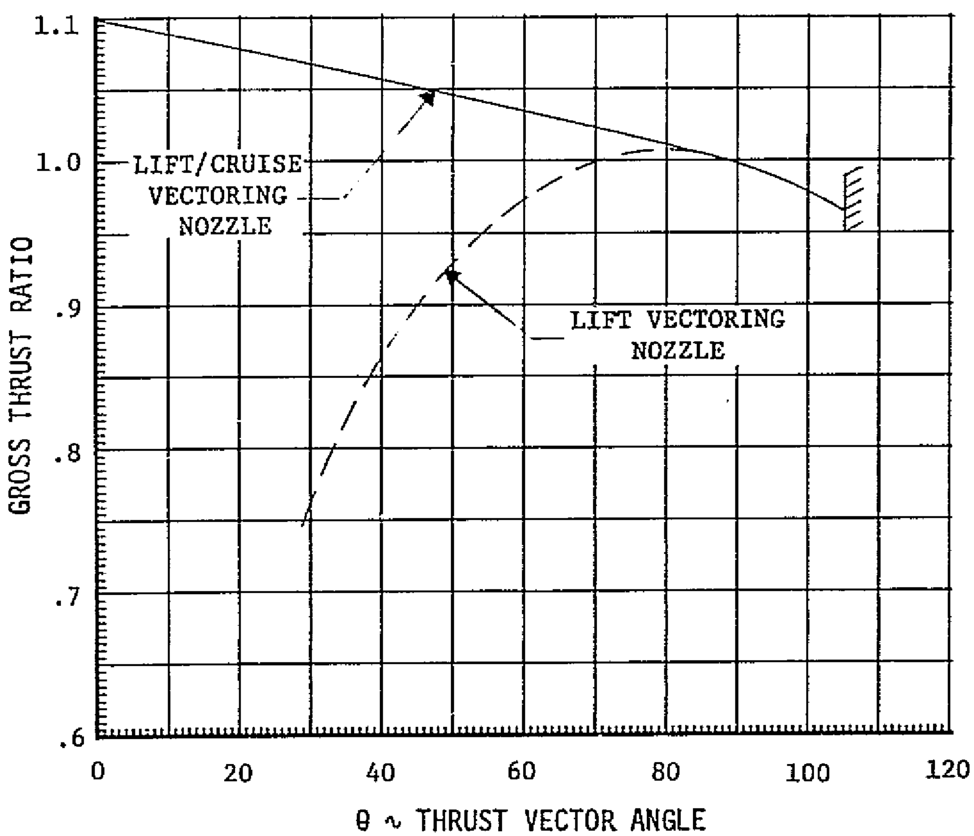
MECHANICAL RTA

ESTIMATED PRESSURE RECOVERY FOR STATIC PERFORMANCE

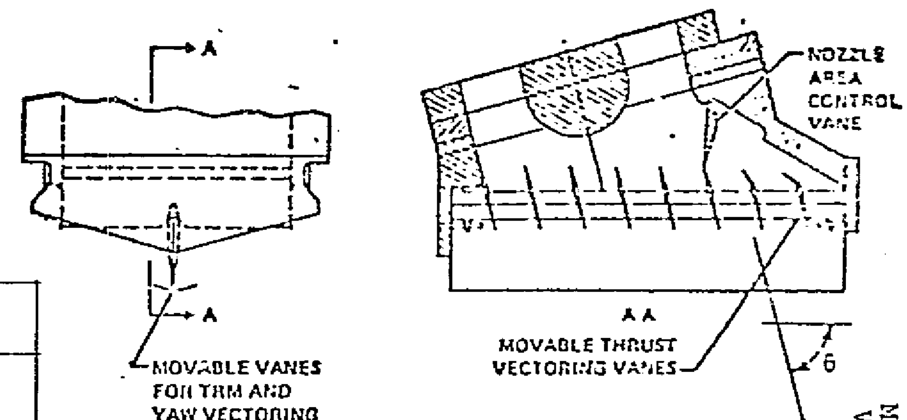


VECTORING SYSTEM PERFORMANCE

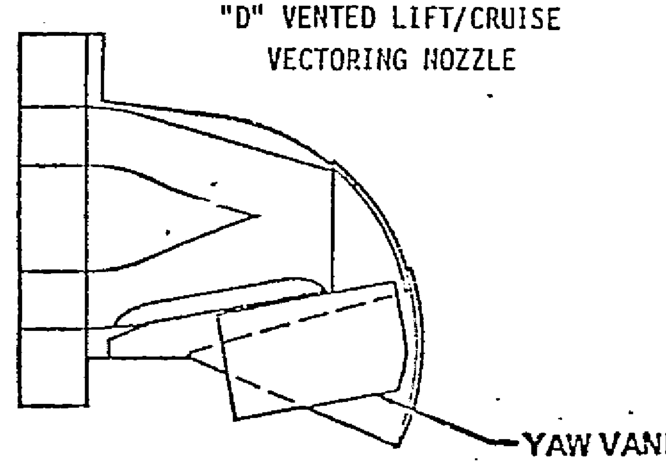
GROSS THRUST VARIATION DURING VECTORING



INTERNAL CASCADE LIFT VECTORING NOZZLE



"D" VENTED LIFT/CRUISE VECTORING NOZZLE



