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AD NUMBER
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USAAVLABS TECHNICAL REPORT 66-51

ADVANCED LIFT FAN SYSTEM (LFX) STUDY (U)

By

Harris C. True

Lawrence J. Volk

July 1966

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

CONTRACT DA 44-177-AMC-341(T)

GENERAL ELECTRIC COMPANY

CINCINNATI, OHIO

SEP 29 1966

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FORT EUSTIS, VIRGINIA 23604

(U) This report has been reviewed by the U. S. Army Aviation Materiel Laboratories. The results are published for the exchange and dissemination of information pertaining to studies of an advanced lift fan propulsion system.

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Task 1M121401D14415
Contract DA 44-177-AMC-341(T)
USAAVLABS Technical Report 66-51
July 1966

ADVANCED LIFT FAN SYSTEM (LFX) STUDY (U)

by

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Lawrence J. Volk

Prepared by

General Electric Company
Flight Propulsion Division
Advanced Technology & Demonstrator Programs Department
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FORT EUSTIS, VIRGINIA 23604

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(U) ABSTRACT

This report presents results of conceptual design and analytical studies leading to identification of an advanced (1968-1970) lift fan propulsion system applicable to a U.S. Army V/STOL surveillance and target acquisition mission.

The successful XV-5A flight research vehicle is used as the progenitor of a family of tip turbine fan-in-wing aircraft from which mission and aircraft design analyses point out an advanced turbojet as the logical core engine gas producer from a stable of engines either available or under active development. These analyses also define optimum lift fan objectives in terms of dimensions and performance.

Results of conceptual lift fan design studies are presented in a comparison with objectives.

Critical technology requirements are identified and recommendations for an exploratory development effort are defined.

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(U) FOREWORD

The program was conducted during the period 1 July 1965 to 31 January 1966 under U.S. Army Aviation Materiel Laboratories Contract DA 44-177-AMC-341(T) by the Lift Fan Systems Operation of the General Electric Company's Advanced Engine and Technology Department. The Lift Fan Systems Operation has been engaged in active development of lift fan V/STOL propulsion since 1958.

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(U) SYMBOLS

a	dimension from center of gravity to propulsion component mount, inches
A/A	air angle, degrees
AR	aspect ratio
b	dimension from center of gravity to propulsion component mount, inches
c	dimension from center of gravity to propulsion component mount, inches
CG	center of gravity
C_{V13}	fan thrust coefficient
D-factor	compressor diffusion factor
DGW	aircraft design gross weight, pounds
D_{max}	wing fan maximum diameter, measured in the direction of the wing chord, inches
D_{TT}	turbine tip diameter, inches
F_A	aft force exerted on propulsion components, pounds
F_N	net thrust, pounds
f_o	aircraft parasite area or drag factor, square feet
F_S	side force exerted on propulsion components, pounds
F_V	vertical force exerted on propulsion components, pounds
ID	inside diameter of scroll inlet, inches
K	ratio of wing fan disc loading to wing loading
L	lift, pounds
L_N	net lift or nominal lift, pounds
L/W	aircraft lift/weight ratio, the ratio of total uninstalled lift on a sea level standard day to vertical takeoff gross weight

L_1 — lift of fan receiving flow during power transfer, pounds

L_2 — lift of fan losing flow during power transfer, pounds

L_3 — blade tang load, based on bearing stress, pounds

L_4 — blade tang load, based on average tensile stress, pounds

M — flight Mach number

M_{ABS} — compressor inlet absolute Mach number

M_{EX} — compressor stator discharge absolute Mach number

M_{IN} — compressor blade inlet absolute Mach number

M_P — pitch moment, inch-pounds

M_R — roll moment, inch-pounds

M_{REL} — compressor inlet relative Mach number

M_{R1} — turbine inlet relative Mach number

M_{R2} — turbine discharge relative Mach number

M_Y — yaw moment, inch-pounds

M_1 — turbine inlet absolute Mach number

M_2 — turbine discharge absolute Mach number

$M_{5.5}$ — turbine stator discharge absolute Mach number

N_F — fan speed, revolutions per minute

PT — power transfer

P_P — fuselage fan pressure ratio

P_W — wing fan pressure ratio

$P_{5.1}$ — core engine discharge total pressure, pounds per square inch

q — dynamic pressure, pounds per square foot

R_H — compressor hub radius, inches

R_H/R_T	compressor radius ratio
r/S	radians per second
r/S^2	radians per second^2
R_T	compressor tip radius, inches
R_1-R_6	propulsion system component mount reactions, pounds
S	wing planform area, square feet
TE	trailing edge
TPLC	total pressure loss coefficient
$T_{5.1}$	core engine discharge total temperature, degrees Rankine
U	turbine pitch line wheel speed, feet per second
U_{FT}	fan tip speed, feet per second
$U_{FT}/\sqrt{\theta}$	corrected fan tip speed, feet per second
VAC	variable area control system
V_{R1}	turbine inlet relative velocity, feet per second
V_{R2}	turbine discharge relative velocity, feet per second
V_1	turbine inlet absolute velocity, feet per second
V_2	turbine discharge absolute velocity, feet per second
W_{CREW}	weight of the aircraft crew, pounds
W_{DV}	weight of the diverter valves, pounds
W_E	weight of the core engines, pounds
W_{FE}	weight of the aircraft fixed equipment, pounds
W_{NF}	weight of the nose fan, pounds
W_{PROP}	weight of the propulsion system, pounds
W/S	aircraft wing loading, pounds per square foot
W_{STR}	weight of the aircraft structure, pounds

W_{TF}	weight of the tail fan, pounds
W_{WF}	weight of the wing fan, pounds
$W_{5.1}$	core engine discharge flow, pounds per second
$W\sqrt{\theta}/\delta$	compressor corrected flow, pounds per second
α	aircraft angle of attack, degrees
α_1	scroll nozzle angle, degrees
α_R	aircraft angular acceleration about the roll axis, radians per second^2
β_1	compressor stator inlet absolute air angle, degrees
β_1'	compressor blade inlet relative air angle, degrees
β_2	compressor stator discharge absolute air angle, degrees
β_2'	compressor blade discharge relative air angle, degrees
δ	ratio of ambient pressure to sea level standard day ambient pressure
$\Delta P/q$	compressor static pressure rise coefficient
Γ	turbine discharge swirl angle, degrees
γ_1	turbine inlet flow angle, degrees
γ_2	turbine discharge flow angle, degrees
η_{AD}	compressor adiabatic efficiency
θ	exit louver deflection angle, degrees
$\dot{\theta}$	aircraft angular velocity about the roll axis, radians per second
$\ddot{\theta}$	aircraft angular acceleration about the roll axis, radians per second^2
$\dot{\phi}$	aircraft angular velocity about the pitch axis, radians per second
$\ddot{\phi}$	aircraft angular acceleration about the pitch axis, radians per second^2

$\dot{\Psi}$ aircraft angular velocity about the yaw axis, radians per second

$\ddot{\Psi}$ aircraft angular acceleration about the yaw axis, radians per second^2

$\bar{\omega}$ compressor total pressure loss coefficient

$\bar{\omega}_{10}$ compressor inlet total pressure loss, expressed as a percentage of the local dynamic pressure

$\bar{\omega}_{11}$ compressor discharge total pressure loss, expressed as a percentage of the local dynamic pressure

(C) SUMMARY (U)

(U) Conceptual design and analytical studies were conducted to identify an advanced lift fan propulsion system (designated LFX) for the 1968-1970 time period, applicable to a U.S. Army surveillance and target acquisition mission.

(U) The objectives of the LFX propulsion system were based on previous contract parametric studies, aircraft design studies by airframe manufacturers, mission analyses, and the experience gained from the successful U.S. Army XV-5A flight research vehicle. The mission analyses and previous parametric studies were used to evaluate different levels of core engine performance and technologies and to determine optimum and desired characteristics for the convertible lift propulsion system. These studies identified a logical choice for the core engine and defined the lift fan objective dimensions and performance.

(U) The LFX propulsion system as conceived consists of two wing lift fans, two fuselage lift fans, and two core engines with diverter valves. Preliminary aerothermodynamic design and preliminary mechanical design were conducted on the wing lift fan. The principal features of the design follow:

(U) The turbine is a full admission turbine. The nominal arc of admission is 280 degrees. During power transfer, the arc of admission can vary between 200 and 360 degrees. The compressor is an advanced version of the X353-5 lift fan used in the XV-5A. The compressor pressure ratio is 1.253, and the tip speed is 946 feet per second. The front frame features a single fore-and-aft main strut and two secondary struts. A fabricated take-apart design is used which enables the use of selective materials and improves maintainability. The fore-and-aft main strut protrudes above the upper wing surface. The door actuation system is a worm gear drive, designed to minimize the holes in the hub. The rotor is straddle-mounted, a "high-flex" design, and is mechanically and dynamically similar to the J85/LF2. The rotor is mechanically designed for continuous operation at 115 percent of the nominal flow condition speed. The maximum allowable speed for intermittent operation is 120 percent, and the maximum speed for rotor containment is 130 percent. The scroll is also a take-apart design. A four-point mounting system is used. The power transfer system is located on the inboard section of the scroll. The rear frame is a composite structure of take-apart design. The rear frame is hung from the scroll, and remains concentric with the rotor. The exit louvers are similar in form to the XV-5A system, but are aerodynamically balanced to reduce the actuation requirements and weight.

(C) The design values of the principal performance parameters are compared to the objective values below (U):

	Objective	Design
Fan nominal pressure ratio	1.25	1.253
Fan tip speed, ft/sec	900-1000	946
Fan nominal flow, lb/sec	484	492
Turbine nominal flow, lb/sec	49.9	49.4
Fan tip diameter, inches	55.6	56.2
Turbine tip diameter, inches	61.2	63.1
Fan efficiency, percent	86	86.5
Turbine efficiency, percent	86	84.1
Fan inlet loss, $\bar{\omega}_{10}$, percent	10	6
Fan exit loss, $\bar{\omega}_{11}$, percent	4	5
Fan thrust coefficient, C_{V13}	.99	.99
Diverter valve leakage, percent	1.0	1.0
Uninstalled lift, guarantee, lb	10,225	10,480
Uninstalled lift, average, lb	10,540	10,800
Average lift per pound of turbine flow, lb/lb/sec	211.5	212.1

(U) In general, the design values of performance met or exceeded the objectives. The turbine efficiency was two points below the objective value. Techniques available for improving the turbine efficiency include increasing the design wheel speed, decreasing the leakage, and increasing the nominal arc of admission.

(U) The first full design cycle resulted in a weight of 674 pounds. This was primarily an extrapolation of J85/LF2 technology to the LFX design. The second design cycle involved materials changes and design changes, and resulted in a weight of 570 pounds. The design weights and the objective weights are compared below:

	Objective	First Design Cycle	Second Design Cycle
Front frame, lb	116	134	127
Rotor, lb	177	165	165
Scroll, lb	86	155	133
Rear frame, lb	87	150	100
Exit louvers, lb	40	70	45
	506	674	570

(U) The design weight of 570 pounds is 64 pounds above the objective weight.

(U) The preliminary design studies have identified areas of critical technology, that is, areas in which there is a need for advancement in lift fan technology to insure compatibility with the advanced core engines of the 1970 time period. As in the advancement of any technology, there are areas requiring detailed aerothermodynamic and mechanical design improvement, component verification, and materials development. These

areas are listed, and suggested study or testing is discussed. These critical technology areas include weight, installation size, turbine efficiency, compressor performance, fan doors, fan power modulation for control, composite structures, insulation, diverter valves, and scale-model and full-scale testing.

(U) Also examined were possible trade-offs and modification potential to minimize installation interface problems. Alternate designs are discussed for the diverter valve, scroll, exit louvers, frames, door actuation system, and wing cavity cooling. Layout drawings for three-engine aircraft are given should a change in mission require such a system.

(U) Recommendations are discussed for an exploratory development effort leading to a full-size demonstrator fan. Three phases of effort are suggested: Phase 1, continue the LFX studies and design-in-depth; Phase 2, final design component testing and manufacture; and Phase 3, demonstrator factory and wind tunnel testing.

(C) This program showed that two GE1/J1B turbojets will do the primary mission operating from a 90-degrees Fahrenheit/2500-foot vertical takeoff environment and cruising entirely at 0.9 Mach flight speed. Two LFX wing fans of 1.25 pressure ratio are the best system choice when all factors are considered. Choice of fuselage fan pressure ratio has only small effects on aircraft gross weights and performance and can therefore be an aerodynamic scale of the wing fan. The LFX system can perform the primary mission requirements at an aircraft vertical takeoff gross weight of about 20,000 pounds. Significant increases in payload or range can be obtained by reducing the Mach 0.9 flight speed requirement.

(U) The secondary mission, with a vertical takeoff at 95 degrees Fahrenheit/6000 feet, requires three full-size advanced GE1 core engines. There is no apparent advantage to using scaled engines or flat-rated fans.

(U) Objective lift performance of 10,540 pounds has been met within a 57-inch fan tip diameter. Objective fan weight without power transfer was established at 506 pounds, yielding a lift to weight ratio of 20.8. Initial preliminary design based on J85/LF2 mechanical design technology produced a fan weight of 674 pounds, including power transfer capability, and a lift of 10,800 pounds. Lift to weight ratio was 16.0. Additional design study aimed at reducing static component weights indicates feasibility of a 570-pound fan, including power transfer capability, with no performance compromise. This produced a lift to weight ratio of 19.0. Changes were limited to selection of better materials, minor improvements in properties of titanium sheet, and a revision in the connection of front and rear frame struts. Lift to weight ratios beyond 19 require radical departure from proven aerodynamic concepts and structural arrangements. A lift to weight ratio of 21 to 23 is

estimated to be an upper limit for the LFX fan design concept in the 10,000-pound size range. The achievement of these levels requires considerably increased aerodynamic risk coupled with liberal use of beryllium and composite materials.

(U) It should be emphasized that the context of lift to weight ratio as used here is conservative in comparison to other V/STOL lift propulsion devices. For example, the fan described above as 19 lift to weight ratio is capable of continuous full admission operation at 23 lift to weight ratio without exceeding any design limits.

(C) DEFINITION OF TARGETS AND OBJECTIVES (U)

(U) Objective targets and goals for the LFX propulsion system were established on the basis of previous contractor experience, mission studies, study information from aircraft manufacturers, and XV-5A experience.

(C) MISSION ANALYSIS (U)

(U) The performance of lift fan propulsion systems was evaluated in two specified missions. The first mission, called the LFX primary mission, was used to identify the core engine-lift fan system which became the basis of the LFX design activity. The secondary mission was used to compare the performance of the GE1/J1B cycle and an advanced GE1 core engine cycle in a different application. These two missions will be discussed separately.

(U) Definition of the LFX Primary Mission

The LFX primary mission was defined by:

1. The mission will be done on a sea level standard day.
2. The aircraft installed lift to weight ratio is 1.23.
3. The fuel allowance for warm-up and takeoff is based on 5 minutes at maximum power.
4. The cruise range is 200 nautical miles, at a cruise Mach number between 0.7 and 0.9.
5. The fuel allowance for landing and reserve is based on 3 minutes at maximum power.
6. The desired payload is 2500 pounds.

(U) Calculation Procedure for the Primary Mission

A computer program was written for this study, which performed the complete mission analysis calculation. For each case run, the program sized and weighed the propulsion system, calculated the wing geometry to contain the fan, calculated the aircraft drag and component weights, and determined the required mission fuel and the resulting payload capability. Two equal-size pitch fans having fan tip diameters of 30 inches were used. The core engines which were used included the J85/J1A1, the GE1/J1B, and an advanced GE1. Both full-size and scaled core engines were used.

(U) Ground Rules and Assumptions for the Primary Mission

1. All bulletin values of specific fuel consumption are increased 5 percent.
2. There is a crew of two, weighing a total of 400 pounds.

3. The aircraft uses a four-fan arrangement consisting of two wing fans, one nose fan, and one tail fan.

4. The wing was scaled from the wing described in reference 18 by maintaining the same geometric relationship between the wing and the fan, as illustrated in Figure 5.

5. The ratio of wing fan disc loading to wing loading has a minimum allowable value of 7.5 (This is an extrapolation of XV-5A experience, to insure the capability of transition to 1.2 stall speed.)

6. There is no range credit or fuel debit for airplane accelerations and decelerations between mission segments.

7. All cruise performance is done with two core engines.

8. Core engine cruise performance includes the thrust loss due to diverter valve effects given in reference 1.

9. The required angular acceleration rates for aircraft control are: pitch axis, 0.53 radian/second^2; roll axis, 1.8 radians/second^2; and yaw axis, 0.4 radian/second^2.

10. Diverter valve weights are scaled proportional to size.

11. Core engine weights are scaled proportional to size if smaller than the design size, and proportional to size to the 1.2 power if larger than the design size.

12. The core engine and lift fan performance and weight data used in this study are documented in reference 7.

13. The aircraft design gross weight is equal to the takeoff gross weight less 1000 pounds.

14. The aircraft parasite drag factor (f_o) was scaled from the data of reference 18 using the relation:

$$f_o = 2 + 0.0104S$$

The symbol S in the above equation denotes the wing planform area. The compressibility drag rise at a flight Mach number of 0.9 was assumed to be 25 percent.

15. The ratio of aircraft uninstalled lift to vertical takeoff gross weight is 1.37, based on the sum of these four factors:

Basic lift to weight at sea level standard day:	1.23
Installation allowance:	0.05
Control and trim allowance:	0.04
Reingestion allowance:	0.05
	1.37

16. The aircraft component weights are scaled from those given in reference 18. The aircraft weight statement is given in Table I.

TABLE I (U)
AIRCRAFT WEIGHT STATEMENT - PRIMARY MISSION

STRUCTURE	
Fuselage	11.8% DGW
Wing and Tails	6.8 lbs/ft^2 S
Landing Gear	5% DGW
Engine Section	50 lbs
Surface Controls	4.5% DGW
	W_{STR}
PROPULSION	
Engines	W_E
Controls and Accessories	65 lbs/engine
Inlet and Exhaust	400 lbs
Wing Fans (2)	W_{WF}
Installation and Ducting	150 lbs
Nose Fan (1)	W_{NF}
Installation and Ducting	215 lbs
Tail Fan (1)	W_{TF}
Installation and Ducting	300 lbs
Diverter Valves	W_{DV}
Lube and Fuel Systems	300 lbs
Trapped Fluids	100 lbs
	W_{PROP}
FIXED EQUIPMENT	
Hydraulic/Electrical Systems	5% DGW
Electronics	275 lbs
Instrumentation and Controls	50 lbs
Air Conditioning and Anti-Ice	200 lbs
Crew Furnishings	520 lbs
Crew Gunfire Protection	100 lbs
	W_{FE}
CREW	
Two Men	400 lbs
OPERATING EMPTY WEIGHT	$(W_{STR} + W_{PROP} + W_{FE} + W_{CREW})$

(C) Results for the Primary Mission (U)

(C) Effect of Core Engine Cycle on Mission Performance (U)

(C) Figure 1 shows the payload capability of three core engines in the primary mission. The J85 family of core engines is too small for this application. The largest J85 engine, the J85/J1A1, is too small even when scaled on one-and-one-half size. The GE1/J1B is a good choice for this application. A GE1/J1B aircraft with 1.3 wing fan pressure ratio can do the mission, and aircraft with wing fan pressure ratios of 1.2 or 1.25 offer a design margin. The advanced GE1 core engines are too large for this application, and offer growth capability to the GE1/J1B system.

(C) Full-Size GE1/J1B Core Engines (U)

(U) Figure 2 shows the payload capability of an aircraft powered by two full-size GE1/J1B core engines in the primary mission. The data are presented as lines of constant values of wing fan pressure ratio. For each line, the value of pitch fan pressure ratio, indicated by the symbols, is varied.

(C) The maximum value of payload is 3400 pounds, at a vertical takeoff gross weight of 21,700 pounds, and is obtained by using a wing fan pressure ratio of about 1.2 and a pitch fan pressure ratio of about 1.15. The required value of 2500 pounds of payload is obtained at a vertical takeoff gross weight of 19,600 pounds by using a wing fan pressure ratio of about 1.3 and a pitch fan pressure ratio of 1.15 or 1.2.

(C) Scaled GE1/J1B Core Engines (U)

(U) Figure 3 shows the effect of engine scaling for an aircraft powered by two GE1/J1B core engines in the primary mission. The upper curve shows the vertical takeoff gross weight capability as a function of engine size. The lower curve shows the corresponding payload capabilities of these scaled engine systems.

(C) Figure 4 summarizes the comparison of full-size and scaled GE1/J1B core engines. The full-size engine system has the payload and vertical takeoff gross weight capabilities shown by the dashed lines. Scaling the engines to the size required to meet the 2500-pound payload requirement yields the solid lines of required engine size and vertical takeoff gross weight capability. The minimum size of the GE1/J1B in a two-engine aircraft which can do the primary mission is 0.88, at a vertical takeoff gross weight of 18,900 pounds, using a 1.2 wing fan pressure ratio.

(U) Wing Loading Requirements

(U) Shown in Figure 5 is the wing from reference 18, which was used as the reference wing in this study. The computer program was written to scale

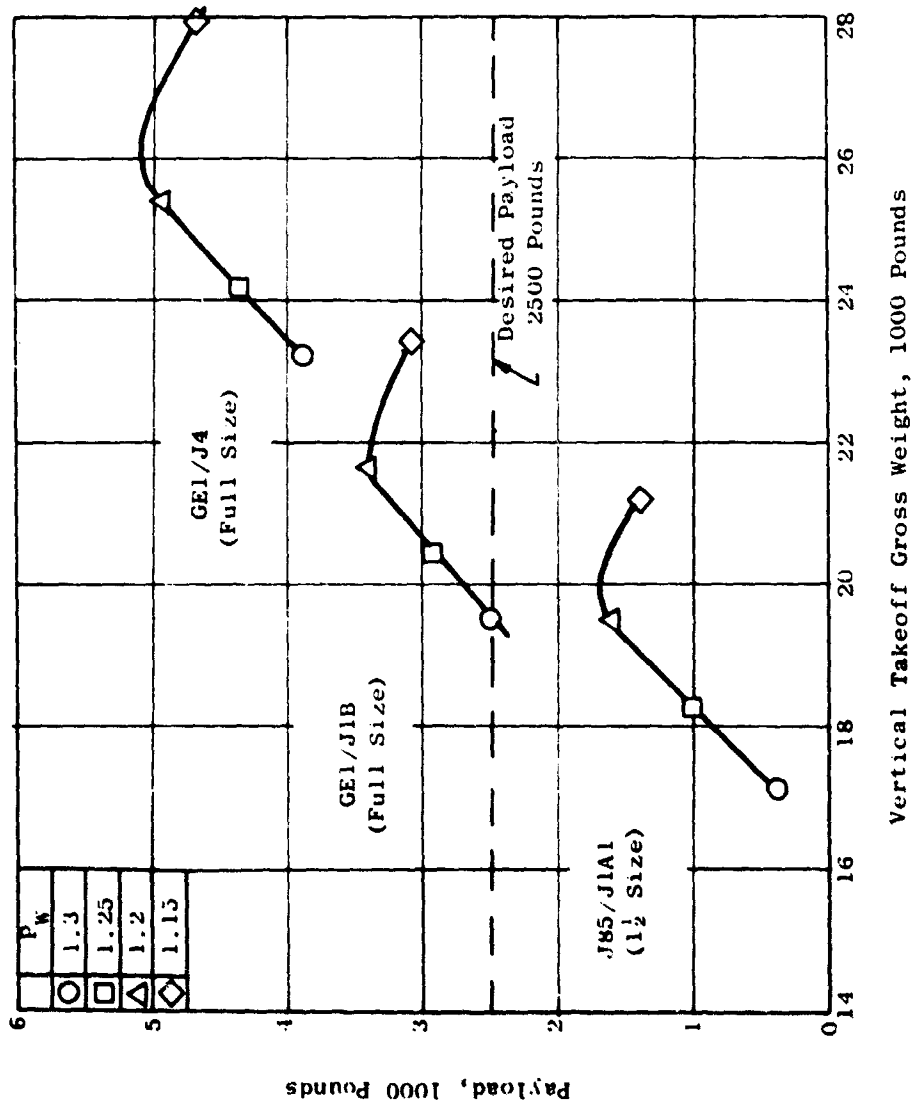

Figure 1. (C) Payload Versus Vertical Takeoff Gross Weight for Three Core Engines in the Primary Mission. (U)

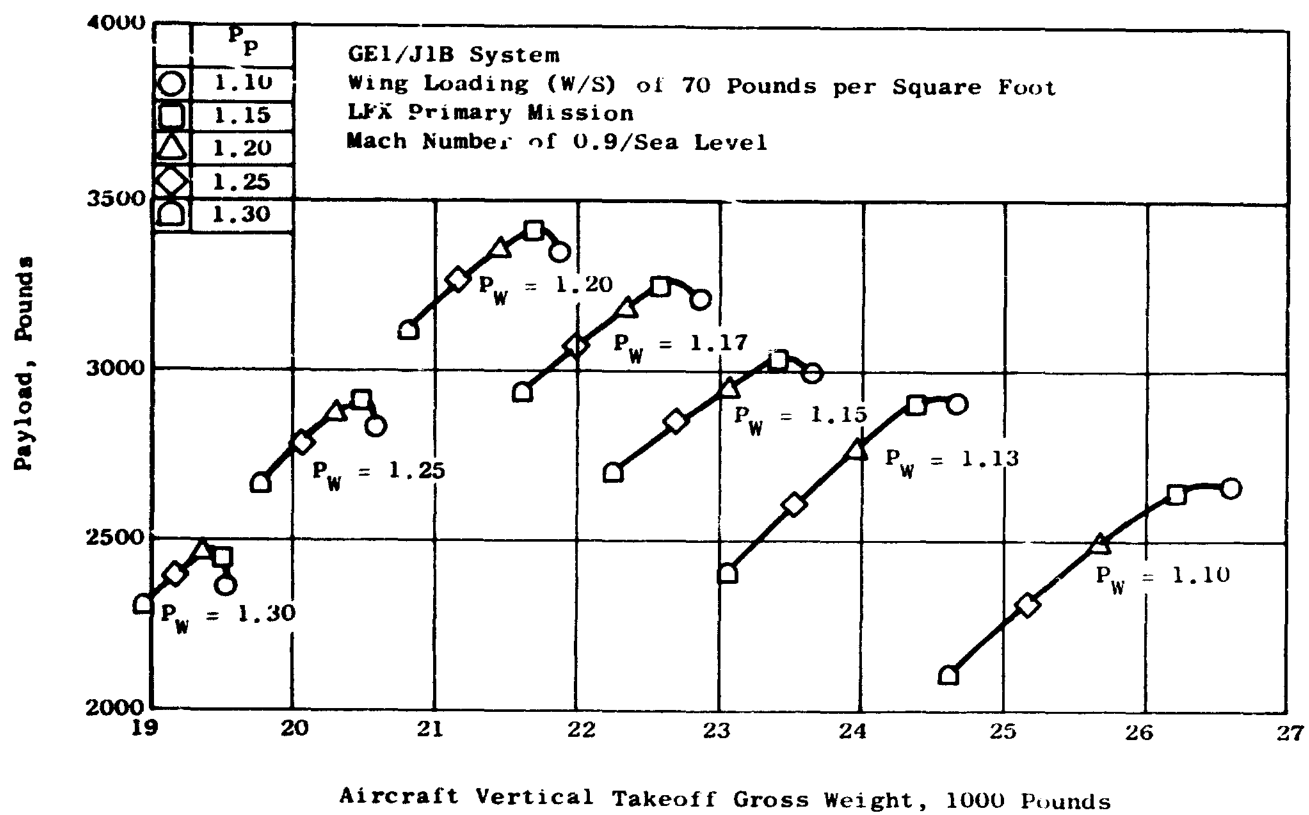

Figure 2. (C) Payload Versus Vertical Takeoff Gross Weight for Full-Size GE1/J1B Core Engines in the Primary Mission. (U)

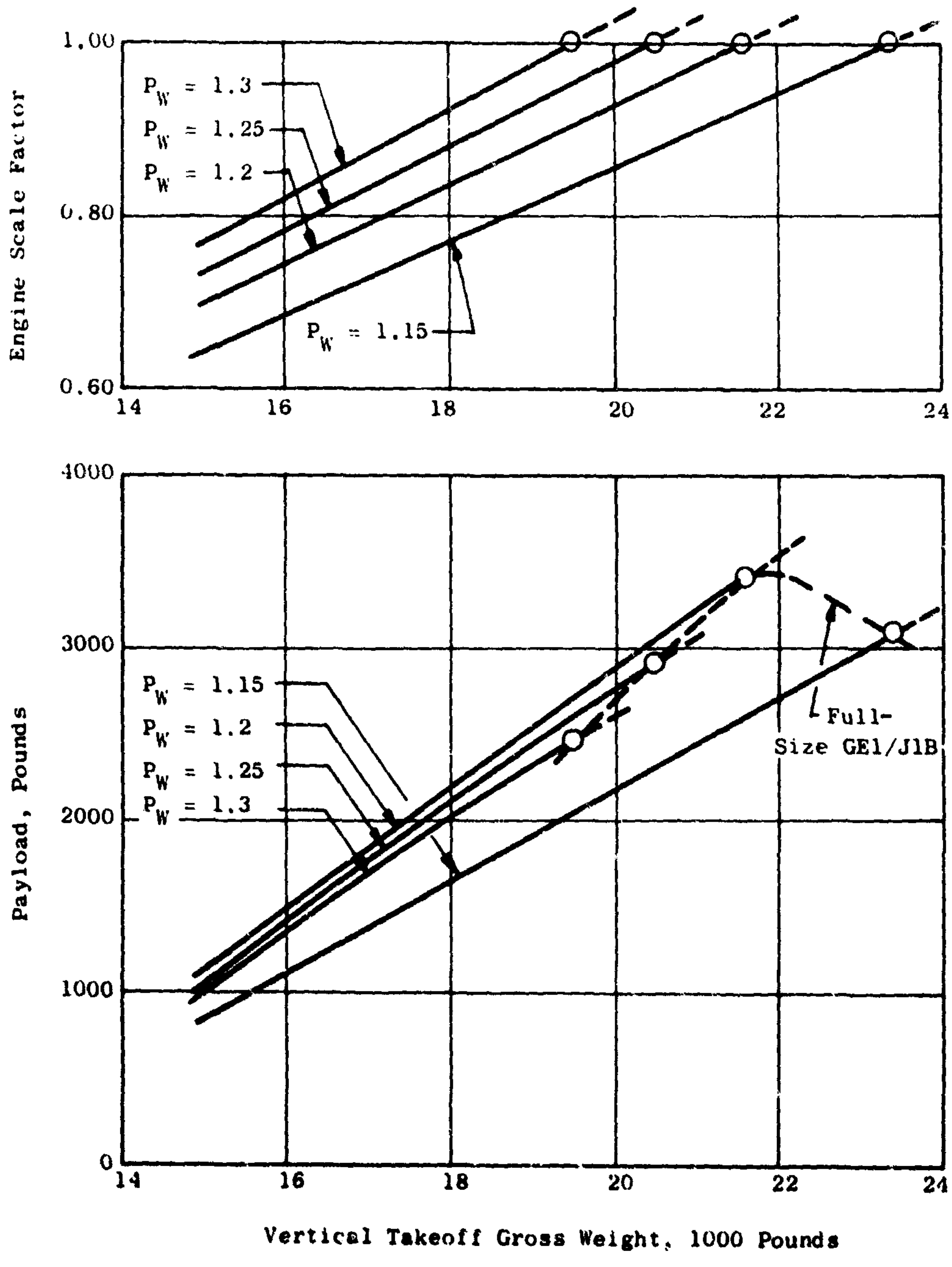

Figure 3. (C) GE1/J1B Scaled Engine Performance in the Primary Mission. (U)

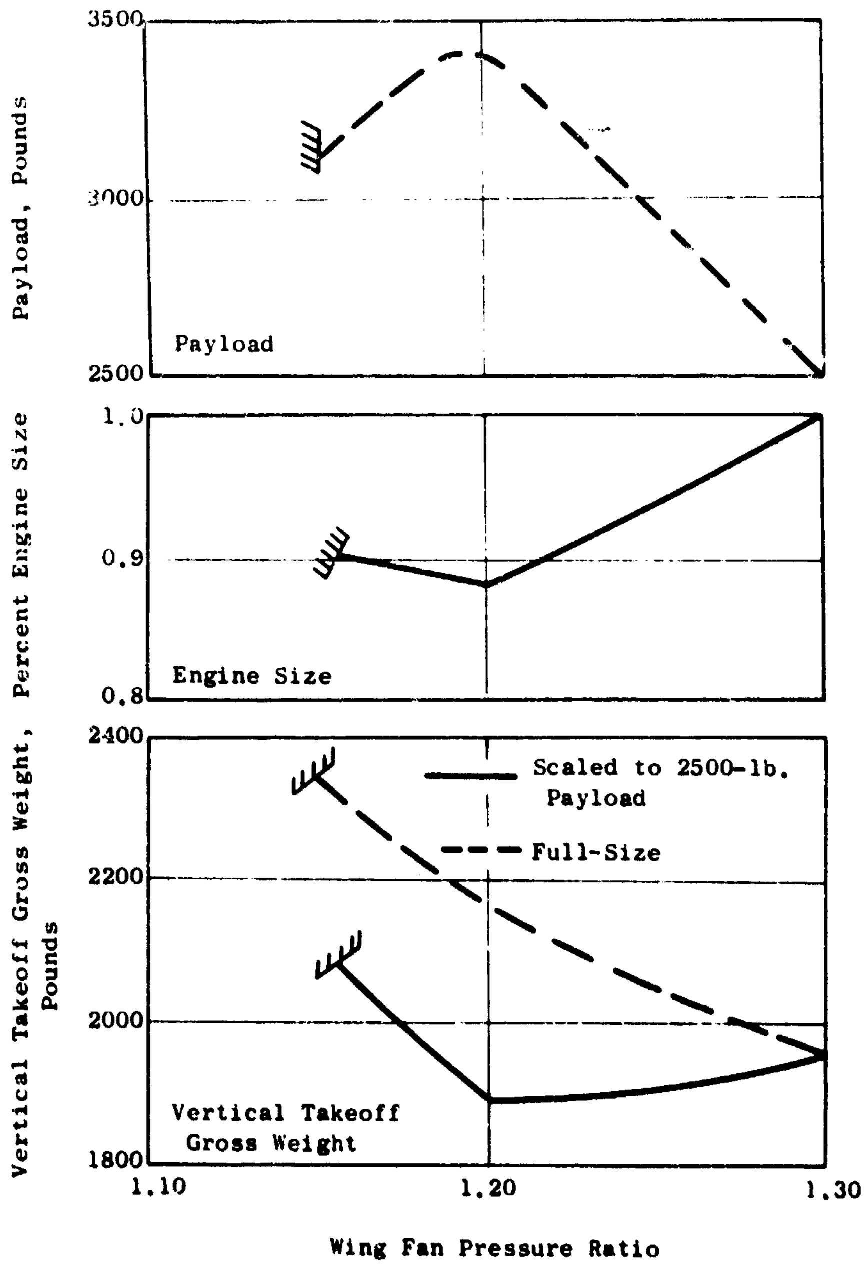

Figure 4. (C) Performance Comparison Between Scaled and Full-Size GE1/J1B Core Engines in the Primary Mission. (U)

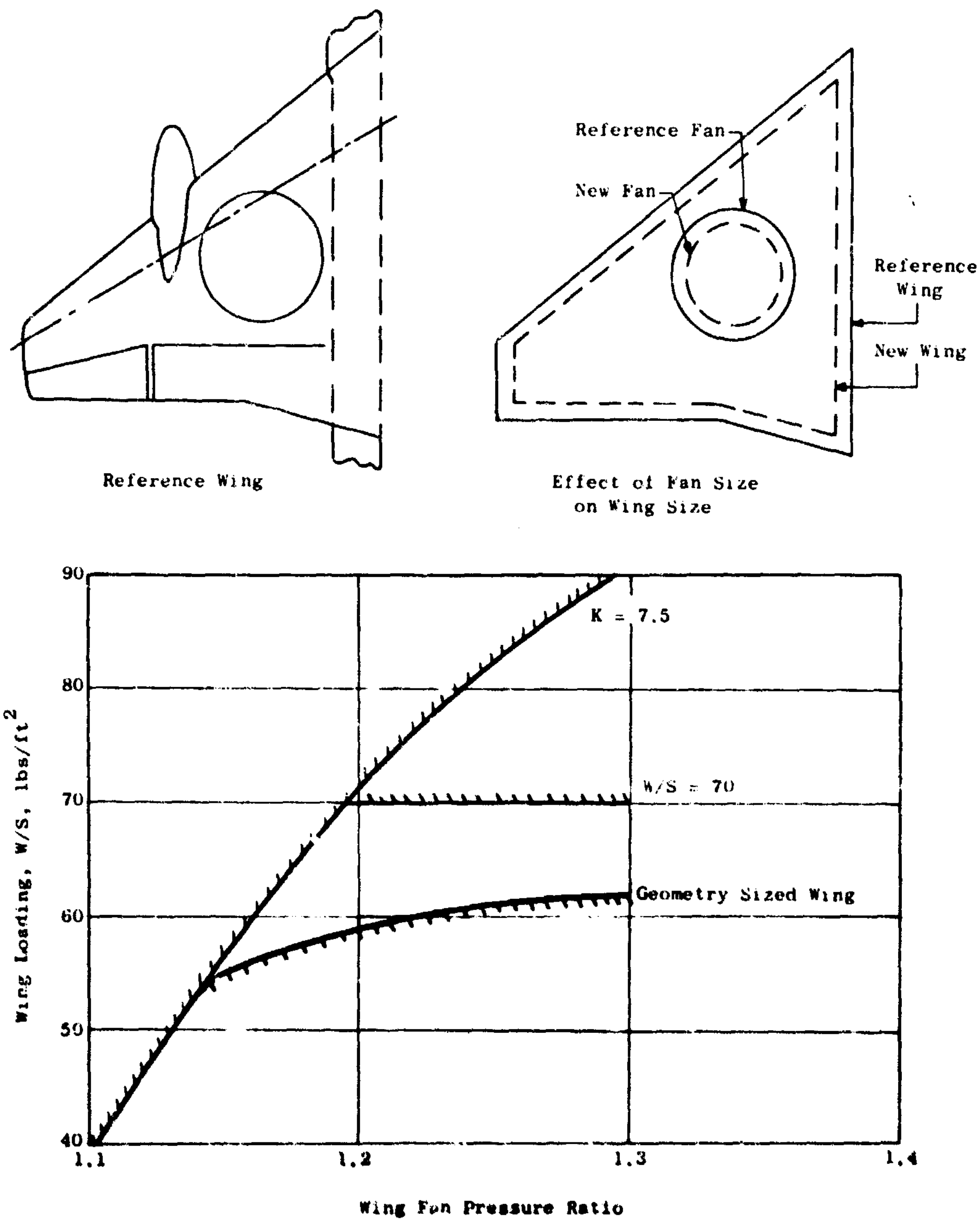

Figure 5. (U) Aircraft Wing Loading Requirements in the Primary Mission.

the wing geometry to the fan as shown. The resulting wing loading capability for this geometry-sized wing is shown on the curve. Also shown on the curve are values of wing loading based on the additional requirement of assumption 5: that the ratio of wing fan disc loading to wing loading (K) must have a minimum value of 7.5. A third line for W/S = 70 pounds per foot^2 is also shown. The wing loading can be increased to 70 pounds per foot^2 by allowing the wing geometry to change. The selection of W/S = 70 was arbitrary, but should represent a practical upper limit.

(U) The wing loading for a practical LFX vehicle, then, will have a value within the region on the curve bounded by the K = 7.5 line on the left, with the W/S = 70 line and the geometry line as upper and lower limits.

(U) Effect of Wing Loading on Mission Performance

Figure 6 shows the performance in the primary mission for an aircraft powered by two full-size GE1/J1B core engines. Shown in the upper curve is the effect of wing loading on the payload capability for a Mach 0.9 flight speed. The line for W/S = 70 represents a line drawn through the peak values of the curves shown in Figure 2. The lines converge on the left to a single line where K becomes the limiting parameter. It is seen that the wing fan pressure ratio for which the maximum payload occurs is a function of wing loading.

Shown on the lower half of Figure 6 is the payload capability as a function of wing fan pressure ratio and mission flight speed, for two wing designs: W/S = 70 and W/S based on geometry. The effect of increases in mission flight speed is to reduce the payload capability because of the higher fuel requirements.

(U) Cruise Thrust Requirements

The aircraft were sized only by the available lift. Figure 7 shows the resulting cruise thrust requirements of the two-engine GE1/J1B aircraft in the primary mission at a flight Mach number of 0.9.

The left-hand ordinate shows the required thrust per engine in pounds; the right-hand ordinate shows this as a percentage of the military thrust. Two lines are shown, for a W/S of 70 pounds per foot^2 and for the geometry-sized wing. This figure shows that aircraft using wing fan pressure ratios less than 1.15 (above 23,000 pounds vertical takeoff gross weight) are thrust-limited. Aircraft using wing fan pressure ratios between 1.2 and 1.3 have a good cruise thrust match and a reasonable acceleration capability.

(U) <u>Definition of the LFX Secondary Mission</u>

The LFX secondary mission was defined by:

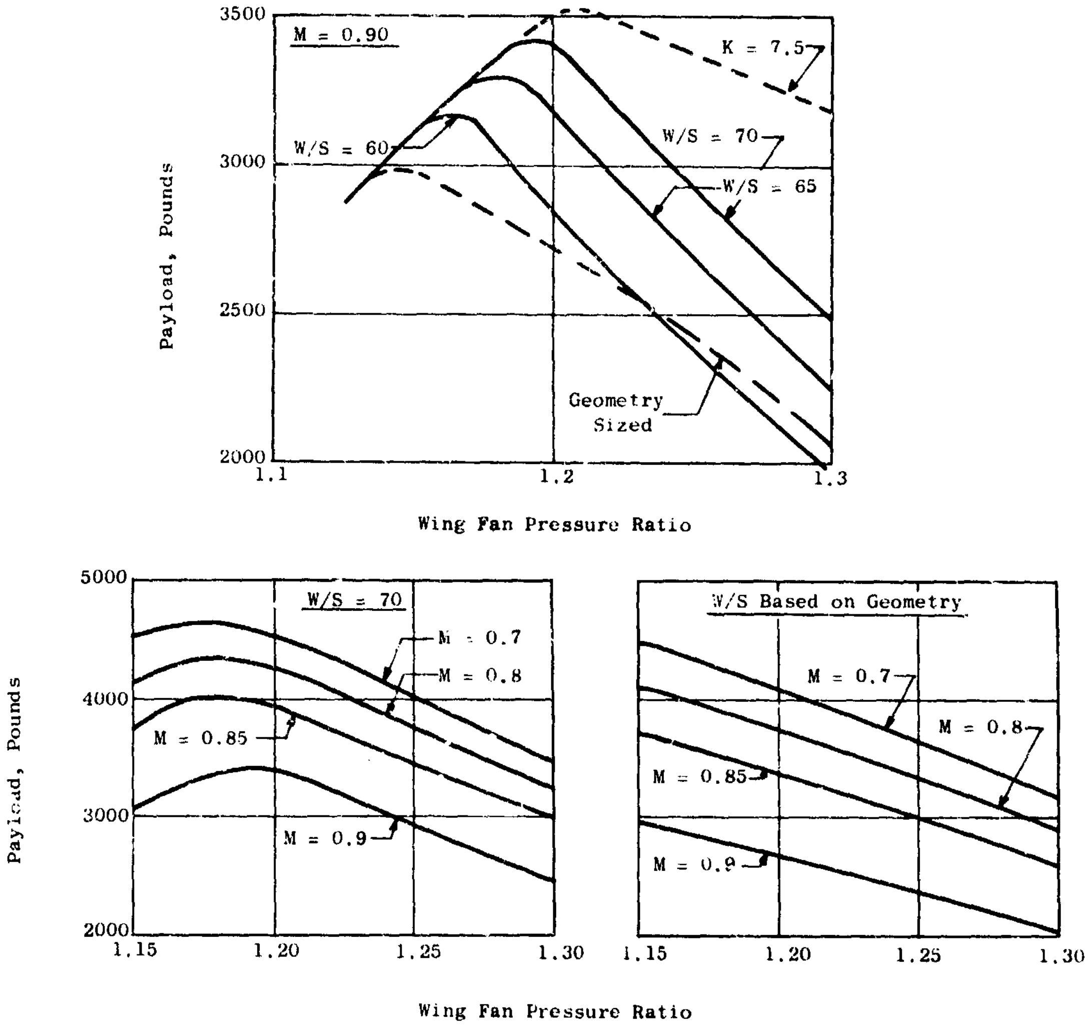

Figure 6. (C) Effect of Wing Loading and Flight Speed on GE1/J1B Core Engine Performance in the Primary Mission. (U)

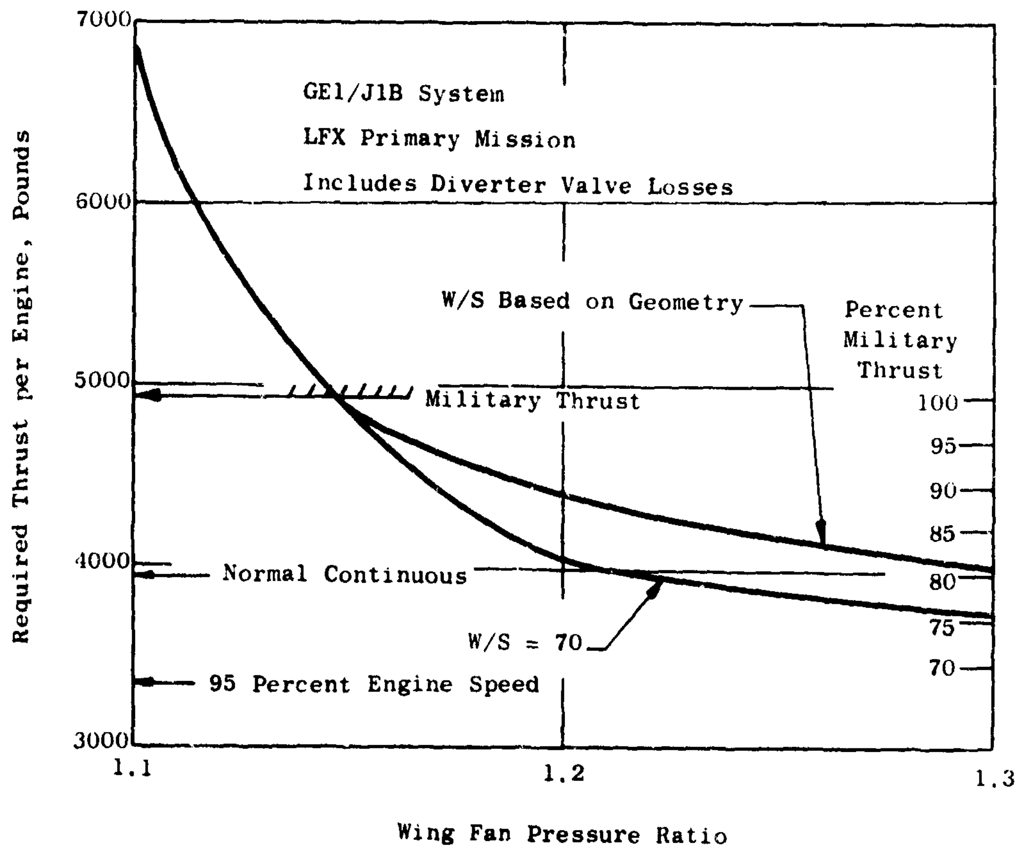

Figure 7. (C) Cruise Thrust Requirements for the GE1/J1B Core Engine in the Primary Mission. (U)

1. The aircraft lift capability is to be calculated at an environment of 95 degrees Fahrenheit/6000 feet.

2. The fuel allowance for warm-up, takeoff, landing and reserve is equivalent to the amount of fuel required for 10 minutes of hover on a sea level standard day at the vertical takeoff gross weight.

3. The aircraft will cruise for 30 minutes at 300 knots, and for 30 minutes at 450 knots, at sea level.

4. The aircraft will loiter for 20 minutes at best endurance speed at sea level.

5. The desired payload is 2800 pounds, plus an assumed 30 percent installation weight, or 3640 pounds total.

(U) Calculation Procedure for the Secondary Mission

The computer program used for the primary mission was rewritten for this study. Included in the program were the aircraft control force requirements and the temperature and altitude effects on lift. The program matched the size of the pitch fans to the control force requirements shown in Figure 20.

This study used aircraft powered by two core engines (both full-size and scaled) and three core engines (full-size, with the third engine being used only for fan mode operation), using the GE1/J1B and an advanced GE1.

The secondary mission vertical takeoff environment was given as 95 degrees Fahrenheit/6000 feet. This study presents mission capability for 95 degrees Fahrenheit/6000 feet; 95 degrees Fahrenheit/4000 feet; 90 degrees Fahrenheit/2500 feet; and sea level standard day.

Two lift fan design points (sea level standard day and 95 degrees Fahrenheit/6000 feet) are used to determine the effect of fan design point on payload capability.

(U) Ground Rules and Assumptions for the Secondary Mission

The first 10 ground rules and assumptions listed previously for the primary mission are also applicable to the secondary mission. To avoid repetition, they will not be repeated here. The additional ground rules and assumptions are as follows:

1. Core engine weights are scaled proportional to size if smaller than the design size, and proportional to size to the 1.26 power if larger than the design size. (The scaling exponent 1.26 reflects the most recent information for GE1 engine scaling.) Core engine installation (inlet and exhaust) weight is scaled proportional to engine size.

2. The lift fan performance and weight data used in this study are documented in reference 14. The installed fan diameter (maximum

diameter) is shown in Figure 8 as a function of fan size and pressure ratio, and was obtained from reference 2.

3. The aircraft design gross weight is equal to 95 percent of the vertical takeoff gross weight. (The original assumption that the design gross weight is equal to the vertical takeoff gross weight less 1000 pounds is not valid for this secondary mission, because of the larger size of the aircraft.)

4. The aircraft parasite drag factor (f_o) was scaled from the data of reference 18, using the relation:

$$f_o = 3.64 + 0.0085S$$

The symbol S in the above equation denotes the wing planform area. (The larger aircraft sizes for the secondary mission explain the higher drag factors.)

5. The required ratio of the aircraft uninstalled lift to vertical takeoff gross weight is shown in Figure 24 as a function of vertical takeoff environment and gross weight. This ratio is the product of four factors: the effect of vertical takeoff temperature and altitude on thrust (from reference 1); installation allowance (5 percent); vertical acceleration allowance (5 percent); and aircraft control requirements (Figure 23).

6. The aircraft component weights are scaled from those given in reference 18. The aircraft weight statement is given in Table II. (Five differences in assumptions can be noted between the weight statements for the two missions. Specifically: the fuselage weight and the tail-fan installation weight are larger because of the larger aircraft and fans used; for convenience, the landing gear pod weight has been separated from the wing weight; the instrumentation and controls allowance has been assumed to be independent of payload and therefore has been increased; and the crew gunfire protection allowance has been increased to 200 pounds.)