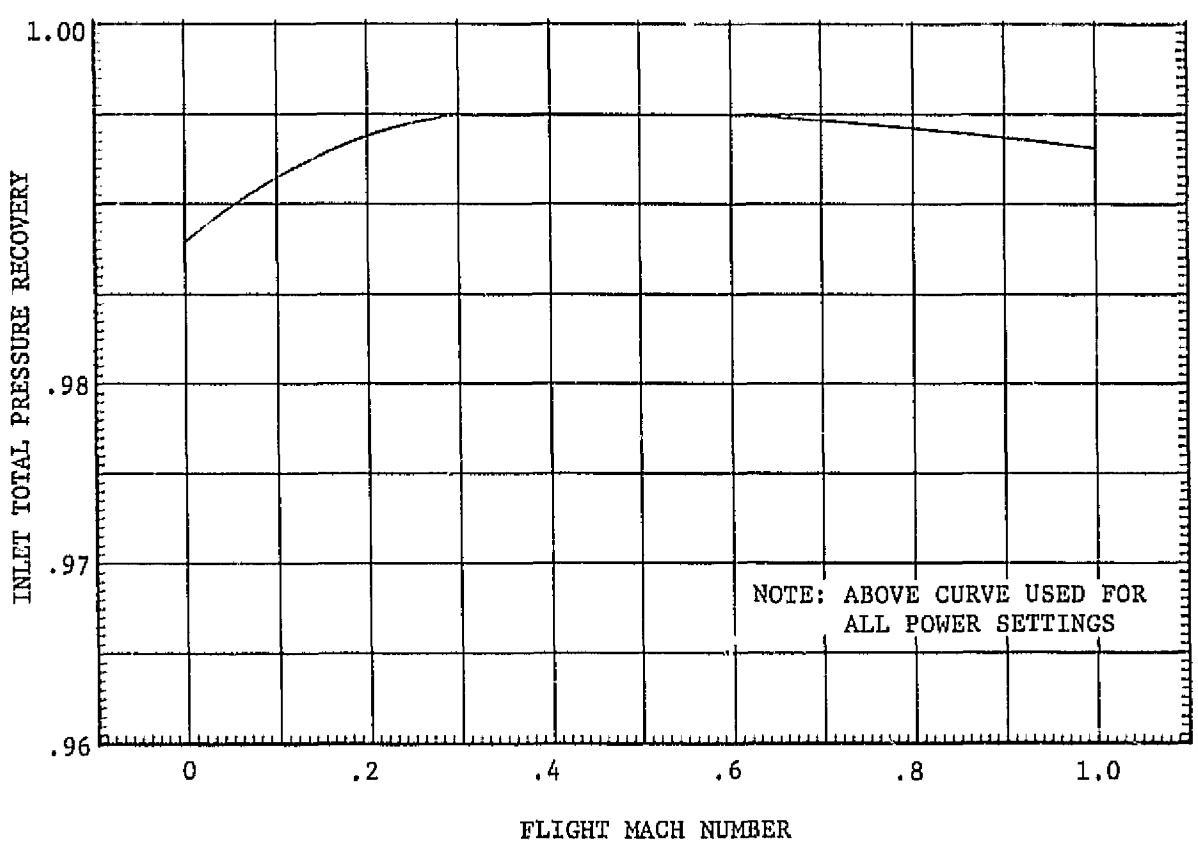
90° VECTOR POSITION

MECHANICAL RTA
PROPULSION SYSTEM INSTALLATION FACTORS
CONVENTIONAL FLIGHT

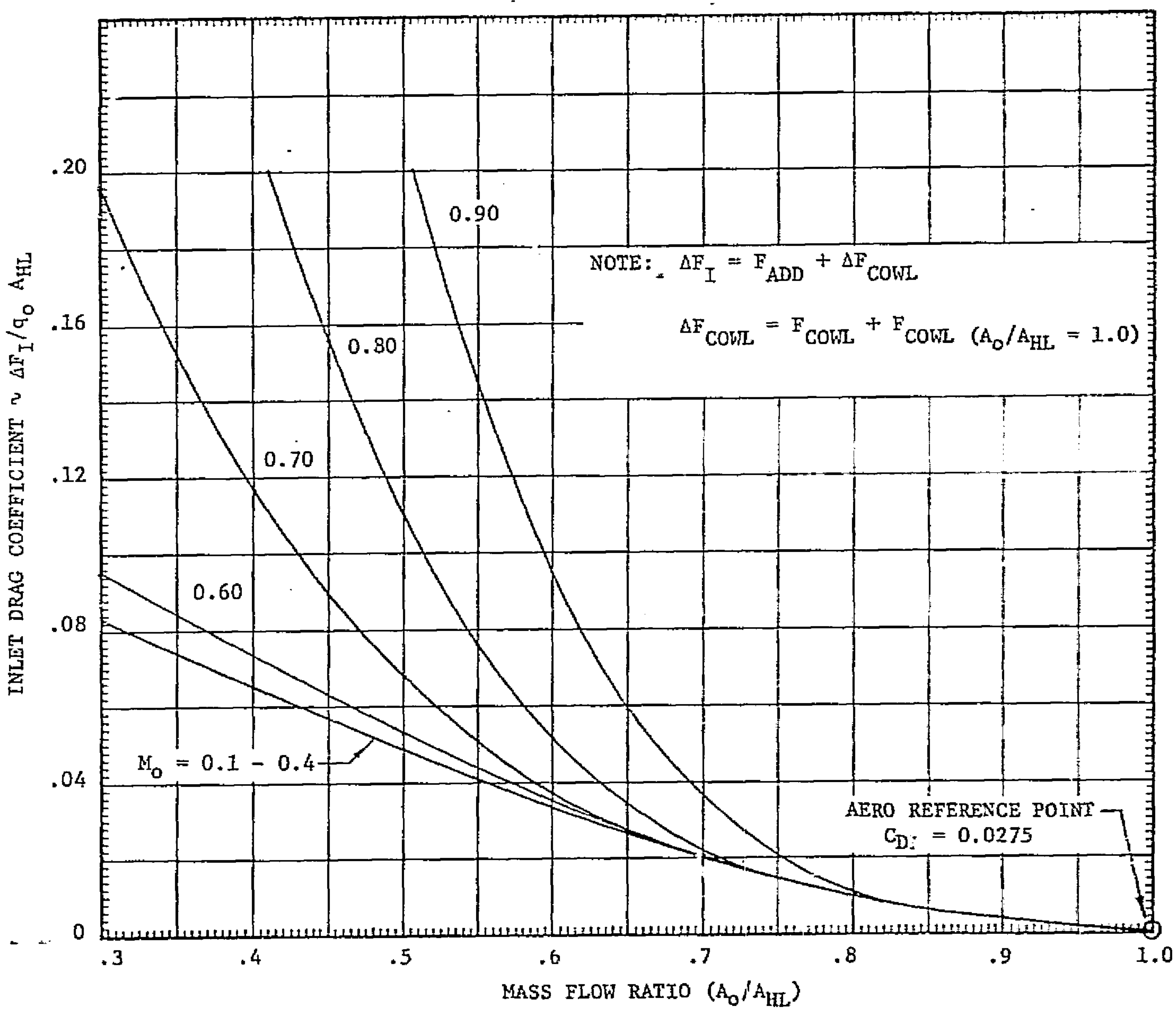
Component	Cruise
Gas Generator	
Pressure Recovery[d]	See Page 16
Compressor Airbleed[a] (lb/sec)	0.10
Horsepower Extraction[a]	40[c]
Nozzle Thrust Coefficient	.98
L/C Fan and Nacelle	
Pressure Recovery	See Page 16
Duct Pressure Loss (Fan Cold Stream)	(b)
Nozzle Thrust Coefficient	.98
Lift Fan System	
Pressure Recovery	---
Nozzle Thrust Coefficient	---
Additional Performance Allowances	
Net Thrust Derate	(b)
Center Gearbox	
Horsepower Extraction	0
Horsepower Loss	(b)

Notes: (a) Per Engine
(b) Furnished by Engine Co.
(c) Supercharged Engine
(d) Non-Supercharged Engine