(C) Results for the Secondary Mission (U)

(C) The GE1/J1B Core Engine (U)

(U) Figure 9 shows the payload capability versus vertical takeoff gross weight for two and for three full-size engines. Each curved line represents constant vertical takeoff temperature and altitude. Each symbol represents a different value of wing fan pressure ratio; these are connected by straight lines. Each of these curved lines represents 25 calculated points, 5 values of wing fan pressure ratio, and 5 values of pitch fan pressure ratio, where only the optimum values of pitch fan pressure ratios are used. These optima occurred for pitch fan pressure ratios between 1.1 and 1.15. The change in payload with vertical takeoff temperature and altitude follows directly from the effect shown in

TABLE II (U)
AIRCRAFT WEIGHT STATEMENT - SECONDARY MISSION

STRUCTURE	
Fuselage	12% DGW
Wing and Tails	6.46 lbs/ft^2 S
Landing Gear	5% DGW
Landing Gear Pods	100 lbs
Engine Section	25 lbs/engine
Surface Controls	4.5% DGW
	W_{STR}
PROPULSION	
Engines	W_E
Controls and Accessories	65 lbs/engine
Inlet and Exhaust	400 lbs
Wing Fans (2)	W_{WF}
Installation and Ducting	150 lbs
Nose Fan (1)	W_{NF}
Installation and Ducting	300 lbs
Tail Fan (1)	W_{TF}
Installation and Ducting	300 lbs
Diverter Valves	W_{DV}
Lube and Fuel Systems	300 lbs
Trapped Fluids	100 lbs
	W_{PROP}
FIXED EQUIPMENT	
Hydraulic/Electrical Systems	5% DGW
Electronics	275 lbs
Instrumentation and Controls	150 lbs
Air Conditioning and Anti-Ice	200 lbs
Crew Furnishings	520 lbs
Crew Gunfire Protection	200 lbs
	W_{FE}
CREW	
Two Men	400 lbs
OPERATING EMPTY WEIGHT	$(W_{STR} + W_{PROP} + W_{FE} + W_{CREW})$

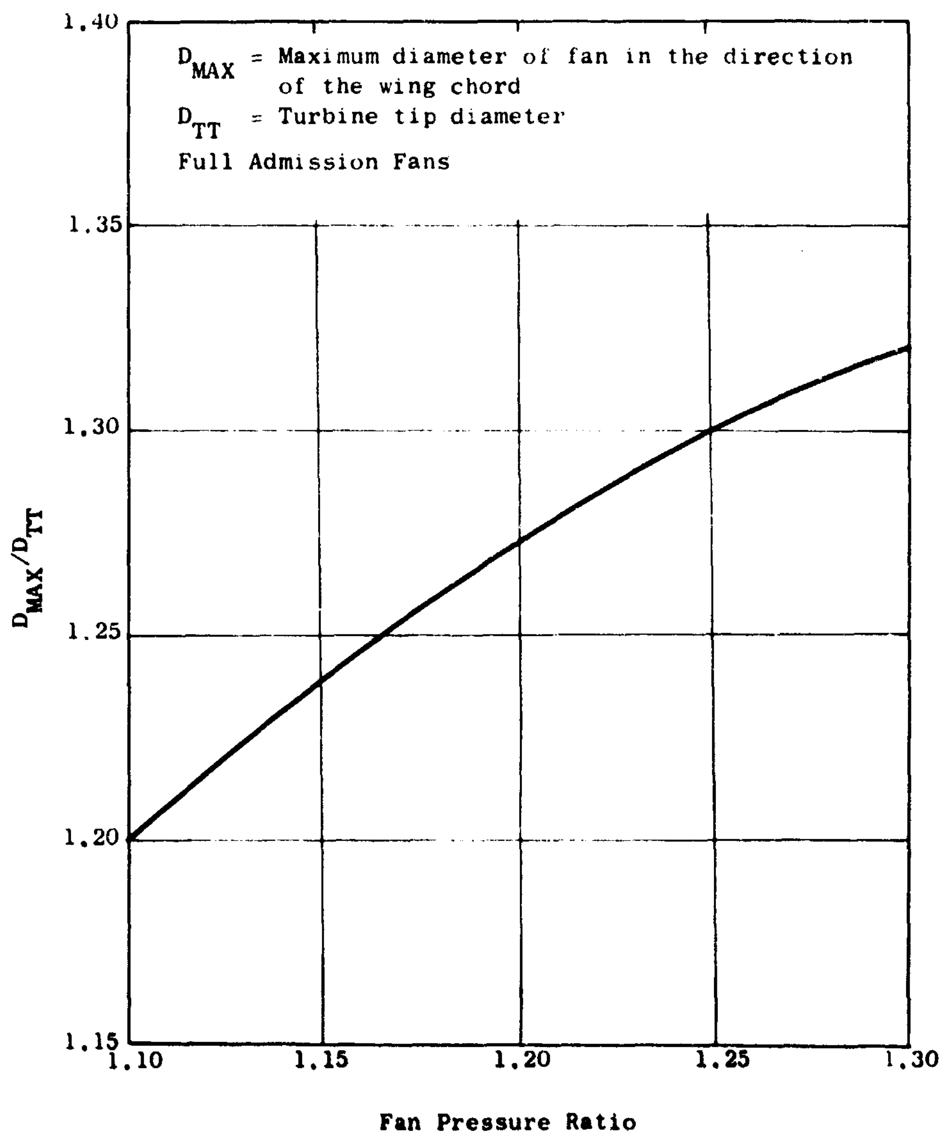

Figure 8. (U) Fan Maximum Diameter Versus Wing Fan Pressure Ratio.

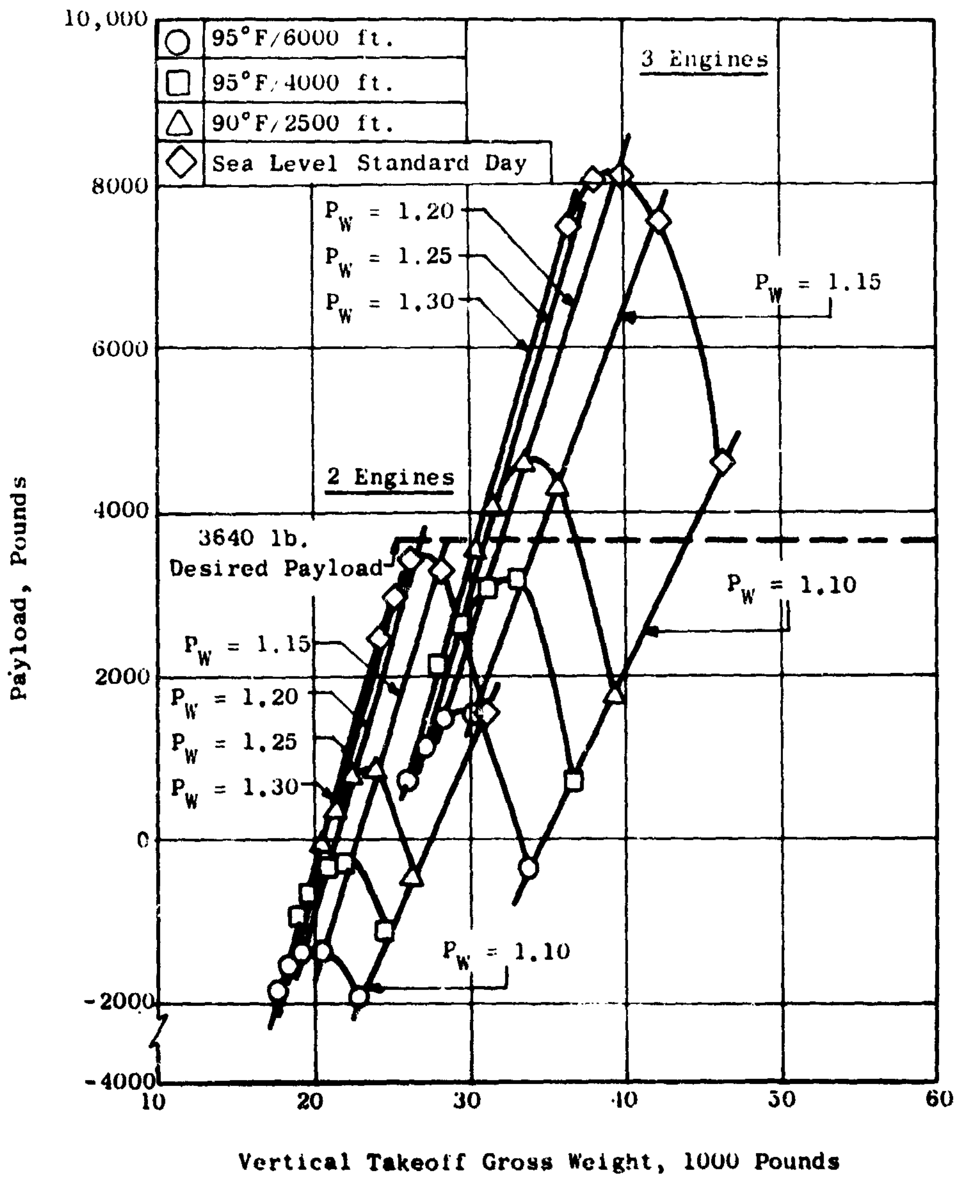

Figure 9. (C) Payload Versus Vertical Takeoff Gross Weight for Full-Size GE1/J1B Core Engines in the Secondary Mission. (U)

Figure 24. The sharp drop-off of the 1.1 wing fan pressure ratio points is caused by the over-sized wing required by the assumption that the ratio of wing fan disc loading to wing loading has a minimum value of 7.5.

(C) The desired payload is 3640 pounds. Neither a two-engine nor a three-engine GE1/J1B aircraft can do the mission at the specified vertical takeoff environment of 95 degrees Fahrenheit/6000 feet. A two-engine GE1/J1B aircraft can almost (3400 pounds payload) do the mission on a sea level standard day. A three-engine GE1/J1B aircraft can do the mission at a vertical takeoff environment of 90 degrees Fahrenheit/2500 feet, and exceeds the payload requirements on a sea level standard day.

(U) The Advanced GE1 Core Engine

Figure 10 shows the payload capability versus vertical takeoff gross weight for two and three full-sized engines. The previous discussion of Figure 9 applies here; the same trends and effects are present. The two-engine aircraft exceeds the mission requirements only on a sea level standard day. A three-engine aircraft can do the mission at the required environment of 95 degrees Fahrenheit/6000 feet. The minimum vertical takeoff gross weight to do the secondary mission at 95 degrees Fahrenheit/6000 feet, is 35,000 pounds, using a 1.2 wing fan pressure ratio and three advanced GE1 core engines. At vertical takeoff environments less severe than 95 degrees Fahrenheit/6000 feet, the three-engine aircraft are very large and are beyond the area of interest for this study.

(C) Scaled Engines (U)

(U) Figure 11 shows the payload versus vertical takeoff gross weight for scaled engines for both core engine cycles. Two vertical takeoff environments are shown: 95 degrees Fahrenheit/6000 feet; and 95 degrees Fahrenheit/4000 feet. The curved lines at the bottom of the scaled engine lines are for full-size engines. The optimum wing fan pressure ratio for scaled engines was found to be 1.2. Table III summarizes the results for 3640 pounds payload.

TABLE III (C)
PERFORMANCE OF SCALED ENGINES IN THE SECONDARY MISSION (U)

Core Engine	Vertical Takeoff Temperature/Altitude (°F/ft.)	Required Vertical Takeoff Gross Weight (lbs.)	Required Engine Size
GE1/J1B (2)	95/6000	42,000	2.18
	95/4000	36,000	1.75
Advanced GE1 (2)	95/6000	38,000	1.68
	95/4000	34,000	1.38

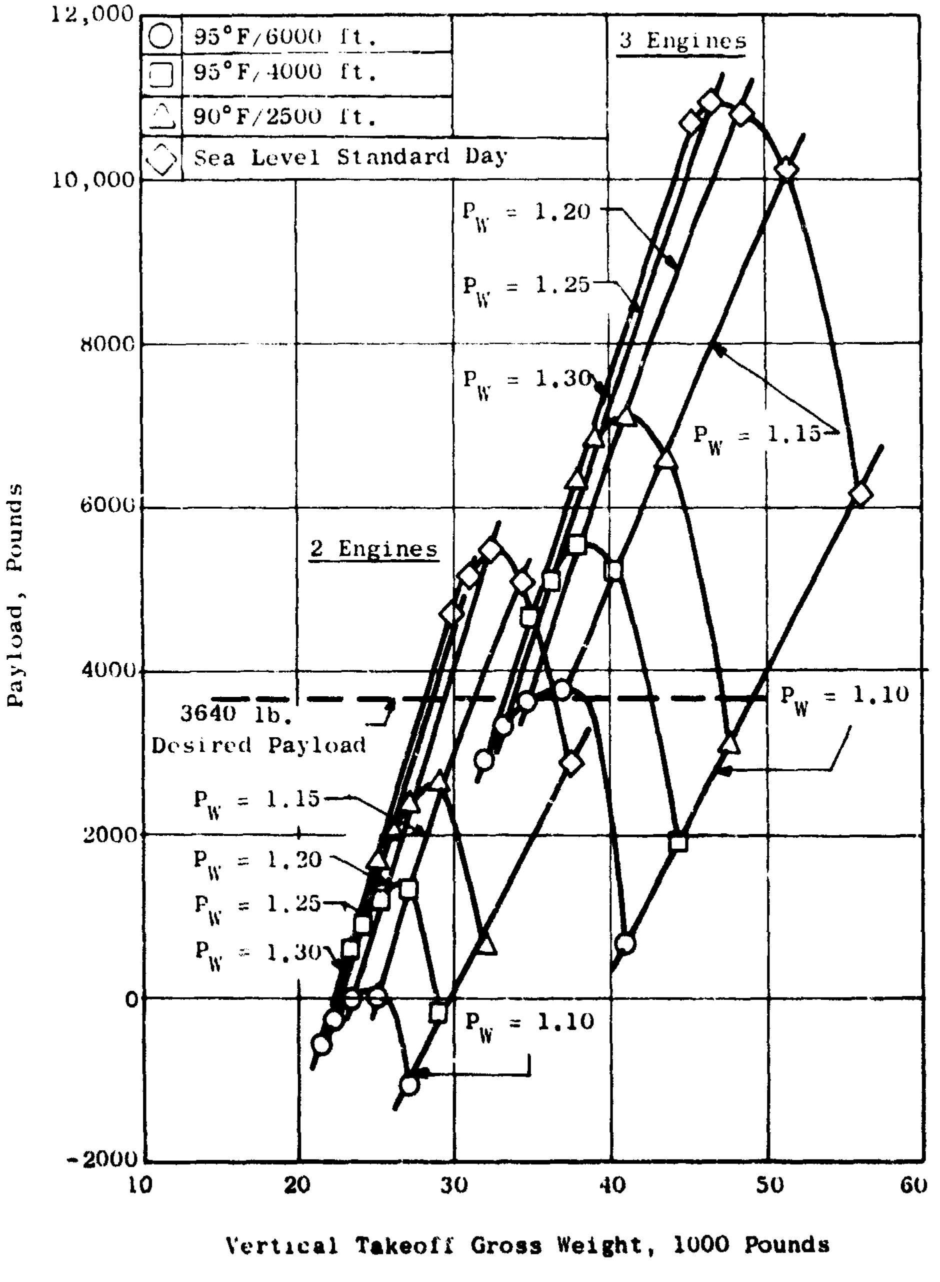

Figure 10. (C) Payload Versus Vertical Takeoff Gross Weight for Full-Size Advanced GE1 Core Engines in the Secondary Mission. (U)

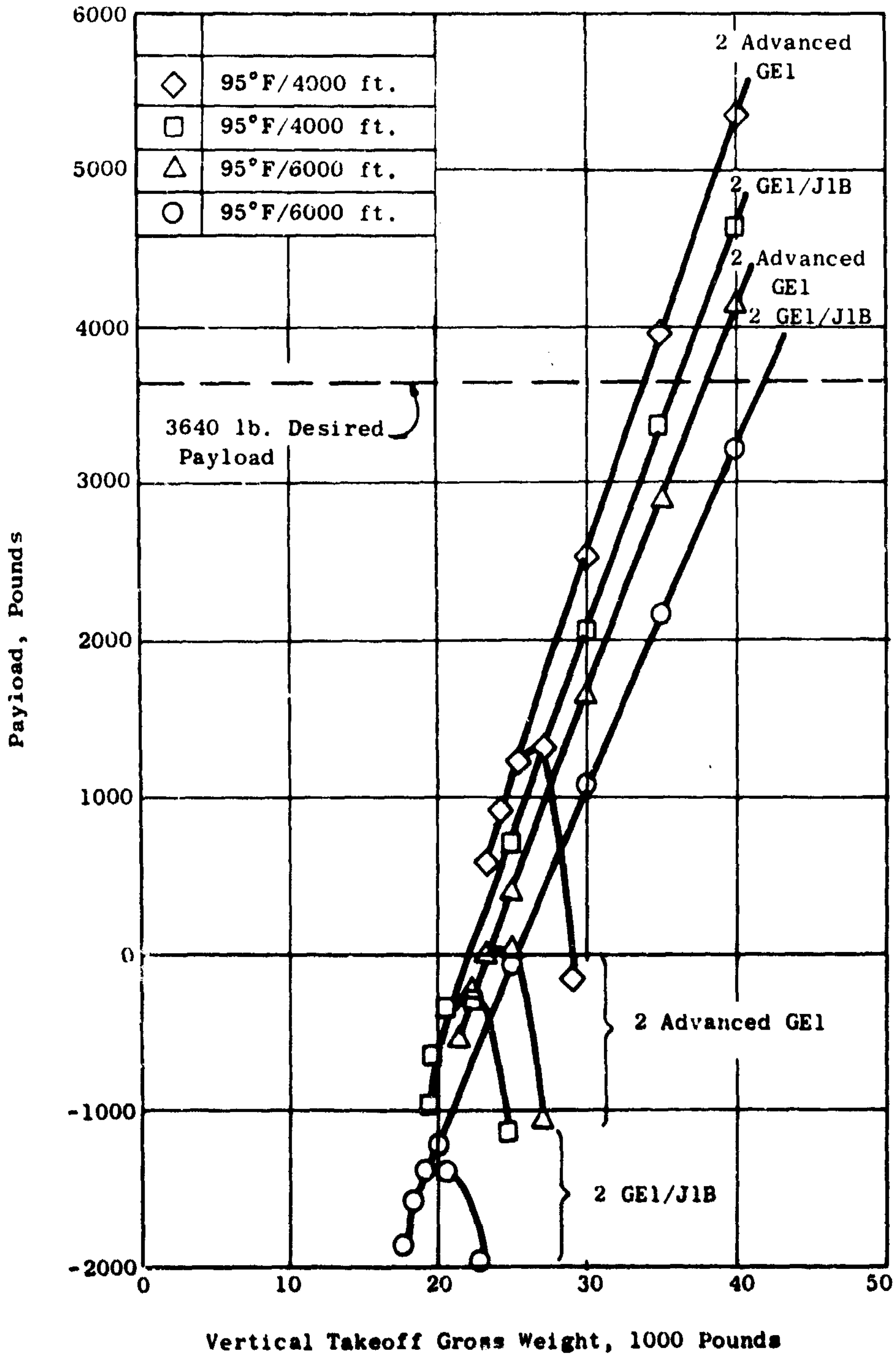

Figure 11. (C) Payload Versus Vertical Takeoff Gross Weight for Scaled Engines in the Secondary Mission. (U)

(C) Table III shows that two GE1/J1B core engines need to be scaled to unrealistically large sizes to do the mission. The table also shows that two advanced GE1 core engines need to be scaled to 1.68 size to do the mission at 38,000 pounds vertical takeoff gross weight; while from Figure 10, three full-size engines can do the mission at only 35,000 pounds vertical takeoff gross weight. Therefore, there is no advantage to using scaled engines in the secondary mission.

(U) Effect of Fan Design Point

The effect of the lift fan design point on mission performance was investigated using the GE1/J1B core engine, by comparing two fan design points: sea level standard day, and 95 degrees Fahrenheit/6000 feet. The fan performance and weight for each design point were determined using existing computer programs for lift fan design point cycle analysis and for lift fan weight calculations. The core engine discharge conditions of flow, pressure and temperature at the two design points were inputs to the programs.

The sea level standard day fans can be derated to run off-design at 95 degrees Fahrenheit/6000 feet by using the thrust lapse data of reference 1. The 95 degrees Fahrenheit/6000 feet design point fans cannot be uprated using the data of reference 1 to gain lift at less severe temperature and altitude conditions. These fans are essentially flat-rated, in that they have been stress-limited to the 95 degrees Fahrenheit/6000 feet conditions of loading, and must be run with reduced core engine power settings when operated in less severe vertical takeoff environments.

The study showed that there is some weight savings from using fans designed for the exact vertical takeoff environment rather than using off-loaded fans that have been designed for a more stringent requirement. Figure 12 shows the payload versus vertical takeoff gross weight comparison of the two fan design points. The flat-rated fans gain about 200 pounds payload. A similar result is expected using advanced GE1 core engines.

The payload gained by using flat-rated fans is not sufficient to enable the GE1/J1B aircraft to do the mission. Further, the approximately 200 pounds of payload gained are not without penalty. Because the vertical takeoff gross weight capability of the flat-rated fans is constant, there is no over-load capability in less severe vertical takeoff ments. If the aircraft is to be in a 95 degrees Fahrenheit/6000 feet environment for only a small part of its operating time, a flat-rated vertical takeoff gross weight capability may not be desirable.

(C) Comparison of the Primary and Secondary Missions (U)

(U) Figure 13 compares the mission performance of the primary and secondary missions. Shown are the performance of the three core engines from Figure 1 for the primary mission, and the performance of the two core

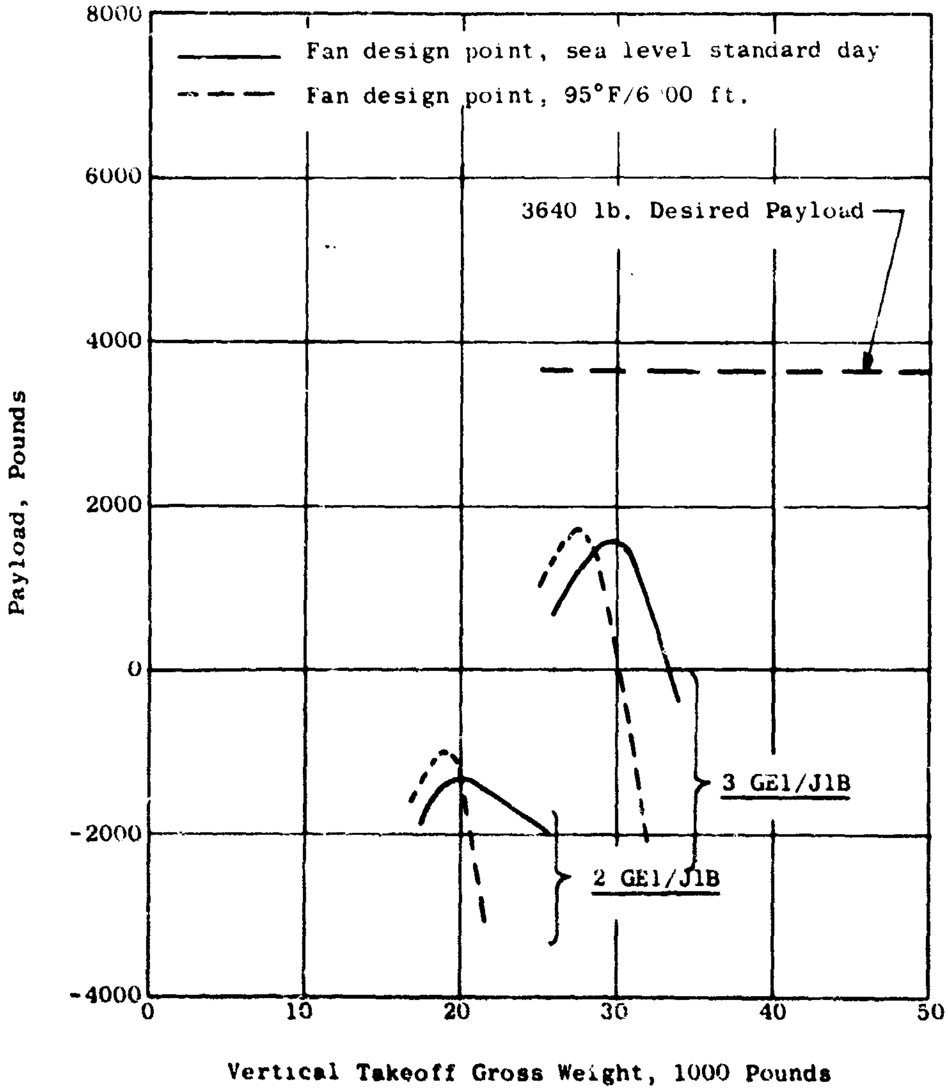

Figure 12. (C) Comparison of Sea Level Standard Day Fans and 95°F/6000-Ft. Fans in the Secondary Mission. (U)

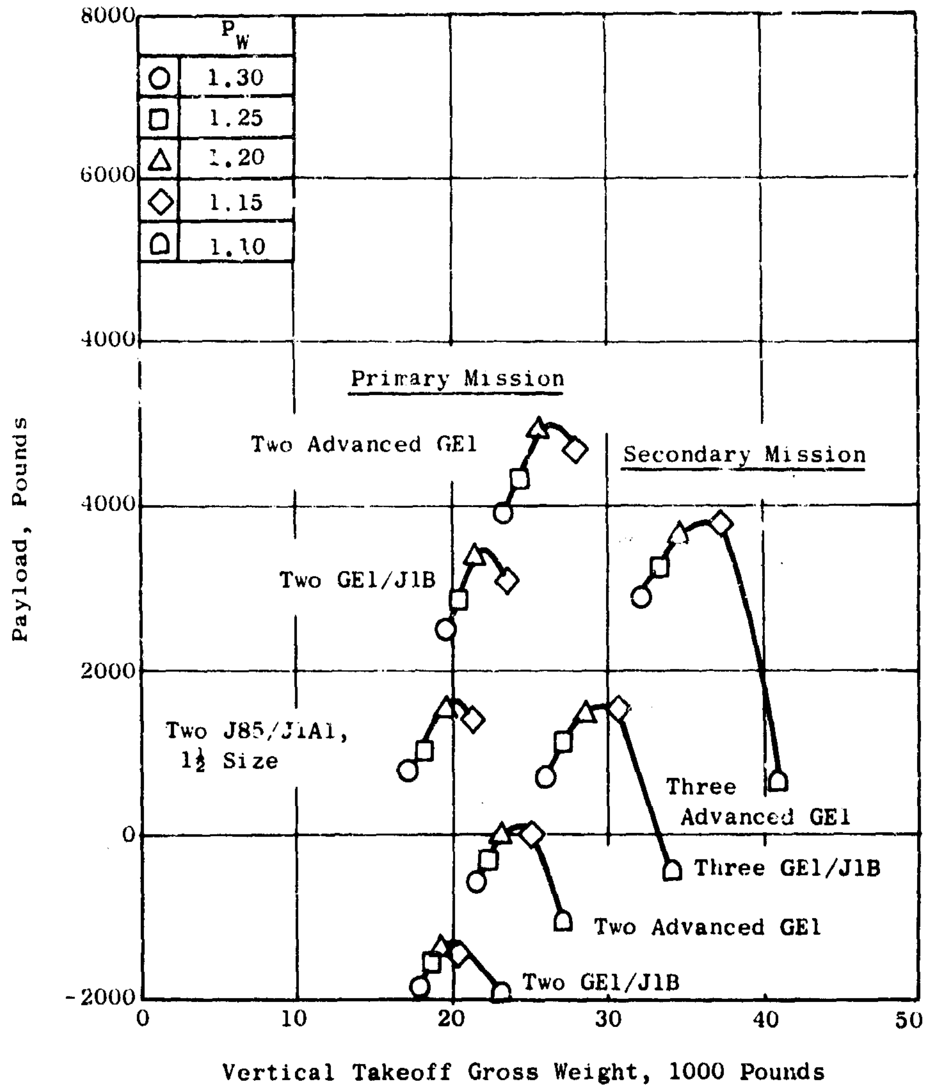

Figure 13. (C) Summary of Mission Analysis Results, Payload Versus Vertical Takeoff Gross Weight. (U)

engines at 95 degrees Fahrenheit/6000 feet from Figures 10 and 11 for the secondary mission.

(C) For the aircraft powered by two GE1/J1B core engines, the difference in payload capability between the two missions is about 4500 pounds. There is a payload penalty of about 2000 pounds for changing the vertical takeoff environment from 90 degrees Fahrenheit/2500 feet to 93 degrees Fahrenheit/6000 feet. (The lift to weight of 1.23 for the primary mission is equivalent to a 90 degrees Fahrenheit/2500 feet environment.) The remaining difference of 2500 pounds is due primarily to the additional fuel required for the secondary mission. A small part of this 2500-pound difference is due to changed ground rules. These changes included reaction control requirements, the pitch fan sizing technique, and aircraft component weight changes.

(U) CORE ENGINE SELECTION

The mission studies showed that the GE1/J1B core engine cycle was well matched to the LFX primary mission requirements. Therefore, the GE1/J1B was selected as the core engine for the LFX propulsion system.

(U) PERFORMANCE AND CYCLE OBJECTIVES

The performance objectives listed here were determined by means of the mission analysis previously described, and by the use of a design point computer program for the GE1/J1B core engine.

The mission analysis showed the optimum wing fan pressure ratio for this mission to be 1.2, with some reduction of payload or range resulting as fan pressure ratio is either increased or decreased from this value. However, a range of fan pressure ratios from 1.16 to 1.30 will yield performance to complete the specified mission with either a higher-than-specified payload or longer-than-specified range. A design point pressure ratio of 1.25 was selected to permit use of a turbine power transfer device for control. The use of power transfer can drive the fan pressure ratio up to approximately 1.3 for either control capability or additional fan-powered aircraft acceleration capability.

The wing fan objective design point is:

Fan nominal pressure ratio	1.25
Nominal turbine flow, lb/sec	49.9
Nominal fan flow, lb/sec	484
Fan tip diameter, inches	55.6
Turbine tip diameter, inches	61.2
Uninstalled average lift, pounds	10,540

The fuselage fan objective design point is:

Fan nominal pressure ratio	1.25
Nominal turbine flow, lb/sec	17.3

Nominal fan flow, lb/sec	164
Fan tip diameter, inches	33.1
Turbine tip diameter, inches	35.7
Uninstalled average lift, pounds	3583

The following component efficiencies and losses are specified objectives at the design point for both wing fan and fuselage fan:

Maximum admission arc, degrees	360
Fan efficiency, percent	86
Pressure ratio	1.25
Turbine efficiency, percent	86
Diverter valve leakage, percent $W_{5.1}$	1
Diverter valve and scroll pressure loss, percent	11
Inlet loss, $\bar{\iota}_{10}$, percent	10
Exit loss, $\bar{\iota}_{11}$, percent	4
Thrust coefficient, C_{V13}	0.99
Compressor tip speed, ft/sec	900 to 1000
Average lift per pound of tip turbine flow, lb/lb/sec	211.5

(C) WEIGHT OBJECTIVES (U)

Component	Component Weight (pounds)	System Weight (pounds)
Core engine	655	1310
Diverter valve	130	260
Wing fan	506	1012
Fuselage fan	154	308
Controls and instrumentation	16	16
		2906

(U) Core engine weight includes fixed jet nozzle. Lift fan weights include mounts, bellmouth, exit louvers, 360-degree scroll with variable area, insulation and actuation linkage. Pitch fan weight includes a 360-degree scroll, variable area, and insulation, but excludes mounts.

(U) MECHANICAL DESIGN OBJECTIVES

The system mechanical design shall be consistent with:

1. Operating limits of 115 percent continuous operations, short time overspeed to 120 percent, and rotor containment to 130 percent speed.

2. Maneuver loads specified in Figures 57 and 58.

3. Mission and life requirements as specified under Operational Objectives.

4. Control modulation as specified under Control Requirements Objectives.

5. Stresses calculated on basis of minimum stock thickness.

6. Weights calculated on basis of nominal stock thickness.

7. Centrifugal field loads and stresses based on maximum calculated part weight.

8. Use of materials available during a time period compatible with initial production in 1967. Use of materials not yet commercially available is permitted and encouraged, but predicted properties must be available from metallurgy for advanced state-of-the-art materials. Use of plastics, epoxies and other lightweight materials should be considered.

9. Wing fans designed for both clockwise and counterclockwise rotation. Pitch fans will be designed for counterclockwise rotation, looking into the fan inlet.

10. Use of maximum leading edge sections consistent with required aerodynamic performance to reduce effects of foreign and domestic object damage.

11. General mechanical design criteria, such as those available in the Flight Propulsion Division design practice manuals.

12. Anti-icing provisions for the wing and fuselage fan bellmouth and bulletnose.

13. Use of grease-lubricated bearings together with a simple, quick means for replenishing the lubricant.

14. Fan design based on rotor-stator concept. Every effort will be made to achieve the thinnest fans compatible with this aerodynamic design concept. Provisions will be considered for improved crossflow ram recovery.

15. Rotor blades and turbine sectors designed such that individual blades and sectors can be removed and replaced without rotor teardown.

16. Consider the summary of XV-5A experience on pages 45 through 50 for design improvement guides.

(U) OPERATIONAL OBJECTIVES

1. Fans shall be designed for 1000 mission hours between overhauls and 10,000 mission hours total life. Replacement of rotor blades, turbine sectors, bearings, etc., and repair of other components shall be permitted to attain the total mission hours. Effect of

changing criteria to 250 mission hours between overhauls and 2000 hours total life shall be examined.

Diverter valves shall be designed for 500 hours between overhauls and 1000 hours total life. Effect of changing life criteria to decrease weight or cost will be identified. Range to be examined shall be not less than 50 hours nor more than 10,000 hours.

Mission Segment	Percent Total Life	Core Engine Power	Fan Power
Takeoff	14	Maximum	Maximum
Cruise	69	95 percent (normal)	-
Landing	08	Maximum	Maximum
Ground checks	09	95 percent (normal)	80 percent (normal)

One percent of fan life will be at single engine inlet conditions. Two percent of fan life will be with partial admission resulting from power transfer.

2. Fan operating limits shall be from minus 30 knots to plus 130 knots equivalent airspeed at altitudes from sea level to 10,000 feet density altitude, with fan running at power levels up to maximum power. Side translations up to 30 knots at flight speeds from minus 30 knots to plus 30 knots will be considered.
3. Maneuver loads - Fans shall be capable of continuous operation with the loads and accelerations shown in the upper half of Figure 57 occurring once in each 0.1 fan operating hour. Fans shall be capable of operation with the loads and accelerations shown in the lower half of Figure 57 occurring once in each 1.0 fan operating hour.
4. Diverter valves should be designed to withstand temperatures up to 950 degrees centigrade with engine airflow and turbine discharge pressures equivalent to ground idle for periods of time up to 15 seconds (engine light-off).
5. Desired engine features include self-start capability, up to one-and-one-half percent of compressor discharge flow for customer bleed, and up to 60 shaft horsepower through customer power take-off. Other requirements will be given as a part of the preliminary specifications.

(U) RELIABILITY OBJECTIVES

Reliability requirements for the lift fan systems have been requested from airframe manufacturers. The achievement of satisfactory reliability depends very largely on the use of sound design concepts.

Specific reliability levels are based on systems performing their design function for the period of operational time in the operating condition to which the systems were designed.

Objective levels of maintainability are shown in Figure 14 as allowable time between overhaul versus cumulative operating hours.

Objective levels of reliability are shown in Figures 15 and 16 as mean time between unscheduled removal and mean time between in-flight power loss.

(U) CONTROLS REQUIREMENT OBJECTIVES

Preliminary requirements for aircraft control have been estimated for an aircraft having two equal-size wing fans and two equal-size pitch fans. Figures 17 and 18 show the assumed values of aircraft pitch axis and roll axis inertias. Figure 19 shows the assumed values of fuselage length and the moment arms of the pitch fans and wing fans as functions of aircraft gross weight. The wing fan was assumed to have a displacement aft of the aircraft center of gravity equal to 13.2 percent of the wing mean aerodynamic chord. Pitch fan flow transfer capability of 50 percent was assumed.

Figure 20 shows the required pitch fan nominal lift for each pitch fan as a function of the aircraft gross weight and the pitch control acceleration rate. Because the two equal-sized pitch fans were assumed to be equally spaced from the aircraft center of gravity, there will be a lift difference between the two pitch fans equivalent to the nominal lift shown for zero acceleration on Figure 20.

Figure 21 shows the wing fan differential lift required to produce the roll acceleration rates as a function of the aircraft gross weight. The wing fan differential lift is the roll control force, and is defined as the difference in lift between the two wing fans.

Figure 22 shows the normalized performance of two equal-size fans during power transfer. Shown are the lift of the fan receiving flow (L_1/L_N) and the lift of the fan losing flow (L_2/L_N). The sum of these two terms is the available system lift, and the difference in these two terms is the available control force. Also shown for convenience is a line for the ratio of available control force to available lift.

The calculation of the total system lift loss as a function of aircraft gross weight was performed by first sizing the pitch fans using the requirements of Figure 20 and then sizing the wing fans using the remaining engine discharge flow. The lift loss due to full pitch control (0.53 radian/second^2) was then determined and compared to the lift loss for full roll control (1.8 radians/second^2). The lift loss for full roll control was found to be higher than the lift loss for full pitch control. Figure 23 shows the lift reserve required for roll control as a function of gross weight.

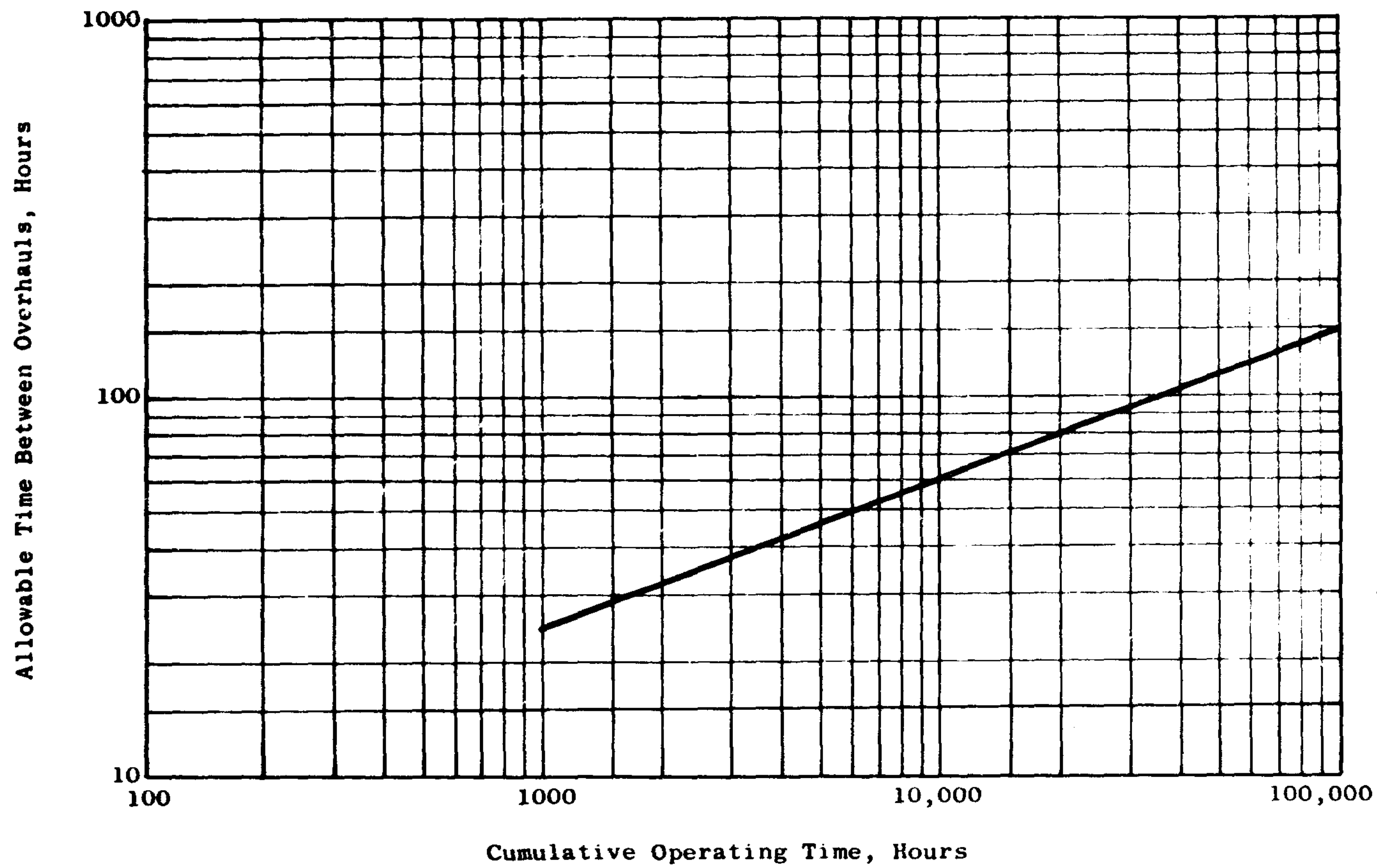

Figure 14. (U) Objective Overhaul Period.

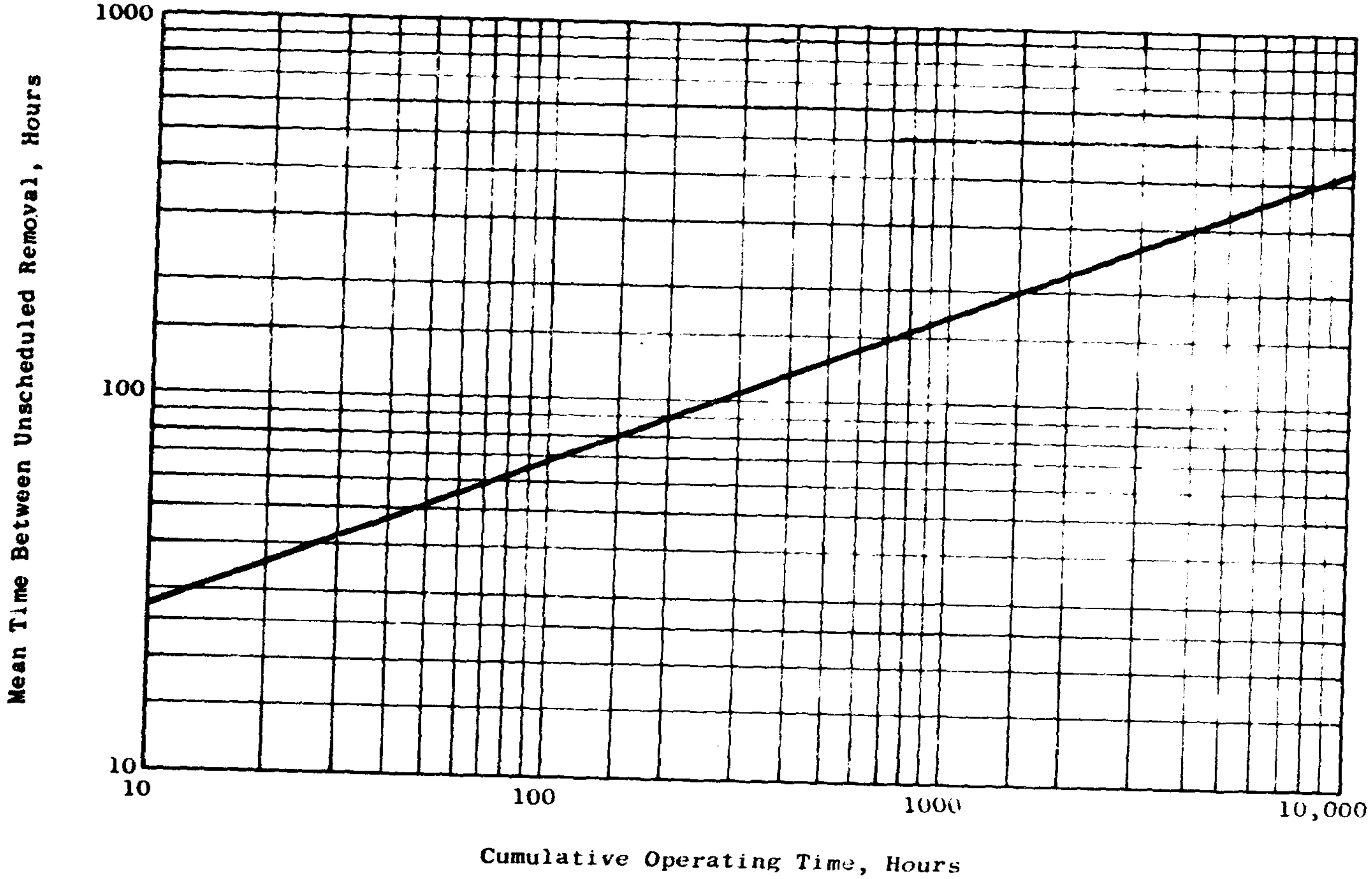

Figure 15. (U) Objective Unscheduled Removal Rate.

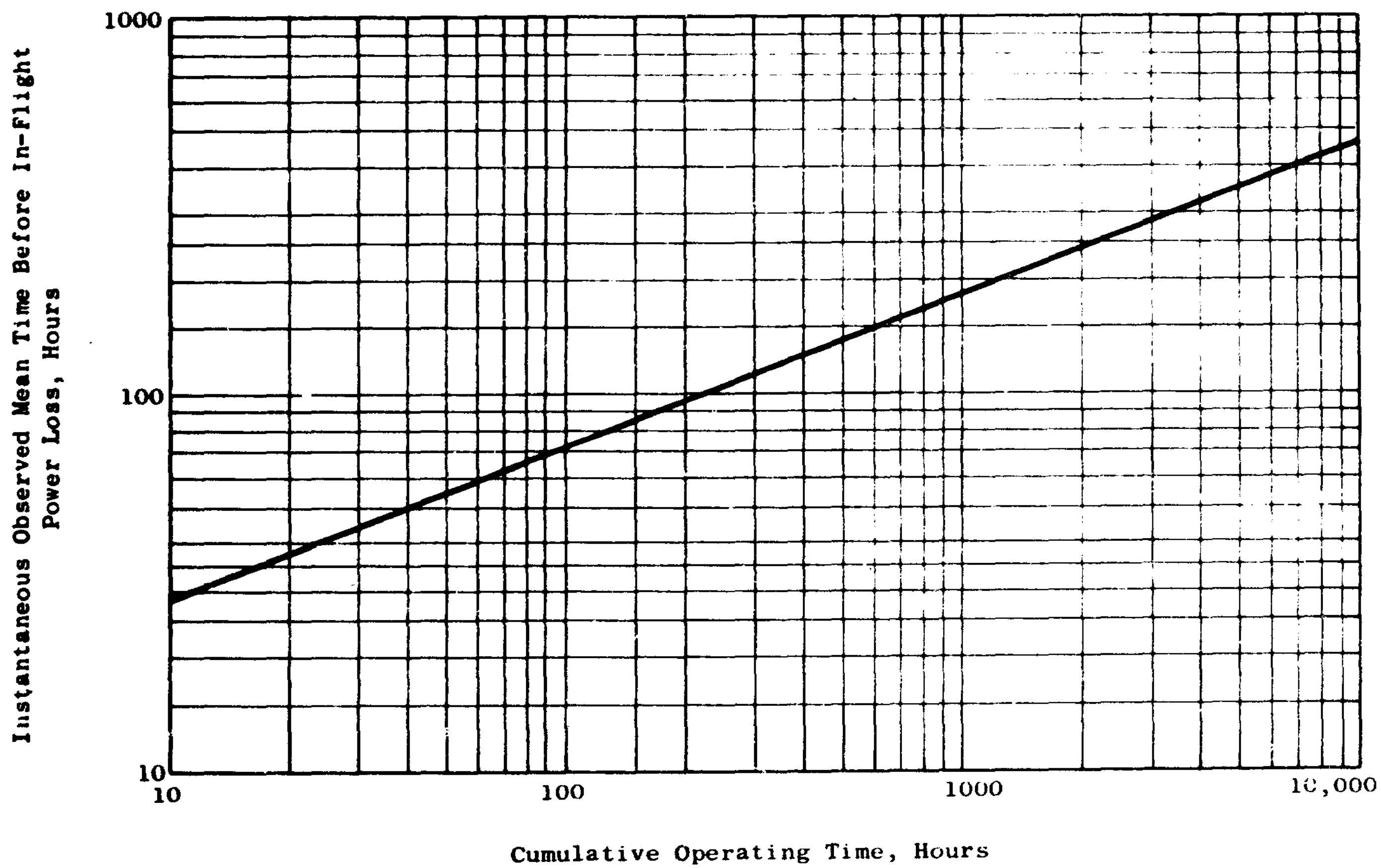

Figure 16. (U) Objective Operational Reliability.

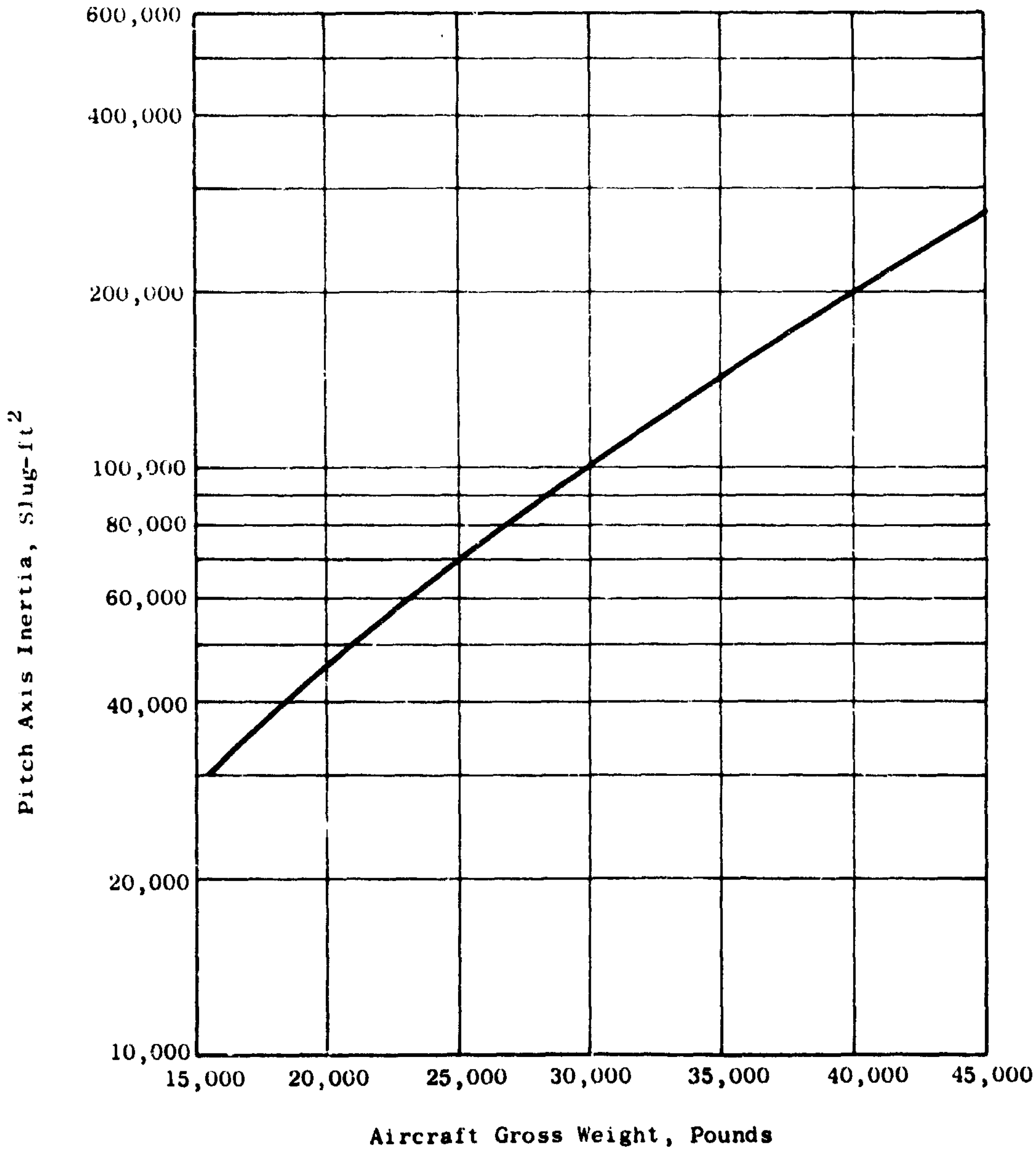

Figure 17. (U) LFX Parametric Aircraft Pitch Axis Inertias.