MECHANICAL RTA
ESTIMATED INLET TOTAL PRESSURE RECOVERY
CONVENTIONAL FLIGHT



ESTIMATED INLET DRAG CHARACTERISTICS

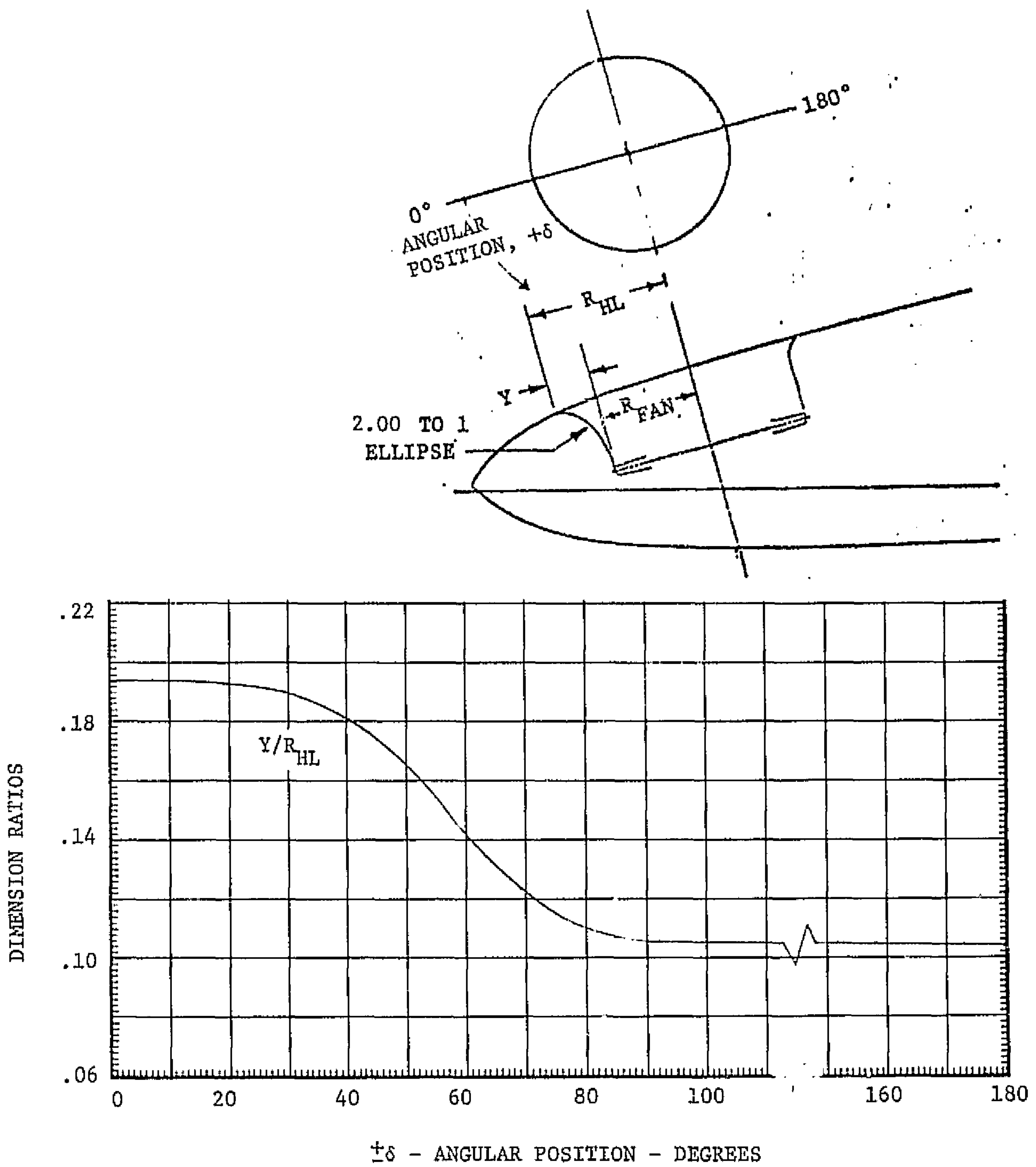


NOZZLE DRAG

NO NOZZLE DRAG INCLUDED IN NPF

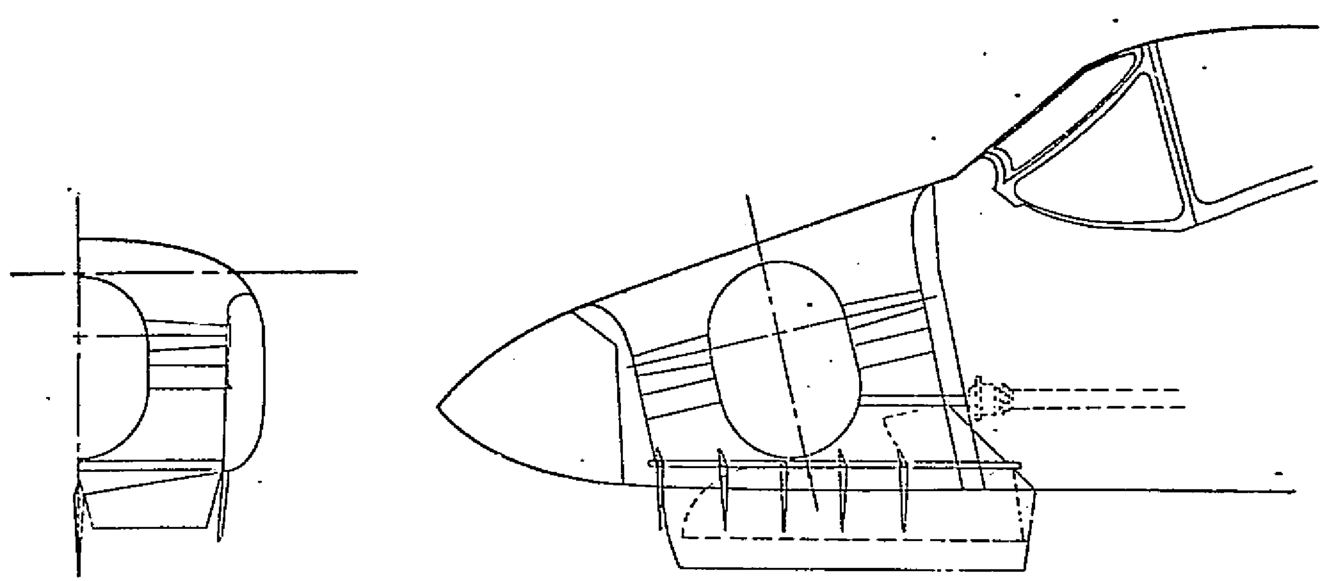
III. PHYSICAL DIMENSIONS AND WEIGHTS

LIFT FAN INLET DEFINITION



MECHANICAL RTA
NOSE FAN UNIT DESIGN GUIDELINES

- INTERNAL GEOMETRY
  - INLET LIP CONTOUR = 2:1 ELLIPSE
  - CONTRACTION RATIO = 1.45
- INLET AREAS
  - A_{HL} = 3584 IN.^{2}
  - A_{TH} = 2472 IN.^{2}