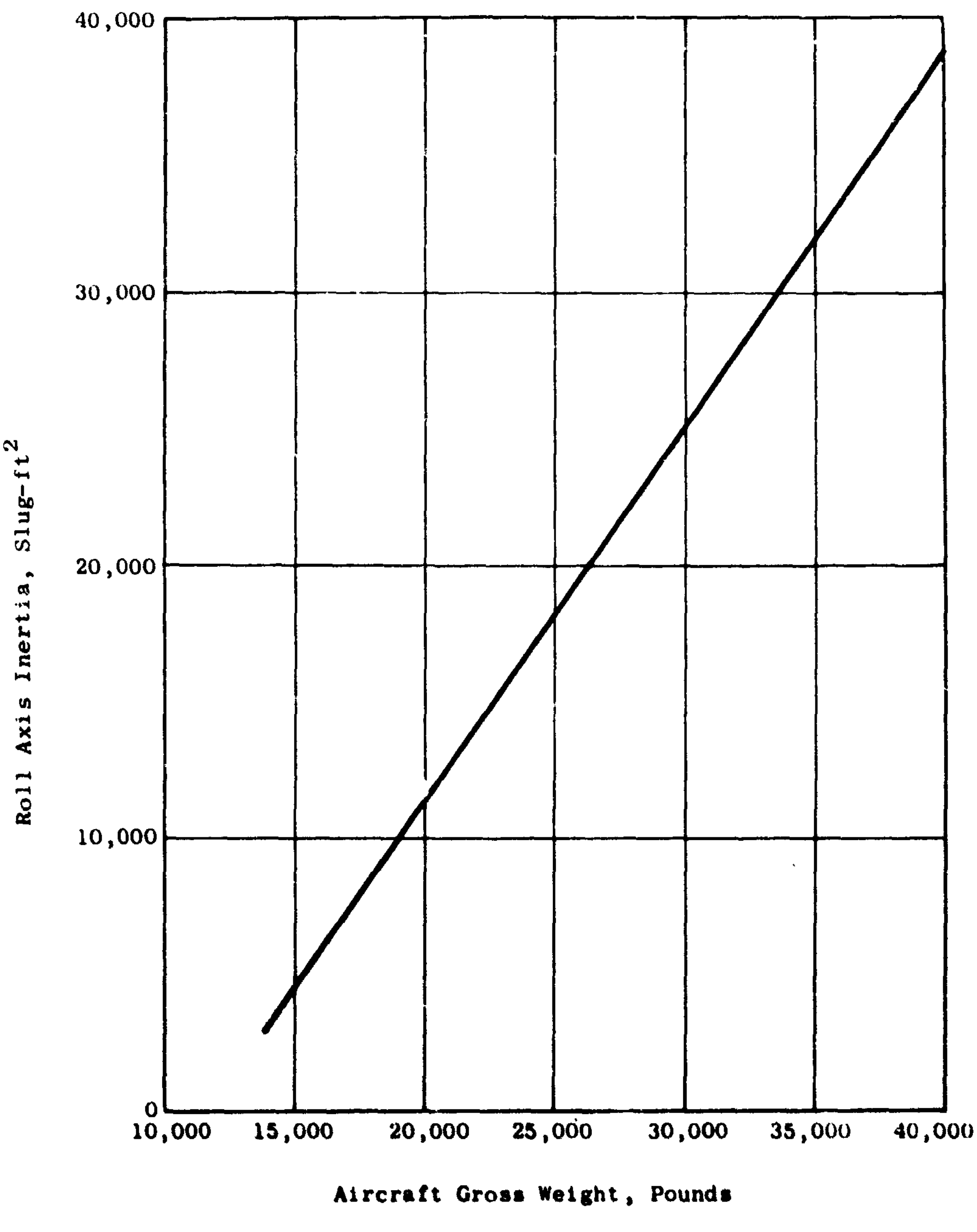

Figure 18. (U) LFX Parametric Aircraft Roll Axis Inertias.

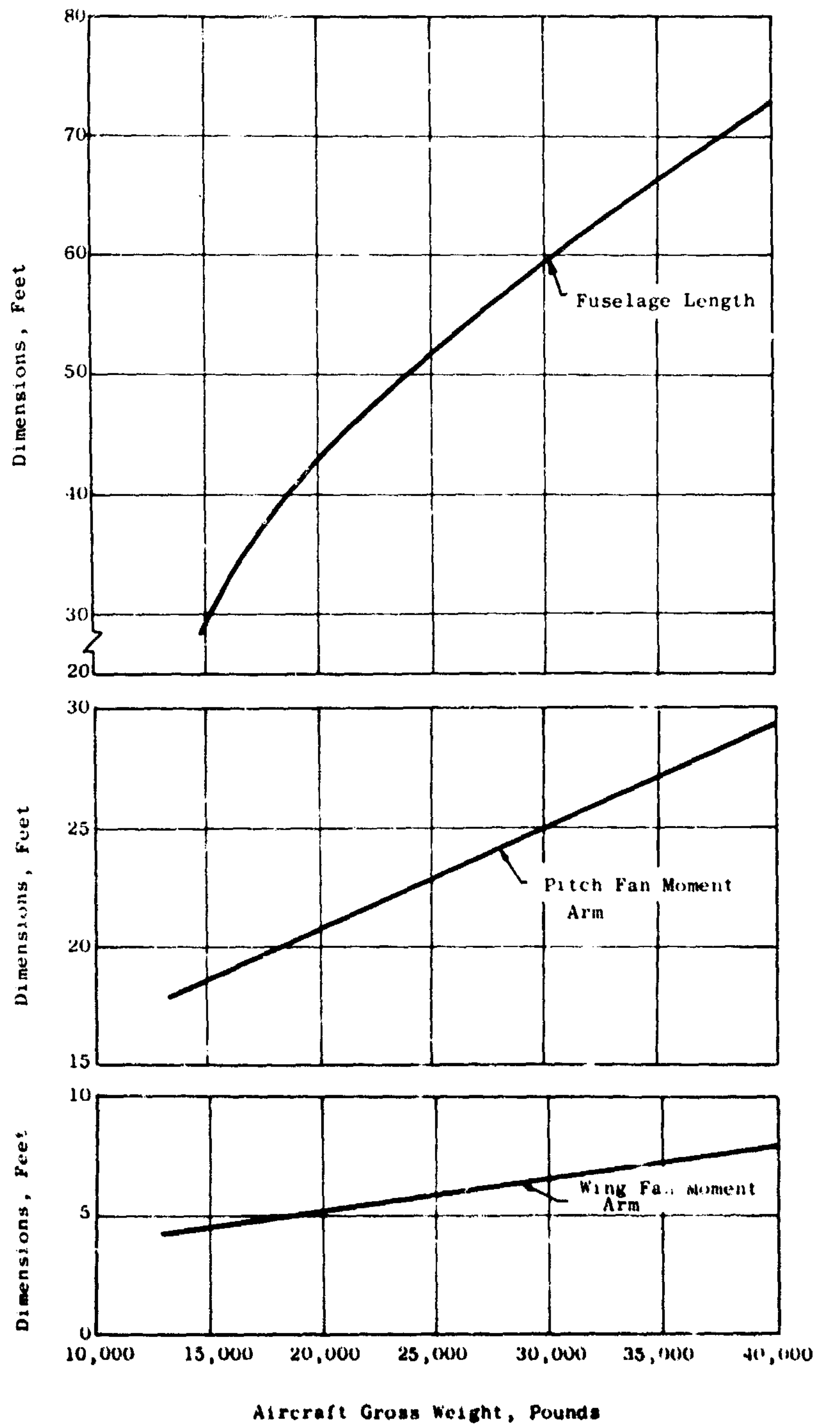

Figure 19. (U) LFX Parametric Aircraft Dimensions.

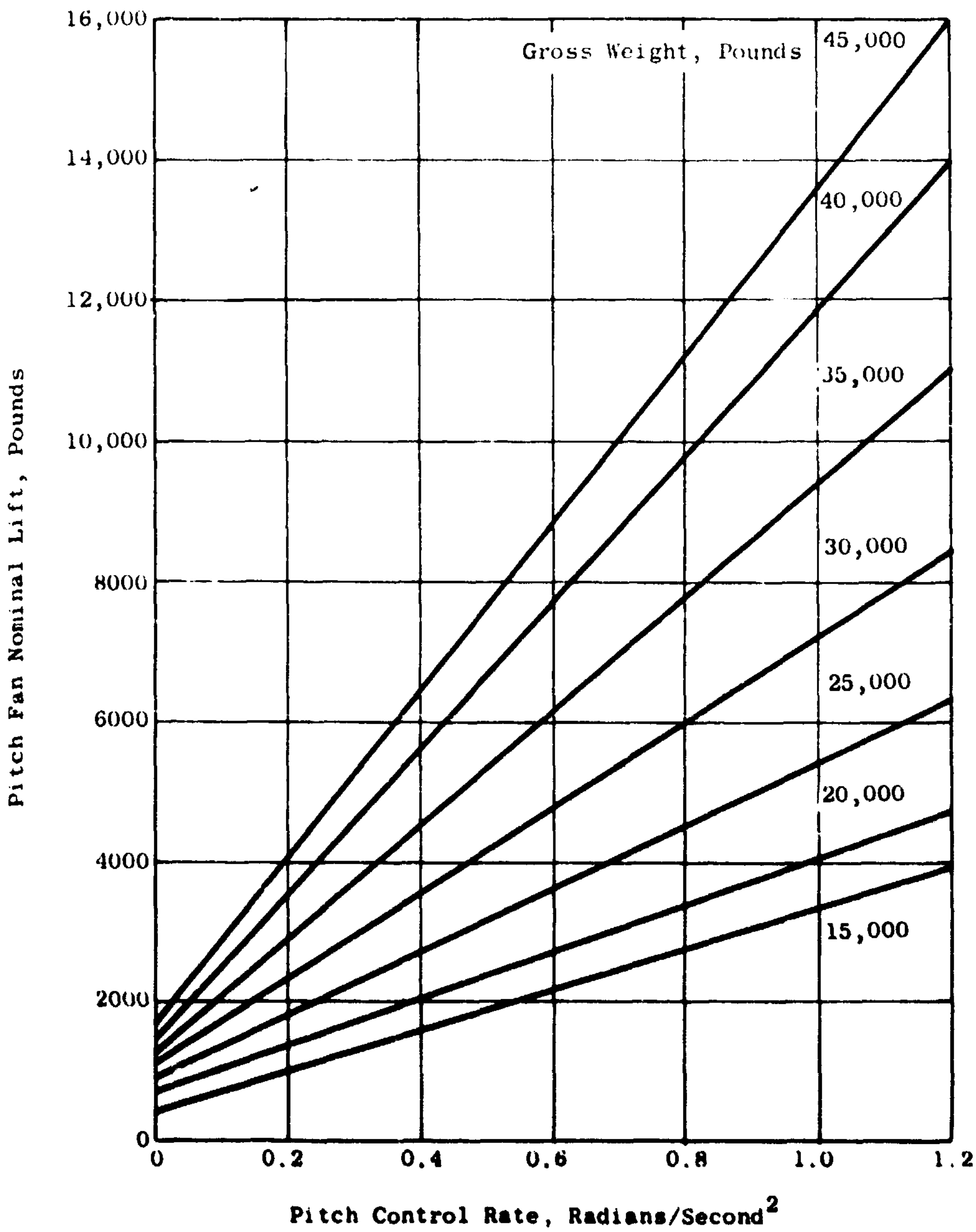

Figure 20. (U) LFX Parametric Aircraft Pitch Fan Nominal Lift.

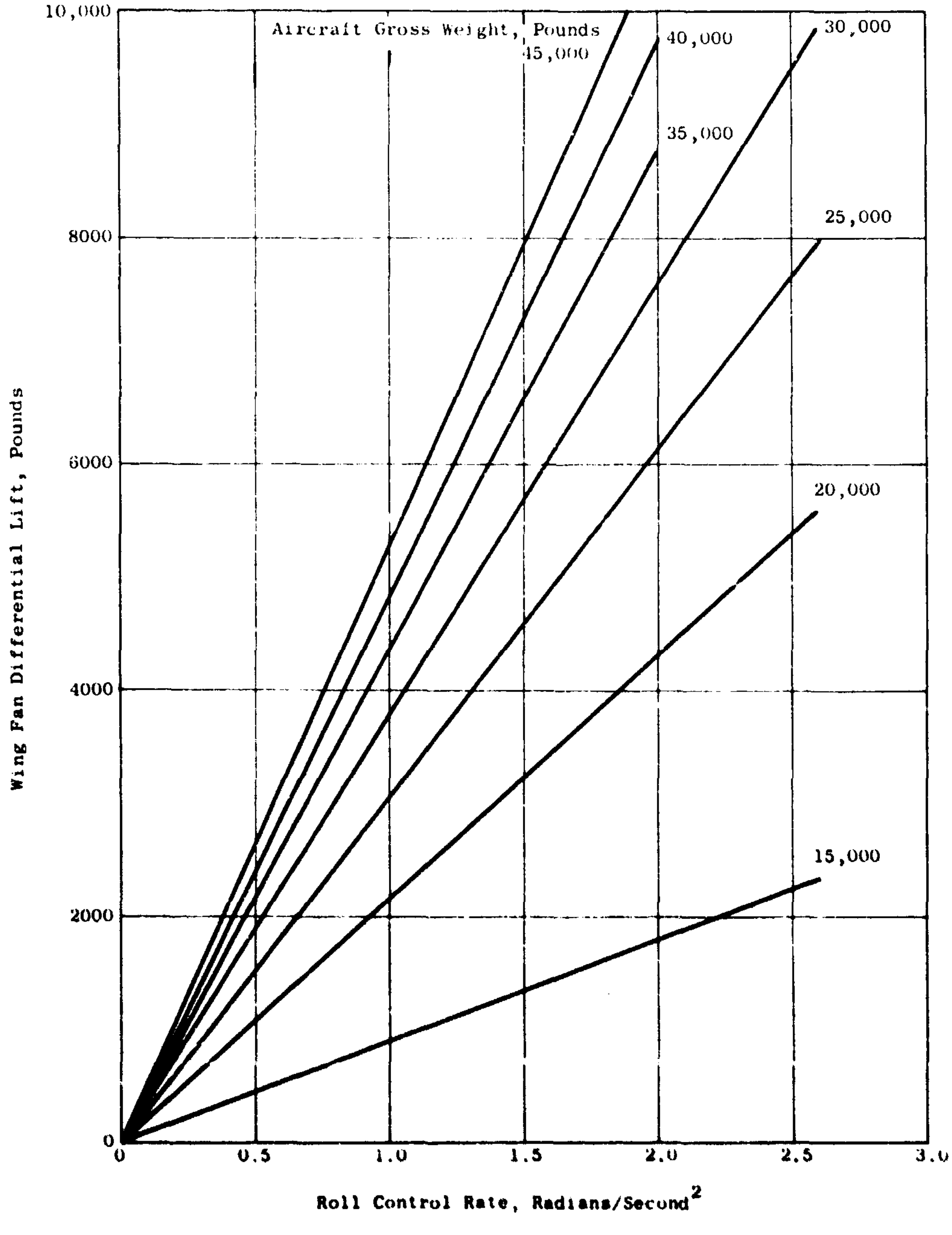

Figure 21. (U) LFX Parametric Aircraft Thrust Modulation for Roll.

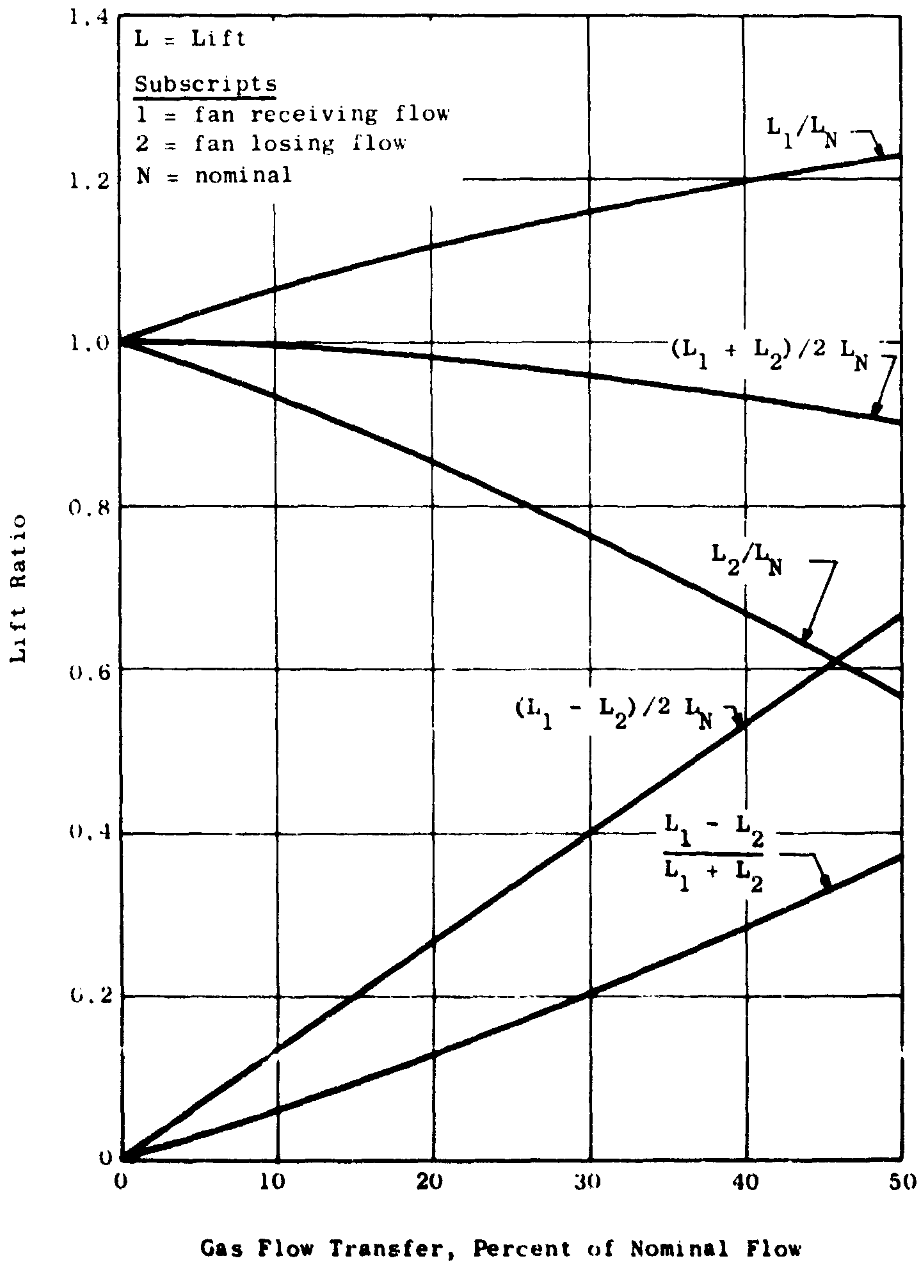

Figure 22. (U) Twin Fan Performance with Power Transfer.

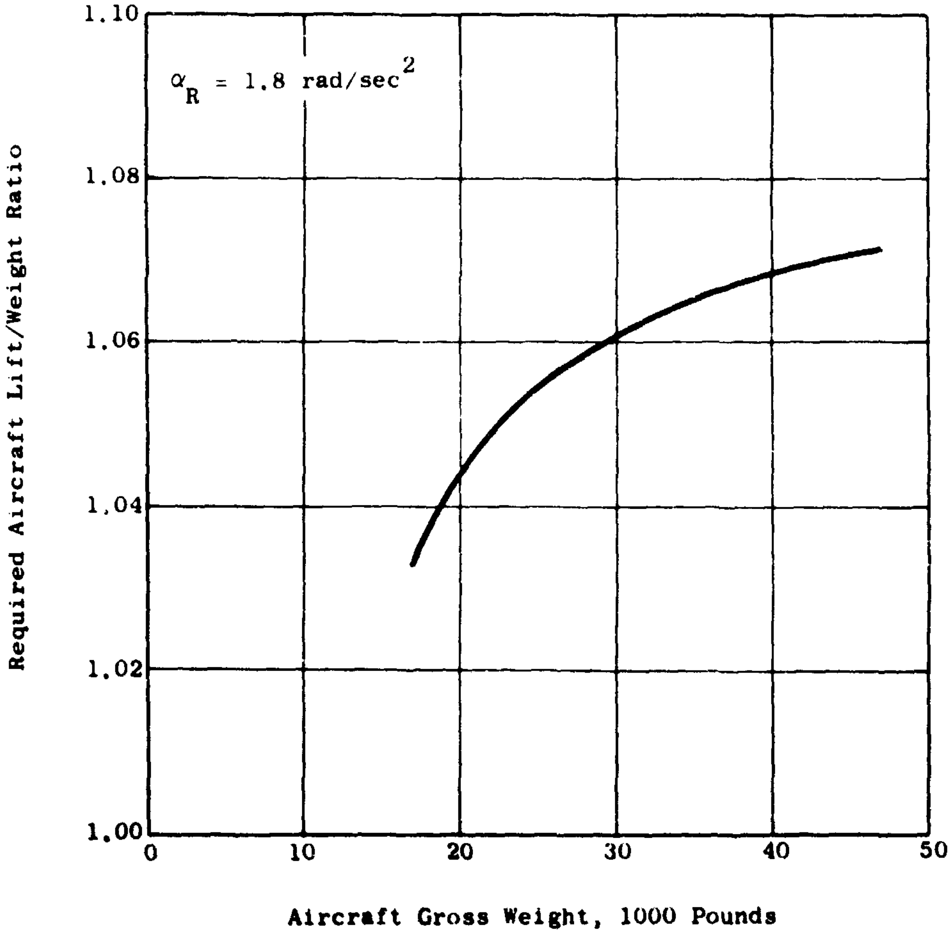

Figure 23. (U) Lift Reserve for Roll Control Versus Aircraft Gross Weight.

Figure 24 shows the total required lift to weight ratio as a function of vertical takeoff gross weight and vertical takeoff environment. This lift to weight ratio is defined as the ratio of uninstalled lift at sea level standard day to vertical takeoff gross weight, and is the product of four terms: the lift reserve for roll control from Figure 23, the vertical takeoff environment effect on lift from reference 1, the assumed 5-percent installation penalty, and the assumed 5-percent margin for vertical acceleration.

(U) SUMMARY OF XV-5A EXPERIENCE

The purpose of this summary is to aid in furnishing guidelines for future lift fan design, and specifically for the LFX studies and specifications.

(U) The J85-5A Core Engines

1. The location of the engine inlets on the XV-5A contributed to good operating characteristics and to problems. The high inlets minimized the ingestion of foreign objects into the core engines during jet mode flight and during fan mode flight. However, the location of the inlets permitted ingestion of the fan turbine discharge gases during low-speed fan-powered flight in ground effect. The engine variable geometry schedule was modified to permit operation within the area of hot gas reingestion, but at a loss of approximately 2 percent of lift at static conditions.

Recommendations

a. Increase the stall temperature tolerance of the core engines.

b. Locate engine inlets in areas not subject to hot gas inflow.

c. Include identification of the fan turbine discharge flow path over the operating range of the aircraft in component and model testing for future installations.

d. Study techniques for controlled directional discharge of the fan turbine exhausts, not only to minimize core engine reingestion but also to aid in reducing aircraft insulation and high temperature material requirements.

2. The service accessibility to the XV-5A engines is such that fuel filters and oil sumps are almost inaccessible on one engine while the engine is installed in the aircraft. Additional accessibility problems were generated when a horizontal fire wall was added after the initial configuration design was completed. Other similar areas requiring increased maintenance time lie in the fuel control adjustments, throttle adjustments and variable geometry adjustments.

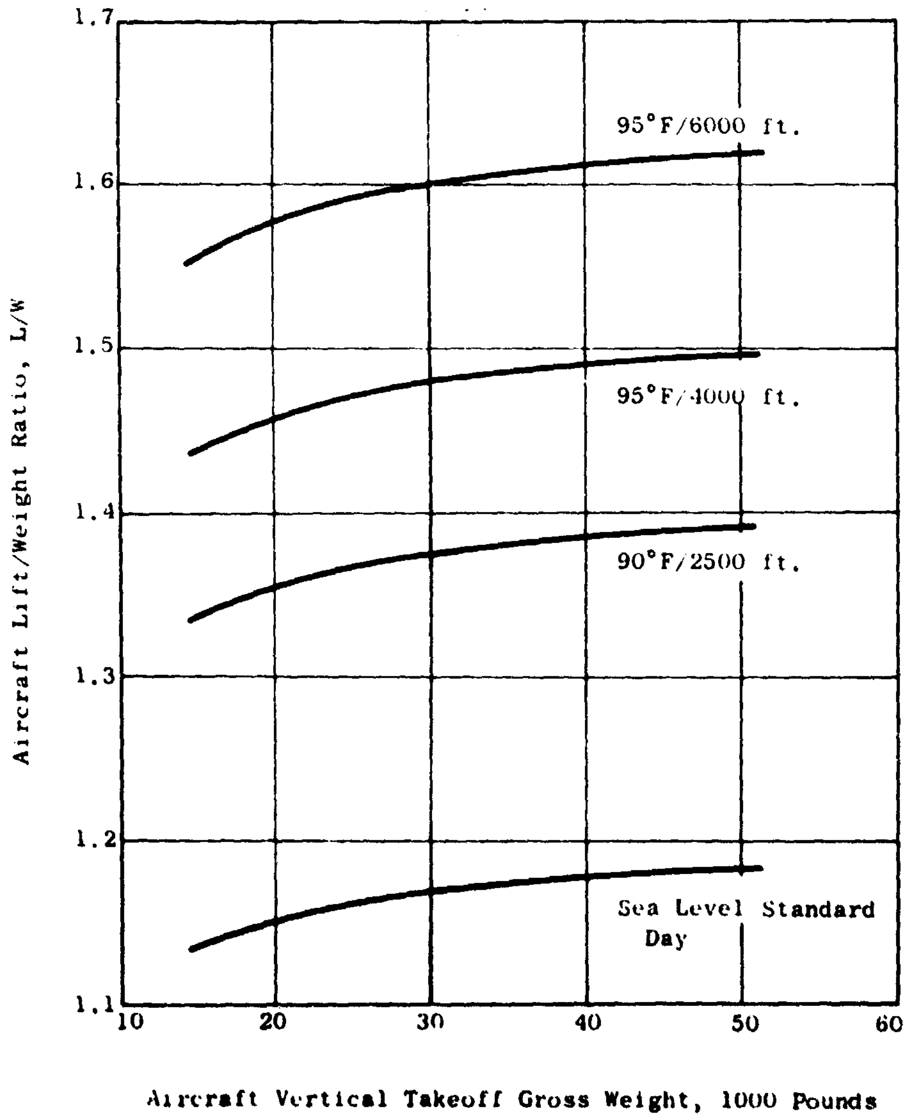

Figure 24. (U) Aircraft Lift to Weight Ratio Versus Vertical Takeoff Gross Weight and Takeoff Environment.

Recommendations

a. Study the feasibility of using left-handed and right-handed engines through relocation of accessories.

b. Closely coordinate installations design in the aircraft with the airframe and engine manufacturer.

c. Have the engine manufacturer design and fabricate the fire walls to eliminate interface and fit-up problems.

(U) Diverter Valves

The diverter valves were initially designed and fabricated for somewhat less severe cycle temperatures and pressures than those encountered in the XV-5A. Partly as a result of this disparity between design and operating environment, the diverter valves have had to undergo field repair and modification. The use of an alloy which could be weld repaired without post-weld heat treatment has facilitated work not otherwise feasible.

Selection of a diverter valve position switch with high temperature and high loading capabilities was necessary. Even so, switch failures occurred frequently, owing to electrical overload or mechanical reasons.

Diverter valve door seals were designed for approximately 1 percent of flow leakage. This leakage, although not in excess of limits, has caused aircraft heating problems in the jet mode and some slight lift loss in the fan mode.

Recommendations

a. Use design criteria commensurate with the expected growth capabilities of the core engine. Include core engine starting temperatures.

b. Evaluate seal design through component test and development. Additionally, define leakage in terms of pressure and flow. Evaluate thermal growth effects on the seals and locate actuators so that thermal growth diminishes seal clearance.

c. Perform proofing tests on components to exceed any predicted load and temperature during operational use. Tests should include cycling under operational pressure and temperature and simulated thermal shock due to engine starts, in-flight flameouts or other critical loading.

d. Remove position indicators from immediate vicinity of the hot valve body. A redesign of the position switch to effect use of rotary switches should be considered.

e. Consider capability for cross-coupling two or more diverter valves.

(U) Pitch Fan X376 (PF-1)

The pitch fan in the XV-5A represents second-generation mechanical and aerodynamic design. It has demonstrated the capability of developing a higher lift to weight ratio than the original LF-1 lift fans.

Although no part of the XV-5A propulsion system suffered identifiable "foreign" object damage, lift fan, pitch fan and core engine rotors suffered "domestic" object damage from screws, bolts, nuts and other aircraft or propulsion system parts. The pitch fan rotor was damaged beyond benching type repairs and had to be removed.

Nozzle area adjustment requires removal of the pitch fan to change scroll blocking plates.

The packed bearings in the pitch fan have demonstrated satisfactory performance. However, there may be some inherent psychological advantages in a system that can be more readily inspected.

The magnetic reluctance revolutions-per-minute pickup has worked well.

Recommendations

a. Evaluate trade-offs in aerodynamic design to decrease sharpness of compressor leading edges. In addition, as hardware is defined mechanically, perform component tests to accurately extend the benching limits for repair of foreign object damage. The ability to repair superficial foreign object damage by benching operations is invaluable to field maintenance.

b. Provide scroll area adjustment capability while fan is installed in the aircraft. This requirement may be eliminated in a power transfer scroll.

c. Provide means for bearing inspection without teardown of fan.

d. Consider, during the detail design:

   1) Better integration of insulation with the fan assembly; design of bosses specifically to aid in insulation mounting.

   2) Improvement of inspection features by self-contained front frame and scroll seal assembly.

   3) Provision of inspection features for bearings.

e. Add additional overspeed capabilities during design. Fan should be capable of running at speeds up to 10 percent higher than maximum operating speed without sustaining damage.

(U) Lift Fan X353-5 (LF-1)

The lift fan as installed in the XV-5A aircraft had a background of extensive running time at static conditions and some dynamic condition

running in wind tunnel installations. During the XV-5A ground tests and flight tests, several changes were incorporated into the lift fans to improve reliability, performance and maintainability. Some of these changes included substituting steel circular inlet vanes for the original aluminum vanes, increasing the stiffness of the exit louvers by doubling the skin thickness, and increasing the load capacity of the exit louver actuating rod arms by both beef-up of material and component design change.

Since these changes were made as an expediency to keep the downtime at a minimum, and were limited by the existing hardware configurations, new designs could improve both the design function and the parts.

The LF-1 maximum rotating speed is very close to the maximum required fan speed for aircraft conversion from fan mode flight to jet mode flight. Because it would be easy for a pilot to overspeed a fan rotor inadvertently, a speed warning and overspeed cutback system was installed in the XV-5A. This system worked as designed and enabled the pilot to maintain near maximum fan rotational speeds without the necessity for constant monitoring. However, the overspeed cutback could be potentially hazardous. It should be eliminated through additional rotor speed margin.

The assignment of responsibility for interface items between aircraft and propulsion system was made by mutual agreement between the airframe manufacturer and the propulsion system manufacturer. Experience has indicated that other division of interface responsibilities might be desirable and that improved communication during the installation design and development would be helpful.

Recommendations

a. Design fans for power absorption capabilities up to maximum available power at maximum vector angle and airspeed equivalent to 1.2 times the stall speed of the aircraft in the preconversion mode (approximately 130 knots indicated airspeed).

b. Design overspeed capability equal to ten percent higher than nominal design rotational speed without damage.

c. Utilize best mechanical design to improve weight of circular turning vane (if required), exit louvers, exit louver actuating system and frames.

d. Consider scroll area nozzle changes as a field requirement and design appropriate accessibility to the area trim devices. Permanently mark the area trim devices to identify direction and magnitude changes as a function of adjustment.

e. Provide for field inspection of exit louver actuating arms and other actuating devices.

f. Design mounting bosses or other devices to aid in locating and mounting insulation.

g. Provide for component testing to simulate both static and dynamic loads.

Detail design recommendations listed below are peculiar to the X353 system and may not be applicable to other system designs. However, they are included here as possible thought provokers for use when other system designs are in process.

a. Provide inlet vane external attachment at the bulletnose to improve interchangeability and to simplify removal.

b. Rotate speed pickup attachment spanwise to aid in easier removal and replacement.

c. Provide a floor at the "record player" or bottom of the bulletnose to prevent loose objects from falling through to the rotor.

d. Provide plumbing attachments and guides in the bulletnose.

e. Consider the vertical wall on the rear frame as a permanent part of front frame instead, to permit the rear stator and exit louvers to be removed from the aircraft with the fan installed. This would also permit rotor exchange.

f. Study scroll seal detail design to achieve improvement in sealing.

g. Integrate bottom wing fairing with the fan rear frame.

h. Relocate internal insulation for better external accessibility.

i. Segment the blade retainers for removal and make the platform tab a separate part for repairability.

j. Reevaluate differential thermal growth between the rear frame and the scroll.

k. Mark the blade moment weights permanently on the blades or bucket carriers.

l. Provide positive installation reference points for the engines and the fans.

(U) COST OBJECTIVES

A preliminary design (Figure 54) was submitted to the General Electric Company's Large Jet Engine Department Manufacturing Operation for costing of both development and production fans. The items defined in the critical areas requiring further study (Critical Technology Definition) were such that a precise, meaningful cost estimate of production quantities of LFX systems was not obtainable. However, it is reasoned, on the basis of studies on previous fan configurations, that LFX lift fans should be available in lots of 1000 or more fans at between $50,000 and $100,000 per fan, depending on features incorporated to meet future requirements. This estimate should be resolved to a more accurate number at the conclusion of the "Design in Depth" (Phase 1.2) proposed as the LFX contract continuation.

Development fans have been estimated at approximately $500,000 per unit in lots of less than five. Reference is made to Figure 86 in the Exploratory Development Effort Section, which shows funding requirements versus years for a program leading to an LFX demonstrator.

(C) PRELIMINARY SPECIFICATIONS (U)

(U) This section follows the format and paragraph numbering sequence of the military specification document, reference 17. Only the applicable paragraphs of reference 17 have been included here.

1. (C) Scope (U)

(U) 1.1 Scope: This preliminary specification presents performance and installation requirements for an advanced lift fan system using a high-energy core engine representative of 1968 to 1970 technology. Data presented are estimated minimum performance figures except where otherwise noted.

The data in this specification represent a summary of LFX studies and are not intended to supercede or replace other issued summaries of data on the core engine or fan systems. This summary does not constitute a commitment on the part of the General Electric Company.

(U) 1.2 Classification: The LFX propulsion system is a conceptual design study of a convertible propulsion system for installation in a class of aircraft suited for surveillance and target data acquisition. The LFX system as conceived includes two wing lift fans, two fuselage lift fans and two core engines with diverter valves. Both wing fans and both fuselage fans have thrust modulation capability to provide aircraft control. Performance data in this specification are for one wing fan and one core engine unless otherwise specifically stated.

(C) 1.3 Basic Core Engine: The turbojet engine cycle selected for use in this specification is one which is representative of a family of lightweight high-performance engines. The designation is GE1/J1B. Weights and performance are representative of those which could be achieved in the 1968 to 1970 time period for production quantities.

The core engine is the basic unit of the General Electric "building block" concept of engine technology. It features high specific energy gas discharge. Estimated turbine discharge conditions on a sea level standard day at takeoff power are:

$W_{5.1}$ = 68.2 pounds per second

$T_{5.1}$ = 1847 degrees Rankine

$P_{5.1}$ = 52.6 pounds per square inch absolute

(U) 1.4 Wing Lift Fan: The wing lift fan cycle was selected to meet objectives identified through mission analysis. The wing lift fan is designed to accept 73.2 percent of the gas flow from one core engine at nominal conditions. The wing lift fan cycle is the one which was used for preliminary mechanical design and sizing.

(U) 1.5 Fuselage Lift Fan: The fuselage lift fan features an aerodynamic conceptual design scaled from the wing lift fan. Design flow to the fuselage lift fan at nominal conditions is 26.8 percent of the turbine discharge flow from the core engine. The fuselage lift fan performance data in this specification are scaled from the wing lift fan. Weights shown are objective weights taken from the previously published parametric study data of reference 1.

2. (U) Applicable Documents

2.1 The following specifications and publications shall be used as a guide in the design of the fan systems:

Reference 17 - Military: MIL-E-5007B - Engines, Aircraft, Turbojet - General Specification for

Reference 10 - General Electric: LFX Memorandum Number 65-8, LFX Objectives

3. (U) Requirements

3.1 Performance Characteristics: The ratings and curves shown are based on the terms and standard conditions defined in reference 17 as modified herein, and on the use of a fuel having a lower heating value of 18,400 Btu per pound. These data indicate estimated levels of uninstalled performance for the propulsion system under standard conditions. The core engine performance does not include losses for inlet, for shaft power extraction or for customer bleed. Flow leakage at the diverter valve is assumed to be one percent of the gas flow. A fixed-area nozzle, trimmed to attain the rated exhaust gas temperature, has been assumed for the jet mode. Lift mode performance includes losses for diverter valve, ducting, scrolls, inlets, and exit louvers. No external effects, such as reingestion or ground effect, are included.

3.4.1 Fuel: The core engine selected for this study shall be compatible with either JP4 or JP5 fuel.

3.4.5 Estimates: The estimated design point data for the fans are listed below. Estimated performance data for the core engine are shown in Figures 25 through 32, inclusive. Estimated performance data for the wing lift fan are shown in Figures 33 through 50, inclusive. The wing fan design point is:

Fan nominal pressure ratio	1.253
Nominal turbine flow, lb/sec	49.4
Nominal fan flow, lb/sec	492
Fan tip diameter, inches	56.2
Turbine tip diameter, inches	63.1
Uninstalled lift, guarantee level, lb	10,480
Uninstalled lift, average, lb	10,800

The fuselage fan design point is:

Fan nominal pressure ratio	1.253
Nominal turbine flow, lb/sec	18.12
Nominal fan flow, lb/sec	180
Fan tip diameter, inches	34.0
Turbine tip diameter, inches	38.2
Uninstalled lift, guarantee level, lb	3844

The wing fan design point component efficiencies and losses are:

Maximum admission arc, degrees	360
Nominal admission arc, degrees	280
Minimum admission arc, degrees	200
Fan efficiency, percent	86.5
Turbine efficiency, percent	84
Diverter valve leakage, percent $W_{5.1}$	1
Diverter valve and scroll pressure loss, percent	8
Fan inlet loss, $\bar{\omega}_{10}$, percent	6
Fan exit loss, $\bar{\omega}_{11}$, percent	5
Fan thrust coefficient, C_{V13}	99
Fan tip speed, ft/sec	946
Lift per pound of tip turbine flow, lb/lb/sec	212.1

3.4.5.1 Performance Correction Curves: Data for correcting performance for diverter valve effects and nonstandard day conditions are shown in Figures 51, 52, and 53.

3.4.6.10 Reverse Thrust: No thrust reverser is provided for turbojet mode operation. Negative thrust in the fan mode can be achieved by forward vectoring of the fan efflux.

3.4.15 Measured Gas Temperature: The measured allowable exhaust gas temperature shall be 1387 degrees Fahrenheit for continuous operation in either the turbojet mode or the fan mode.

3.7 Drawings and Diagrams: The following General Electric Company drawings form a part of this specification:

LFX Fan Design - Figure 54
LFX System Study - Figure 55
LFX Installation Study - Figure 56

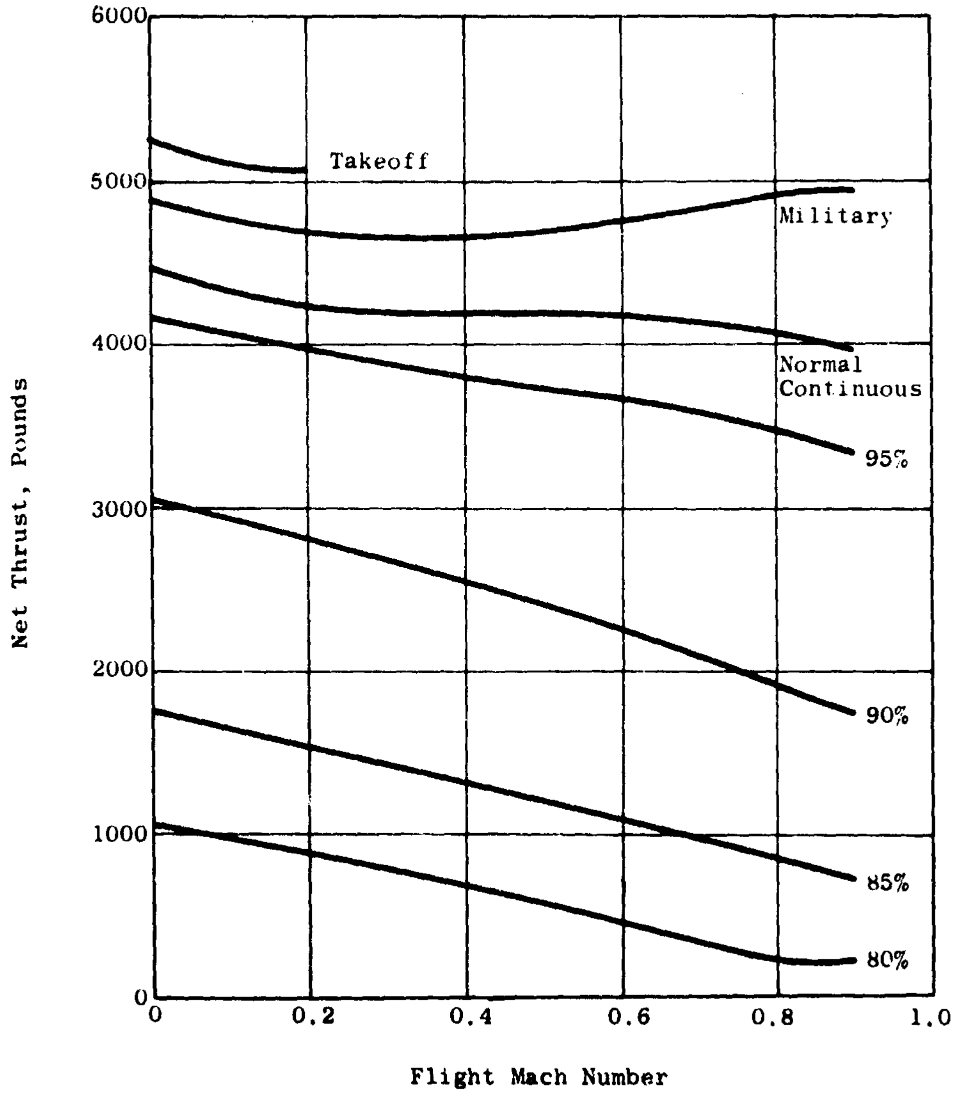

Figure 25. (C) GE1/J1B Core Engine Turbojet Net Thrust Versus Flight Mach Number at an Altitude of Sea Level. (U)

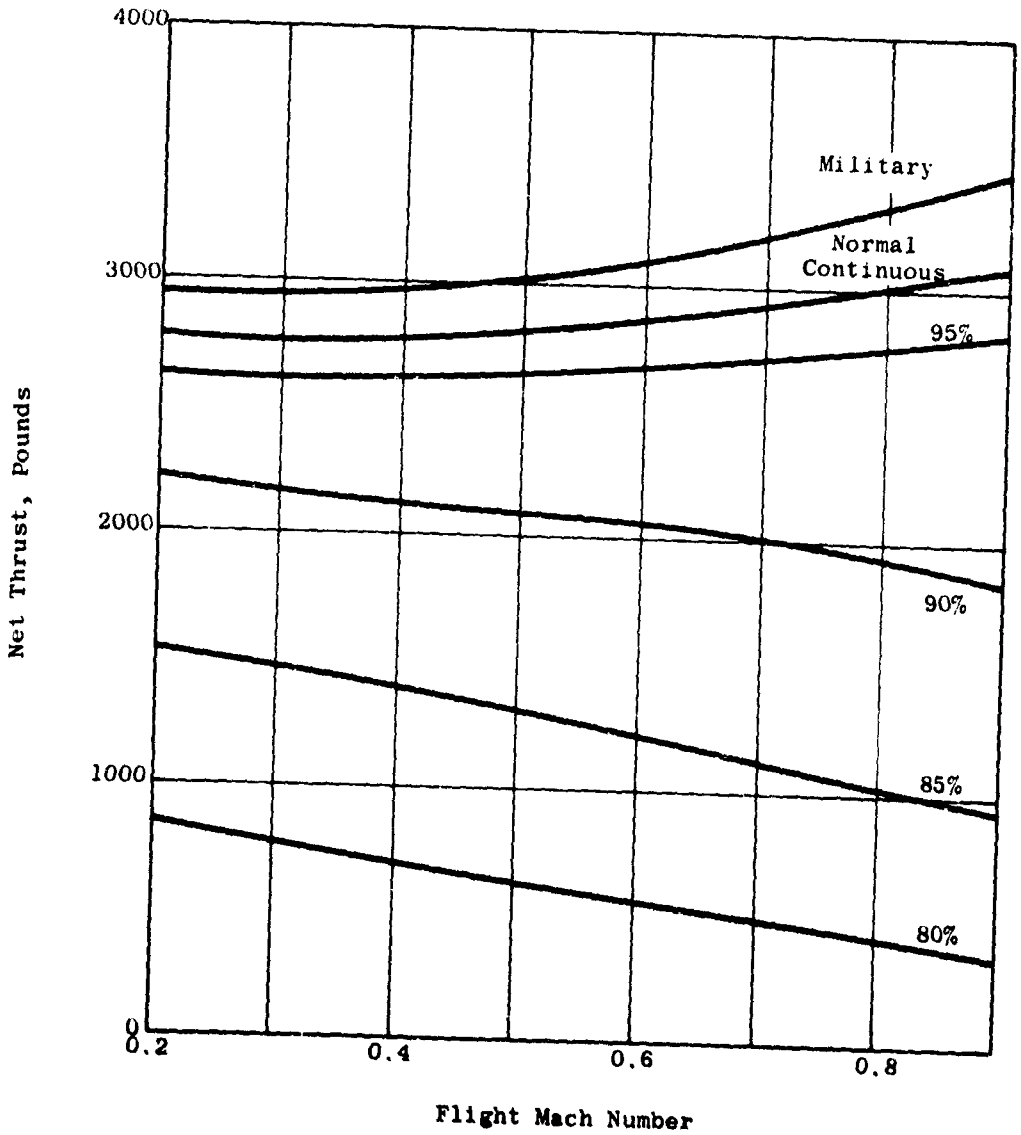

Figure 26. (C) GE1/J1B Core Engine Turbojet Net Thrust Versus Flight Mach Number at an Altitude of 15,000 Feet. (U)

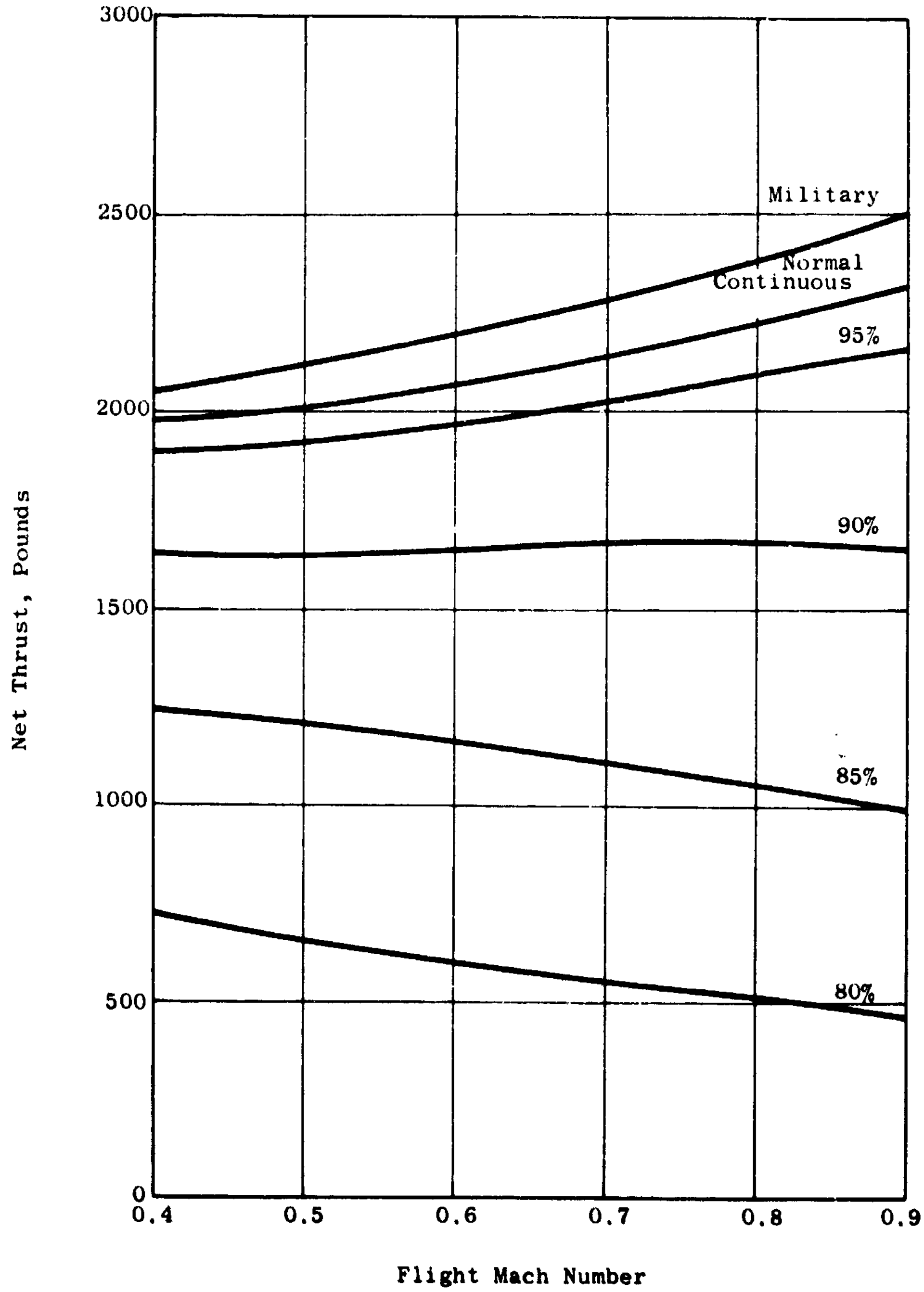

Figure 27. (C) GE1/J1B Cor(ngine Turbojet Net Thrust Versus Flight Mach Number at an Altitude of 25,000 Feet. (U)

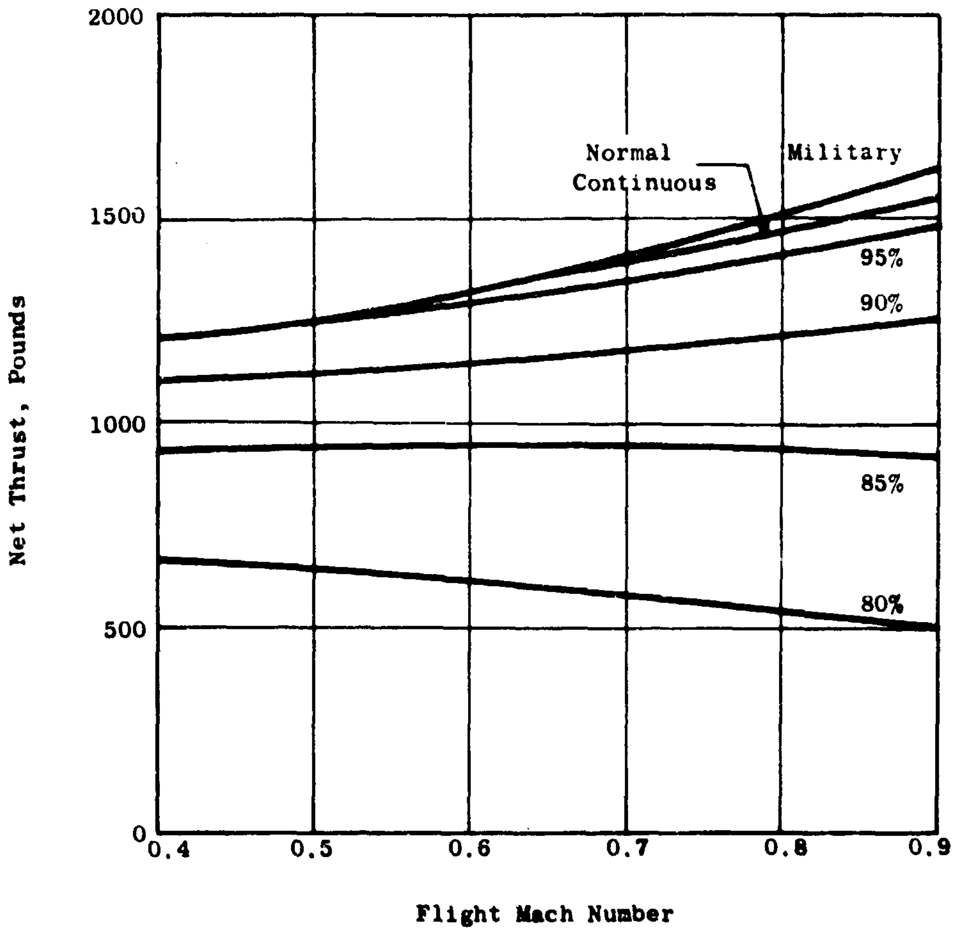

Figure 28. (C) GE1/J1B Core Engine Turbojet Net Thrust Versus Flight Mach Number at an Altitude of 36,089 Feet. (U)

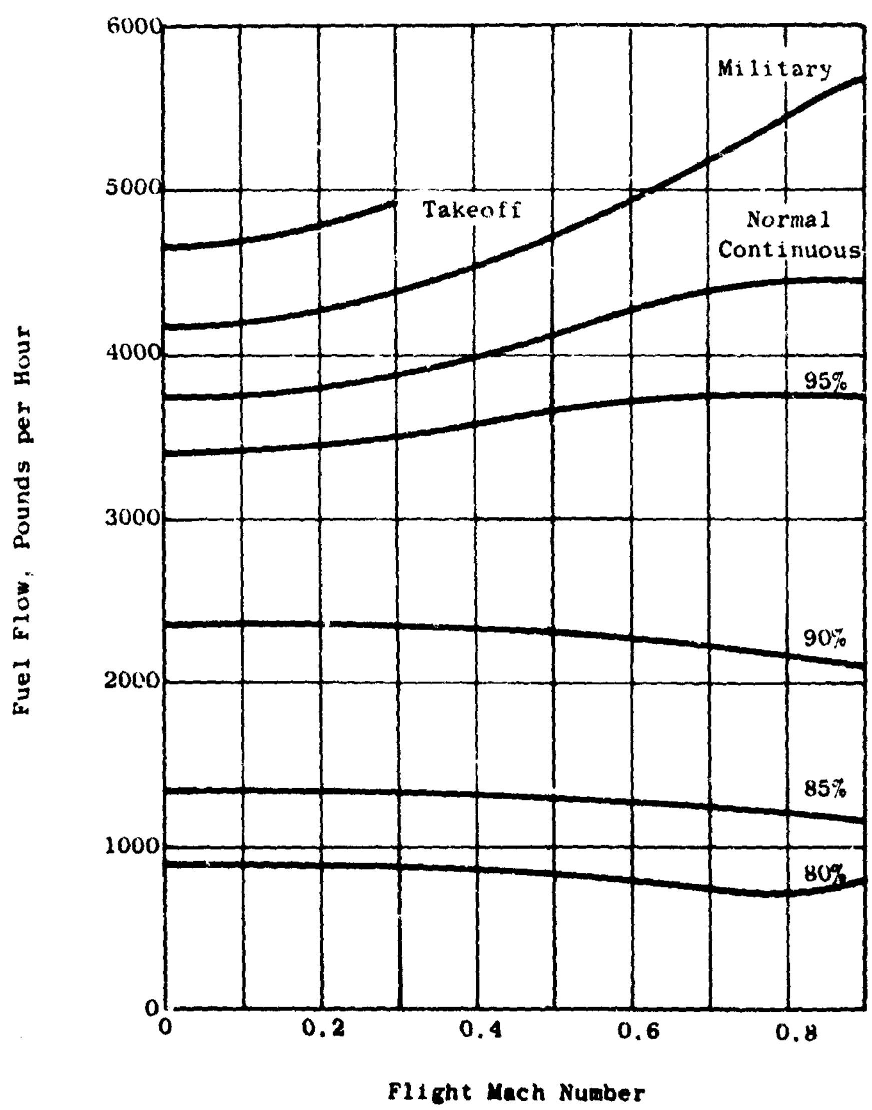

Figure 29. (C) GE1/J1B Core Engine Fuel Flow Versus Flight Mach Number for One Core Engine at an Altitude of Sea Level. (U)

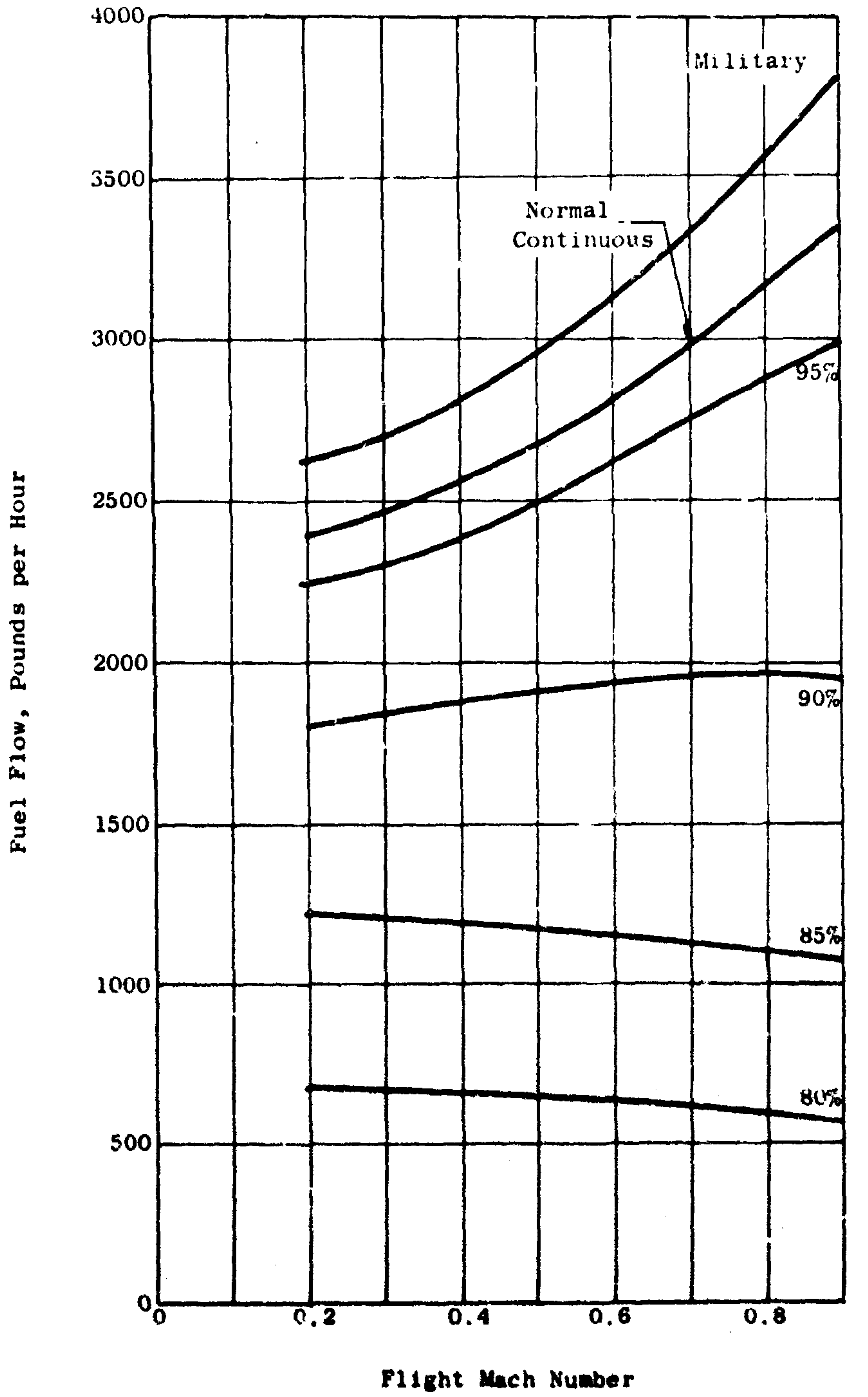

Figure 30. (C) GE1/J1B Core Engine Fuel Flow Versus Flight Mach Number for One Core Engine at an Altitude of 15,000 Feet. (U)

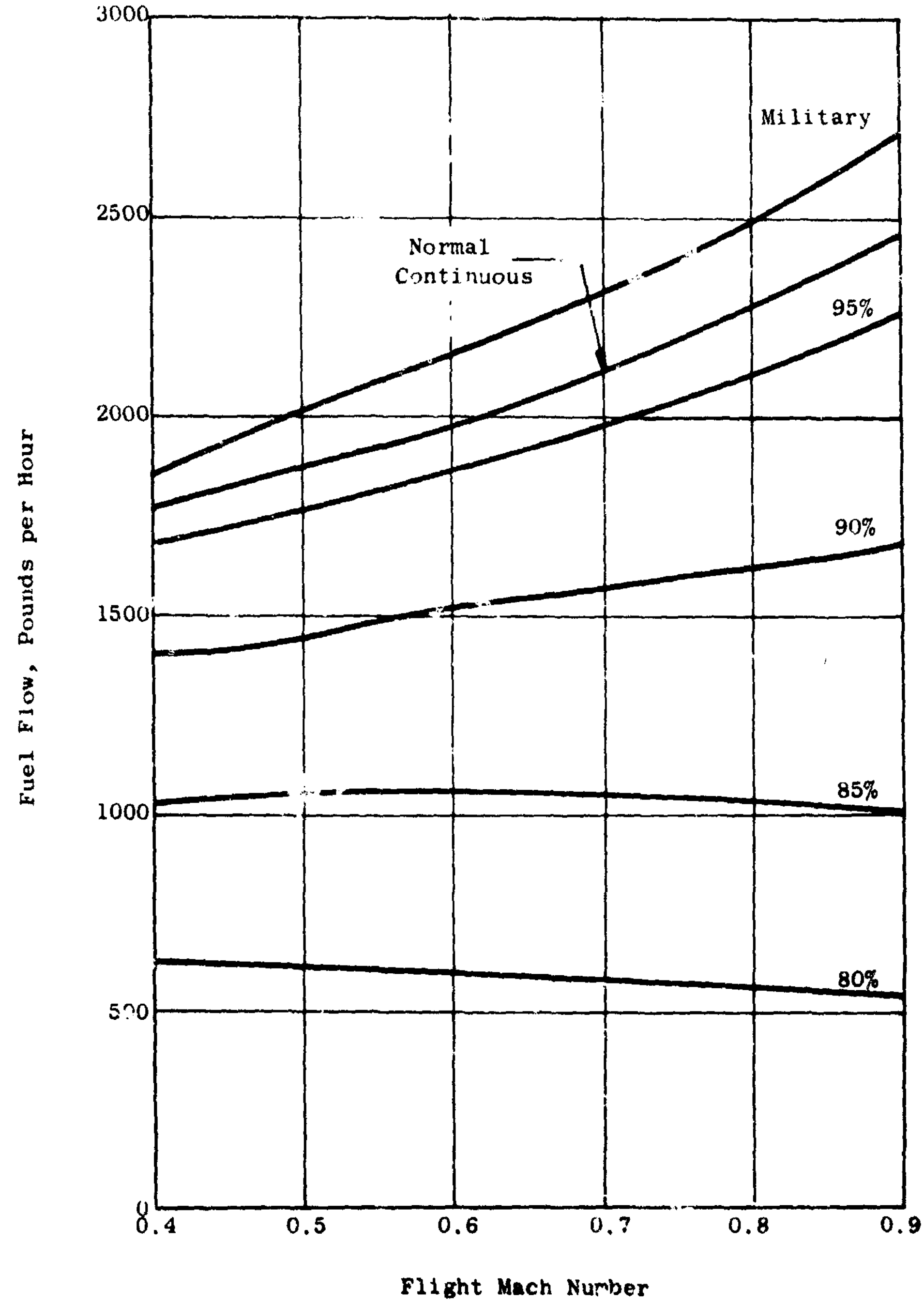

Figure 31. (C) GE1/J1B Core Engine Fuel Flow Versus Flight Mach Number for One Core Engine at an Altitude of 25,000 Feet. (U)

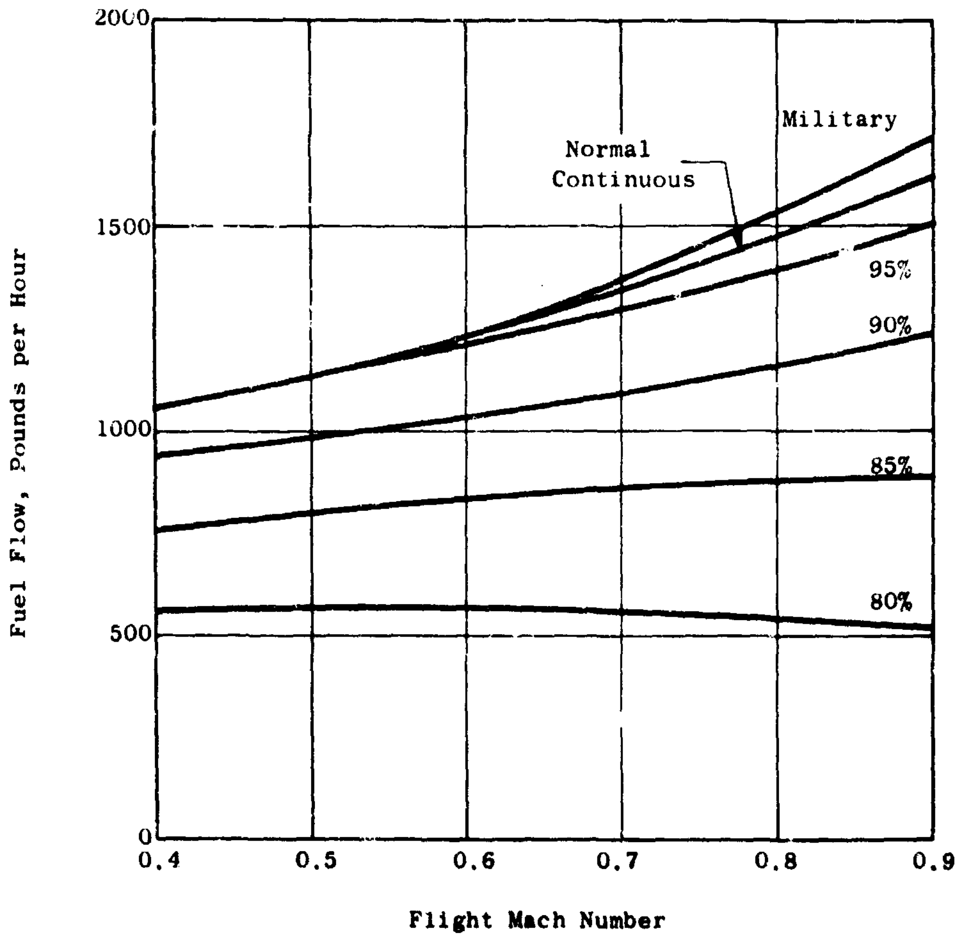

Figure 32. (C) GE1/J1. Core Engine Fuel Flow Versus Flight Mach Number for One Core Engine at an Altitude of 36,089 Feet. (U)

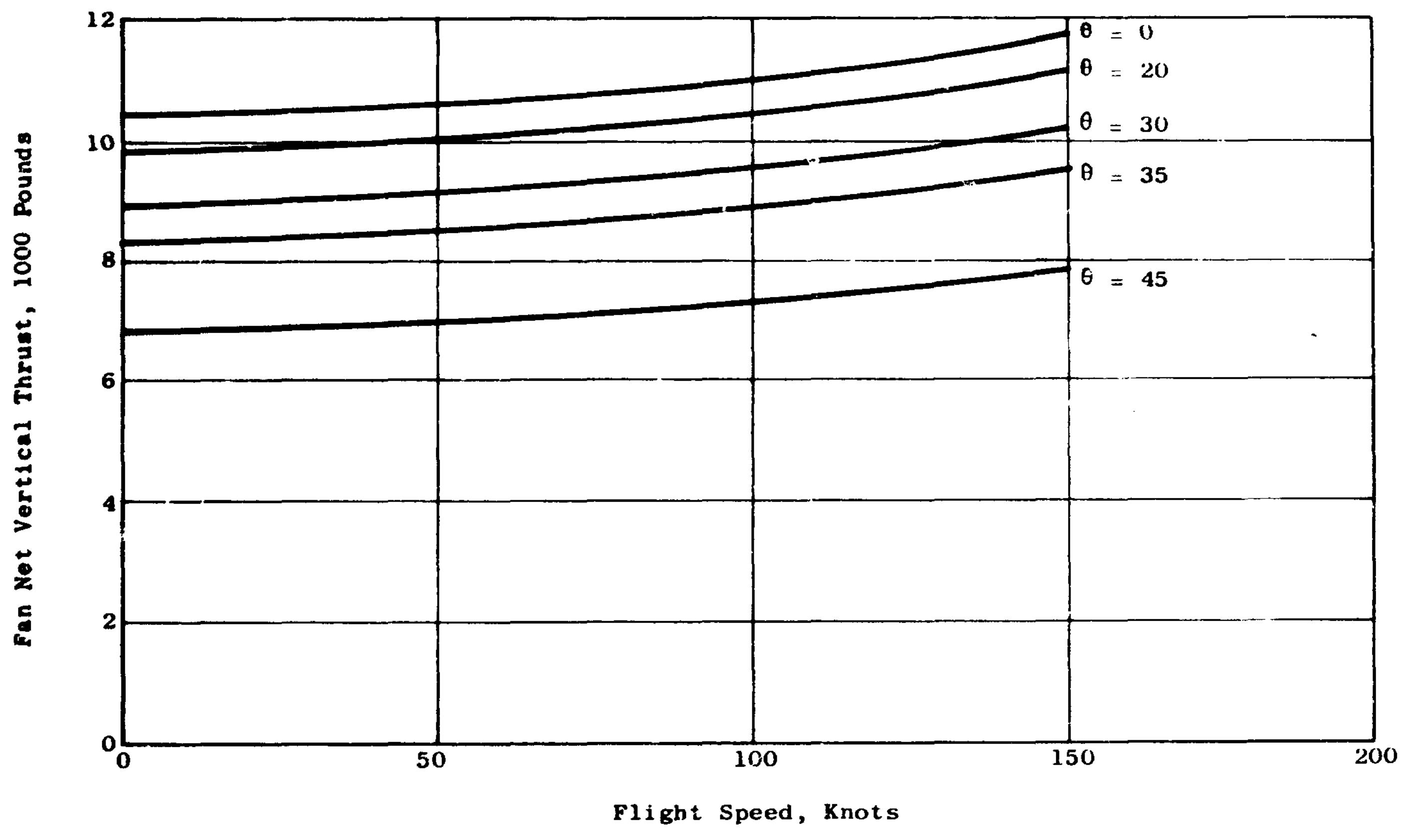

Figure 33. (U) Fan Net Vertical Thrust Versus Flight Speed at Sea Level, for 100-Percent Fan Inlet Recovery and Takeoff Engine Power Setting.

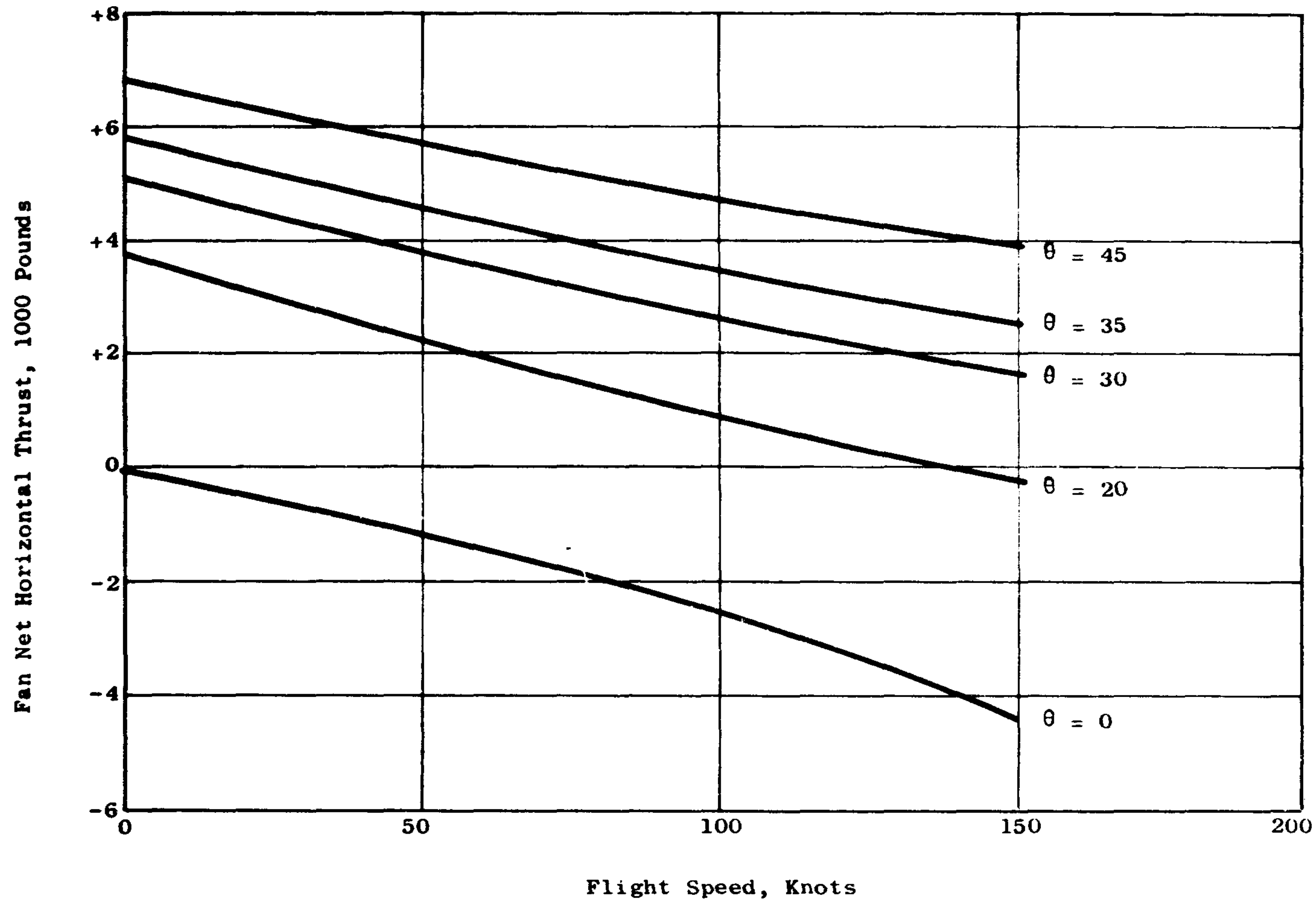

Figure 34. (U) Fan Net Horizontal Thrust Versus Flight Speed at Sea Level, for 100-Percent Fan Inlet Recovery and Takeoff Engine Power Setting.

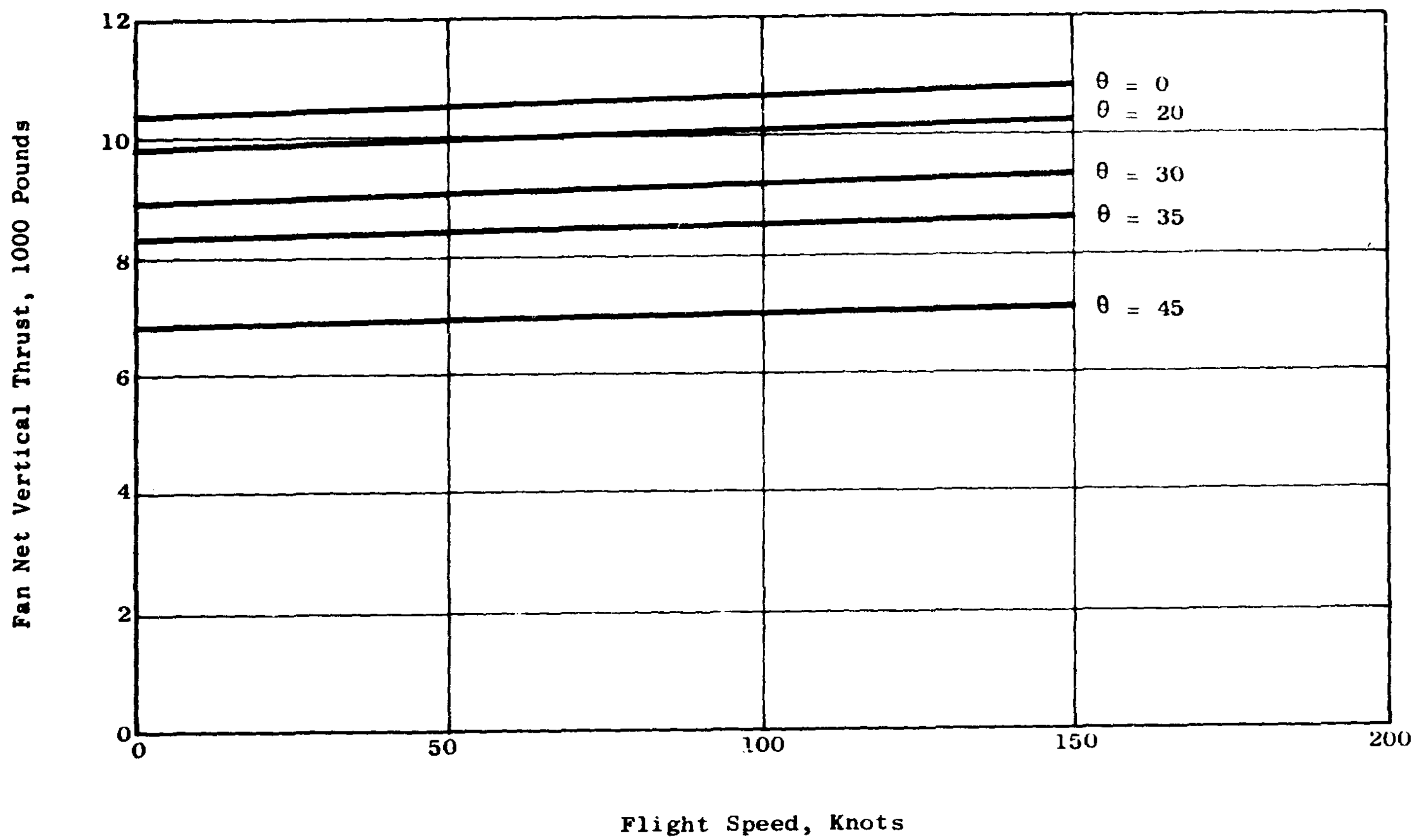

Figure 35. (U) Fan Net Vertical Thrust Versus Flight Speed at Sea Level, for 50-Percent Fan Inlet Recovery and Takeoff Engine Power Setting.

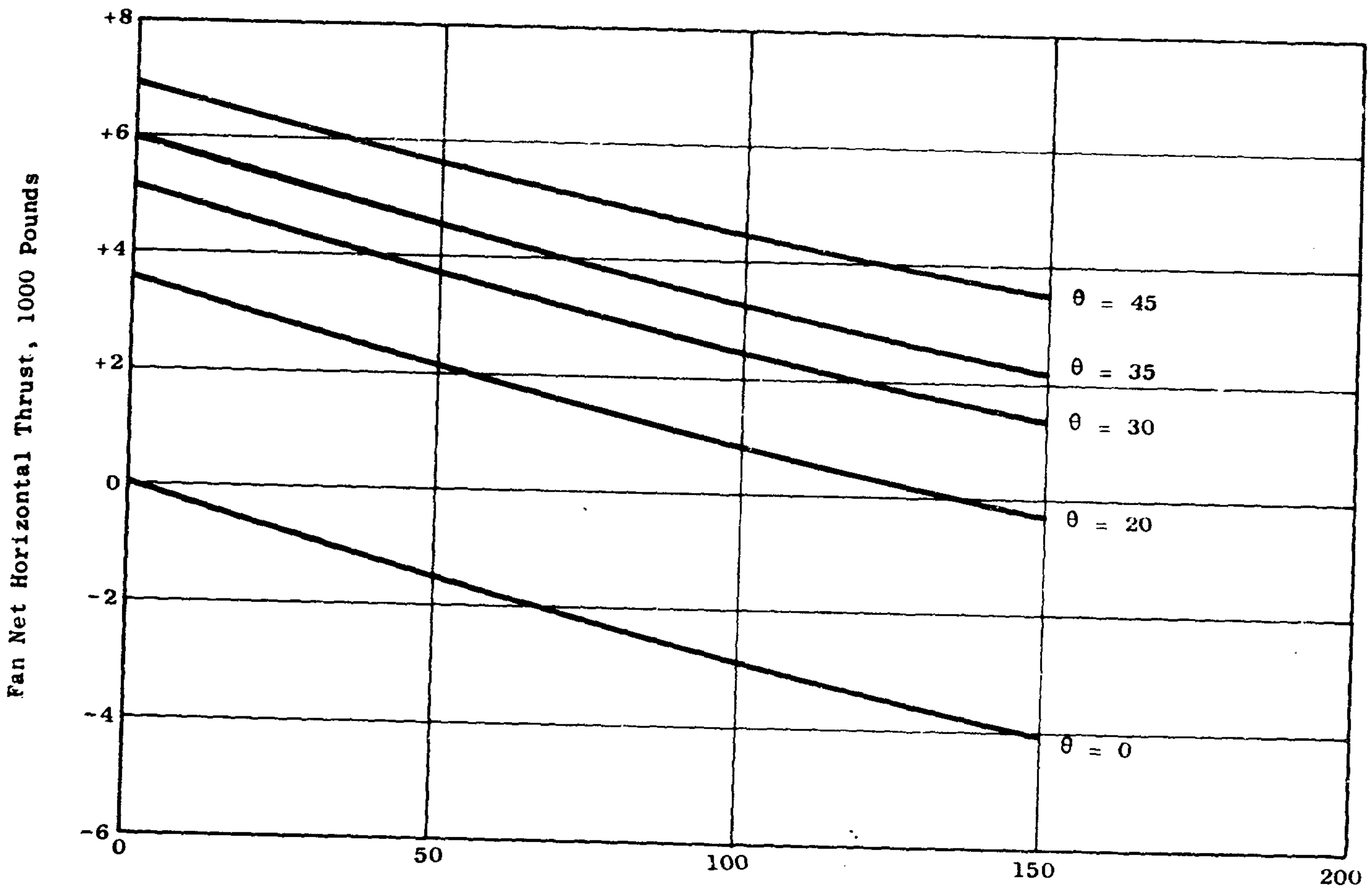

Figure 36. (U) Fan Net Horizontal Thrust Versus Flight Speed at Sea Level, for 50-Percent Fan Inlet Recovery and Takeoff Engine Power Setting.

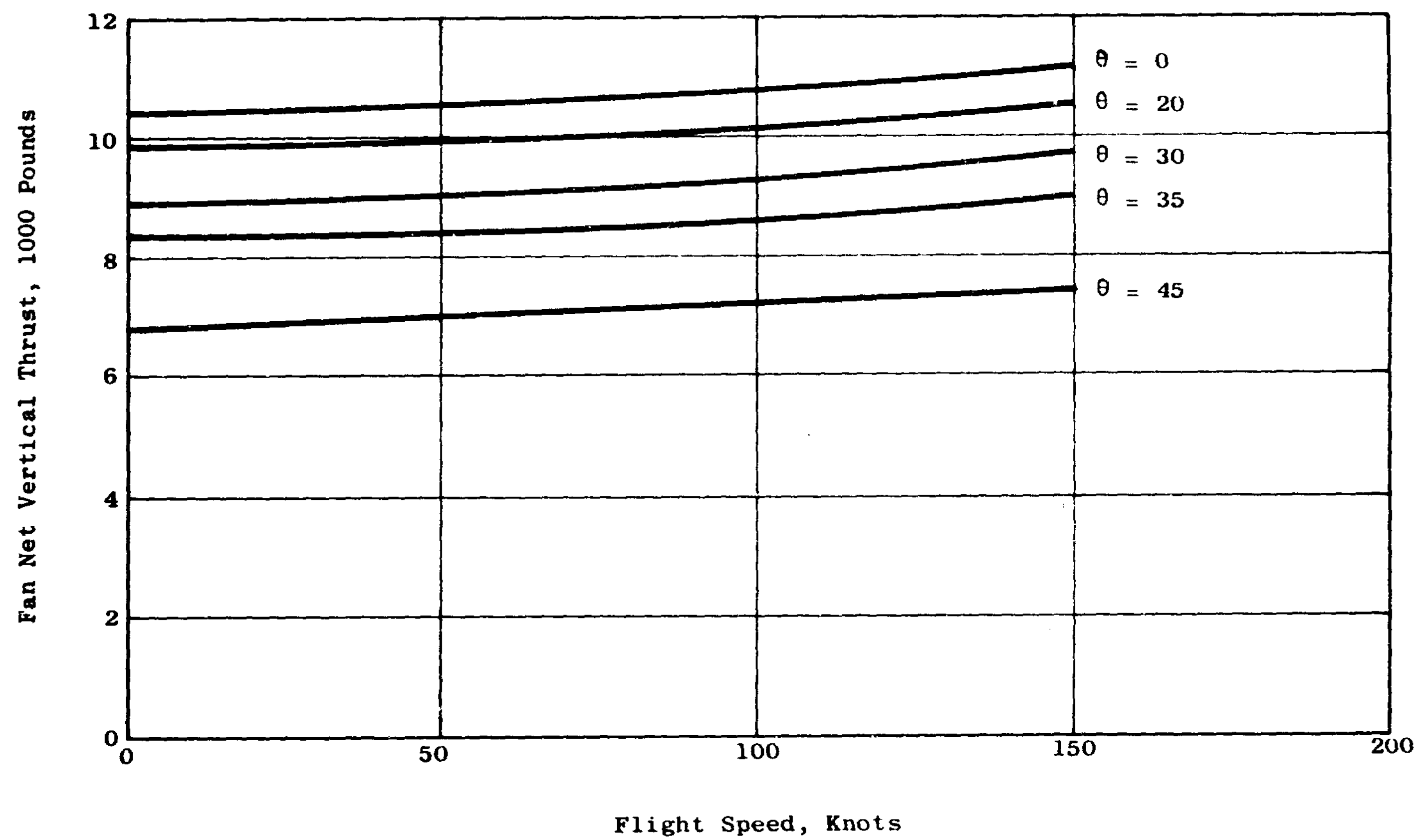

Figure 37. (U) Fan Net Vertical Thrust Versus Flight Speed at Sea Level, for 0-Percent Fan Inlet Recovery and Takeoff Engine Power Setting.

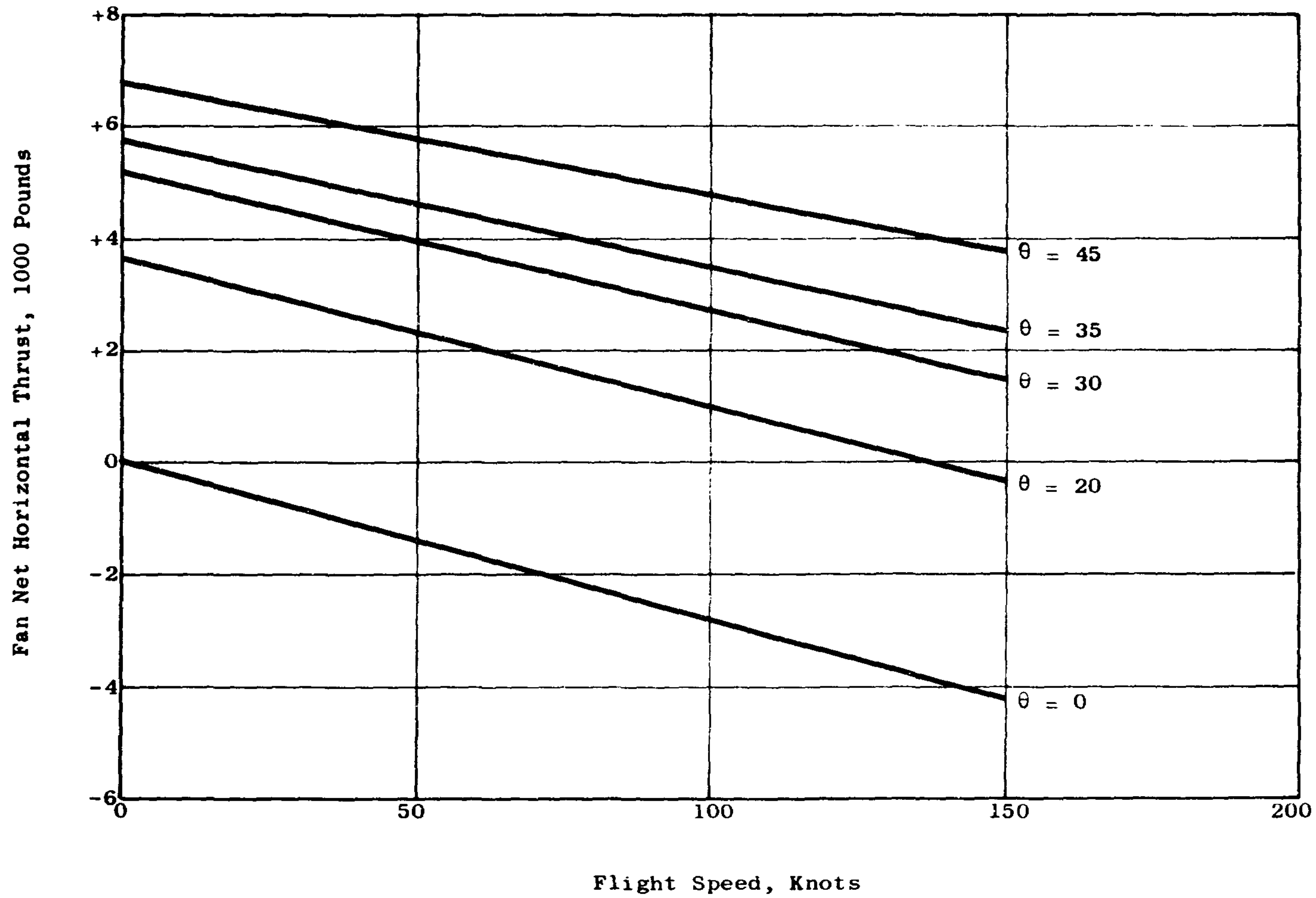

Figure 38. (U) Fan Net Horizontal Thrust Versus Flight Speed at Sea Level, for 0-Percent Fan Inlet Recovery and Takeoff Engine Power Setting.

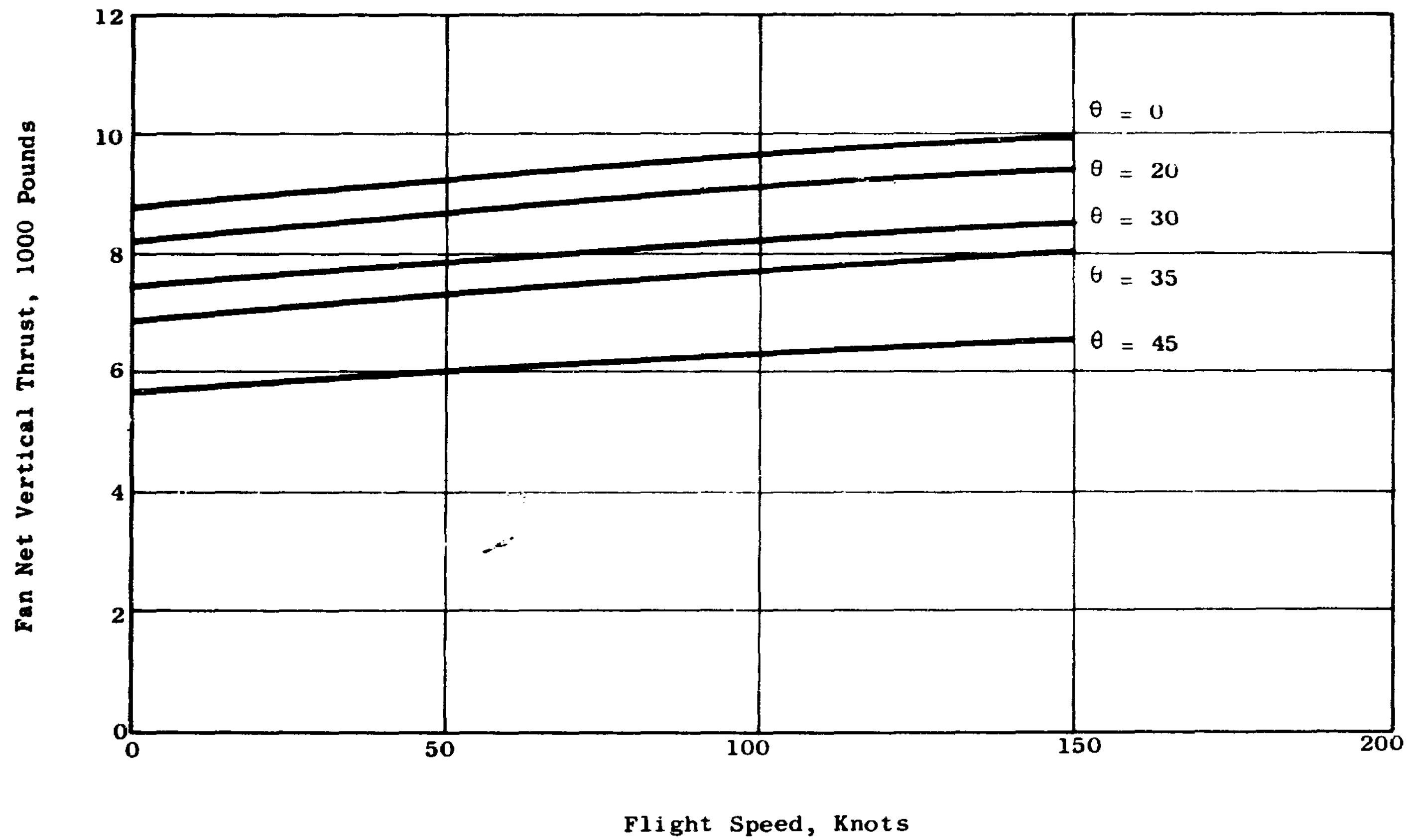

Figure 39. (U) Fan Net Vertical Thrust Versus Flight Speed at Sea Level, for 100-Percent Fan Inlet Recovery and Normal Continuous Engine Power Setting.

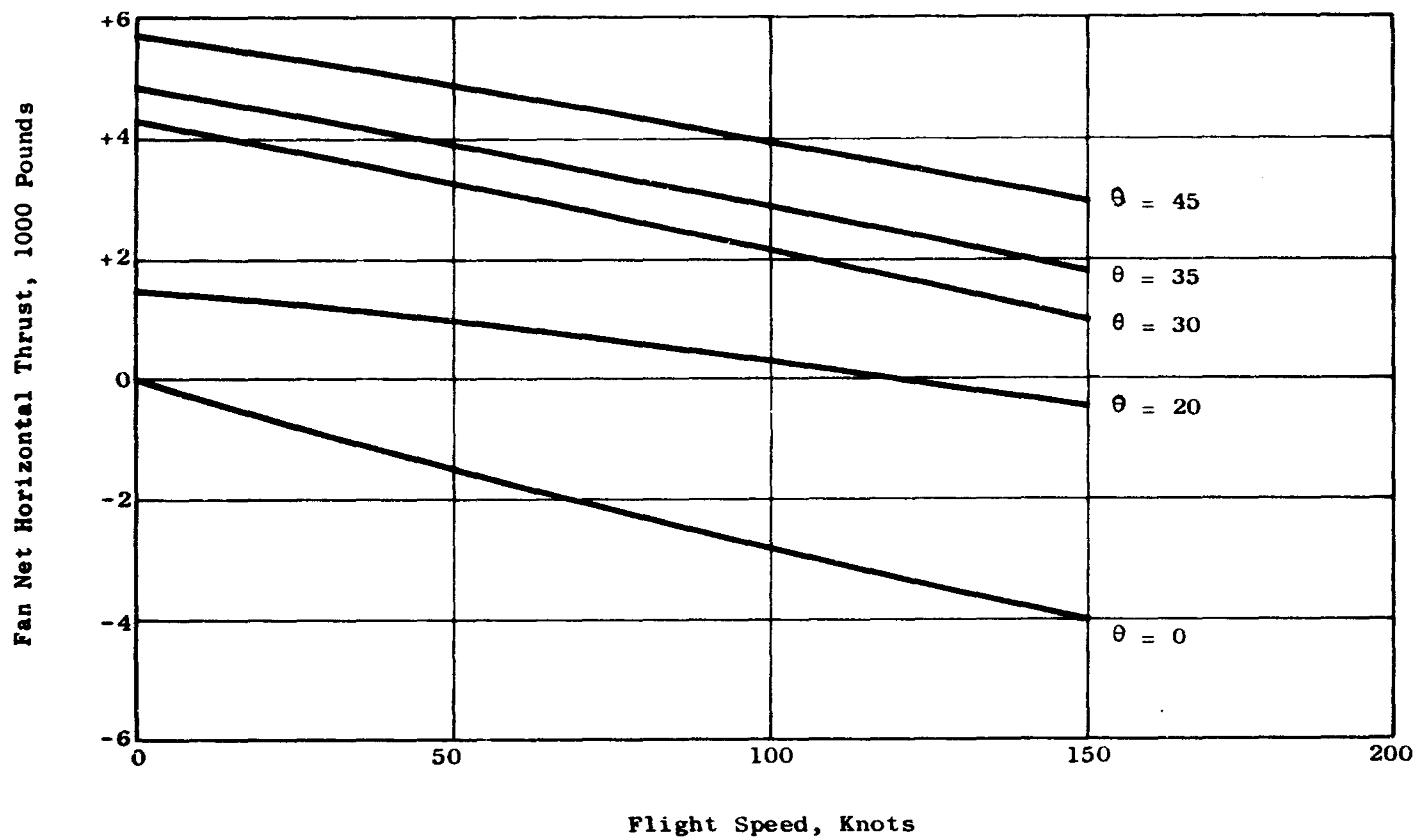

Figure 40. (U) Fan Net Horizontal Thrust Versus Flight Speed at Sea Level, for 100-Percent Fan Inlet Recovery and Normal Continuous Engine Power Setting.