- INSTALLED PERFORMANCE (STATIC)
  - INLET RECOVERY = .988
  - NOZZLE VELOCITY COEFF. = .95
- VECTORING REQUIREMENTS
  - ARTICULATED VANES: $40° \leq \theta \leq 105°$
  - YAW VANES: $\pm 16°$



MECHANICAL RTA
LIFT/CRUISE UNIT DESIGN GUIDELINES

o <u>INTERNAL GEOMETRY</u>

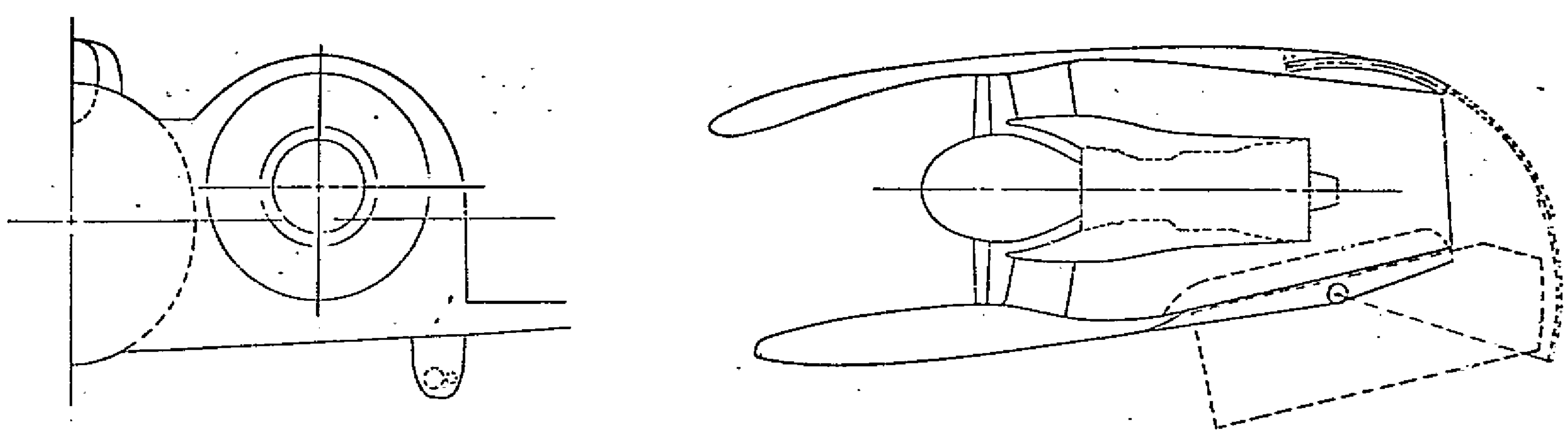
- MAX INTERNAL WALL ANGLE (θ_{MAX}) = 7° @ .5 L_D
- INLET LIP THICKNESS RATIO (Y/R_{HL}) = .10
- INLET LIP CONTOUR = 2:1 ELLIPSE
- LIP LEADING EDGE RADIUS (R_{LIP}) = .05 R_{HL}
- INTERNAL DUCT CONTOUR = CUBIC CONTOUR