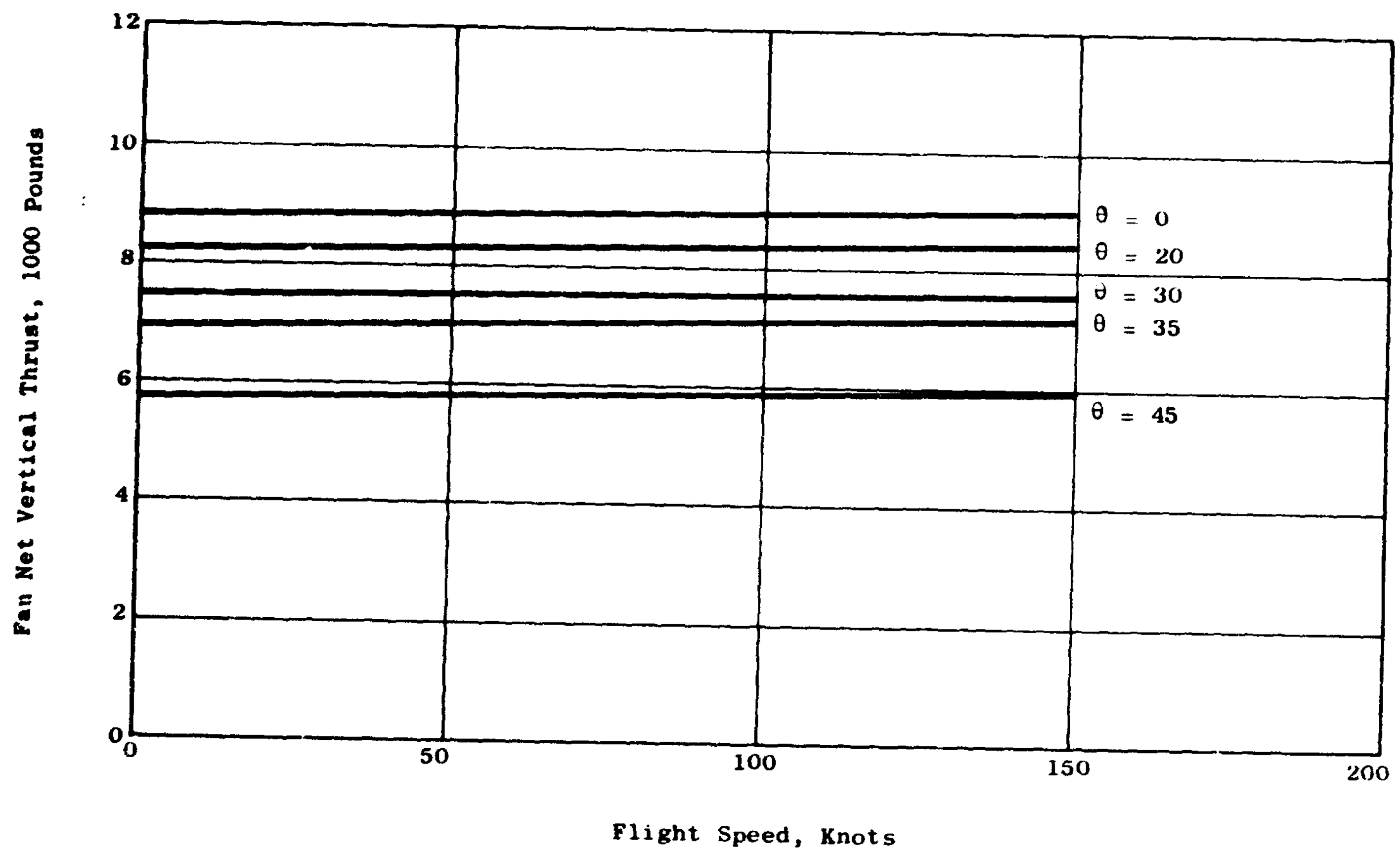

Figure 41. (U) Fan Net Vertical Thrust Versus Flight Speed at Sea Level, for 50-Percent Fan Inlet Recovery and Normal Continuous Engine Power Setting.

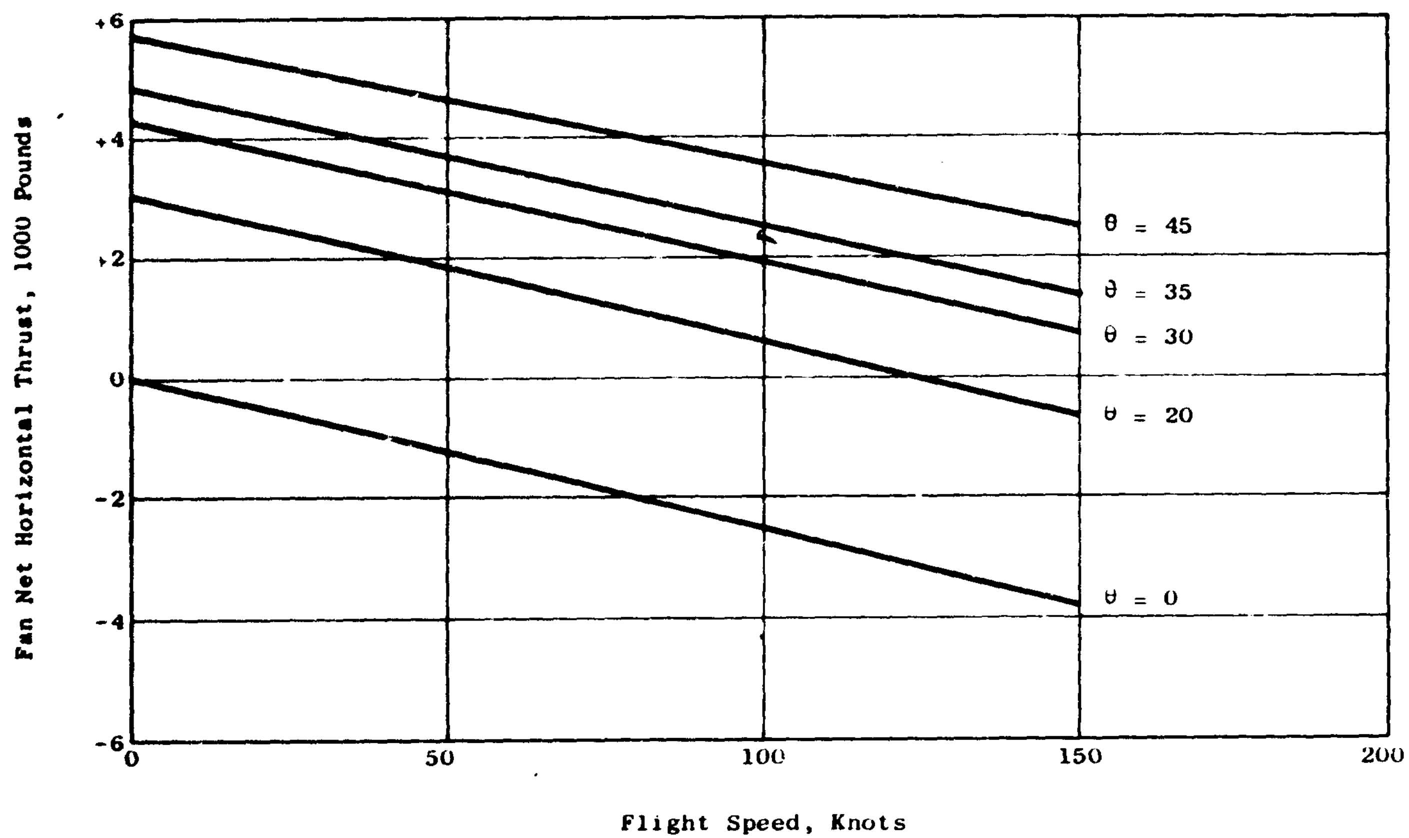

Figure 42. (U) **Fan Net Horizontal Thrust Versus Flight Speed at Sea Level, for 50-Percent Fan Inlet Recovery and Normal Continuous Engine Power Setting.**

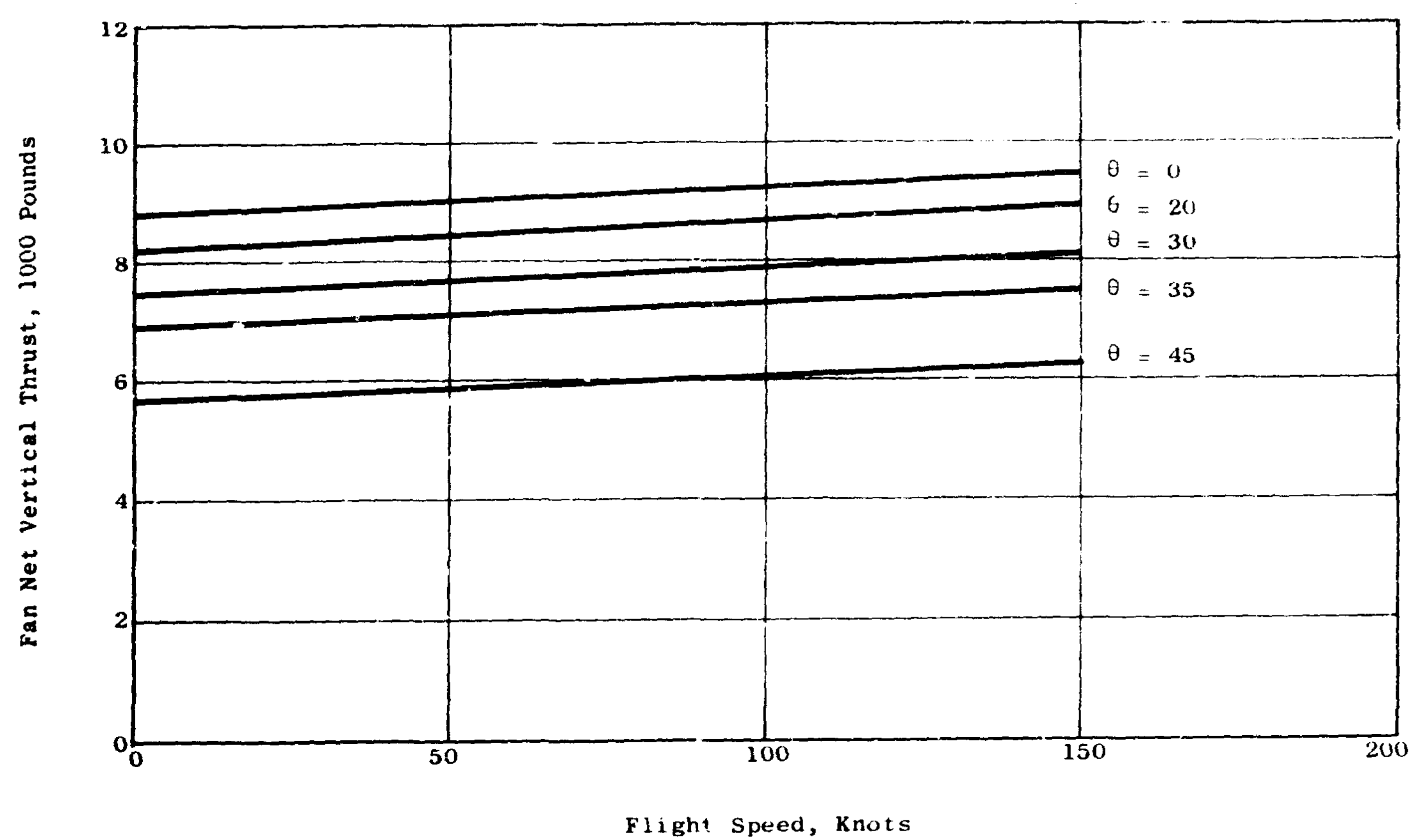

Figure 43. (U) **Fan Net Vertical Thrust Versus Flight Speed at Sea Level, for 0-Percent Fan Inlet Recovery and Normal Continuous Engine Power Setting.**

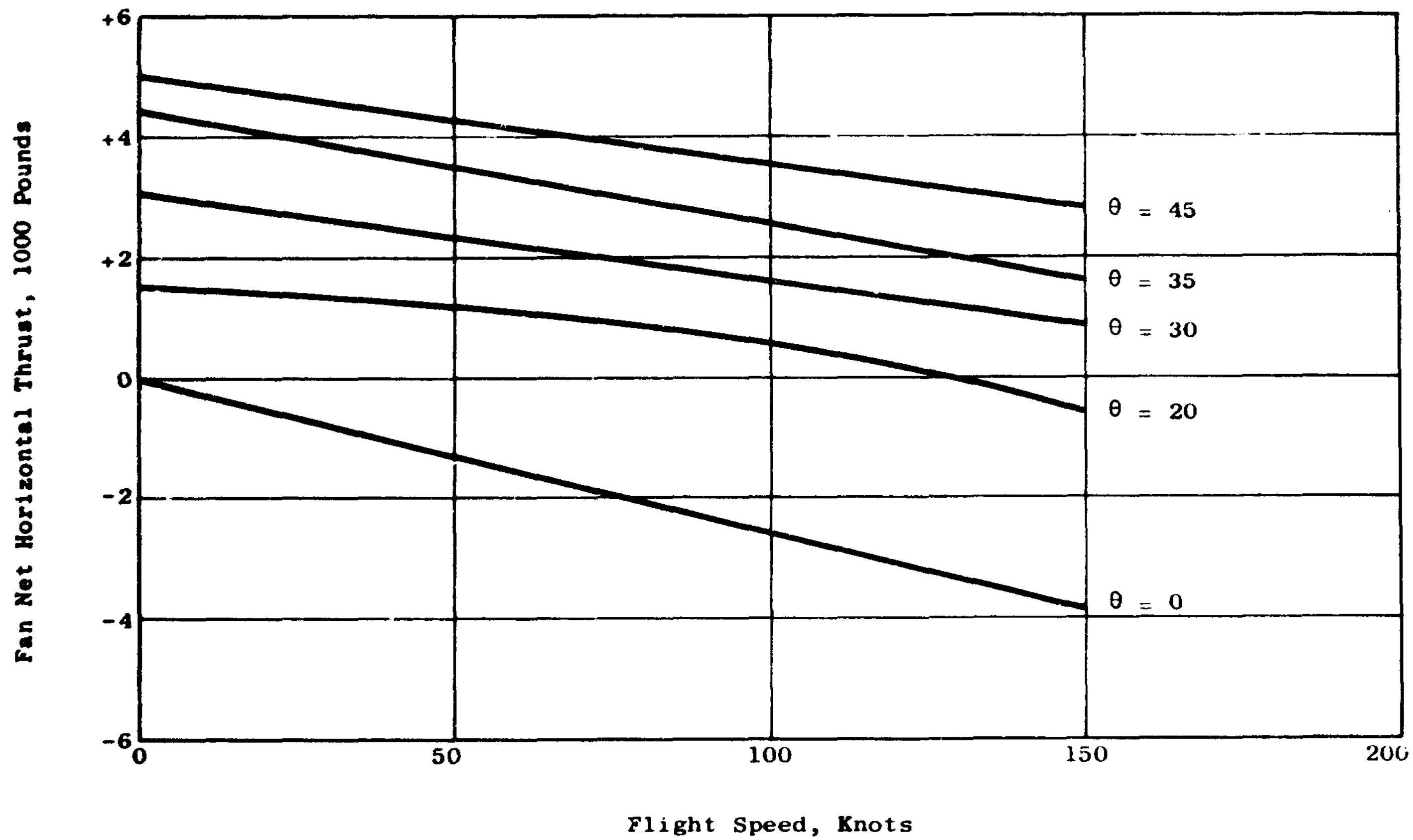

Figure 44. (U) Fan Net Horizontal Thrust Versus Flight Speed at Sea Level, for 0-Percent Fan Inlet Recovery and Normal Continuous Engine Power Setting.

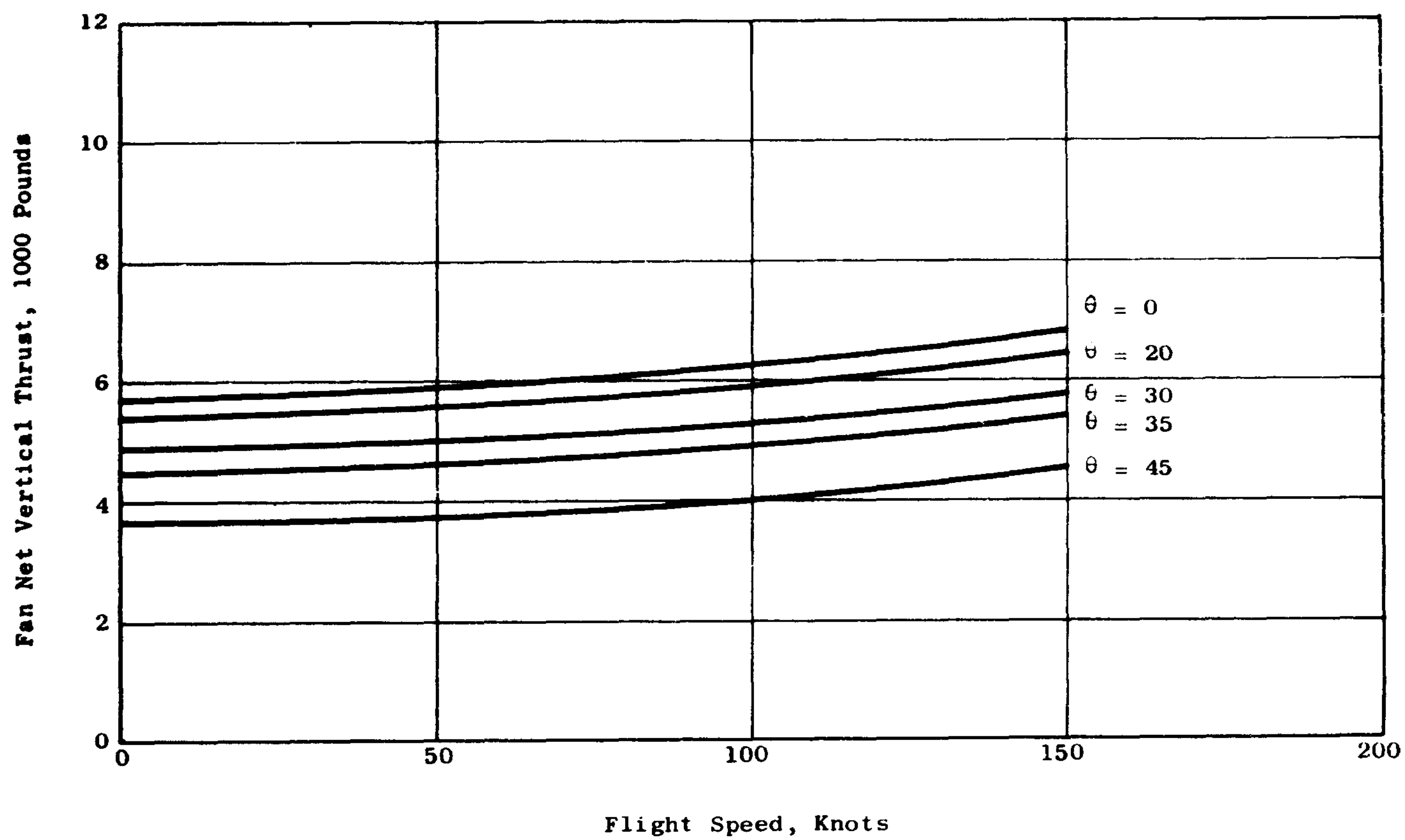

Figure 45. (U) Fan Net Vertical Thrust Versus Flight Speed at Sea Level, for 100-Percent Fan Inlet Recovery and 90-Percent Engine Power Setting.

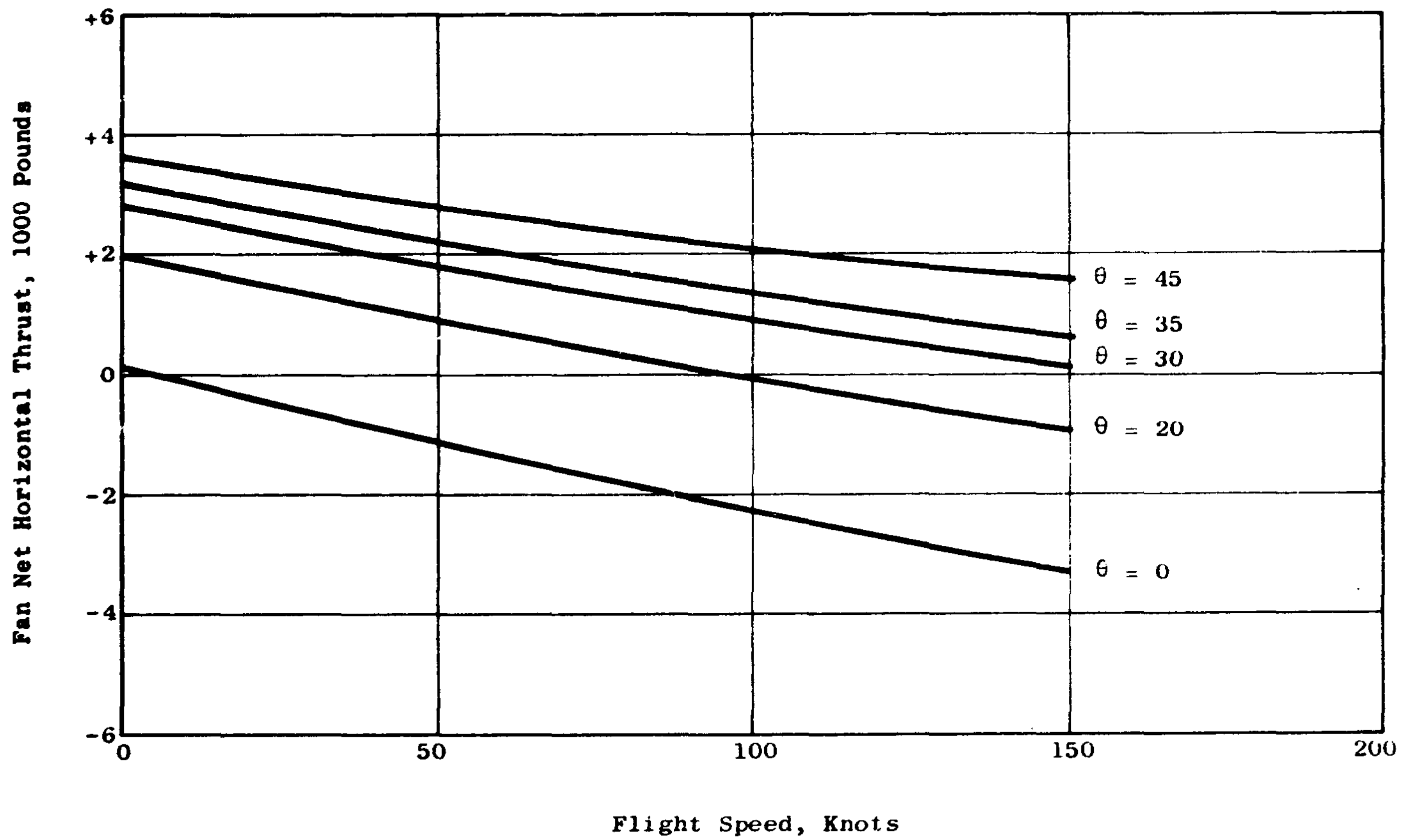

Figure 46. (U) Fan Net Horizontal Thrust Versus Flight Speed at Sea Level, for 100-Percent Fan Inlet Recovery and 90-Percent Engine Power Setting.

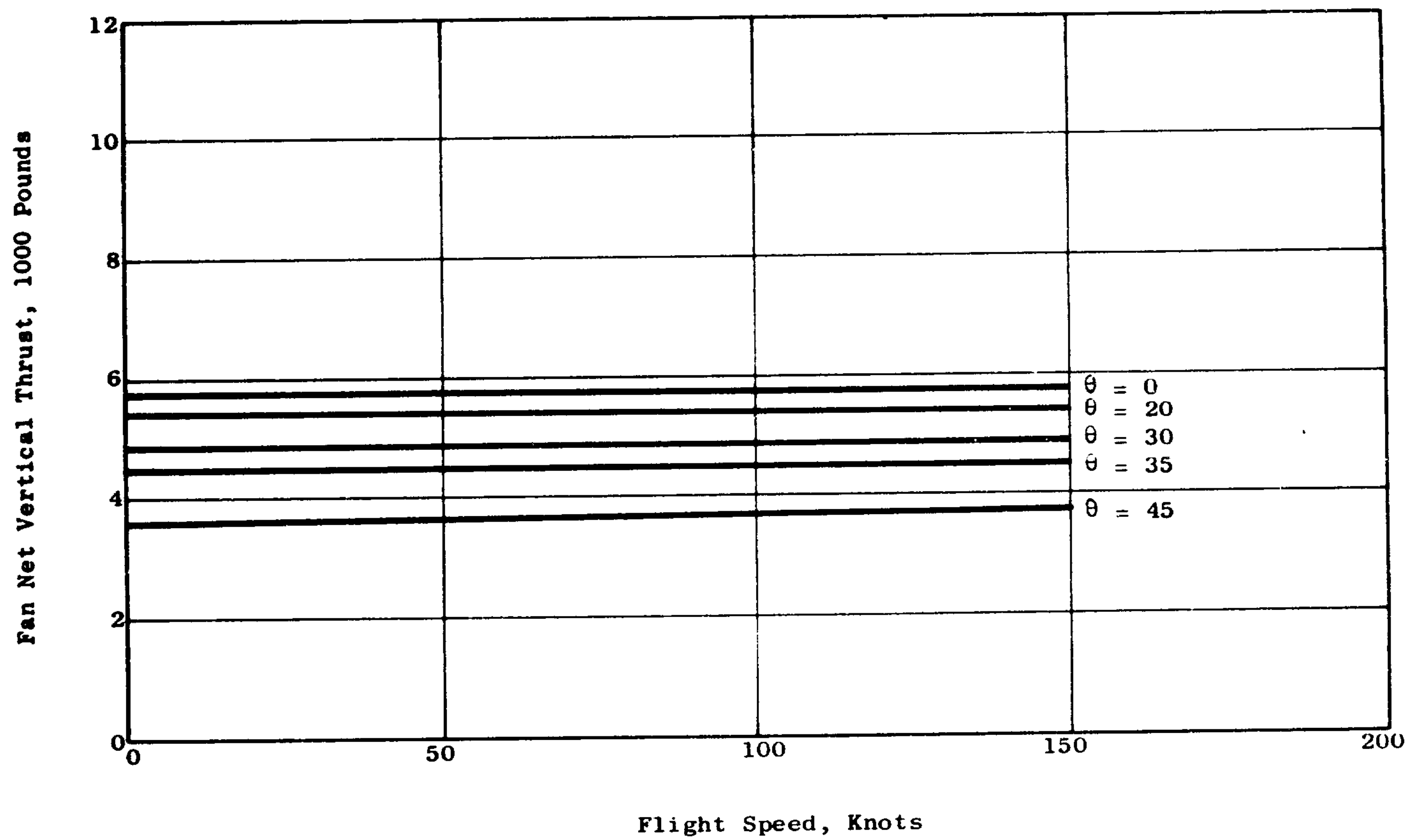

Figure 47. (U) Fan Net Vertical Thrust Versus Flight Speed at Sea Level, for 50-Percent Fan Inlet Recovery and 90-Percent Engine Power Setting.

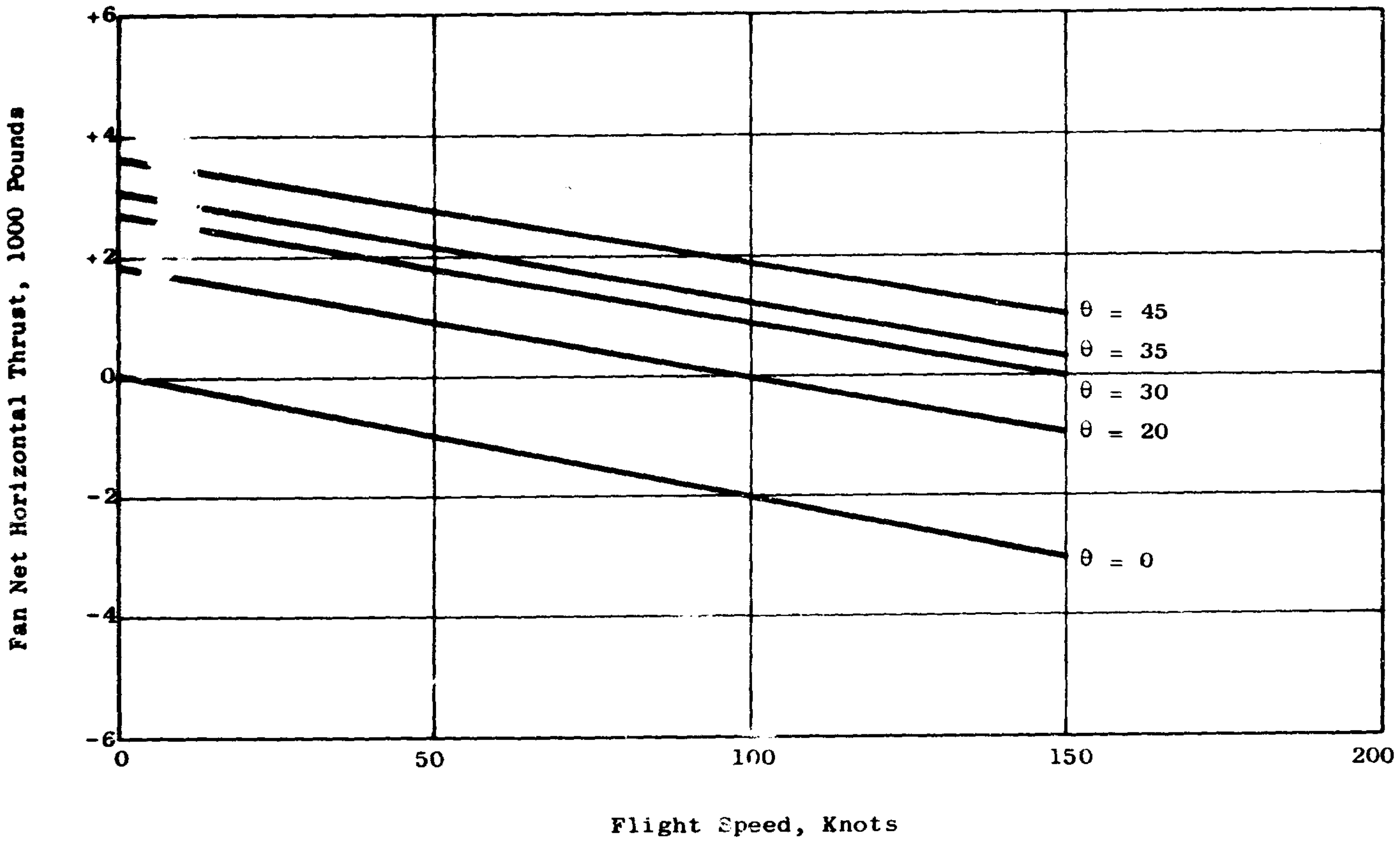

Figure 48. (U) Fan Net Horizontal Thrust Versus Flight Speed at Sea Level, for 50-Percent Fan Inlet Recovery and 90-Percent Engine Power Setting.

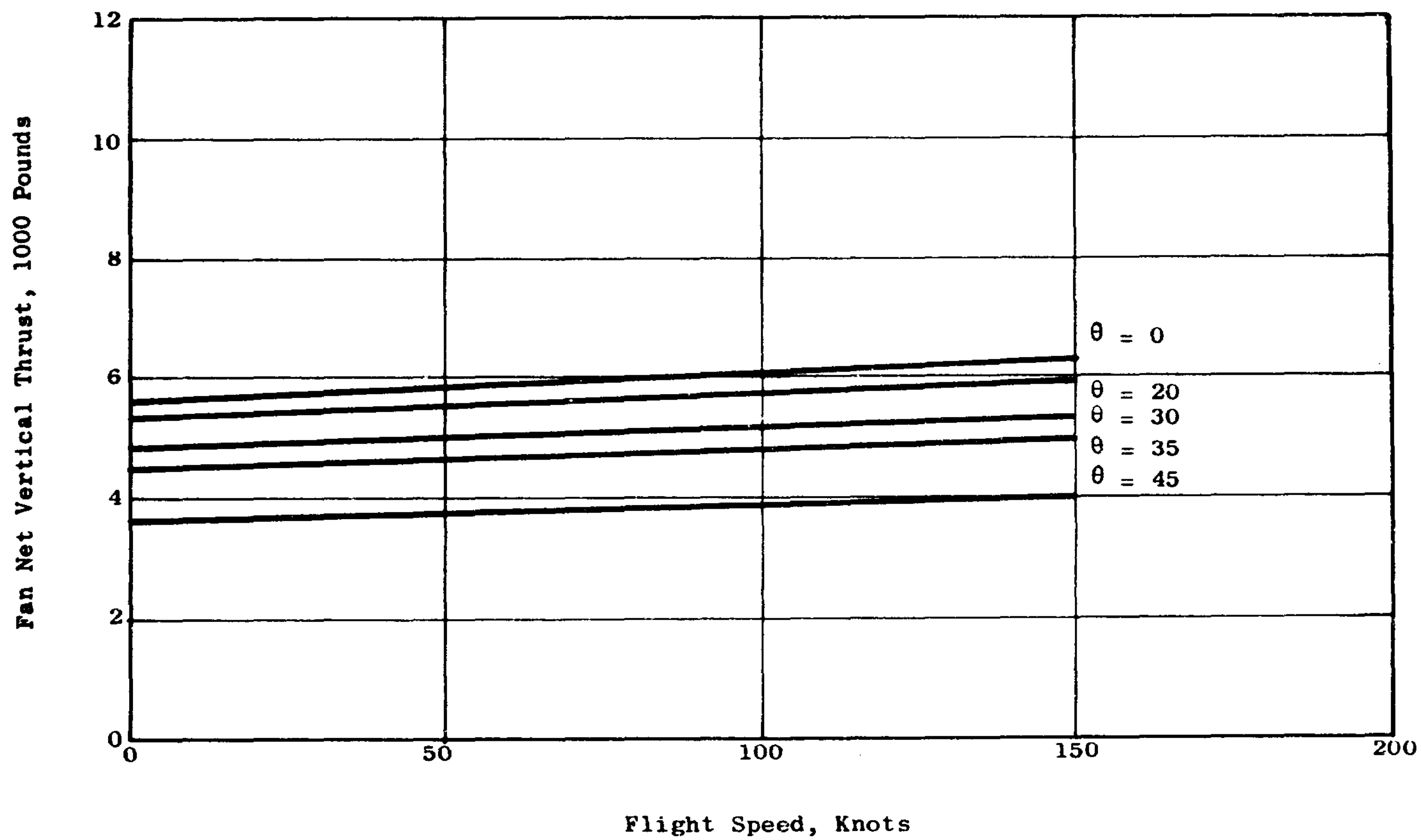

Figure 49. (U) Fan Net Vertical Thrust Versus Flight Speed at Sea Level, for 0-Percent Fan Inlet Recovery and 90-Percent Engine Power Setting.

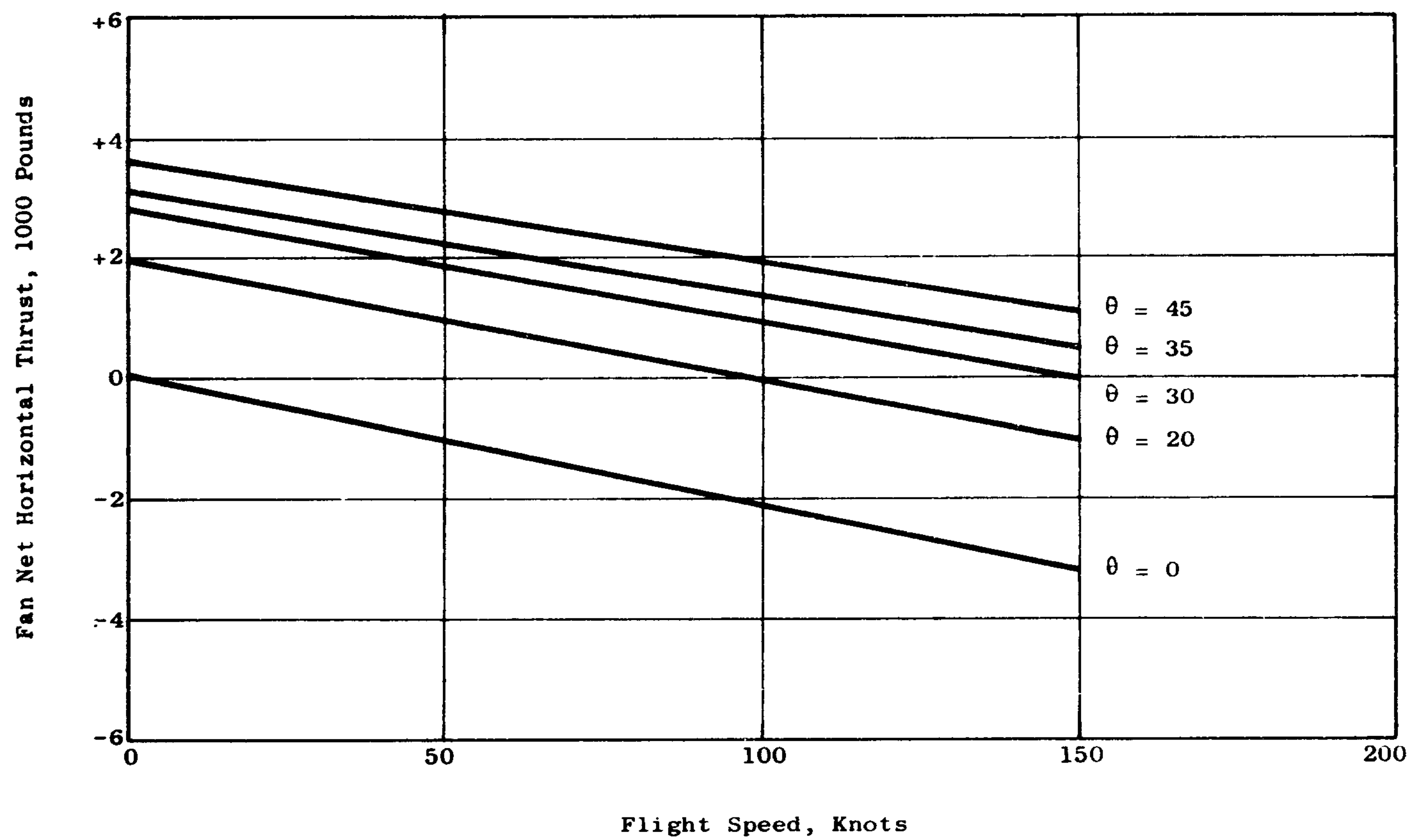

Figure 50. (U) Fan Net Horizontal Thrust Versus Flight Speed at Sea Level, for 0-Percent Fan Inlet Recovery and 90-Percent Engine Power Setting.

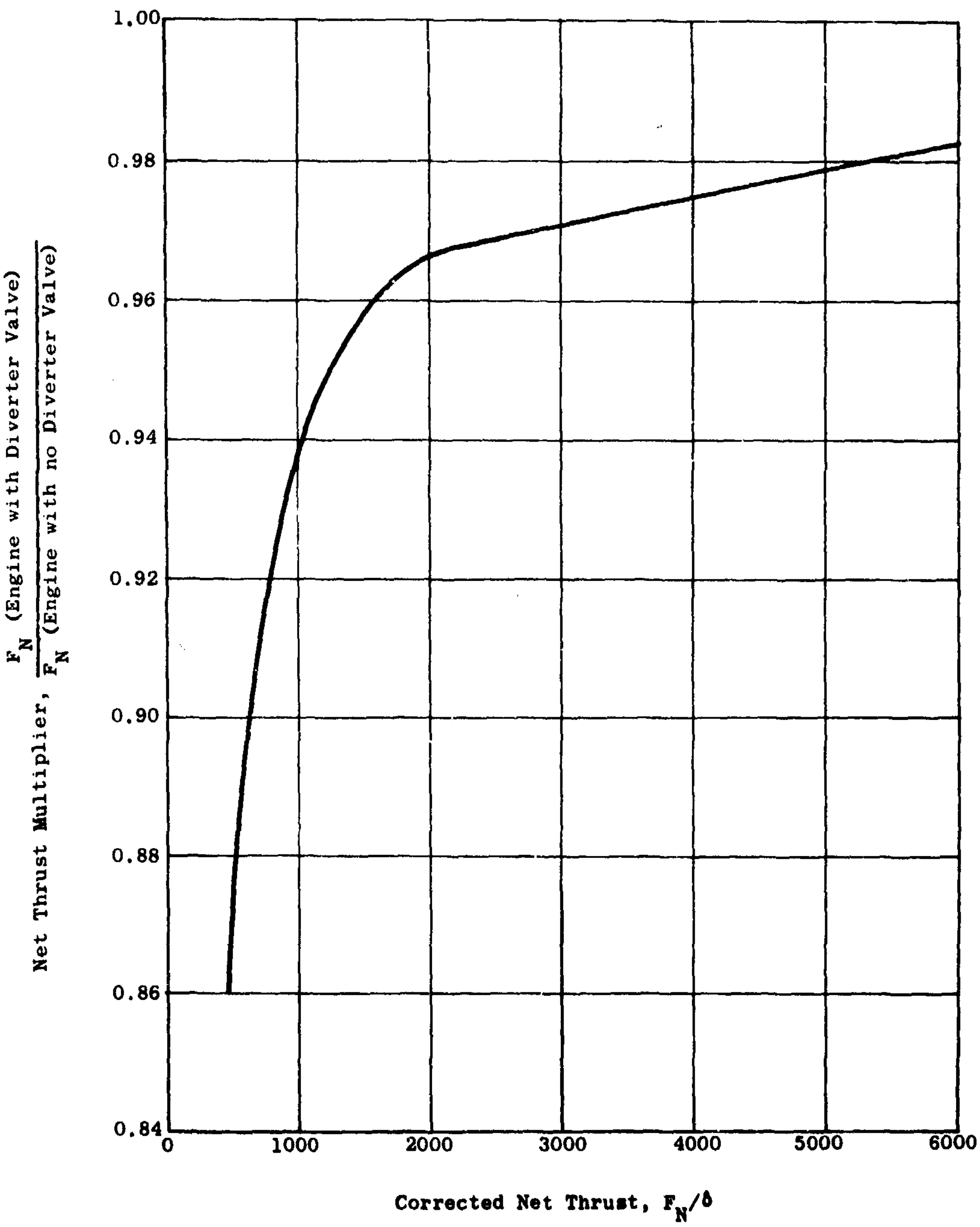

Figure 51. (C) Diverter Valve Correction to GE1/J1B Core Engine Cruise Performance. (U)

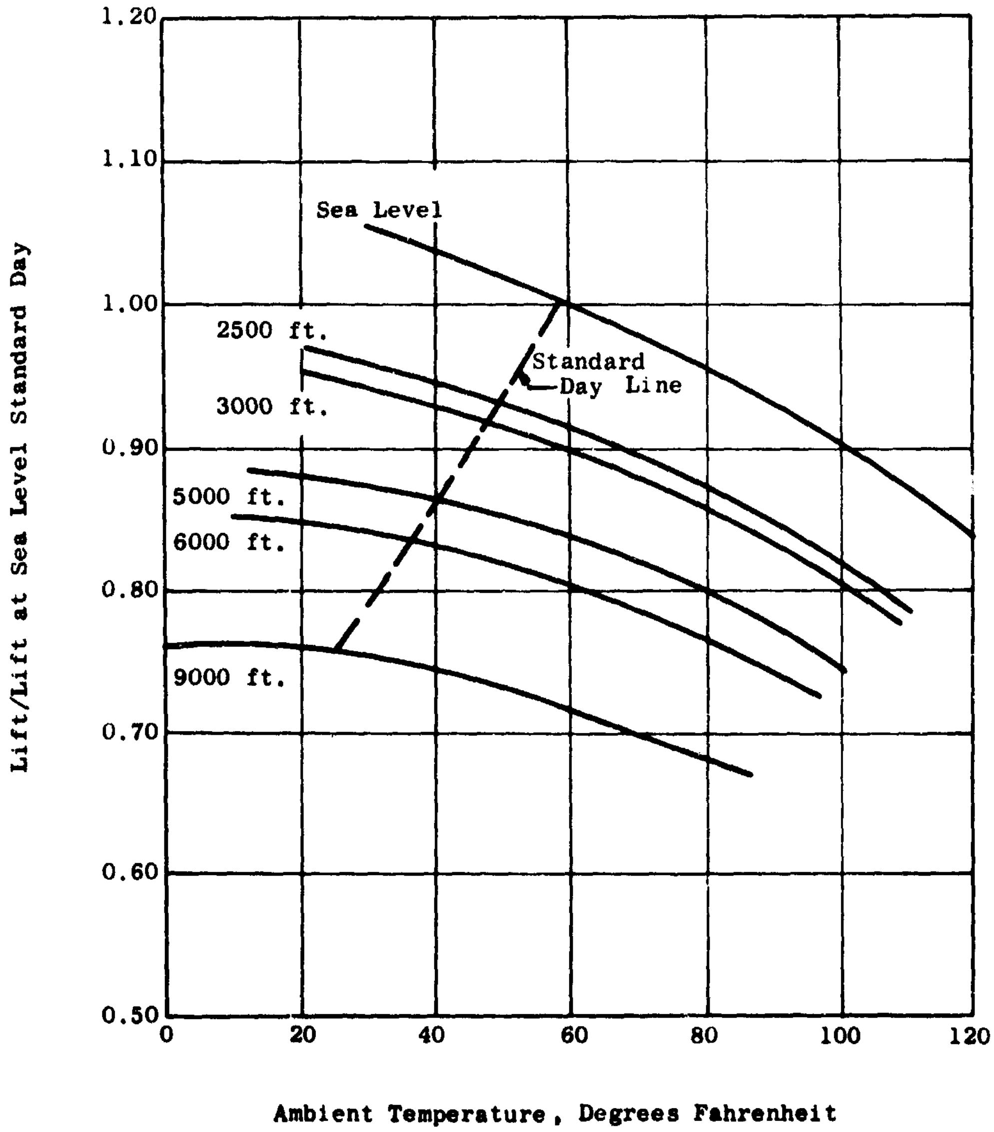

Figure 52. (C) Effect of Ambient Temperature and Altitude on GE1/J1B Lift Fan Performance. (U)

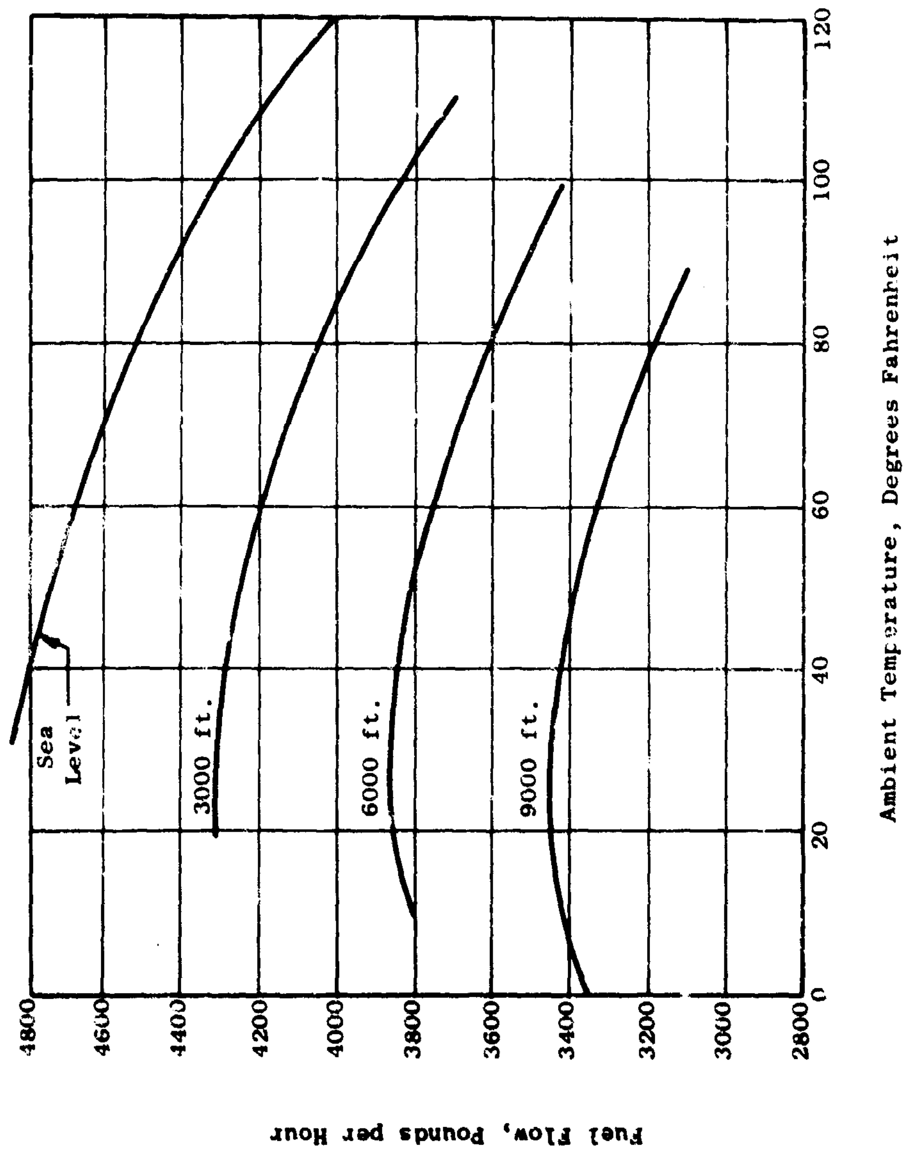

Figure 53. (C) Effect of Ambient Temperature and Altitude on GE1/J1B Core Engine Fuel Flow. (U)

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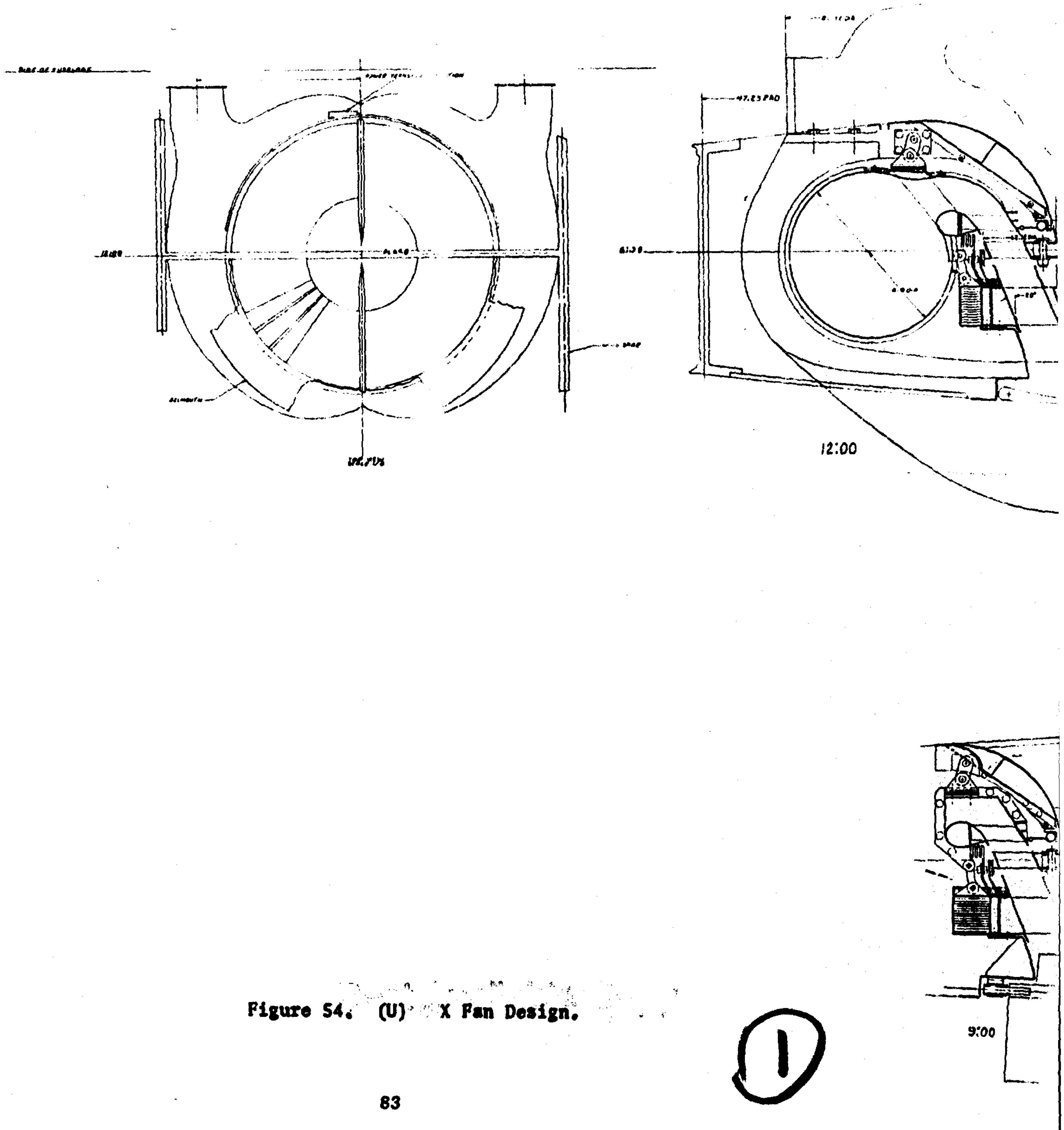
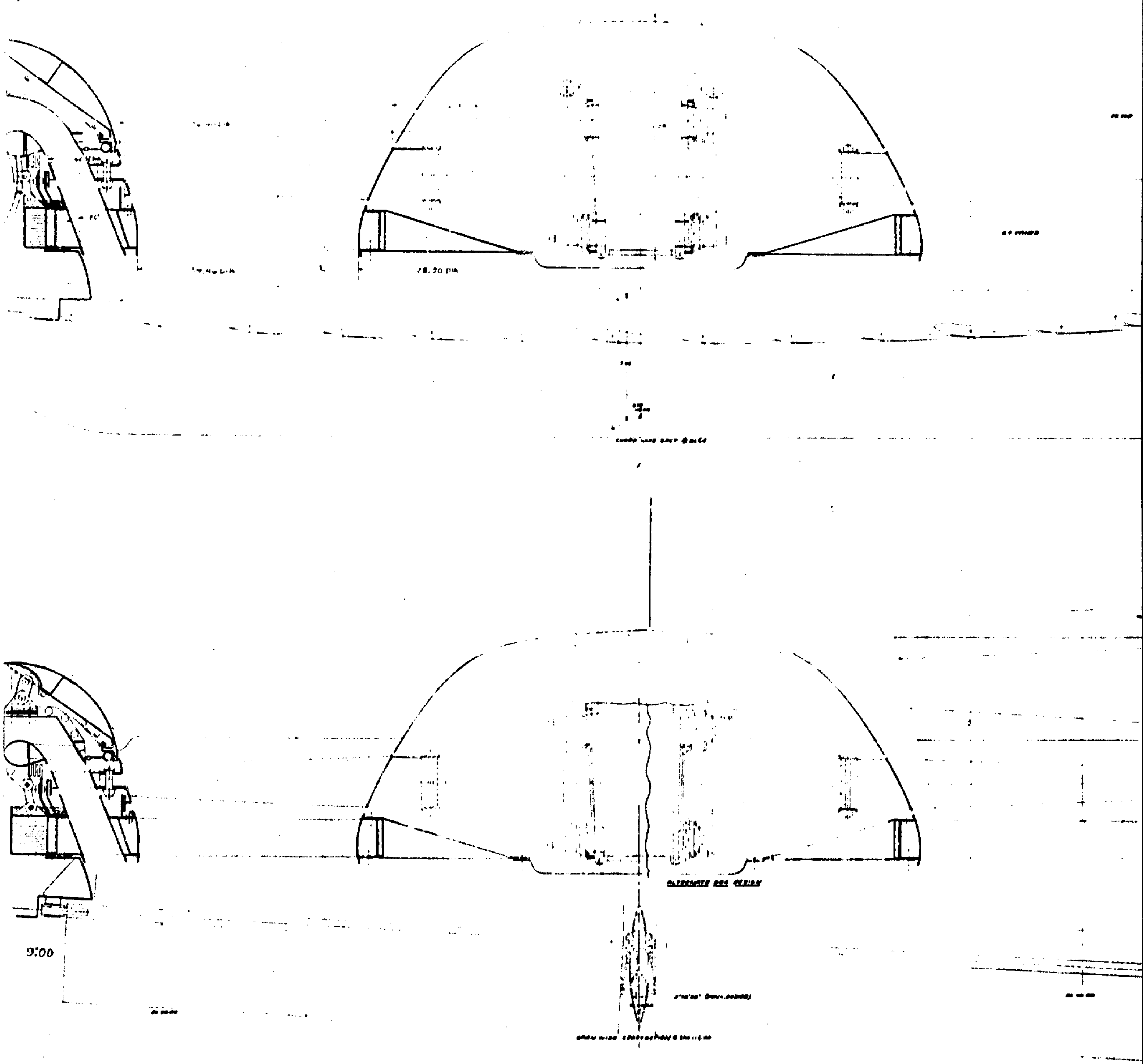


Figure 54. (U) X Fan Design.