o <u>EXTERNAL GEOMETRY</u>

- COWL THICKNESS RATIO (R_{HL}/R_{MAX}) = .85
- COWL FINENESS RATIO [$L_C/(R_{MAX} - R_{HL})$] = 6.0
- COWL CONTOUR = DAC-3 SHAPE

o <u>INLET AREAS</u>

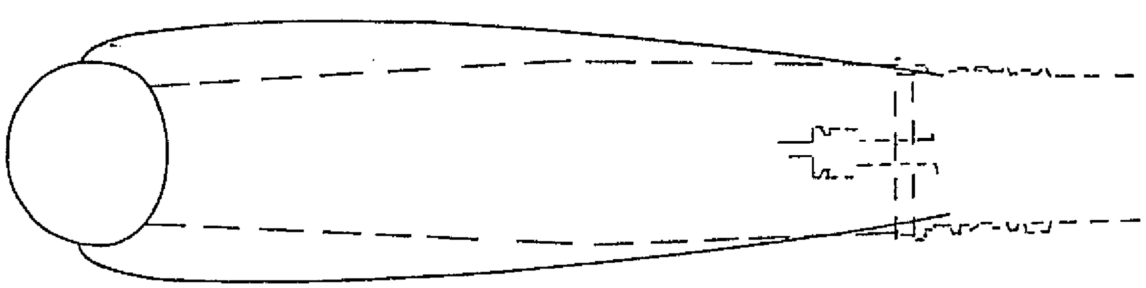
- THROAT = 16.63 FT^2
- HIGHLIGHT = 20.79 FT^2