(1)



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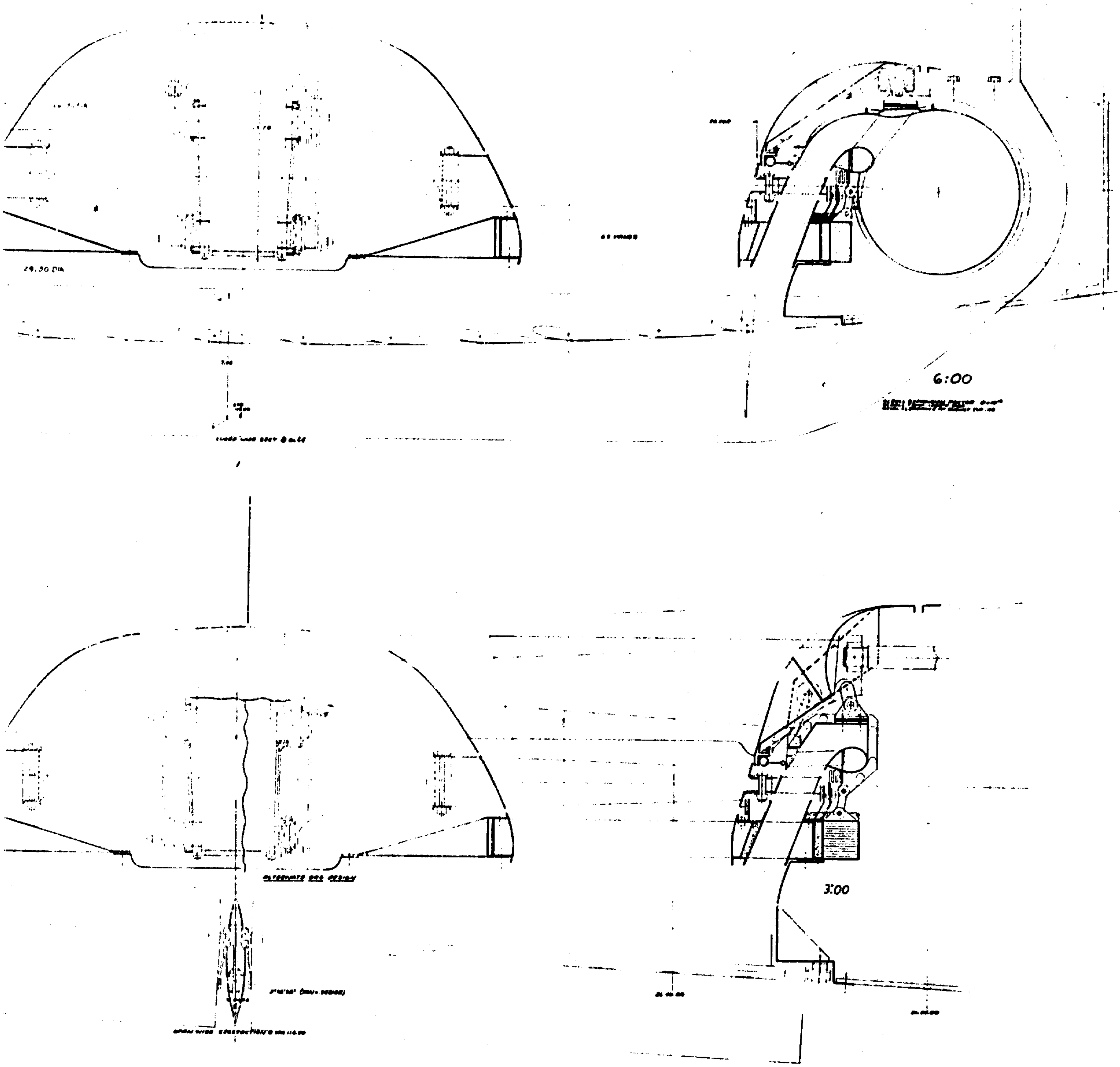

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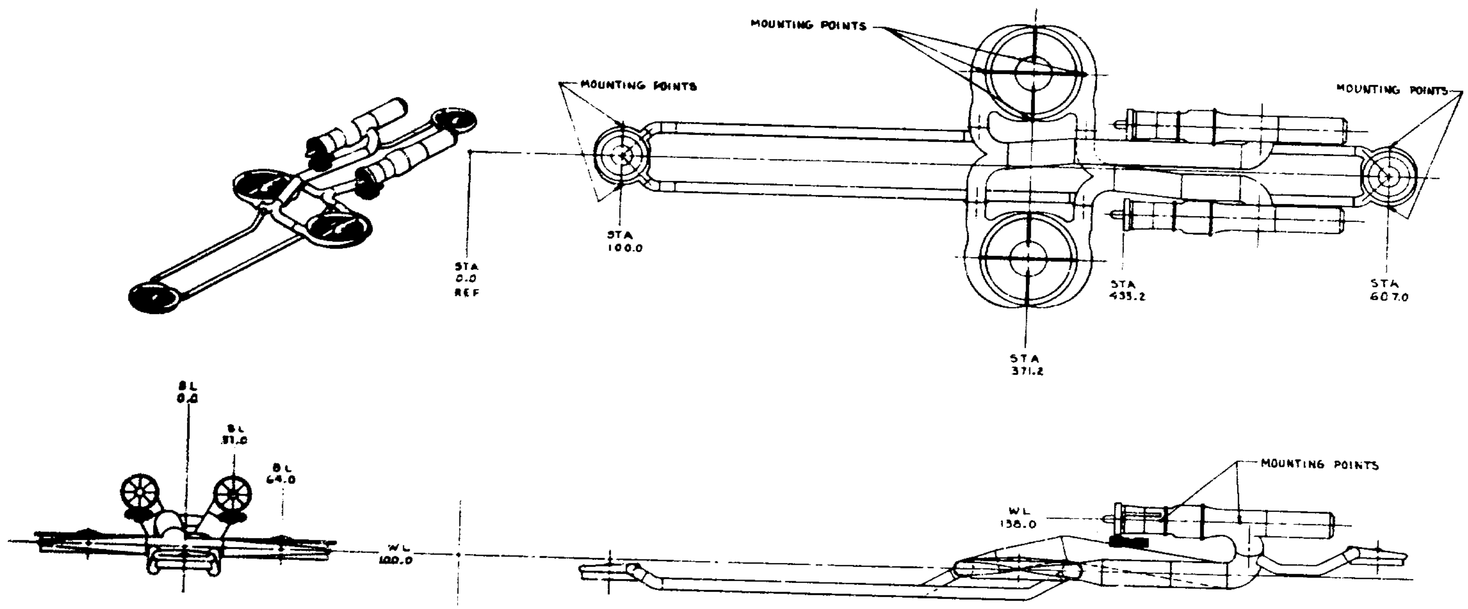


Figure 55. (U) LFX Fan Design.

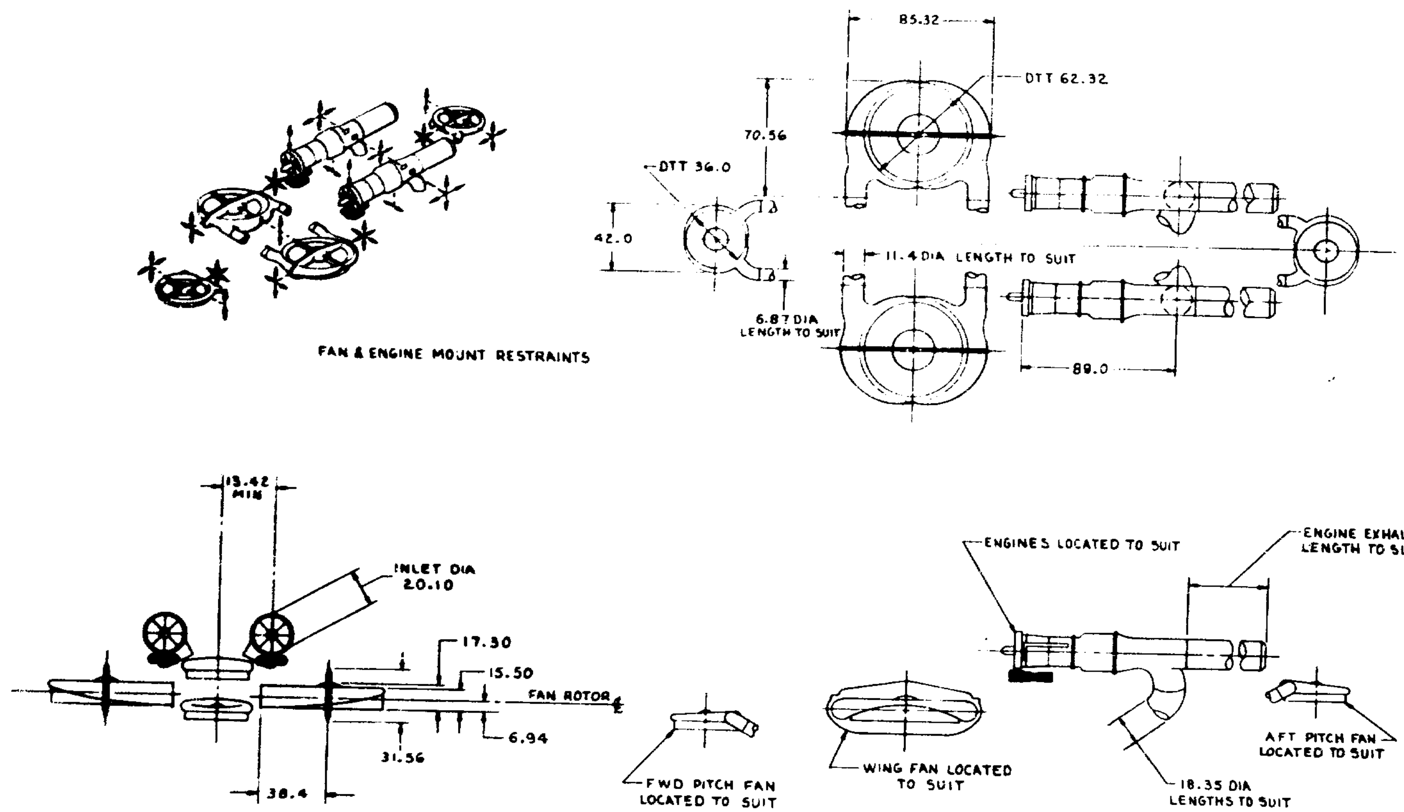


Figure 56. (U) LFX Installation Study.

3.11.1.3 Exit Louver Actuation: The exit louver system shown in Figure 68 features an aerodynamically balanced louver system requiring a single actuator provided by the airframe manufacturer, and located either at the front or rear of the rear frame main strut. Estimated maximum required force is 3600 pounds. Estimated required stroke is 3½ inches.

3.12 Dry Weight of Complete System: The dry weight of the complete LFX system, consisting of two turbojet engines, two diverter valves, two wing lift fans and two fuselage lift fans, is estimated to be 2974 pounds. Table IV shows the estimated weight of major components. Weights shown include power transfer scrolls for both the wing fan and fuselage fan, and exit louvers for the wing fans. Mount hardware for the fan, engine, and diverter valve, as well as all pneumatic ducting and flexible joints between the diverter valve exit flange and the scroll inlet flange, is considered aircraft components. Hardware weights are not included in the table.

3.14 Flight Maneuver Forces: Objective conditions for the system while operating in the turbojet mode are shown in Figure 57. Objective conditions for the fan mode are shown in Figure 58.

3.16.3.3 Mount Loads: Reaction forces at the power plant component mounting points are given for flight maneuver forces and moments. These applied forces and moments are positive in the directions shown in Figure 59. The mount loads given in the following tables are positive in the directions shown in the diagram accompanying each table, and act on the propulsion system component.

Table V shows the mount loads for the wing lift fans. Where two signs are given, the upper sign refers to the right-hand wing lift fan and the lower sign refers to the left-hand wing lift fan. When viewed from above the aircraft, facing forward, the left-hand wing lift fan rotates counterclockwise and the right-hand wing lift fan rotates clockwise.

Table VI shows the mount loads for the fuselage lift fans. Where two signs are given, the upper sign refers to the nose lift fan and the lower sign refers to the tail lift fan. The fuselage lift fans rotate counterclockwise when viewed from above the aircraft.

Table VII shows the mount loads for the core engines, using an assumed mounting system. Other mounting arrangements are possible. The arrangement will be determined by the final airplane configuration. The core engines rotate clockwise when viewed from the rear of the engine, looking forward.

The aircraft angular velocity induces a gyroscopic moment in the propulsion system mounts. Using the sign convention of Figure 59, the directions of these induced gyroscopic moments are shown in Table VIII. The polar inertia and angular velocity of the propulsion system components are shown in Table IX.

TABLE IV (C)
WEIGHT SUMMARY (U)

	Group Weight
Wing Fan Group	
Rotor	
Front frame, including bellmouth inlet and with provisions for mounting closure doors	
Rear frame, with exit louvers and linkage, but without actuator	
Scroll, with power transfer capabilities, but without actuator	
Insulation	
TOTAL	570 Pounds
Fuselage Fan Group	
Rotor	
Front frame, with no bellmouth	
Rear frame, with no exit louvers	
Scroll, with power transfer capability	
Insulation	
TOTAL	154 Pounds
Core Engine-Diverter Valve Group	763 Pounds
TOTAL	1,487 Pounds
TOTAL PER AIRCRAFT	2,974 Pounds

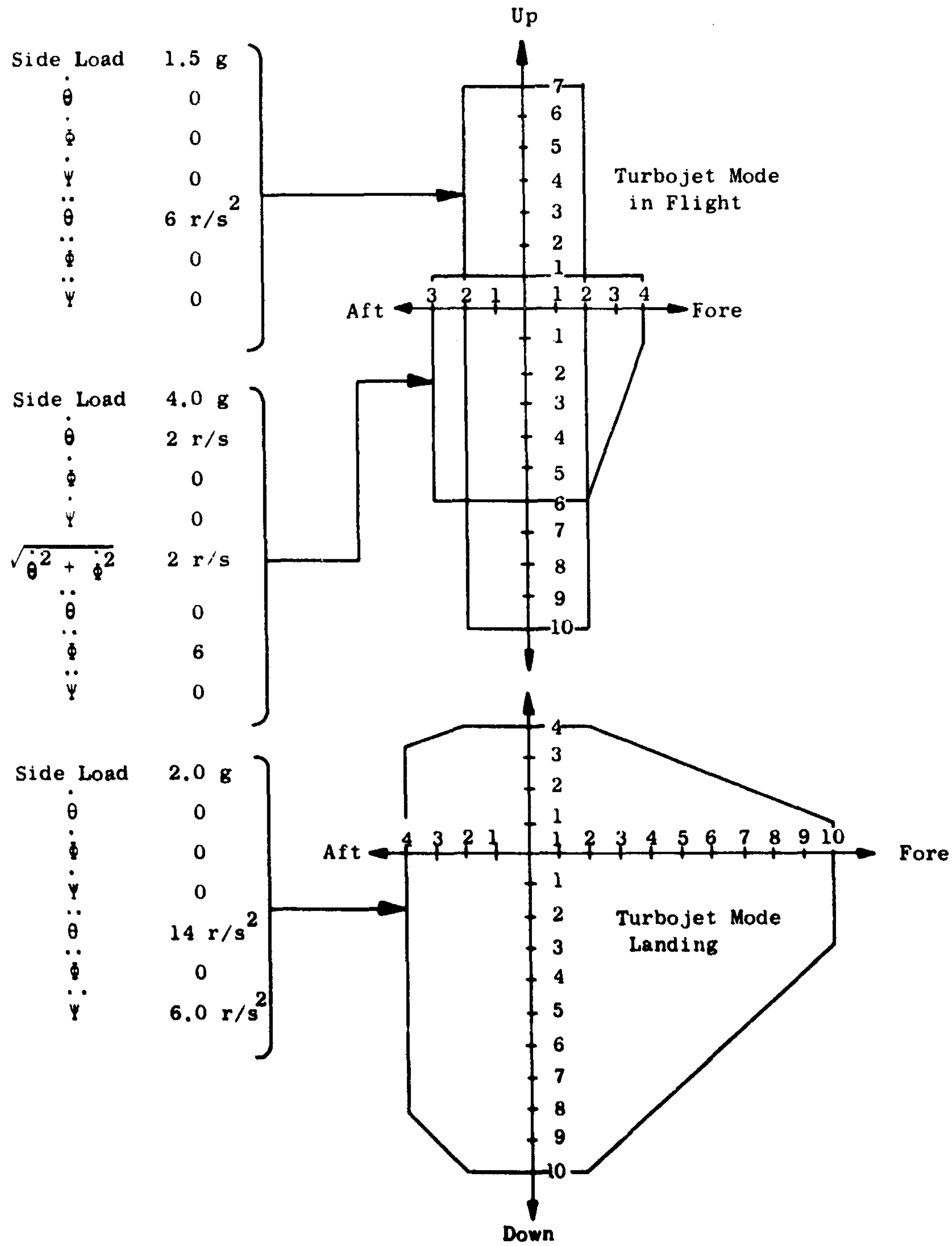


Figure 57. (U) Maneuver Loads - Turbojet Mode.

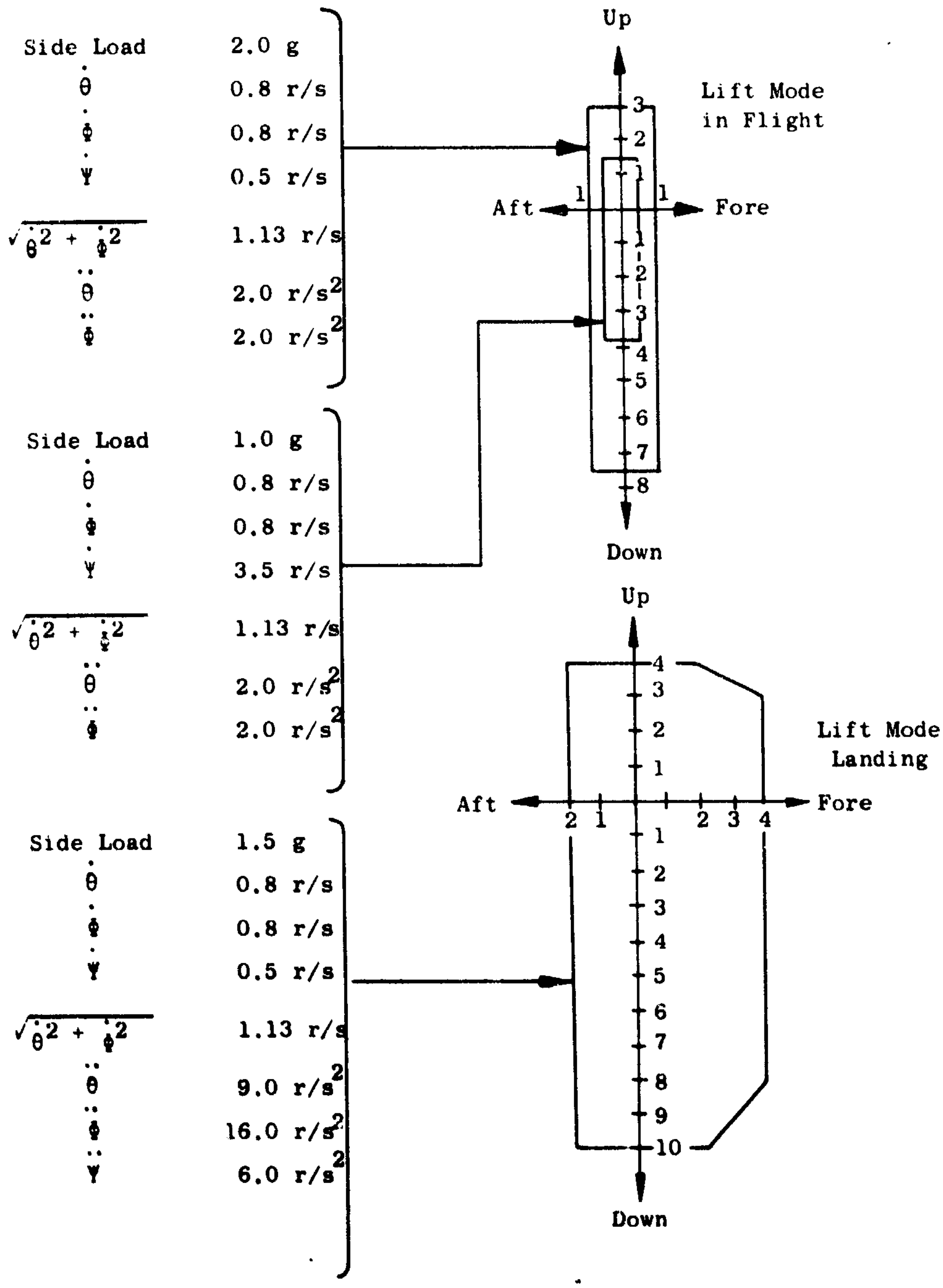


Figure 58. (U) Maneuver Loads - Lift Mode.

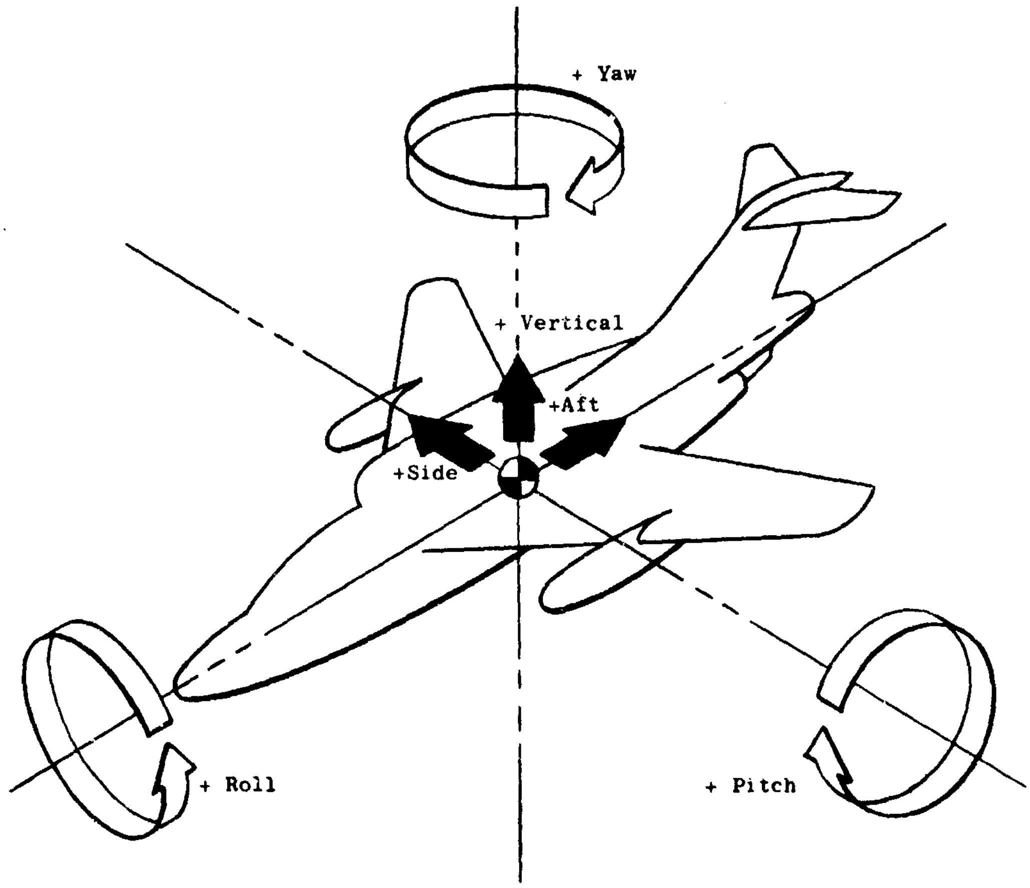
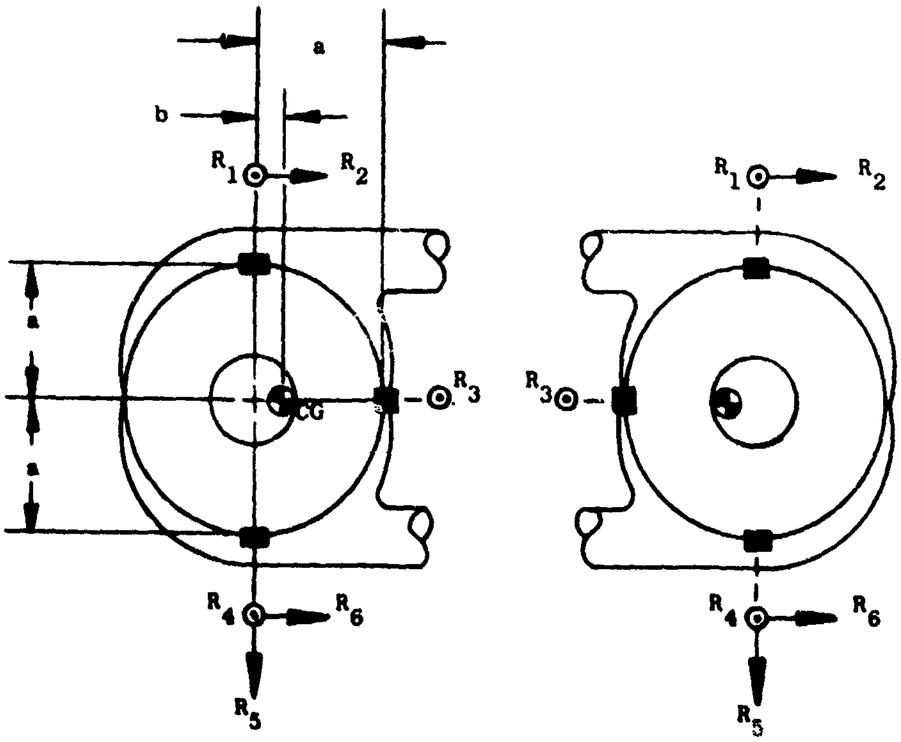


Figure 59. (U) Sign Convention for Aircraft Motion and Propulsion System Mount Loads.

TABLE V (U) WING LIFT FAN MOUNT LOADS						
Applied Force and Moment	Wing Fan Mount Loads					
	R_1	R_2	R_3	R_4	R_5	R_6
Aft force F_A	-	$\pm bF_A/2a$	-	-	$-F_A$	$\mp bF_A/2a$
Side force F_S	-	$-F_S/2$	-	-	-	$-F_S/2$
Vertical force F_V	$\frac{b-a}{2a} F_V$	-	$-bF_V/a$	$\frac{b-a}{2a} F_V$	-	-
Pitch moment M_P	$-M_P/2a$	-	-	$M_P/2a$	-	-
Roll moment M_R	$\pm M_R/2a$	-	$\mp M_R/a$	$\pm M_R/2a$	-	-
Yaw moment M_Y	-	$-M_Y/2a$	-	-	-	$M_Y/2a$



Left-Hand Wing Lift Fan **Right-Hand Wing Lift Fan**

TABLE VI (U)
FUSELAGE LIFT FAN MOUNT LOADS

Applied Force and Moment		Fuselage Fan Mount Loads					
		R_1	R_2	R_3	R_4	R_5	R_6
Aft force	F_A	-	-	$-F_A$	-	-	-
Side force	F_S	-	$-F_S$	$\frac{a-b}{a} F_S$	-	$\frac{b-a}{a} F_S$	-
Vertical force	F_V	$-bF_V/a$	-	-	$\frac{b-a}{2a} F_V$	-	$\frac{b-a}{2a} F_V$
Pitch moment	M_P	$\pm M_P/a$	-	-	$\mp M_P/2a$	-	$\mp M_P/2a$
Roll moment	M_R	-	-	-	$M_R/2a$	-	$-M_R/2a$
Yaw moment	M_Y	-	-	$\pm M_Y/a$	-	$\mp M_Y/a$	-

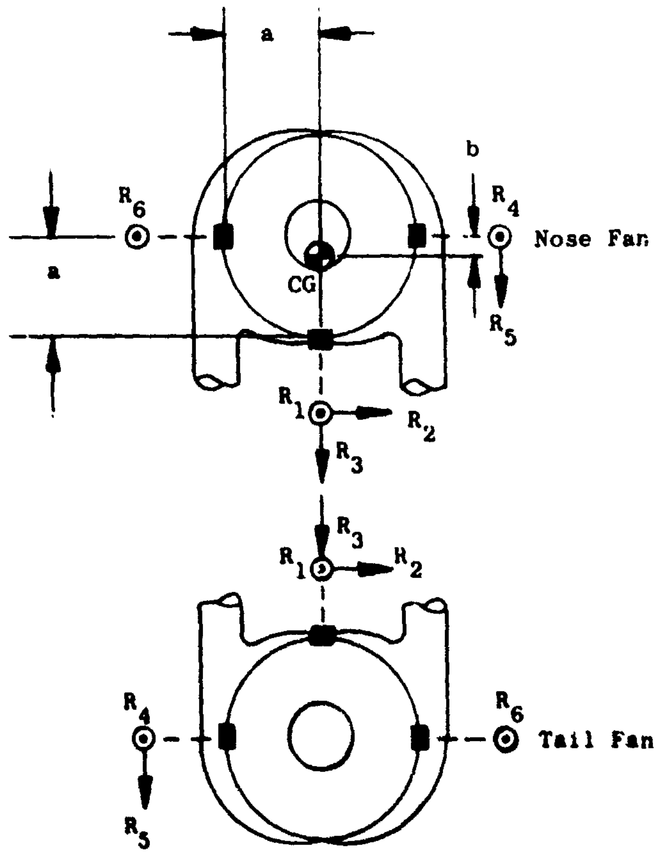
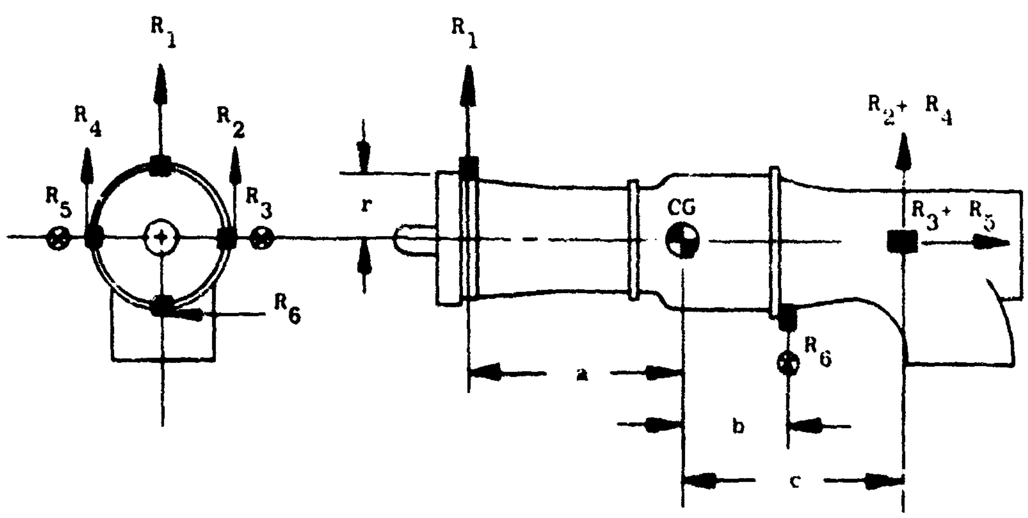


TABLE VII (U) ENGINE MOUNT LOADS						
Applied Force and Moment	Engine Mount Loads					
	R_1	R_2	R_3	R_4	R_5	R_6
Aft force F_A	-	-	$-F_A/2$	-	$-F_A/2$	-
Side force F_S	-	$-F_S/2$	$bF_S/2r$	$F_S/2$	$-bF_S/2r$	$-F_S$
Vertical force F_V	$\frac{-cF_V}{a+c}$	$-\frac{1}{2}\frac{aF_V}{a+c}$	-	$-\frac{1}{2}\frac{aF_V}{a+c}$	-	-
Pitch moment M_P	$\frac{-M_P}{a+c}$	$\frac{1}{2}\frac{M_P}{a+c}$	-	$\frac{1}{2}\frac{M_P}{a+c}$	-	-
Roll moment M_R	-	$\frac{-M_R}{2r}$	-	$M_R/2r$	-	-
Yaw moment M_Y	-	-	$M_Y/2r$	-	$-M_Y/2r$	-



Front View Side View

TABLE VIII (U) GYROSCOPIC MOMENTS				
	Induced Gyroscopic Moments			
Aircraft Angular Velocity	Engines	Wing Lift Fans Left	Wing Lift Fans Right	Fuselage Lift Fans
+ pitch	+ yaw	- roll	+ roll	- roll
+ roll	-	+ pitch	- pitch	+ pitch
+ yaw	- pitch	-	-	-

TABLE IX (U) SPEEDS AND POLAR INERTIAS OF THE PROPULSION SYSTEM COMPONENTS		
Propulsion Component	Polar Inertia in-lb-sec^2	Angular Velocity at Takeoff rev/min
Wing lift fan	177.4	3856
Fuselage lift fan	22.6	6550
Core engine	20.59	14,130

(U) PRELIMINARY DESIGN

CYCLE SELECTION

The mission analysis identified the required fan performance. Previous parametric studies (reference 1) identified the approximate fan sizes and aerodynamic design required to meet this performance.

One of the LFX objectives was to identify the minimum fan size which would meet the performance requirements. Studies were made to determine a design point having an optimum ratio of turbine gas energy to torque input. These studies used parametric compressor designs, with compressor tip speeds between 800 and 1000 feet per second. Variations of turbine discharge Mach number and compressor tip speed are shown in Figure 60. From these data, a design point was selected which had a compressor tip speed of 946 feet per second and a turbine discharge Mach number of 0.50.

TURBINE AEROTHERMODYNAMIC DESIGN

The LFX turbine is a full admission turbine mounted at the tip of the compressor blade. A flow path with a slope of 26 degrees at the bucket platform was selected to match the slope of the compressor rotor tip and to minimize diameter. The nominal arc of admission is 280 degrees. During power transfer, the arc of admission will vary between 360 degrees and 200 degrees.

Below are listed the design parameters for the turbine at the design point conditions. The turbine overall efficiency of 0.841 does not meet the objective efficiency of 0.86. Techniques available for improving the estimated efficiency include increasing the design wheel speed, decreasing the leakage, and increasing the nominal arc of admission.

Inlet total temperature, degrees Rankine	1847
Inlet total pressure, pounds per square inch, absolute	48.39
Inlet gas flow, pounds per second	49.4
Inlet flow function	43.8
Total to static pressure ratio	3.25
Total to total pressure ratio	2.76
Exhaust static pressure, pounds per square inch, absolute	14.88
Design power, horsepower	6653
Design energy, Btu/lb/sec	97.43
Exit axial Mach number	0.50
Overall efficiency	0.841
Pitch wheel speed, feet per second	1016
Velocity ratio	0.400
Work function	1.18
Admission arc, degrees nominal	280
Admission arc, degrees maximum	360
Admission arc, degrees minimum	200

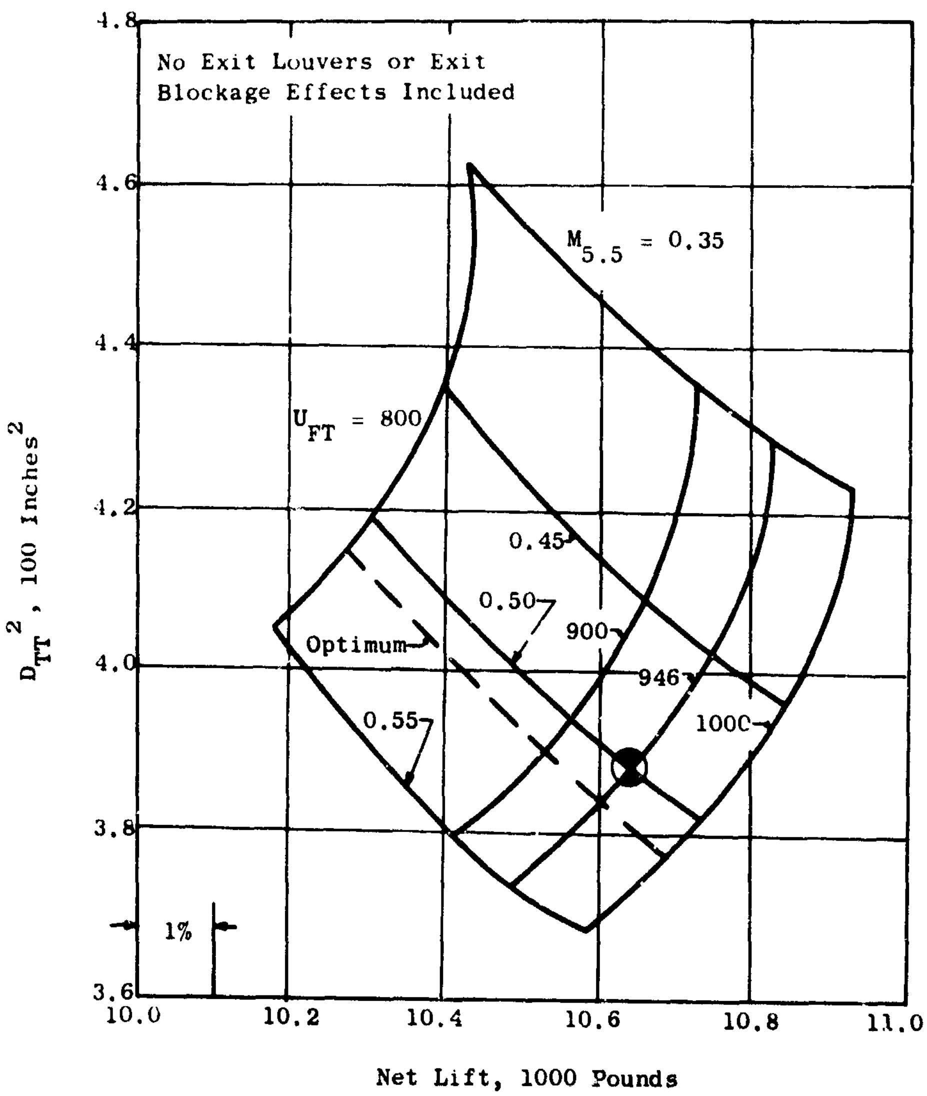


Figure 60. Turbine Speed and Exit Mach Number Variations.

The division of energy available to the turbine is tabulated below. The exhaust energy that is shown is available at the turbine bucket exhaust and does not include a nozzle thrust coefficient.

Shaft power	0.7335
Exhaust energy	0.1256
Nozzle loss	0.0350
Bucket loss	0.0854
Leakage	0.0116
Partial admission loss	0.0083
Swirl loss	0.0006
	1.0000

The overall loss in the turbine scroll and in the ducts to the scroll was calculated using an average value of flow Mach number of 0.25. This Mach number is lower than that used in the XV-5A (0.3), and is commensurate with an 8 percent head loss.

The turbine vector diagram for the design point is shown in Figure 61.

COMPRESSOR AEROTHERMODYNAMIC DESIGN

The LFX fan design represents an advanced version of the XV-5A fan (X353-5). The stage pressure ratio and disc loading were increased by increasing the wheel speed, so no loss in real fan efficiency should result. A comparison of the X353-5 and the LFX design performance is given below:

	X353-5	LFX
Stage nominal pressure ratio	1.115	1.253
Stage efficiency (no leakage)	0.88	0.88
Corrected tip speed, feet per second	720	946
Corrected flow, pounds per second	529	492
Tip diameter, inches	62.5	56.2
Hub-tip radius ratio	0.40	0.477
Aspect ratio (approximate)	5.85	5.8

In general, the risk involved in the advanced LFX is no greater than that in the X353, even though the rotor tip relative Mach number is almost sonic for this higher tip speed. There should be no reduction in stall margin. However, stall may be characterized by a more distinct stall manifestation at this higher Mach number.

A schematic representation of the fan stage is shown in Figure 62. As shown, the annulus convergence is small, requiring an average stator exit Mach number of about 0.505, compared to the average exhaust value of approximately 0.59 when the fan flow is fully expanded to ambient conditions. The primary advantages derived from loading the stator in this manner lie in the reduced Mach number at the exhaust louver leading edges,

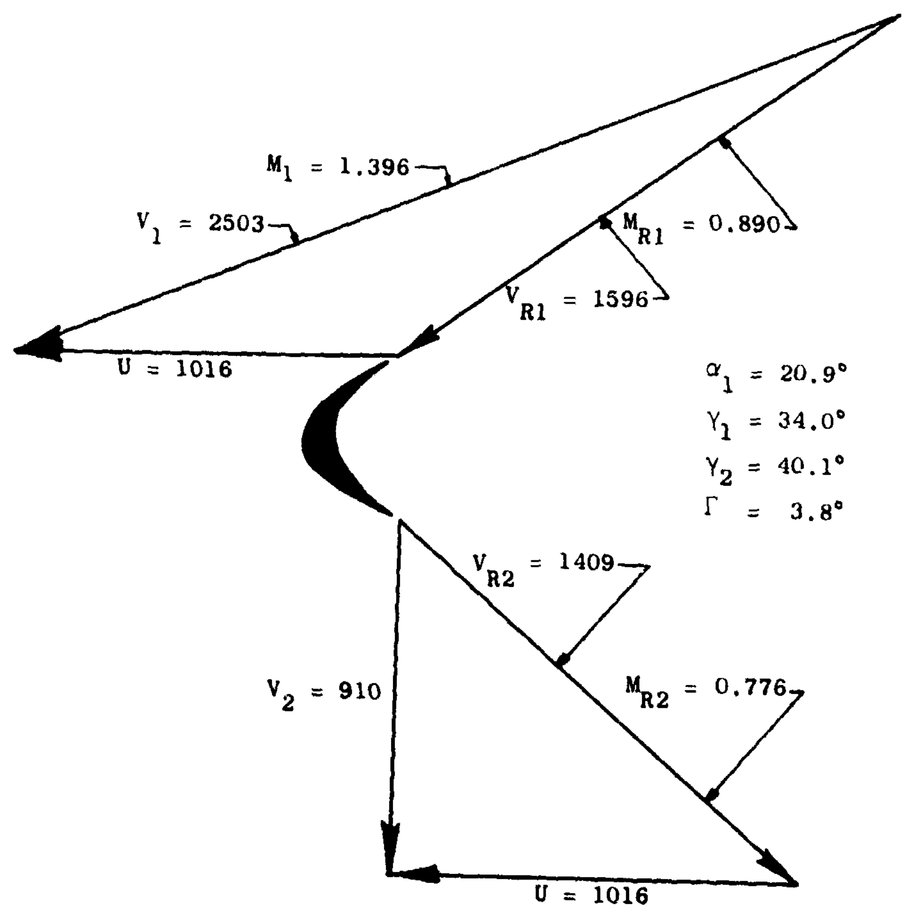


Figure 61. LFX Turbine Vector Diagram.

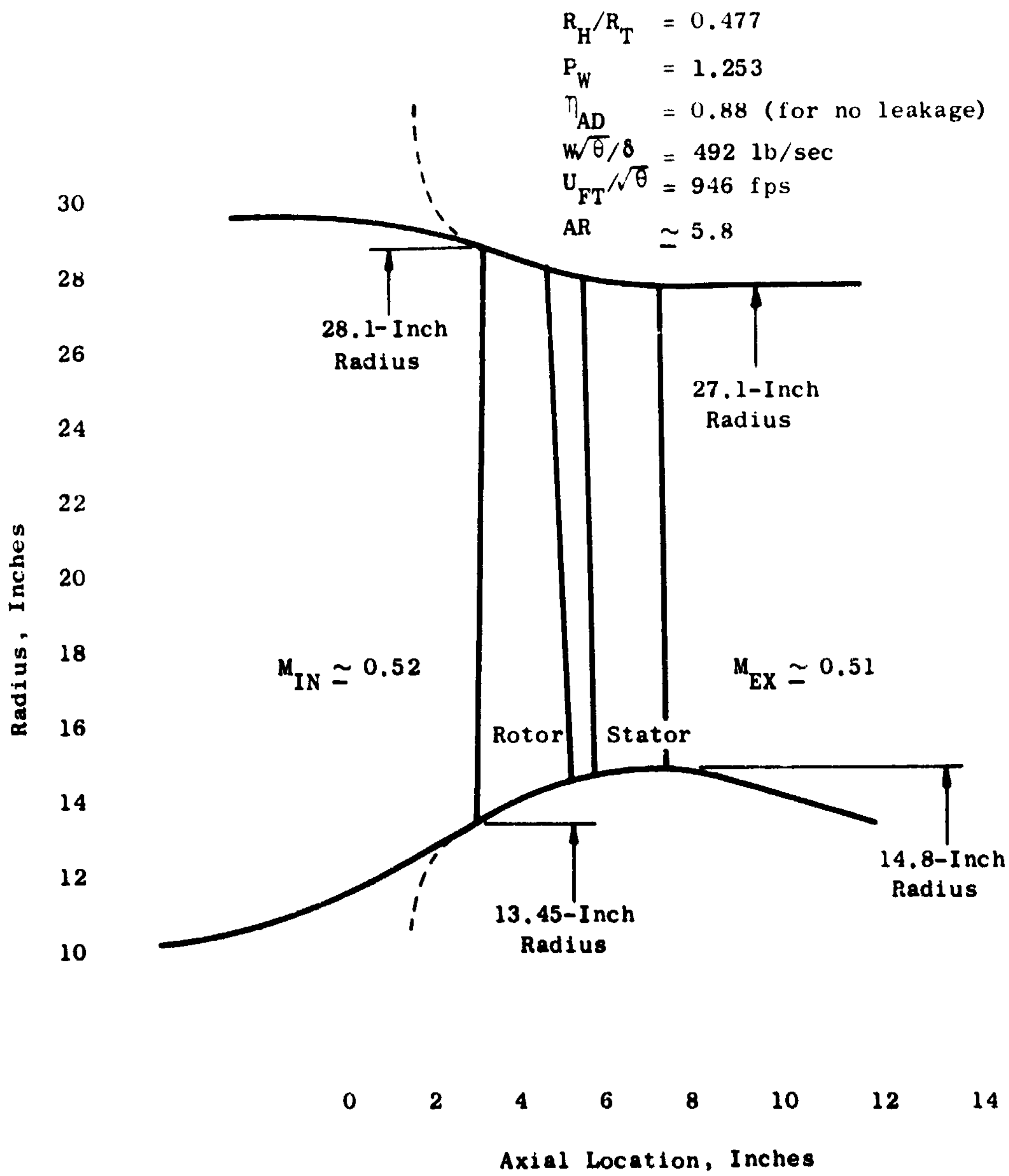


Figure 62. Preliminary Aerodynamic Design - Fan Compressor Flow Path.

and in the ability to maintain higher-than-ambient static pressure at the stator hub trailing edge. These factors, in turn, permit reduced exit louver losses and increase base pressure at the fan exit center body.