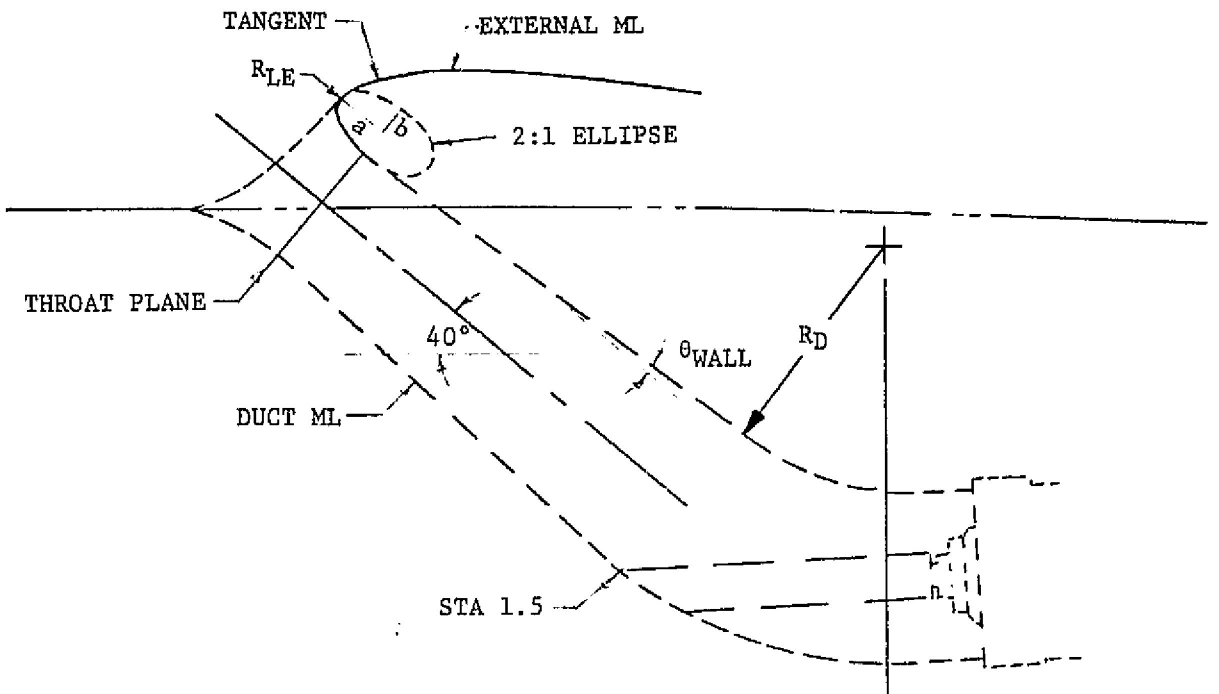


GAS GENERATOR TOP INLET DESIGN CRITERIA

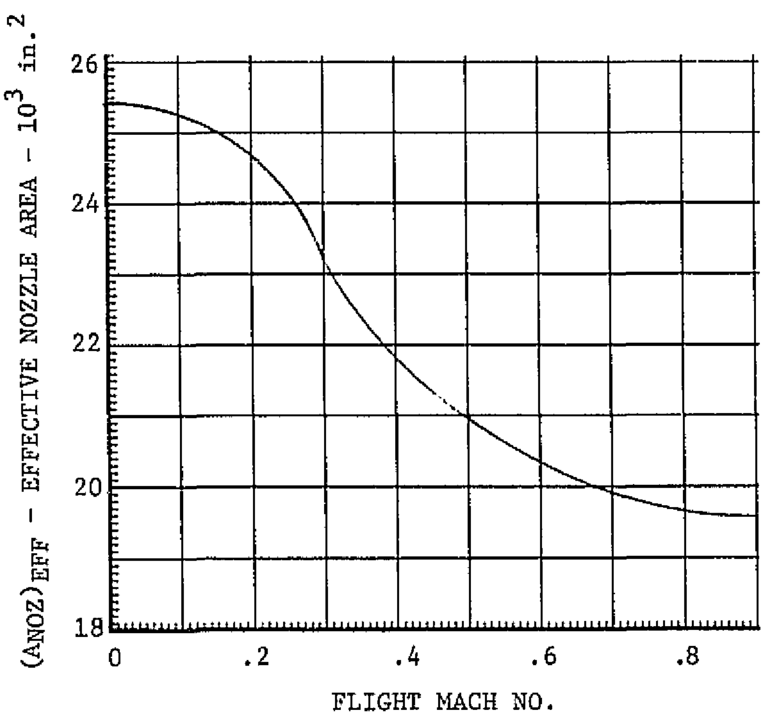
(SEMI-FLUSH INLET DESIGN FOR THIRD ENGINE)

- o INTERNAL GEOMETRY
  - CONTRACTION RATIO $(R_{HL}/R_{TH})^2$ = 2.0
  - INLET LIP CONTOUR = 2:1 ELLIPSE
  - ENGINE FACE DIAMETER (D_E) = 18.4 in.
  - DIFFUSER WALL ANGLE ($2\theta_{WALL}$) = 7°
  - DUCT INSIDE TURN RADIUS (R_D) = 27 in.
  - TURN CONTRACTION RATIO $(R_{STA\ 15}/R_E)^2$ = 1.18
- o EXTERNAL GEOMETRY
  - LIP LEADING CONTOUR (R_{LE}) = b^2/a
  - COWL CONTOUR = DAC-3 SHAPE
- o INLET AREAS
  - A_{HL} = 332 in^2
  - A_{TH} = 166 in^2