The compressor stage characteristics are shown in Figure 63. The rotor and stator loadings, as represented by the so-called D-factor and the dimensionless static pressure rise coefficient $\Delta P/q$, are moderate and are well within the region where very high efficiency has been realized. The design values of $\bar{\omega}$, the total pressure loss coefficient, are dependent upon the values of the rotor and stator loadings and on the values of the rotor and stator Mach numbers. The rotor loss coefficients are probably slightly low, particularly at the tip. However, the stator values are believed to be slightly high throughout, for the current loading and Mach numbers. It is expected that these losses can be rearranged to define more exact variations without lowering the overall efficiency.

As indicated in the LFX X353-5 comparison table, the current design efficiency (based on the loss assumptions of Figure 63) does not include the effects of hot gas leakage from the tip turbine. These effects, based on past experience, could lead to a percent-and-a-half drop in fan efficiency and a corresponding reduction in fan pressure ratio. However, it is believed that this loss could be reduced by directing the seal leakage aft and by leakage area allowance.

The compressor stage performance data and the predicted map of fan performance are shown in Figures 64 and 65, respectively.

The fan inlet and exit configurations are both important in fixing the overall lift system performance. The inlet distortion and losses influence the fan performance directly, while the required fan stall margin is directly related to the effective louver blockage and louver losses at the maximum deflection louver orientation. Assuming that no new exotic louver system (such as the chevron louvers, variable camber louvers, or translating louvers) is introduced, it is recommended that in order to avoid stall, the fan inlet and exhaust system be ultimately defined by experimental evolution of the components. This may necessitate locating the zero-deflection operating point at a compromise position on the fan map in order to allow sufficient stall margin for louver closure. It is felt that only nominal losses in system lift would result from this procedure.

FRONT FRAME MECHANICAL DESIGN

The front frame mounts the fan rotor, the scroll and the wing fan closure doors. It maintains internal static to rotating concentricity, and forms the compressor inlet flow path. It also transmits the lift, thrust, and maneuver loads from the fan to the aircraft.

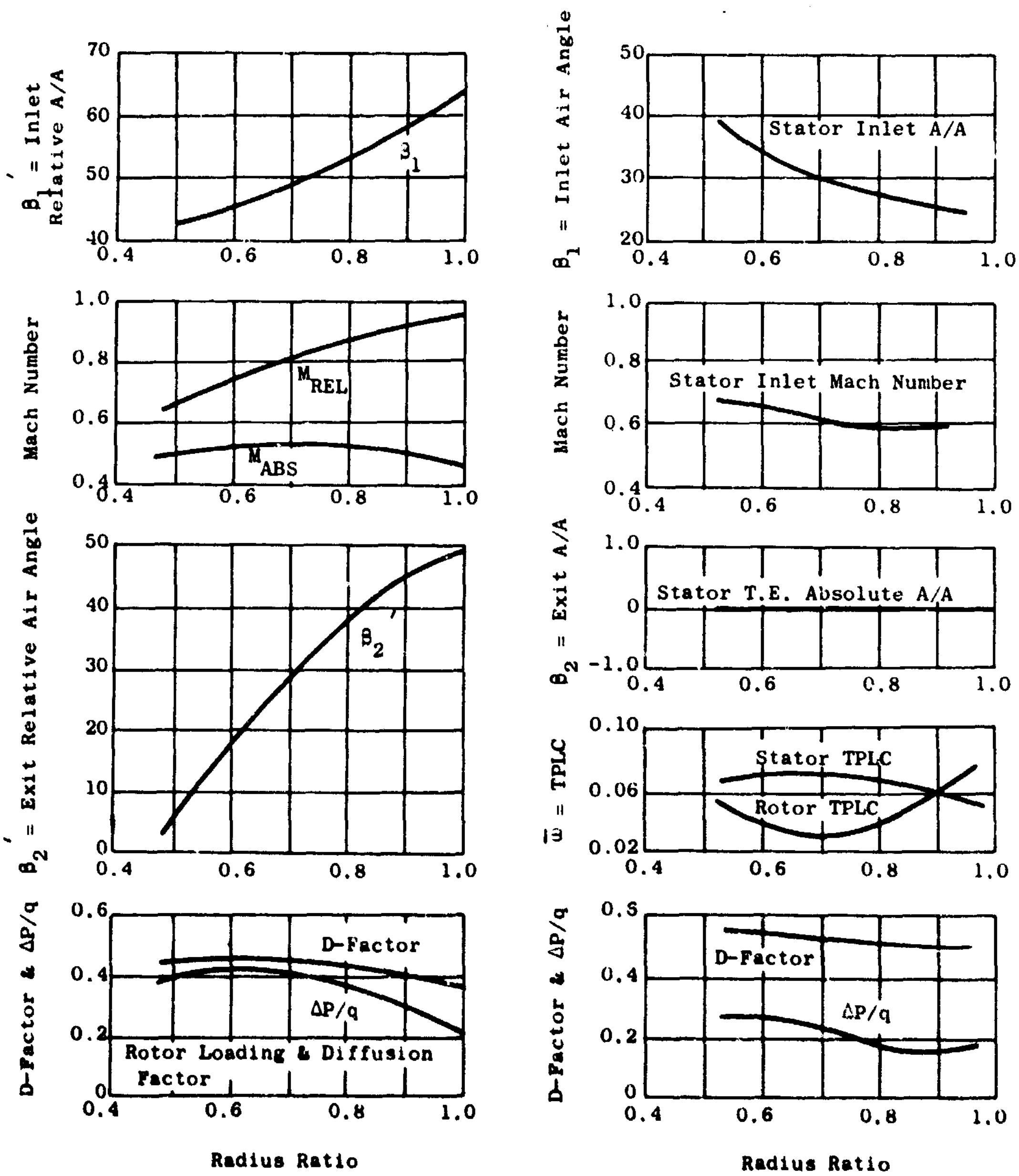


Figure 63. Compressor Stage Characteristics.

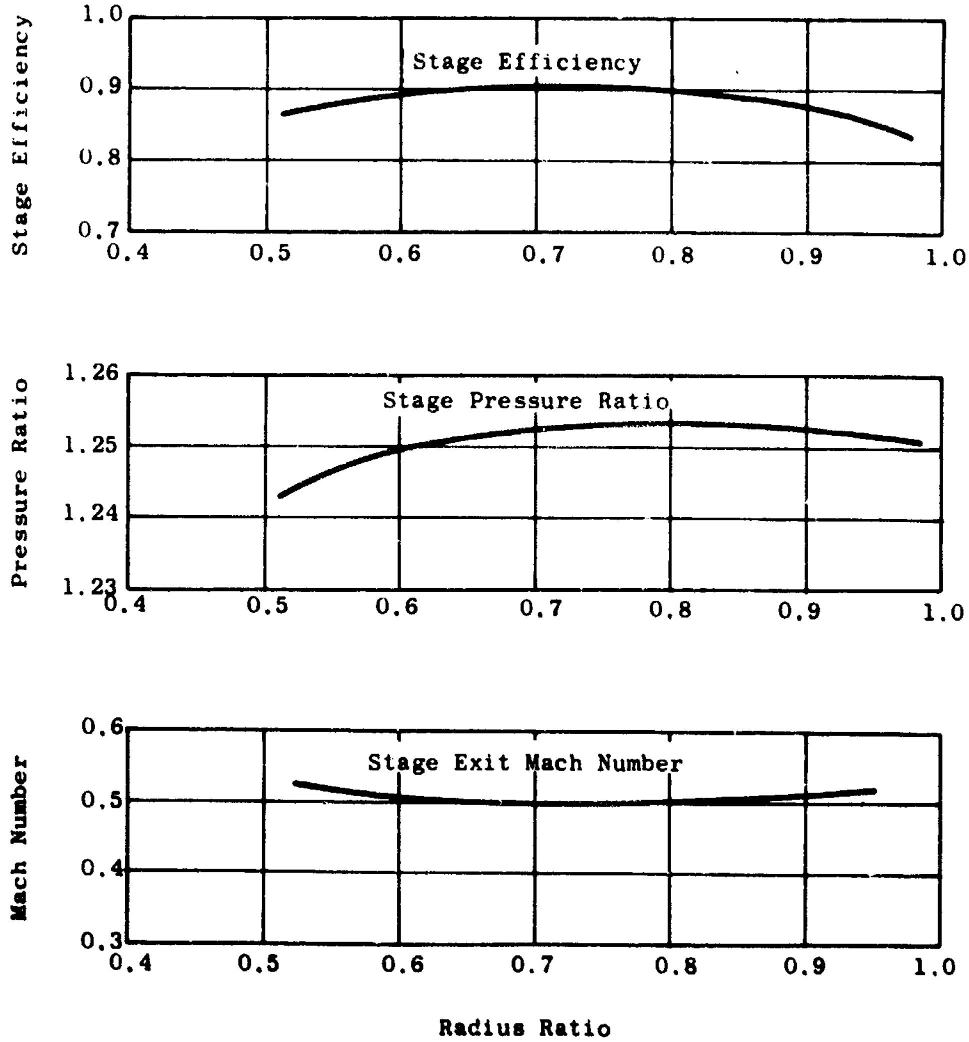


Figure 64. Compressor Stage Performance.

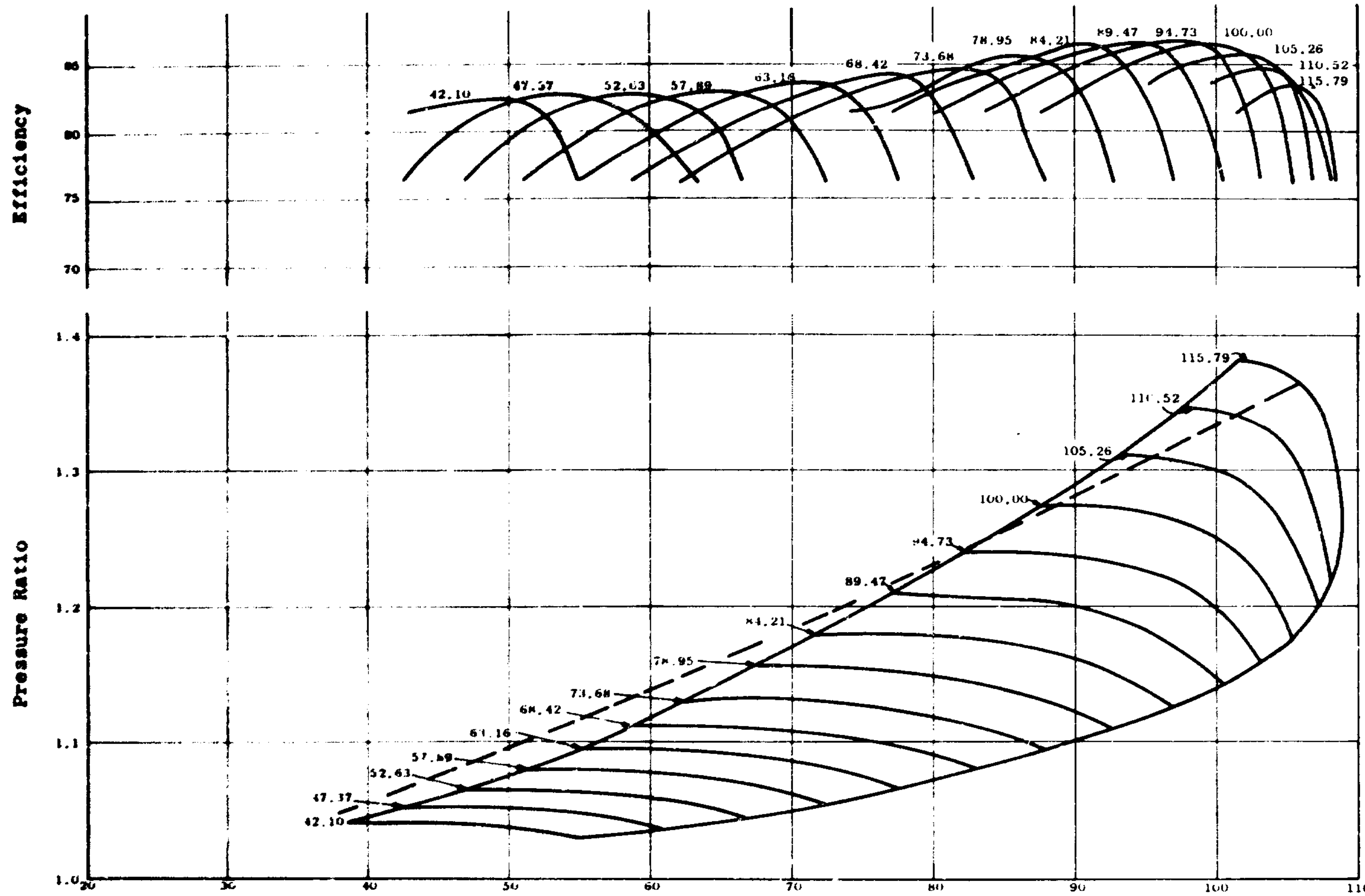


Figure 65. Predicted Compressor Map.

Studies were made of several frame configurations prior to selection of the final LFX design. The selected design features a single fore-and-aft main strut and two secondary struts for gyroscopic loading. The fabricated bolted design enables the selective use of specific materials and improves maintainability. This bolted frame design has about the same weight as an integrally-constructed frame, but has the additional benefit of lower cost.

The frame i supported in three positions in the wing, as shown in Figure 66. The 6 o'clock and 12 o'clock mounts carry the main lift and thrust, while the 3 o'clock mount carries gyroscopic and cross-flow-induced loads. The 9 o'clock mount is shown unattached to the airframe. The 9 o'clock minor strut maintains bellmouth concentricity and rotor-seal clearance. Additional and more detailed load analyses may show a requirement to pin the 9 o'clock mount in order to limit vertical deflection. Front frame loads are listed below. The maximum condition of loading is shown. No maneuver loads are included.

Scroll and rear frame load, pounds	1108
In-plane piston load, pounds	2270
Vertical rotor lift, pounds	6832
Bellmouth load, pounds per inch	19.8
In-plane rotor load	(Maneuver load)
Bulletnose lift, pounds	1050
Moments	
Gyroscopic, inch-pounds	93,500
Cross-flow, inch-pounds	20,000
Induced	(Maneuver load)
Doors, inch-pounds	36,000

The fore-and-aft main strut protrudes above the wing upper surface. The secondary struts are buried in the wing when the fan doors are closed. The struts are sized to deflection criteria, with low stresses as a result. The major strut will limit hub axial deflections to 0.120 inch during a maximum combined lift and maneuver load condition.

The hub is sized to limit rotor tip deflection to 0.060 inch during a maximum maneuver load condition.

The bellmouth and bulletnose are bolted onto the struts. There are no circular turning vanes and no cross-flow vanes.

The materials were selected based on availability, compatibility, and cost. To achieve a low-cost design, cast aluminum was chosen for the major strut. 17-4 PH was used for the hub and for the 3 o'clock strut.

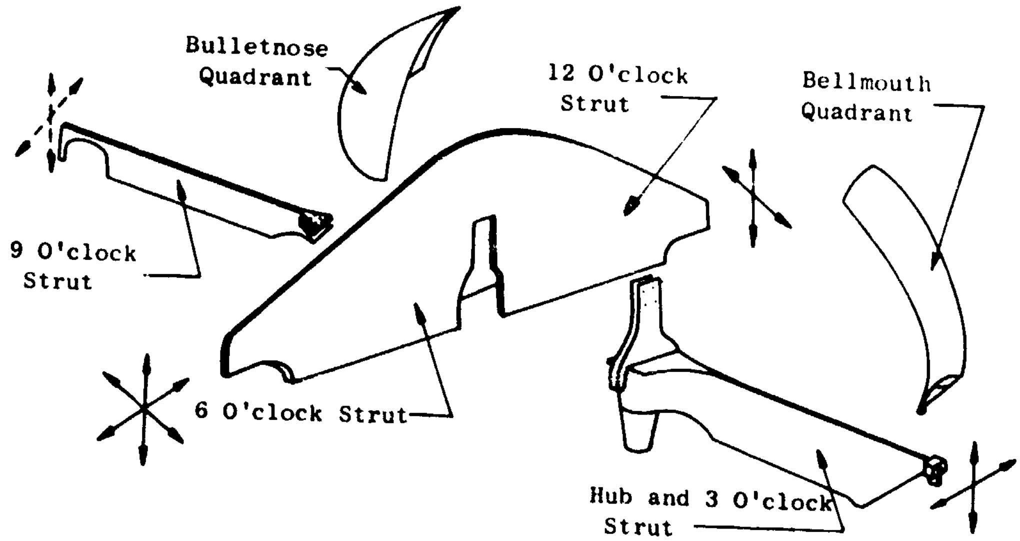


Figure 66. Front Frame.

Preliminary stress analysis of the 9 o'clock strut showed that aluminum was not acceptable; therefore, this strut will probably require use of 17-4 PH. The bellmouth was made from titanium, and the bulletnose can be made either from plastic or from a magnesium alloy.

Table X shows the front frame weights. The preliminary weights are those used in the LFX design review. The final weights reflect the changes made in the final design. This final weight of 127 pounds compares to an objective weight of 116 pounds.

TABLE X
FRONT FRAME WEIGHT

Front Frame Component	Preliminary Weight (Pounds)	Final Weight (Pounds)
Major strut	52	58
3 o'clock minor strut and hub	40	32
9 o'clock minor strut	10	10
Bulletnose	5	5
Bellmouth	22	17
Miscellaneous hardware	5	5
TOTAL	134	127

DOOR ACTUATION SYSTEM DESIGN

An airframer's proposed hydraulically-actuated system is objectionable because it requires cutting a large hole in the hub. A different design, shown in Figure 67, has been identified and appears to be feasible. The objects of the design are to minimize holes in the hub and to transmit loads to the minor strut without influencing the support members of the actuation system.

The actuation system will be the responsibility of the airframe manufacturer. Provisions for attaching the system to the major strut will be made. These will include holes for hinges and shaft passage, and hard points for bolting required members.

This system is similar to the airframer's design, except for the elimination of the gearbox and the addition of fore-and-aft bearing supports. This allows all loads which are transmitted from the hub to the minor struts to pass directly to the struts without going through a gear housing. Four hard points on each side of the major strut are provided for attachment of the fore-and-aft bearing support.

There is only one small hole in the hub area, through which the outboard drive shaft passes. Provisions can also be made for a bearing mount in the hub, if required. The drive shaft from the aircraft is covered by a protective fairing in the airstream area, and follows the minor strut.

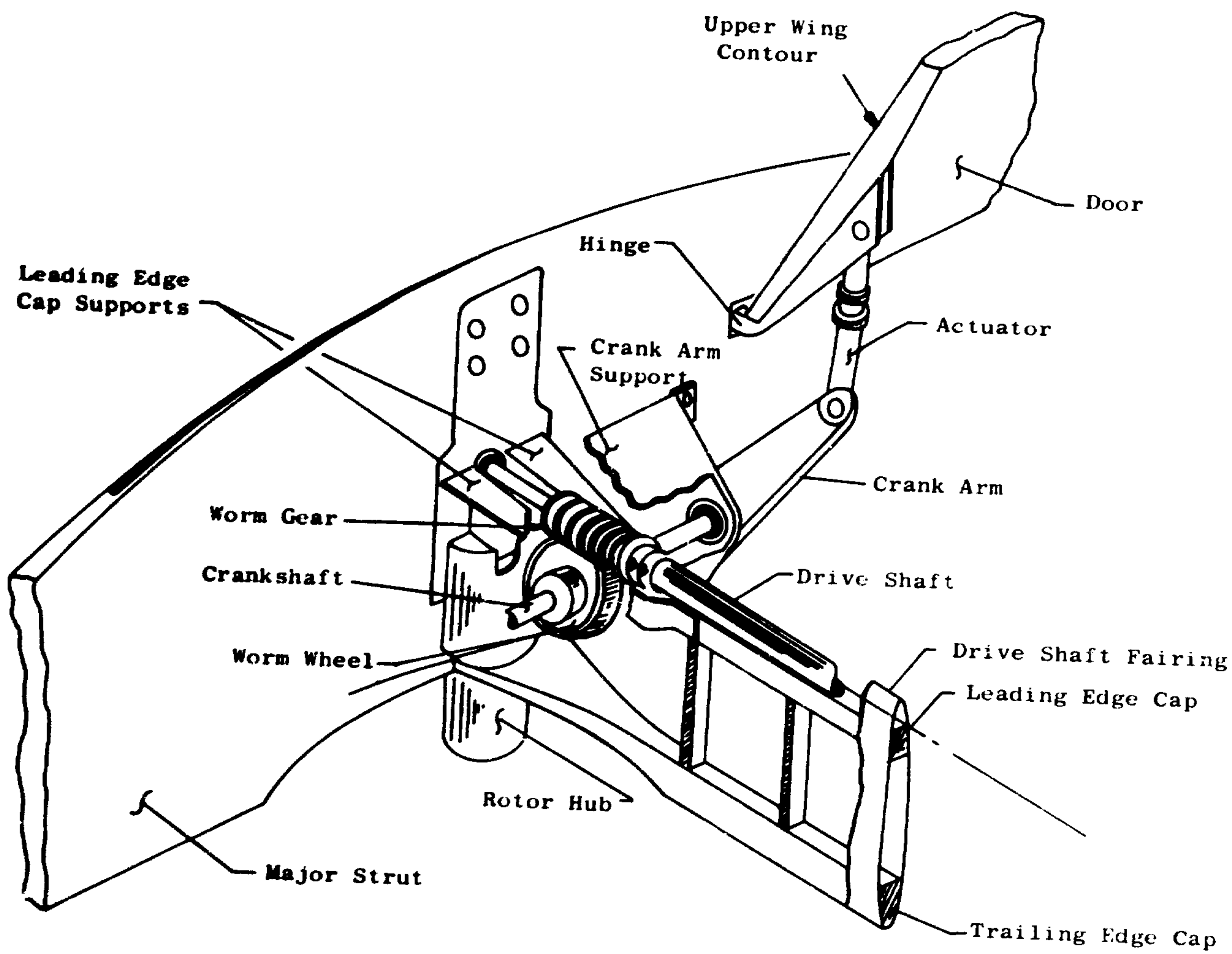


Figure 67. Door Actuation System.

The upper cap support of the minor strut must clear the crankshaft and attach directly to the hub. The leading edge of the minor strut can be almost any shape required to give the required section properties. A final drive shaft and gear selection must be made before any configuration can be selected.

The door hinge design is unchanged from the airframer's design, except for making the door thinner at the top of the hub to allow the bulletnose to pass through and still clear the door reinforcement. It is desirable that holes not be cut into the bulletnose for these reinforcements. The only holes required in the bulletnose are those for actuation arm and crank rod passage. The bulletnose can be made with a removable top for easy access to the actuating mechanism.

As shown in Figure 67, the gears are exposed to the atmosphere. This was done because the upper strut support snakes over the crankshaft. However, a thin sheet-metal covering may be placed between the bearing supports at top and bottom. These gears are used only for intermittent motion and for slow speeds, and are encased in the bulletnose. Therefore, sophisticated sealing is not required.

The installation of the doors and actuating mechanism requires that the rotor be lowered approximately 2 inches in the wing from its present layout position. This is an unresolved interface problem.

REAR FRAME MECHANICAL DESIGN

The rear frame provides the exit stators for the fan and turbine, which remove swirl from the discharge flow. The rear frame also supports the honeycomb seals for the fan and the turbine. The design of the rear frame also permits aspiration of the air in the wing cavity by providing a flow path through the turbine stators into the middle box, where it will be ejected into the fan stream. It is possible to design the rear frame to support the exit louvers. However, this design assumes that the exit louvers are attached to the airframe.

The rear frame is designed so that it can be taken apart (see Figure 68). It consists of an outer box which is a continuous ring structure required for four-point mounting. The stator assembly is constructed in four 90-degree segments which are attached to the outer box and the one-piece inner box. The fan and turbine stators pass completely through the inner box for increased stiffness.

The rear frame is mounted to the scroll by four A-frame type mounts. The mounts, which are located directly under the front frame struts, transmit loads in the axial and tangential directions. This type of mounting acts to keep the rear frame concentric with the rotor, but it makes it difficult to transmit the exit louver loads directly to the airframe.

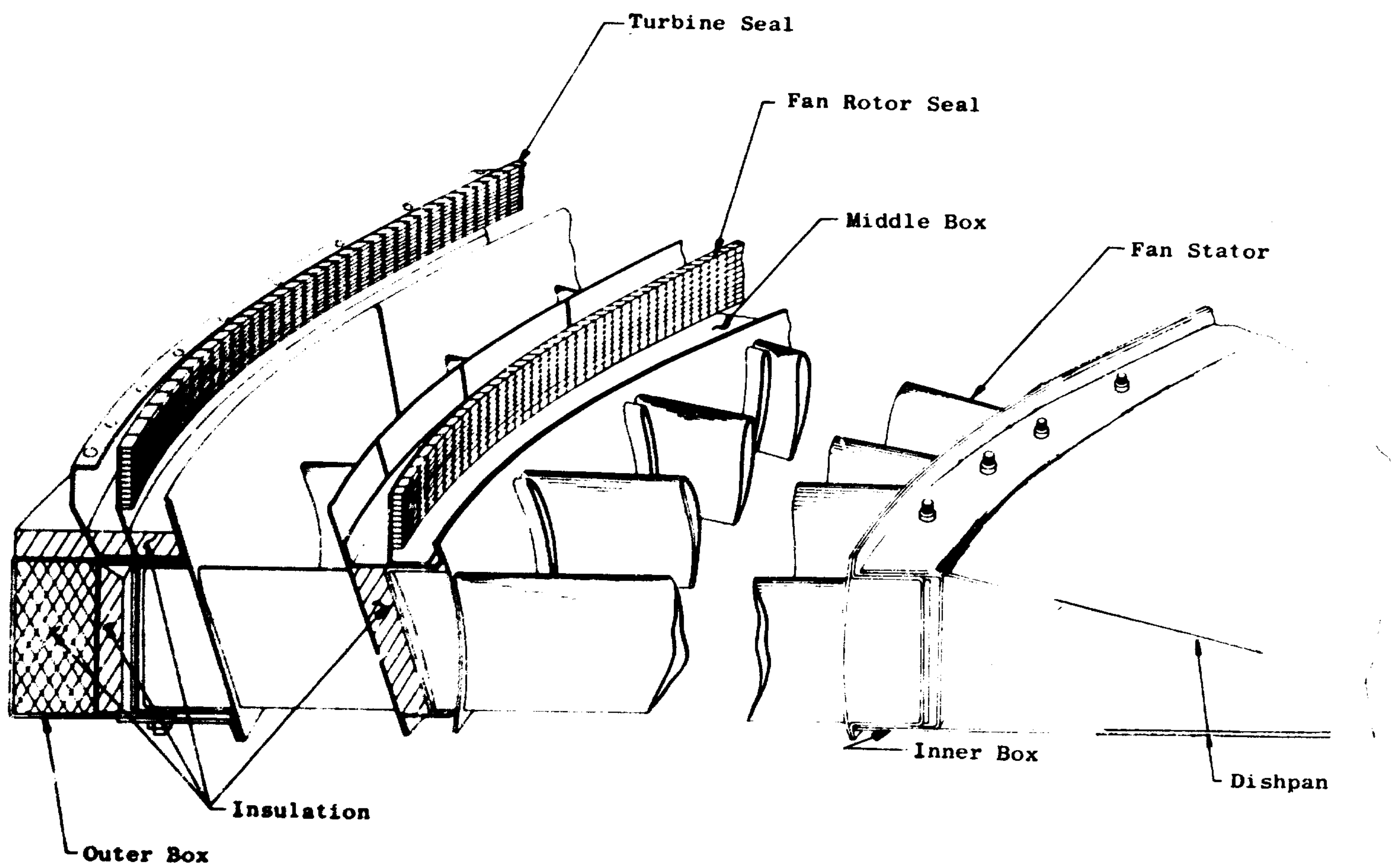


Figure 68. Rear Frame.

The fan seal located on the middle box is on a stable platform, and the pressure drop across it is low. It is not expected to be much of a problem. The turbine seal located on the outer box may be a problem, since it is mounted on the hot section of the frame. It cannot be mounted on the outer casing, because this is saw-slotted for thermal expansion circumferentially and has little stiffness in the radial direction. Therefore, the design shown is an attempt to tie the seal support to the outer box, which has torsional stiffness, and to insulate and cool the box structure to minimize thermal growth.

The loadings on the rear frame components are:

1. Fan stator torque - 131,500 inch-pounds, acting as a horizontally distributed load of 7.5 pounds per inch on the fan stators. This torque is resisted at the four scroll-to-front-frame mounts.
2. Fan stator axial load - 1508 pounds, acting upward as a distributed load of 3.12 pounds per inch on the fan stators.
3. Dishpan axial load - 685 pounds, acting upward.
4. Axial and circumferential loading on the turbine stators due to hot gas flow is assumed to be negligible.
5. Honeycomb seal pressure loading is negligible for the fan seal and is estimated at 1.5 pounds per square inch for the turbine seal.

The estimated temperatures of the rear frame components are:

Dishpan	100 degrees Fahrenheit
Inner box	100 degrees Fahrenheit
Fan stators	100 degrees Fahrenheit
Middle box	Ambient to 1050 degrees Fahrenheit
Turbine stators	1050 degrees Fahrenheit
Outer casing	1050 degrees Fahrenheit
Outer box	250 degrees Fahrenheit

The stresses calculated for the components of the rear frame were based on gas loads and thermal loads. Louver loads were directly taken out by the rear strut. The maximum stresses which were calculated occurred in the following sections:

1. Outer Box
   a. 11,400 pounds per square inch - bending stress due to stator and dishpan lift.
   b. 24,670 pounds per square inch - bending stress due to stator torque.

c. 45,000 pounds per square inch - bending stress due to turbine vane thermal expansion, assuming box temperature of 250 degrees Fahrenheit. If the outer box temperature is 400 degrees Fahrenheit, this disappears.

d. 18,000 pounds per square inch - torsional stress due to turbine stator vertical moment.

2. Turbine stators

a. 61,300 pounds per square inch - bending stress due to transmitting torque from fan stators to outer box.

b. 13,400 pounds per square inch - bending stress due to carrying the fan and dishpan thrust.

3. Middle Box

a. 39,400 pounds per square inch - torsional stress due to fan stator vertical moment.

b. 47,400 pounds per square inch - bending stress due to torque loading of fan stators.

c. 3460 pounds per square inch - shear stress due to torque loading of fan stators.

4. Fan Stators

a. 73,800 pounds per square inch - bending stress due to torque loading.

b. 8010 pounds per square inch - bending stress due to lift loading.

5. Inner Box

a. 12,250 pounds per square inch - torsional stress due to fan stator vertical moment.

b. 6250 pounds per square inch - bending stress due to fan stator torque.

c. 27,000 pounds per square inch - shear stress due to fan stator torque.

Titanium was chosen for the stators and the inner box. Inconel 718 was chosen for the turbine vanes, the middle box, the outer box and the casing. The dishpan and fan honeycomb seals use 321 stainless steel. The insulation can be a glass fiber material. Typical properties for the insulation are 8 pounds per cubic foot density and a heat transfer coefficient of 0.2 Btu/ft/°F/hr.

Table X shows the rear frame weights. These weight estimates are based on the material thicknesses shown in Figure 68. They do not include any

supports for the exit louvers. The preliminary weights are those used in the LFX design review. The final weights reflect the changes made in the final design. The final weight of 100 pounds compares to an objective weight of 87 pounds.

TABLE XI
REAR FRAME WEIGHT

Rear Frame Component	Preliminary Weight (Pounds)	Final Weight (Pounds)
Inner box	10.1	5.4
Fan stators (60)	37.5	19.9
Turbine vanes (60)	9.8	9.8
Middle box	15.9	15.9
Outer box and casing	52.0	36.0
Honeycomb seals	7.0	7.0
Mounts	6.0	6.0
TOTAL	138.3	100.0

EXIT LOUVER MECHANICAL DESIGN

The exit louvers consist of a set of parallel airfoils beneath the rear frame, and are used to turn the discharge flows of the fan and turbine to provide horizontal thrust during fan-supported flight. During conventional turbojet flight, the exit louvers close to form the local lower wing surface beneath the fan. The exit louvers on the XV-5A are differentially vectored to provide aircraft roll control forces. There is no requirement for differential vectoring for the LFX design, because the required aircraft roll control forces are supplied using gas power transfer.

The exit louver system that was selected is similar in form to the XV-5A design. An airfoil with a 7-inch chord was used for the louver. The exit louvers were mounted to the center strut. (Outboard support would require a ring of pivot blocks integrated into the lower wing surface structure.) A single-pin actuation system was selected, with the pin located approximately at the quarter-chord of the louver. This pin location results in aerodynamically balanced louvers, and significantly reduces the actuation loads. The louver load was assumed to be 19 pounds per inch at a 45-degree vector angle. Figure 69 illustrates the louver actuation system.

Titanium was chosen for the louver airfoil. The ends of the louvers which project into the turbine stream are filled with a steel core material to provide the required strength at the exhaust gas temperature of the turbine. The remainder is titanium honeycomb filled.

Table XII shows the louver weights. The preliminary weights are those used in the LFX design review. The final weights reflect the changes made in the final design. The final weight of 45 pounds compares to an objective weight of 40 pounds.

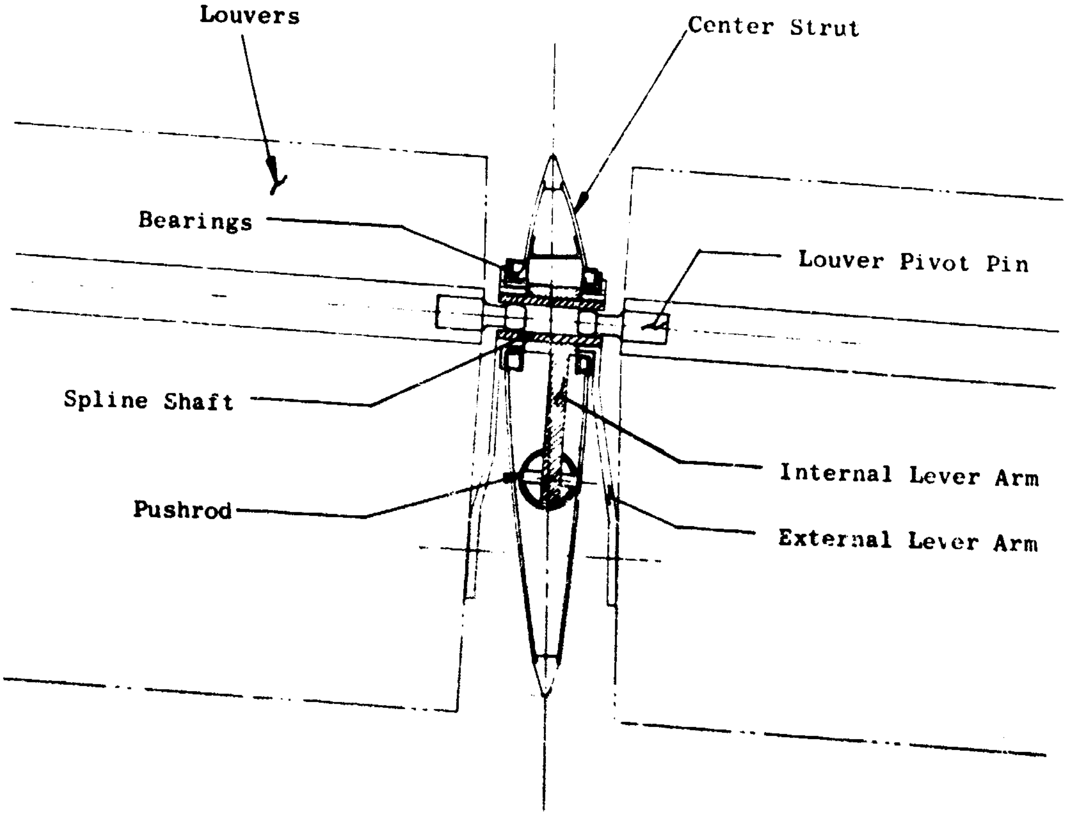


Figure 69. Exit Louver Actuation.

TABLE XII LOUVER WEIGHT		
Louver Component	Preliminary Weight (Pounds)	Final Weight (Pounds)
Skin	55	20
Core		10
End mounts and actuation rod	15	15
TOTAL	70	45

SCROLL MECHANICAL DESIGN

The scroll carries the engine exhaust gas from the duct to the turbine of the lift fan. The scroll nozzle diaphragm is used to vary the energy input to the turbine through variation of the turbine admission area. The four-point mounting system allows the scroll to help support the fan gyroscopic loads, holds the rear frame concentric with the rotor, and transfers rear frame and louver loads through the scroll to the main fan mount.

The scroll is a take-apart design, as shown in Figure 70. This is a definite advantage for manufacturing, accessibility and maintenance. The flow Mach number in the scroll has a design value of 0.25.

Two scroll configurations were investigated. The selected design has the variable area control system located in the inboard portion of the scroll. This variable area control system (VAC) actuates splitter vanes which are located between the nozzles in the nozzle diaphragm. These splitter vanes are actuated in successive steps, with a short control stroke, as shown in Figure 71.

The 360-degree scroll has a seal length that is virtually double the length of the one used on the XV-5A. This, combined with the higher pressure, makes leakage a significant problem area. A bellows-type scroll seal has been proposed for the outer scroll seal as a means of eliminating one of the leakage paths (see Figure 72).

The scroll is mounted to the front frame as shown in Figure 54. This configuration permits unrestricted thermal expansion and transfers all loads to the front frame. It also requires the scroll and rear frame to assist the 9 o'clock minor front frame strut during maneuver and gyroscopic loading.

The mounting of the rear frame to the scroll holds the rear frame concentric with the scroll. Torque exerted on the rear frame stators is transmitted through the rear frame to scroll mounts, where it is counteracted by the nozzle partition torque in the scroll.

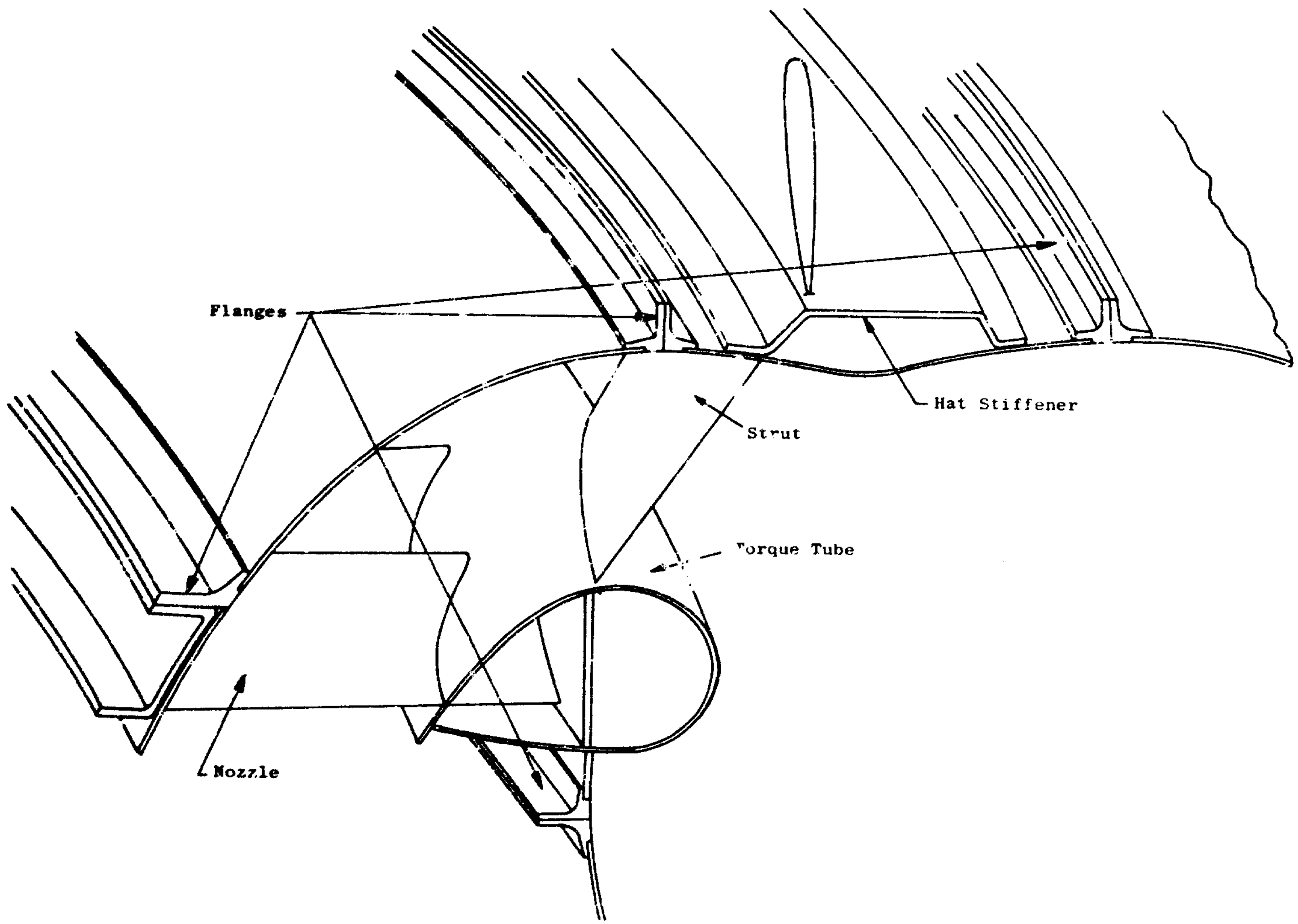


Figure 70. Scroll Take-Apart Design.

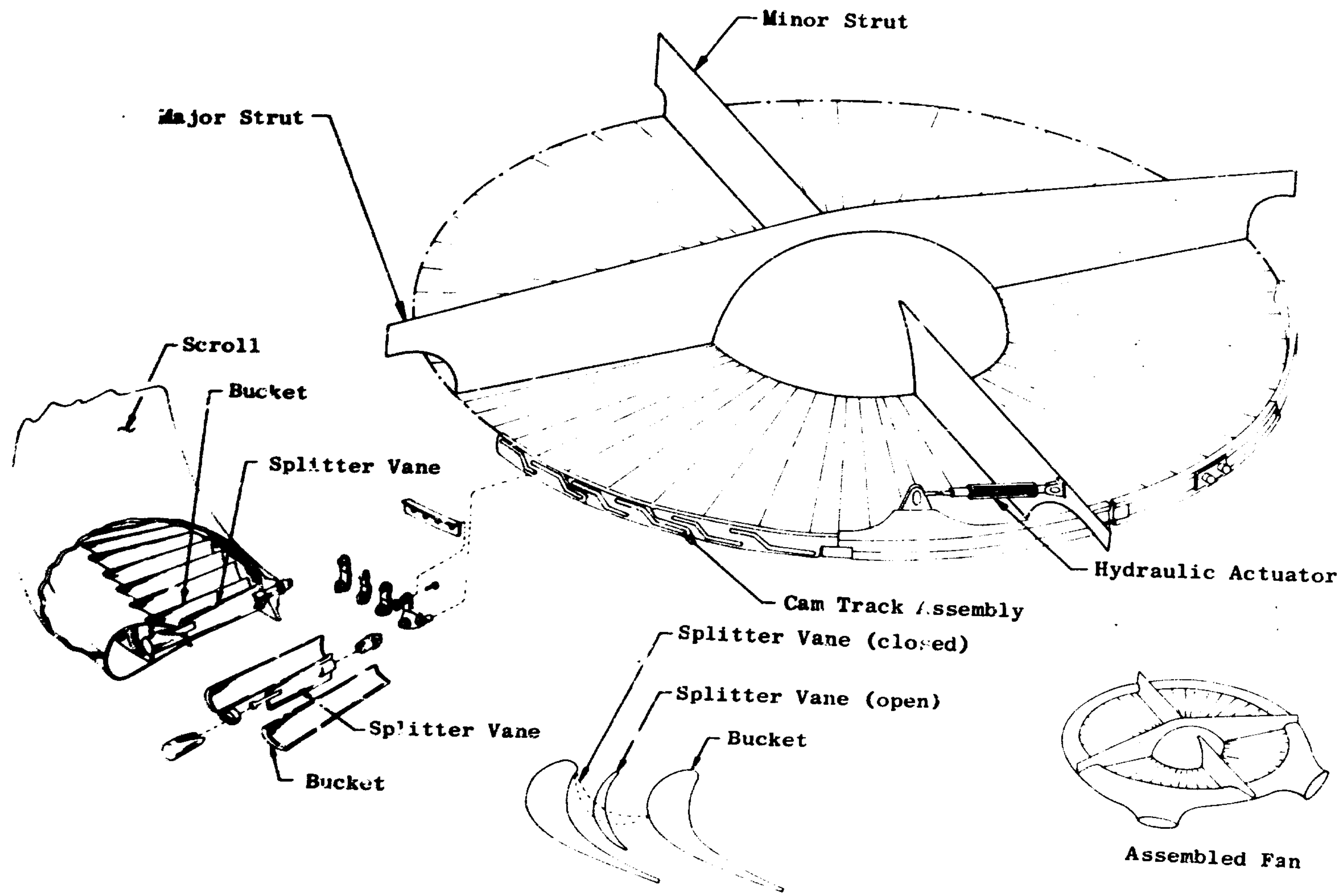


Figure 71. Variable Area Scroll Actuation.

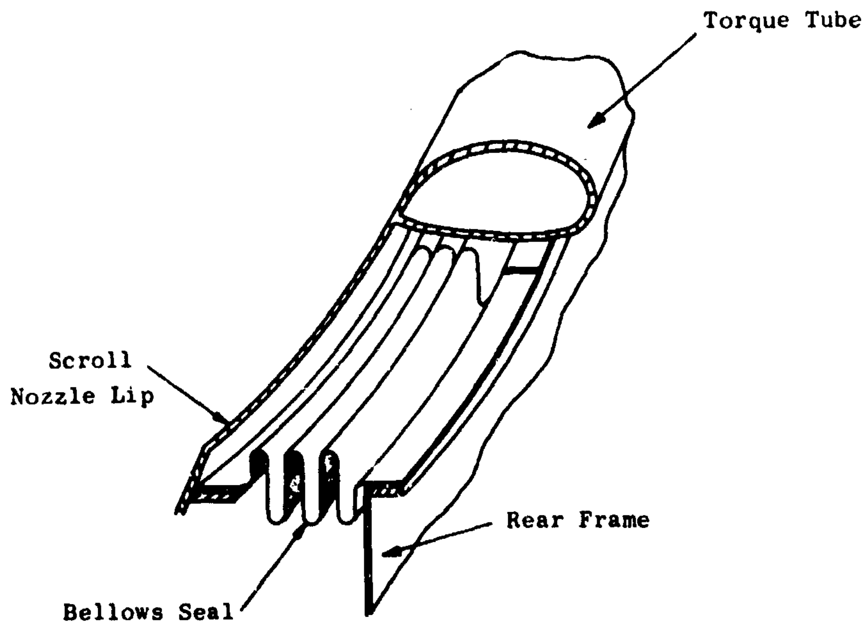


Figure 72. Proposed LFX Scroll Seal.

The scroll loading described below is summarized in Figure 73.

1. Nozzle torque - 135,000 inch-pounds, under full power transfer conditions. This load is applied as a distributed horizontal-circumferential force of 23.4 pounds per inch over 360 degrees of nozzles.

2. Scroll axial force (upward) on nozzles - 2232 pounds, distributed over 360 degrees of nozzles. This is shown in Figure 73 as a distributed load of 11.4 pounds per inch.

3. Rear frame torque - This is approximately equal to the scroll nozzle torque and is in the opposite direction. The load is applied to the scroll links from mount locations under the four struts.

4. Rear frame lift - This is estimated at 2200 pounds and will be transmitted to the scroll by links that will take vertical loads. These links are located at the four major struts.

5. Inlet piston force - 4540 pounds on each inlet. These loads are transmitted through the scroll in a direct line with the 6 o'clock and 12 o'clock strut mounts, where they are taken out.

6. Variable area actuation loads - A circumferential reversing load, not believed to be significant.

7. Maneuver loading - The scroll must carry not only its own weight but also that of the rear frame. It will also assist the 9 o'clock front frame strut in resisting gyroscopic loading.

8. Temperature and pressure loading - The temperature and pressure inside the scroll are assumed to be 1381 degrees Fahrenheit and 48.39 pounds per square inch absolute.

Although cross-sectional areas of the scroll are determined by flow requirements, circular arcs are used throughout in forming these areas, in order to eliminate bending stresses and to use the skins as efficient membrane stress members. The nozzle partitions (included primarily for aerodynamic reasons) and the struts act as mechanical ties at the intersection of these circular arcs, and provide the reactions necessary to maintain the membrane action of the skins. Circumferential sheet-metal hat sections and the inner "torque tube" (included primarily to provide flow characteristics of the nozzle outer diameter) act as double skin attachments for the struts and nozzles; they also act as composite beams which maintain membrane stresses between the nozzles and struts. The maximum stresses shown in Figure 74 are due to pressure only. They will be increased by the effect of rear frame, gyroscopic and piston force loading. They were calculated based on X353-5 section properties. A calculation using the section properties of the LFX nozzles resulted in stresses within the properties of the selected material. A savings in weight would result by making the nozzle partition section and the torque tube about 0.3 inch deeper than shown on Figure 54.

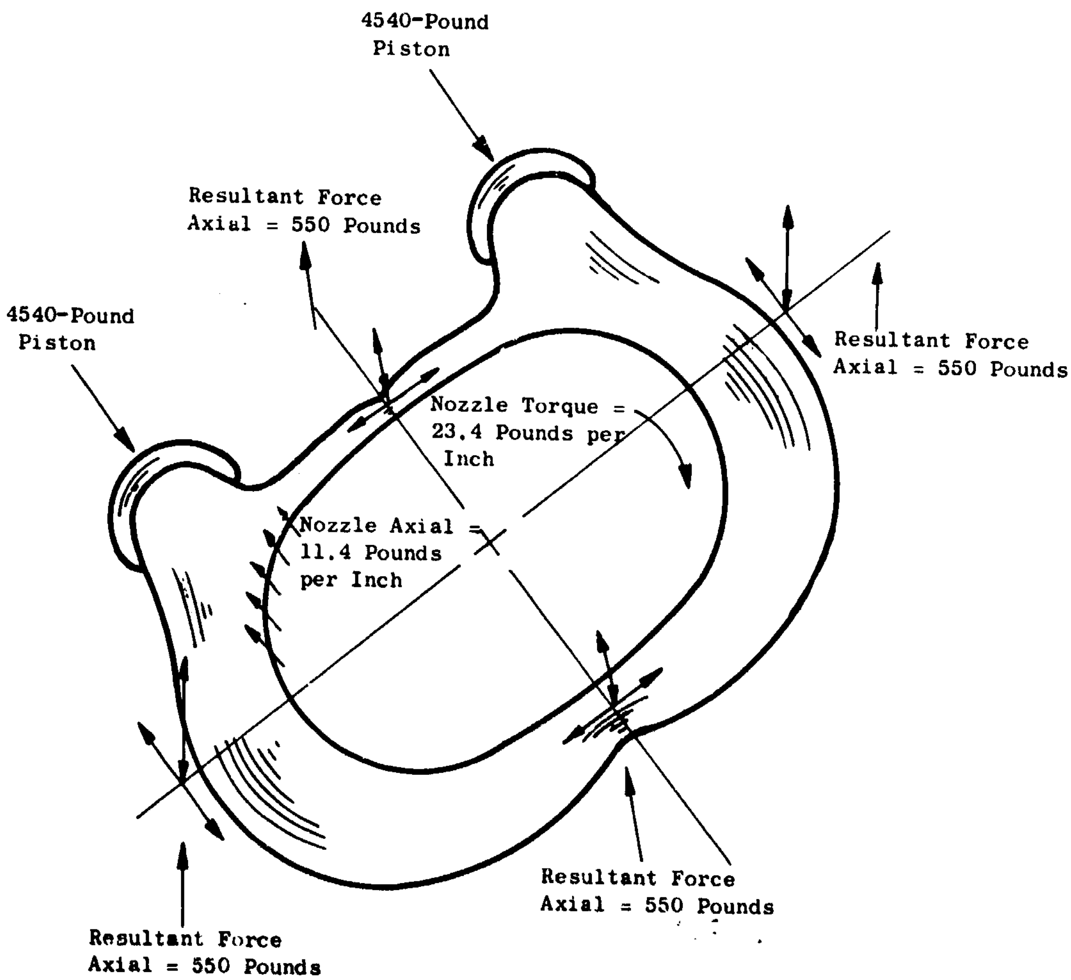


Figure 73. Scroll Loads.