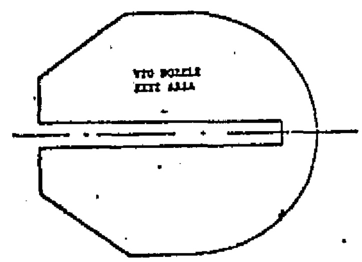


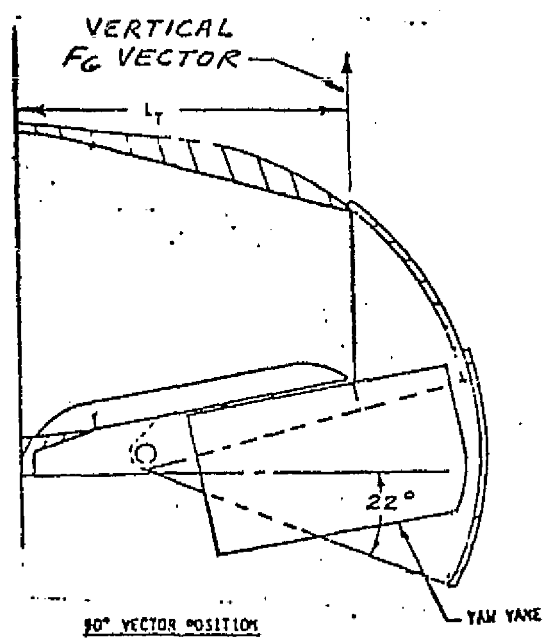


LIFT/CRUISE FAN CRUISE NOZZLE AREAS
PD370-25A MECHANICAL FAN RTA
STANDARD DAY



L/C NOZZLE SIZING CHARACTERISTICS
PD370-25A MECHANICAL FAN SYSTEM





$A_{NOZ_{VTO}}$ = VTO NOZZLE EXIT AREA = 5369 in^2

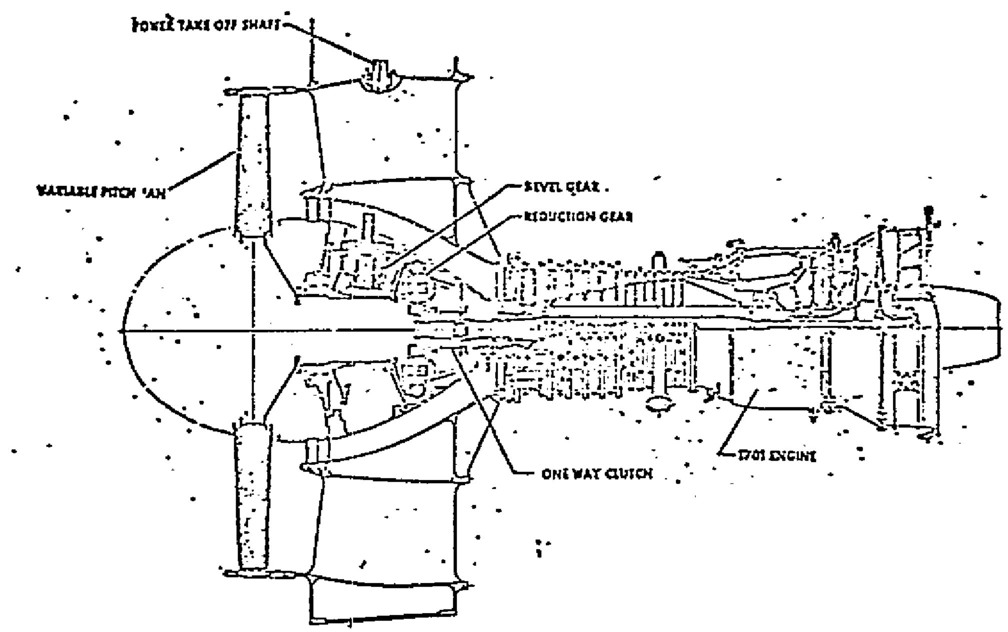
$A_{NOZ_{CRUISE}}$ = CRUISE NOZZLE EXIT AREA = 2200 in^2

L_T = 65.25 in.

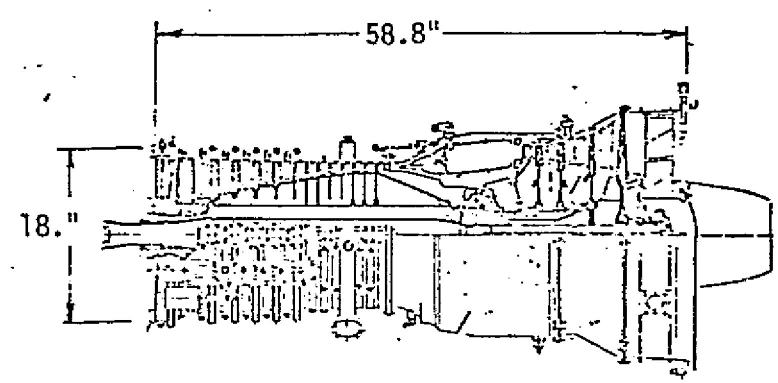
MECHANICAL RTA PROPULSION SYSTEM
LIFT/CRUISE UNIT

UNINSTALLED DESIGN CHARACTERISTICS
S.L.S., 59°F
INTERMEDIATE

COMPRESSOR PRESSURE RATIO	12.3
OVERALL PRESSURE RATIO	14.8
BYPASS RATIO	
PRIMARY AIRFLOW, LB/SEC	
BURNEROUT TEMPERATURE, °F	
FREE TURBINE INLET TEMP, °F	
FAN INLET AIRFLOW, LB/SEC	
HP REQUIRED PER FAN, HP	
COLD STREAM THRUST, LB	
PRIMARY THRUST, LB	
TOTAL THRUST, LB	
TOTAL SFC, LB/HR-LB	
WEIGHT, (L/C UNIT), LB	2361
THRUST/WEIGHT (L/C UNIT)	



XT701 TURBOSHAFT ENGINE



UNINSTALLED ENGINE CHARACTERISTICS
S.L.S., 59°F
INTERMEDIATE

% N_{GG}, % RPM	100.0
CPR	12.3:1
AVG. PR/STAGE	1.21
WA_{GG}, LB/SEC	44.3
TIT, °R	
EGT, °R	2097
SHP	8079
W_{GAS}, LB/SEC	45.4
W_F, LB/HR	3780
WEIGHT, LB	765

MECHANICAL RTA

NOSE FAN UNIT

INSTALLED DESIGN CHARACTERISTICS
S.L.S., 90°F
INTERMEDIATE DRY

FAN DIA., IN.	62
N_{FAN}, % RPM	100
V_{TIP}, FT/SEC	932
W_{FAN}, LB/SEC	597.8
WEIGHT, NOSE FAN, LB	1047
FAN THRUST, LB	9461
THRUST/WEIGHT	9.04
$\Delta\beta$ FAN, DEGREES	-4
HP REQ'D	6159