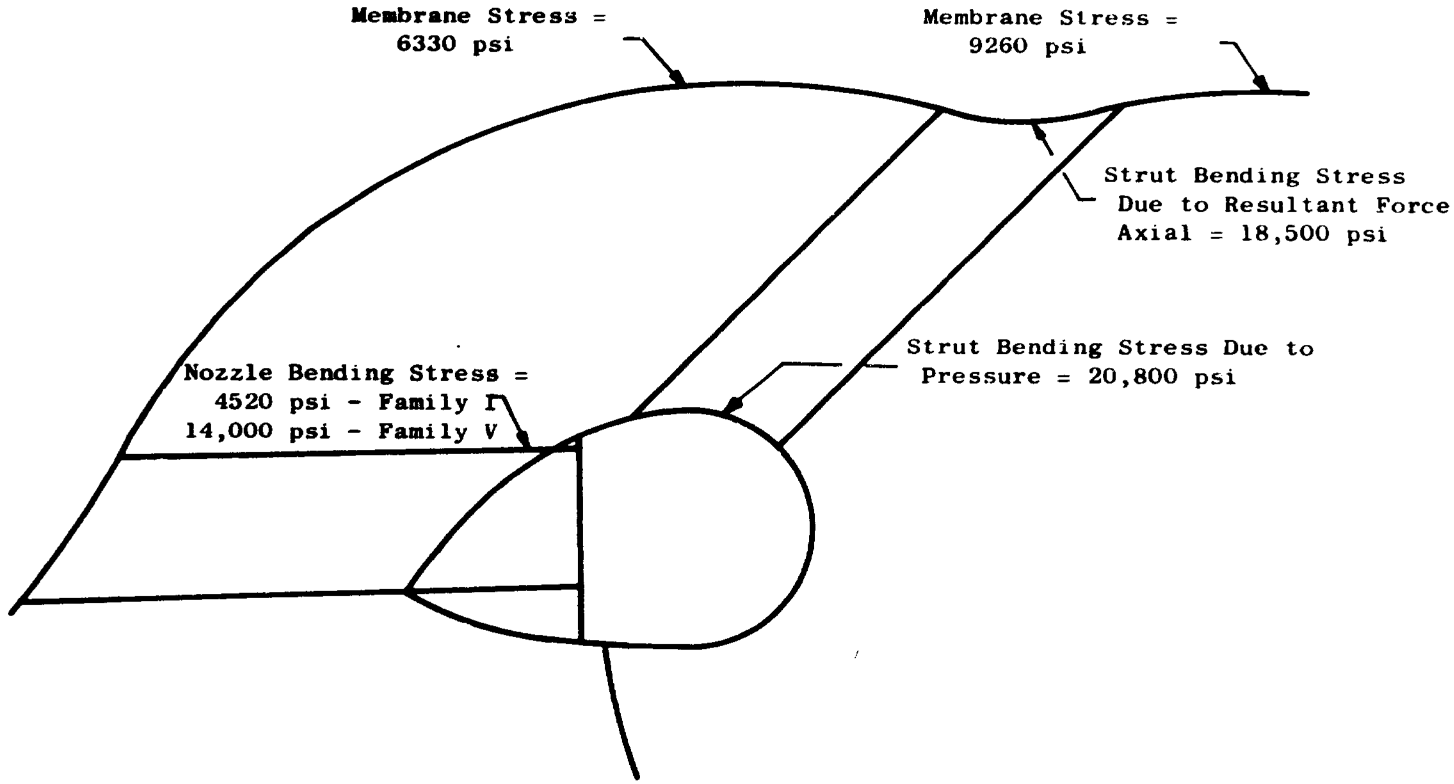


Figure 74. Scroll Stresses Due to Pressure Loading.

Rene' 41 was chosen for the scroll design, based on:

1. Mission life = 1000 hours
2. Operating temperature = 1387 degrees Fahrenheit
3. Stress level = 20,000 pounds per square inch

Table XIII shows the scroll weights. The preliminary weights are those used in the LFX design review. The final weights reflect the changes made in the final design. The final weight of 132.7 pounds compares to an objective weight of 86 pounds.

TABLE XIII
SCROLL WEIGHT

Scroll Component	Preliminary Weight (Pounds)	Final Weight (Pounds)
Nozzles	12.1	9.1
Struts	4.7	4.7
Torque tube	14.4	14.4
Stiffeners	22.2	17.3
Skin	38.0	38.0
Variable area control	25.0	25.0
Flanges	3.1	3.1
Mounts	20.0	5.0
Scroll seals	5.0	5.0
Insulation	5.3	11.1
TOTAL	149.8	132.7

ROTOR MECHANICAL DESIGN

The mechanical design of the rotor was based on the LFX objectives, XV-5A experience, LF-2 operating experience, and established rotor design practices.

The LFX rotor is a straddle-mounted single-stage rotor with a hollow shell disc, high flexural capability compressor blades, and concentrically mounted turbine segments attached to the blade tips. Because the design is mechanically and dynamically similar to the J85/LF2 rotor design, analytical techniques established for the LF2 rotor design have been used.

The aerodynamic design speed of the rotor is 3856 revolutions per minute. The mechanical design speed was defined to be 115 percent of this value, to insure rotor speed tolerance in a variety of operating conditions. The value of 115 percent speed was selected to allow 13-percent overspeed during power transfer or cross-flow and to allow an additional

2-percent overspeed for tachometer accuracy. This mechanical design speed of 115 percent is the maximum speed for continuous operation, and was used to size the bearings and to define the requirements for all life-limited parts.

The rotor deflection criterion and the criterion for non-life-limited parts were based on 120 percent of the design speed. Thus, the rotor can run intermittently at speeds up to 120 percent of the design speed without necessitating a rotor teardown.

Because lift fans are normally operated in close proximity to vital aircraft systems and personnel, an additional criterion of 130 percent speed was selected as the rotor containment speed. Any rotor operation in the speed range between 120 percent and 130 percent would necessitate a teardown and inspection but would not cause destructive failure of the blades or disc.

Past experience has indicated that fundamental axial vibratory modes should occur either at very low speeds where centrifugal loads are small or at speeds which are above the design operating rotor speeds. The six-node (cosine 3θ) axial response of the LFX rotor is below 70-percent revolutions per minute. The four-node mode (cosine 2θ) was intended to be above 135-percent revolutions per minute to avoid onset of natural frequency response stresses at the 130-percent instantaneous rotor containment speed. The calculated cosine 2θ mode occurs at approximately 130-percent speed and will have to be raised.

The LFX rotor has been designed so that the first flexural, second flexural, and first torsional vibratory modes of the blades are all greater than the second harmonic of the geometric passing frequencies. At these frequencies the energy level of the excitation field is of low strength, and the natural damping of the system is sufficient to minimize rotor deflections.

The Campbell diagram, Figure 75, summarizes the important LFX rotor dynamic characteristics.

The LFX blade design uses a comparatively high aspect ratio. This type of blading is commonly referred to as a "high-flex" design. The turbine segments mounted on the compressor blade tips serve as end plates and aid in reducing flutter. An important design criterion in this type of design is the inverse Strouhal number and is called the reduced velocity flutter parameter. The LFX initial aerodynamic design resulted in a blade configuration which had a reduced velocity flutter parameter higher than desirable. The final design is shown on Figure 76. Other representative rotors are shown for comparison.

Several sump configurations were investigated during the course of the LFX design studies. Variables under consideration included type of lubrication, bearing spacing, bearing loading and bearing size. The LFX

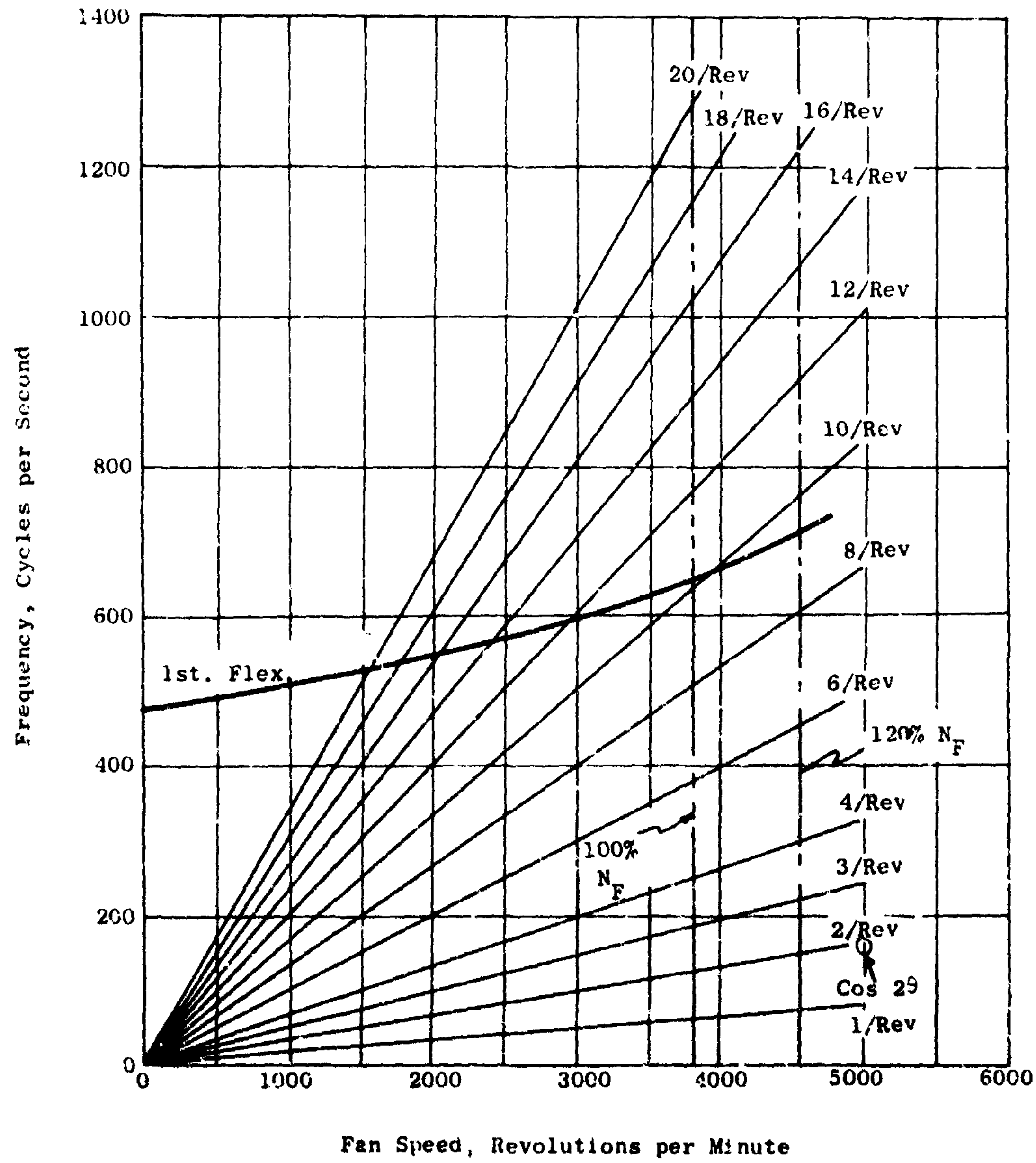


Figure 75. Campbell Diagram.

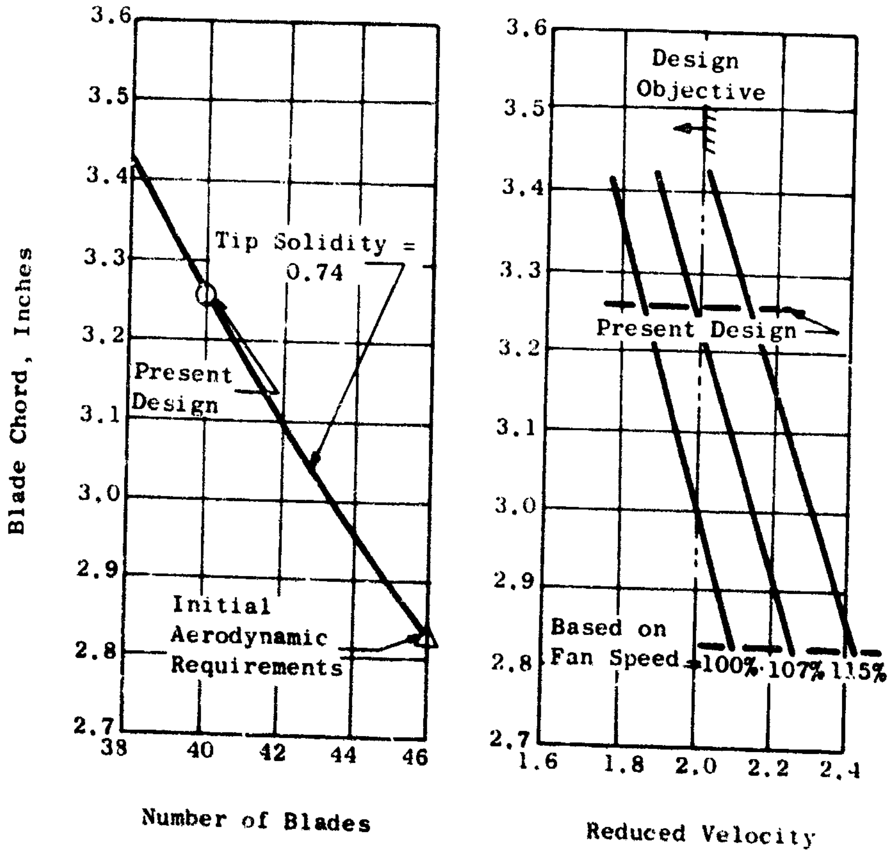


Figure 76. Reduced Velocity Flutter Parameter.

selected configuration features straddle-mounted dual bearings with the thrust bearing and roller bearing similar to the J85/LF2 configuration. The calculated loads used in the bearing studies are shown in Table XIV. The LFX final design bearings are capable of 1000-hours' operating life with a safety factor of 3.

The lift fan has been a consistent pacesetter in the use of grease-packed bearings with lubricant replenishable only by removal of the bearing from the rotor. Although the LFX design objectives called for bearings which were capable of lubricant replenishment without rotor teardown, the LFX design does not show this feature. Using the loads which were calculated for the bearings, there is no advancement of the XV-5A packed bearing state of the art required to meet the LFX requirements. One possibility which has been suggested is revitalization of the lubricant through hypodermic injection. There is no background of lift fan experience for this technique.

TABLE XIV
BEARING LOADS

Maneuver	Condition	Percent Rotor Speed	Percent Time	Resulting Load Radial	Resulting Load Thrust
Idle	Standing	70	30	1767	3040
	Taxi	100	10	4944	6308
			40		
Hover	Full PT (1)	94	4	6231	5962
	50% PT	97	8	5511	6330
	Normal	100	11	3476	6708
	50% PT	104	9	6313	7231
	Full PT	107	8	8033	7500
			40		
Transition	Normal (2)	115	19.9	8520	7300
	α Max. (2)	120	0.05	24029	7300
	50% PT	107	0.05	34108	7300
			20.0		
Static		0	50	1132	2000
Oscillatory		3	50	1132	2000

(1) PT = Power Transfer

(2) α Max. = Maximum Aircraft Angle of Attack

Titanium was used for the compressor blades and disc, and Rene' 41, a high nickel content steel, was used for the buckets and carriers.

Conservative design practice established the allowable and the design stress levels. The turbine bucket carriers are fastened to the blade tangs with a single axial pin (see Figure 77). The maximum allowable value of the blade tang average tensile stress was 25,000 pounds per square inch, which is commensurate with an estimated maximum tang temperature of 700 degrees Fahrenheit. The maximum allowable value of hole bearing stress was equal to 67 percent of the value obtained at 0.2-percent yield, and assumes a stress concentration factor of 1.5.

The calculated rotor component weights are listed below:

Carrier	38.0 lbs
Blades	58.2 lbs
Disc	50.7 lbs
Bearings	12.7 lbs
Miscellaneous	5.4 lbs
Total Rotor Weight	165.0 lbs

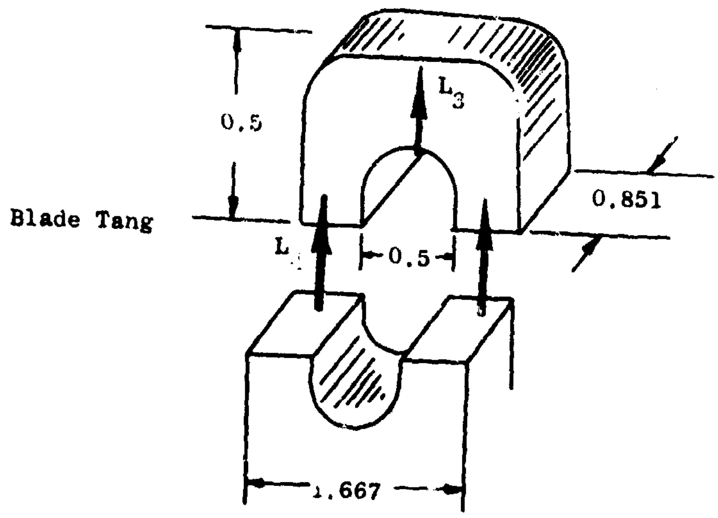
The calculated weight of 165 pounds compares to an objective weight of 177 pounds. The objective weight was based on a weight distribution similar to the J85/LF2 rotor. The calculated polar moment of inertia of the rotor is 177.4 inch-pound-second^2.

During the LFX rotor design investigations, desirable characteristics were identified. One of these was the relatively small turbine bucket height required for the 280-degree arc of turbine admission which was specified in the design. These short stiff buckets minimize the turbine sector weight. No detailed design investigations were conducted with turbines having similar through-flow velocities using smaller turbine admission arcs; however, it is probable that the rotor weight objectives could not have been met using a smaller turbine admission arc.

Potential problem areas and areas requiring further study for design verification are:

1. Thermal gradients were not calculated for the LFX rotor. Tang temperatures in excess of 700 degrees Fahrenheit could dictate the use of steel for blade material, with either a resultant increase in weight, a requirement for composite materials, or consideration of turbine cooling. Thermally induced stresses in bucket carriers may require design changes.
2. The conservative resonance-free rotor speed range objective has not completely been obtained. The goal of cosine 2θ axial vibratory mode response was 135 percent of design speed. The calculated response was at approximately 130-percent speed.
3. The maintainability characteristics of the rotor need further study. The bucket carriers are not a take-apart design. The bearing mounting arrangement will probably require a special

	Number of Buckets per Carrier	18
Carrier	Carrier Weight - Maximum	1.681
	Bucket Chord	1.0



L_3 = 20,879 pounds, based on bearing stress

L_4 = 21,959 pounds total, based on average tensile stress

Figure 77. Blade Tang.

tool for rotor assembly, and might also require heating and chilling the shaft and race mounts for assembly.

4. The reduced velocity flutter parameter is higher than previous designs at the 115 percent value of design speed. Detailed analysis should be conducted as the next step toward a final design.

5. The LFX bearing design does not allow replenishment of grease supply. The possibilities of grease "revitalization" should be thoroughly investigated with appropriate bench component testing.

DIVERTER VALVE MECHANICAL DESIGN

The diverter valve is attached to the core engine at the turbine aft flange. The diverter valve is a hydraulically operated two-position valve whose function is to direct the core engine exhaust to either a fixed-area jet nozzle (for cruise) or to the fans (for lift).

The diverter valve design selected for LFX is shown in Figure 78. It is a double-door design similar to that used in the XV-5A. This so-called double-butterfly door design is basically a blend of a cylindrical duct and a circular elbow, fitted with a flat door which closes the cylinder exit and a curved door which closes the elbow exit. In either cruise or lift mode, one door acts to close the unused exit, while the other door lies parallel to the flow stream.

The diverter valve design currently used in the XV-5A has a turning loss equal to about 3½ percent of the total pressure and has a leakage equal to about 1 percent of the flow. The turning loss for the LFX design is estimated to be the same as for the XV-5A design. The leakage loss in the LFX design could be higher than 1 percent because of the higher discharge pressure of the GE1/J1B core engine, unless an improved seal design is used. One obvious possibility is the use of higher actuation pressures to hold the door seals tighter against the valve wall.

From the available literature, it appears that the operation of dry unlubricated bearings above 1200 degrees Fahrenheit is an advance in the state of the art. However, the diverter valve in the XV-5A is presently being operated at temperatures up to 1300 degrees Fahrenheit, with no apparent difficulty in the bearings. A comparison should be made between high-temperature bearings and air-cooled bearings to determine the effect on performance, weight and cost.

René 41 was selected as the diverter valve material. This is an available heat-treatable alloy that can provide the desired 1000 hours of life at a reasonable weight. The component weights of the diverter valve are shown below, and are based on the material having a 40,000-pounds-per-square-inch, 0.2-percent yield strength for the design life and temperature.

Valve body	52.0 lbs
Forward door	22.0 lbs
Rear door	10.9 lbs
Diffuser	20.0 lbs
Linkage	9.0 lbs
Actuator	5.0 lbs
Insulation	7.5 lbs
Limit switches	0.6 lbs
Miscellaneous	3.0 lbs
Total	130.0 lbs

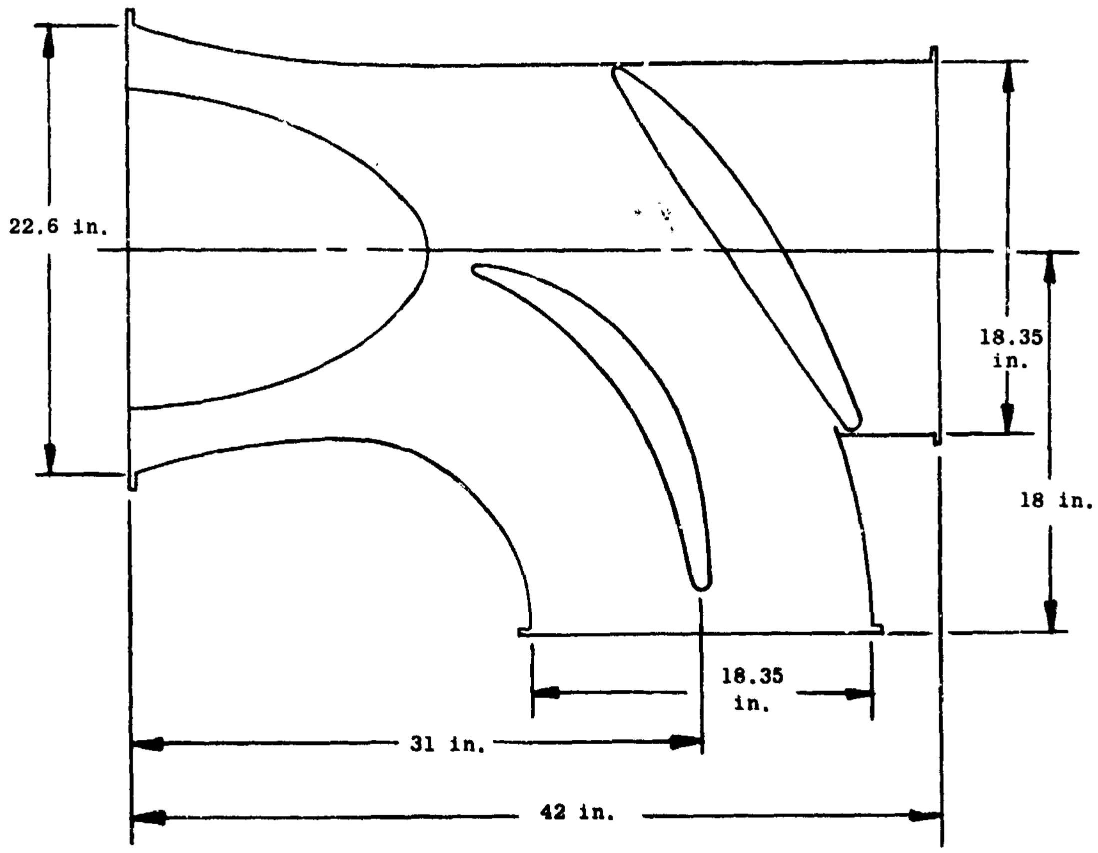


Figure 78. Double Butterfly Door Diverter Valve, Shown in the Lift Mode.

(U) CRITICAL TECHNOLOGY DEFINITION

DEFINITION OF CRITICAL AREAS

The lift fan concept has been demonstrated to be an efficient method of combining vertical takeoff and landing capability with high-speed jet-powered horizontal flight. The XV-5A aircraft and its propulsion system continue to demonstrate the inherent performance and reliability of the convertible lift/thrust system. However, studies performed during the LFX program and others have identified the need for advancement in the aft fan technology in order to insure compatibility with the advanced core engines of the 1970 era. As in the advancement of any technology, there are areas requiring detailed aerothermodynamic and mechanical design improvement, component verification and materials development. Some of these critical technology areas are listed, together with suggested testing or study. Sequence of the items listed is not intended to indicate a priority or to indicate a level of comparative risk.

1. Problem: Weight

   Specifics: The technology identified in the LFX fan design is capable of achieving ultimate lift to weight ratios of approximately 21 to 1. Progress beyond this level will require radically different approaches to the mechanical rotor-frame arrangement. Different thermodynamic approaches may also be required.

   Solution: The problem of reducing weight is inherently related to the problem of increasing performance. Within the framework of a specific structural concept such as frame-rotor-frame, reductions in weight can be achieved through the most efficient utilization of all structural material. This direction in turn leads to concepts of combining functional parts, eliminating parts where possible, and improving material properties. Mechanical ingenuity through applied conceptual design exercises is the single item most likely to yield useful advances in this area. A different approach to improving lift to weight ratio could follow the path of improved aerothermodynamic efficiencies. The LFX design compressor tip speed is 946 feet per second. A 25-percent increase in wheel speed would yield peak turbine efficiency. Higher compressor efficiency may also be available from higher wheel speeds, possibly up to 1300 feet per second. A secondary payoff from this type of design alternative would be a higher airflow and net thrust per unit area of fan disc.

   Tests: No specific tests have been devised to develop mechanical ingenuity. Rather, ingenuity is most frequently a result of applied thinking and multiple path comparisons for specific requirements. Design studies yield the opportunity for such ingenuity.

The higher wheel speed payoff can be identified initially through aerodynamic design paired with appropriate cascade tunnel testing.

Risks: Mechanical ingenuity implies doing something in a manner different from that which has been done previously. Risk may be proportional to the degree of departure from "tried and proven" practices unless sufficient detailed analysis of the specific configurations is performed.

The increase of wheel speeds will increase the centrifugal loads in proportion to the square of the wheel speed. Gyroscopic loading will also increase, as will bearing loads and heating. Increased vibratory susceptibility may require midspan ties or changes in aspect ratio.

2. Problem: Size

Specifics: The LFX design features a 360-degree active scroll arc. Design work and previous investigations indicated that this design concept would yield the minimum rotor diameter for specific core engine energy levels. However, using a reasonably low Mach number for gas flow in the scroll and designing the scroll as a pressure vessel gives an installed fore and aft dimension as much as 37 percent greater than the turbine tip diameter.

Solution: Solutions to the problem of installed fan dimensions have varying degrees of desirability and effect on installed performance. One method of decreasing scroll duct size (and installed fan dimension) is to increase duct Mach number (see Figure 80). However, this method increases internal aerodynamic flow losses and directly decreases the net lift for specific core engine energy levels.

A second method for decreasing scroll duct size is to reduce the active arc of admission. This technique would require longer turbine buckets for equal turbine through-flow Mach numbers, which, in turn, would tend to increase rotor weight and diameter. One possible solution is to increase the turbine through-flow velocities as much as 50 percent over those shown for the LFX. A secondary advantage would be reduced static pressure in the turbine rotor area and, consequently, reduced seal leakage.

A third possible method for decreasing installed fan dimensions lies in the shaping of the scroll to fill the space under the bellmouth and over the turbine rotor more completely. Although this might increase scroll weight through departure from the circular pressure vessel shape, there is a possibility of combining the scroll and the inlet bellmouth to reduce weight.

Tests: Cascade tests of suitable airfoils and flow paths with velocities up to transonic speeds can serve to identify geometrics yielding minimum Mach number effect losses. Improved duct flow through smaller disturbances can be identified.

3. Problem: Turbine Efficiency

   Specifics: The comparatively highly loaded single-stage turbine in the LFX operates at an adiabatic efficiency less than theoretically attainable. Improvement of turbine efficiencies will result in increased net lift for specific core engine energy levels.

   Tests: Effects of seal clearance and scroll/turbine/nozzle mismatch can be identified in cascade tests. Better yet, a rotating rig will enable controlled condition variation.

4. Problem: Compressor Performance

   Specifics: Although General Electric has been operating high pressure ratio (1.8) single-stage axial flow compressors for many years, there is a dearth of data on fans with a radius ratio similar to the tip turbine fans operating at medium (1.3) pressure ratios in cross flow with attendant distortion and nonaxial flow. Effects of the cross-flow velocities associated with necessary aircraft transition speeds may range from excessive performance losses to compressor stall. The additional effects of inlet blockage such as wing fan cover doors, and exit blockage such as exit louvers, need to be identified in sufficient detail to permit compensating the aerodynamic design optimum performance balance for all significant operating conditions.

   Tests: An aerodynamic and mechanical research program is vitally needed to demonstrate the feasibility of high pressure ratio lift fans operating in cross-flow transition flight environment covering the 0- to 200-knot flight regime. In the interest of timely and economical results, a small-scale model fan test program is recommended as the first step in acquiring the needed technology.

   In spite of hundreds of hours of experience available from X376, X353, and 26-inch model fan tests covering static and cross-flow conditions, there can be no confident extrapolation of the successes achieved to the fan pressure ratios of current interest. Whereas the 1.1 pressure ratio fan characteristics could be defined and correlated by means of incompressible flow theory, higher pressure ratio fans with necessarily higher velocity and Mach number flows cannot be treated with such a simplified approach. The differences are analogous to blowers versus compressors.

   Tolerance of XV-5A fans to inlet distortion, bellmouth and hub flow separation, and advancing/retreating blade velocity influences cannot be assumed automatically for higher pressure ratio fans which, by their nature, will be less amenable to variations in airfoil angle of attack. These differences can and will be subjected to available analytical techniques. But the proof required as a prerequisite for large-scale demonstrator or flight hardware problems must be derived from real fan hardware tests with representative pressure ratio, installation influence, and other fan characteristics.

The test program must provide sufficient flexibility to determine lift fan operation and design requirements in submerged wing and platform (fold-out) installations.

The test fan must be able to develop at least 1.25 pressure ratio. It must be large enough for reasonable instrumentation equipment, small enough to drive with available power sources. It must be capable of simulating the effects of a hot tip turbine upon fan performance, if it is not in fact driven by a hot tip turbine.

The test fan must be compatible with inlet and exhaust vectoring investigations. It must be capable of operation within available wind tunnel facilities. It must be sufficiently flexible to permit investigation of various fan concepts, including rotor-stator, inlet guide vane-rotor, and others.

5. Problem: Full-Scale Component Testing

   Specifics: Experience gained through long time development testing has shown that the development of advanced concepts by scale model rig testing alone does not provide development of mechanical design technology. This is due basically to the inability to duplicate satisfactorily the vibratory modes, thermal gradients and transient phenomena associated with a complete operating assembly. A demonstrator fan designed to flight hardware criteria is vital to complete technology development.

   Tests: Design, fabricate and test a full-scale tip-turbine fan with pressure ratio of at least 1.25. Component tests mentioned previously are logical prerequisites for this step.

6. Problem: Fan Covers (Doors)

   Specifics: The problem of obtaining appropriate doors to cover lift fans when the fans are not in use may well be one which can only be solved by tailoring specific installations. The LFX design shows mounting provisions for a split butterfly door similar to that of the XV-5A. Actuation and mounting of the doors through the fan structure is costly in terms of additional weight which performs no function in either flight mode. Other types of door closures should be evaluated to define better potential weights and aerodynamic performance.

   Solutions: Both General Electric and airframe contractors have evaluated the split butterfly door and multiple actuation systems. Other design concepts such as splitting doors at the fan diameter and hinging outboard, raising a door in one piece above the fan inlet, and using "roll-up" designs are conceptually feasible.

   Tests: After evaluation through conceptual design, choose the most promising of the alternate designs and test either in scale model facility or, preferably, in full-scale static and cross-flow testing.

7. Problem: Fan Power Modulation for Control

Specifics: The power transfer scheme shown for the LFX design concept embodies many areas which require verification through testing. These tests will serve to validate both performance and mechanical design approaches. However, the power transfer scheme shown is complicated and, in itself, has aerodynamic losses which result in performance losses.

Solutions: Tests of power transfer scrolls with modulation capability up to at least plus-or-minus 50 percent of nominal thrust should be performed. Tests of other identifiable types of power modulation should be performed. One such technique is embodied in U.S. Patent Number 3,146,590. This concept could be tested with a comparatively simple test rig, as schematically defined in the above patent.

8. Problem: Composite Structures

Specifics: The lift fan concept with its tip-mounted turbine exhausting concentrically around a column of cool fan air develops a requirement for materials capable of withstanding high temperatures in close proximity or contiguous with materials in cold air streams.

Solutions: Materials with widely variant characteristics need to be joined together. Specific examples would be aluminum or titanium in proximity of the turbine discharge area. Design studies and tests of bonding processes, mechanical ties and slip joints should be made.

9. Problem: Insulation

Specifics: The requirement for lightweight low thermal conductivity insulation is ever present where hot turbomachinery operates in close proximity to aircraft materials of aluminum. In addition, in the lift fan concept, it is necessary to achieve an effective insulation of the scroll and ducts. Some of the areas needing insulation are not readily accessible after assembly.

Solution: One possible solution lies in the development and testing of various thixotropic compounds having good insulating qualities. Other potential solutions to the problem lie in better sealing around the fan periphery, in using purge air to cool the resulting cavities, and in using double-wall ducts.

10. Problem: Diverter Valve

Specifics: The diverter valve in the lift fan system must operate continuously in the hot gas stream of the core engine. Seal leakage at the diverter valve doors is doubly harmful in that it causes a performance loss and also causes heating problems in non-active duct areas. Little diverter valve experience exists in either the temperature or pressure range of advanced core engines.

Solution: Tests of various seal concepts, bearings and actuation systems should be performed in atmosphere having pressure and temperature commensurate with or in excess of that shown for the LFX.

(U) TRADE-OFFS AND MODIFICATION POTENTIAL TO MINIMIZE INSTALLATION INTERFACE PROBLEMS

ALTERNATE DIVERTER VALVE DESIGNS

A second diverter valve design is shown in Figure 79. This so-called sugar scoop design has a single scoop-shaped door which rotates to cover the unused diverter valve exit passage. This door forms part of the outer diffuser wall. The diffuser center body is divided into two sections, with the forward section mounted to the valve body and the aft section mounted to the door. This aft section rotates with the door and is always aligned with the flow.

This single-door (sugar scoop) design has a shorter overall length than the double-door design. It is not apparent that this is a requirement for the LFX aircraft; however, the airplane configuration might change and make this second design desirable. This sugar scoop design requires a larger duct turning radius to turn the flow to the fans. This requirement increases the duct volume in the aircraft.

The sugar scoop design has a seal length that is 15 percent longer than the butterfly door seal, which makes leakage a more serious problem in this design.

Turning losses in the two designs are expected to be equal. The butterfly door design has been aerodynamically proven, but testing of the sugar scoop configuration will be required since previous tests of this design did not include a rotating diffuser.

The weights of the two valves are equal if constructed of the same material. Table XV compares the weights of the two designs.

TABLE XV
DIVERTER VALVE WEIGHT COMPARISON

Diverter Valve Component	Butterfly Design (Pounds)	Sugar Scoop Design (Pounds)
Valve body	52.0	48.0
Forward door	22.0	-
Aft door	10.9	-
Scoop	-	30.0
Diffuser	20.0	30.0
Linkage	9.0	5.0
Actuator	5.0	6.0
Insulation	7.5	7.5
Limit switches	0.6	0.6
Miscellaneous	3.0	3.0
TOTAL	130.0	130.1

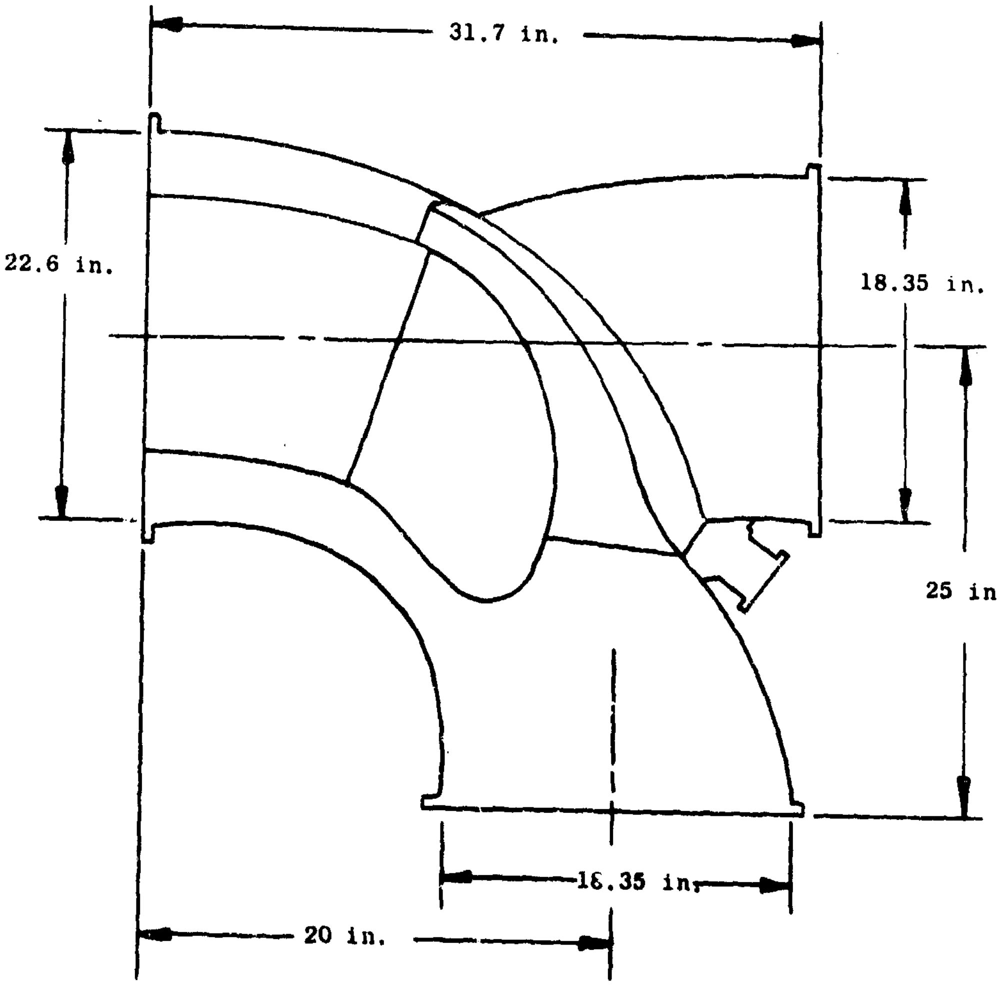


Figure 79. Sugar Scoop Door Diverter Valve, Shown in the Lift Mode.

ALTERNATE SCROLL DESIGNS

Effect of Scroll Mach Number

The present scroll is sized for a 0.25 flow Mach number to meet pressure drop objectives. This scroll design requires a wing spar spacing of 84 inches. Increasing the flow Mach number in the scroll will reduce the size of the scroll, and thus reduce the scroll weight and the required wing spar spacing, but will lower the system performance. Figure 80 shows the effect of scroll Mach number on the required wing spar spacing. For example, increasing the Mach number of 0.6 (an unreasonably high value) will reduce the required spar spacing only 3 inches.

Effect of Arc of Admission

The selected design has a 360-degree arc of admission. No studies were made on the effect of reducing this arc of admission. If the admission arc were reduced, the wing spar spacing might be reduced. However, the tip turbine would become larger, and this would necessitate considerable changes in the fan design.

Location of the Variable Area Control System

Two scroll designs were investigated. The designs differed only in the location of the variable area control system (VAC). Inboard and outboard locations were studied. The selected design had the VAC located inboard.

The location of the VAC in the scroll affects the performance and size of the scroll. Figure 81 compares the scroll dimensions for the two designs. The design having the VAC located outboard has the smaller chordwise diameter, but the best scroll efficiency is obtained by locating the VAC in the inboard portion of the scroll. With this design, the turbine gas does not have to make the sharp turn from the scroll inlet into the inboard scroll segment during normal hover performance. It has been estimated that this turn results in a 0.3-percent pressure loss.

The scroll Mach number has a constant value of 0.25 with this inboard VAC design, regardless of the amount of power transfer. For the VAC outboard design, the flow Mach number is continuously variable, depending on the amount of power transfer. This is a disadvantage because of the increase in scroll pressure losses.

The VAC inboard design will reduce the fuselage heating, but the VAC outboard design will reduce the heating of the landing gear.

The VAC inboard design has shorter actuation lines.

The VAC inboard design is 1.5 pounds heavier than the VAC outboard design because of the bulkier scroll. The VAC inboard scroll has 161 square inches more skin area than the other design.

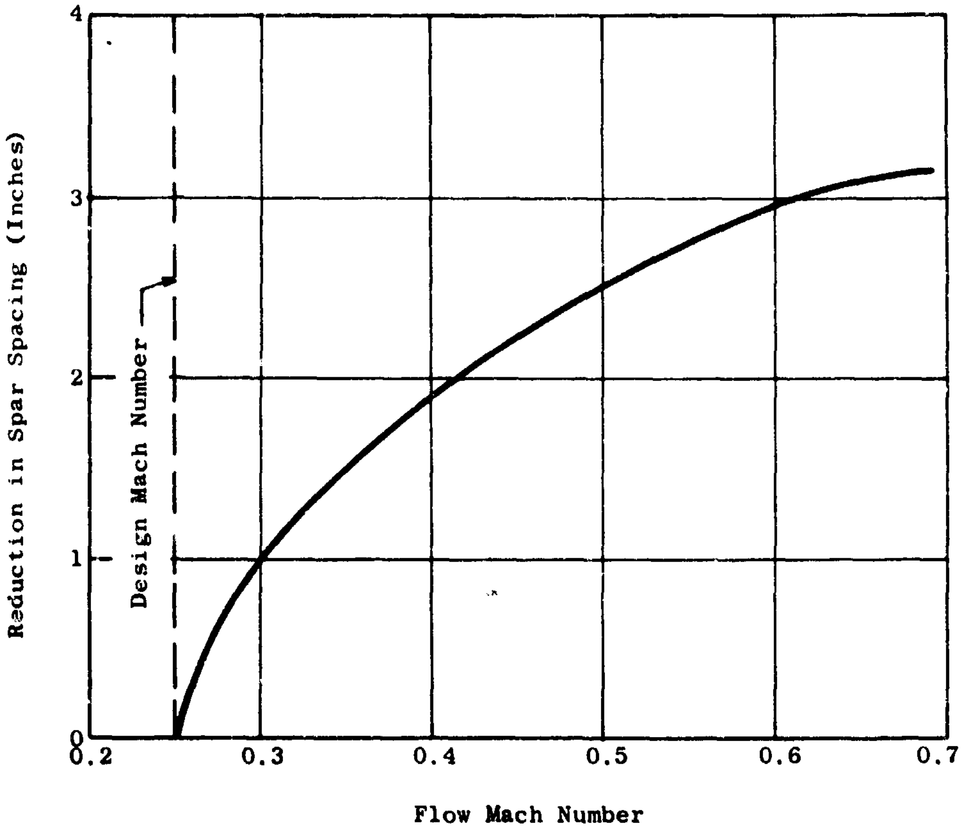


Figure 80. Spar Sparing Versus Scroll Mach Number.

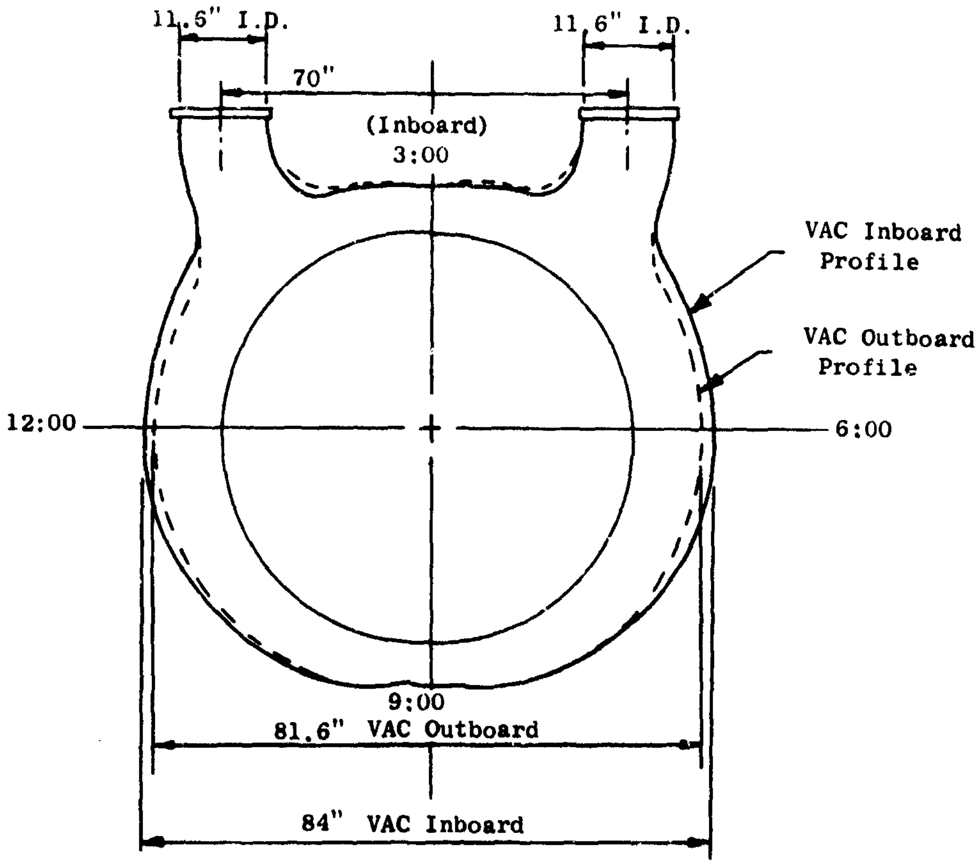


Figure 81. LFX Scroll Size Comparison.

To summarize: The VAC inboard scroll design is aerodynamically more efficient and has shorter actuation lines, but it requires 2.4 inches of additional spar spacing and is 1.5 pounds heavier.

ALTERNATE EXIT LOUVER DESIGNS

Modification to the Present Design

The exit louver design selected for LFX consisted of a cascade of aerodynamically balanced straight airfoils. The aerodynamic balance was obtained by pinning the airfoils at approximately the quarter-chord point. This design reduces the louver loads, but it may require that the fan assembly be raised in the wing to provide clearance for the louver rotation. If interference be a problem, a modification to this design could be made by unbalancing the louvers at the points of interference; that is, by moving the louver pin closer to the louver leading edge only where required. This has not been investigated.

Some consideration was given to adding an expandable rubber boot at the bottom of the fan hub (see Figure 82). This boot would help to reduce the pressure loss under the dishpan associated with straight louvers. The addition of this boot would probably require that the exit louvers be actuated from each side. The value of this boot concept needs to be determined by comparing performance gain to added weight and complexity. No evaluation has been made.

Swept Exit Louvers

The swept louvers, sometimes called chevron louvers, consist of a cascade of V-shaped airfoils. The V-shape of the louvers helps to eliminate the reduction in fan discharge area which occurs with louver vectoring, and thus helps to reduce the throttling of the fan during transition.

Several designs of swept louvers were considered (see Figure 83). The designs used a sweep angle of 30 degrees and had a loading of 3.22 pounds per square inch at a 60-degree vector angle. The results in all cases showed that a center support is required and that the actuation system will be a major design problem. It is not feasible to actuate the louvers from the side, because a center support is required. This would mean placing an actuation system on each end of the louver, which is a redundant system.

Probably the most promising actuating method is to support the louver in the center and on the ends, with the actuator being attached to the center mount (Configuration E on Figure 83). This would necessitate splitting the center of the louver to allow room for the actuator when louvers are in stowed position. The basic airfoil shape and construction are the same as those of the conventional louvers.

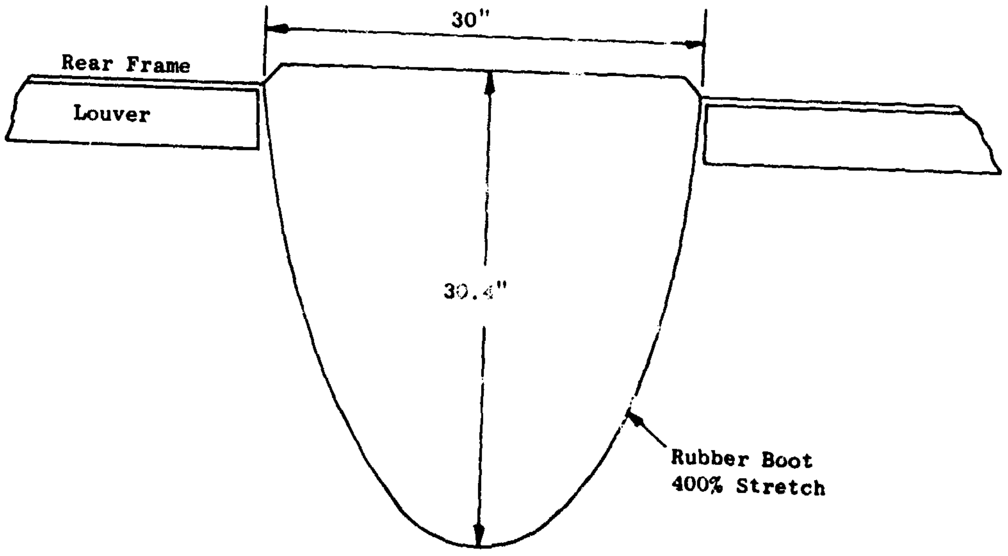


Figure 82. Inflatable Rubber Boot.

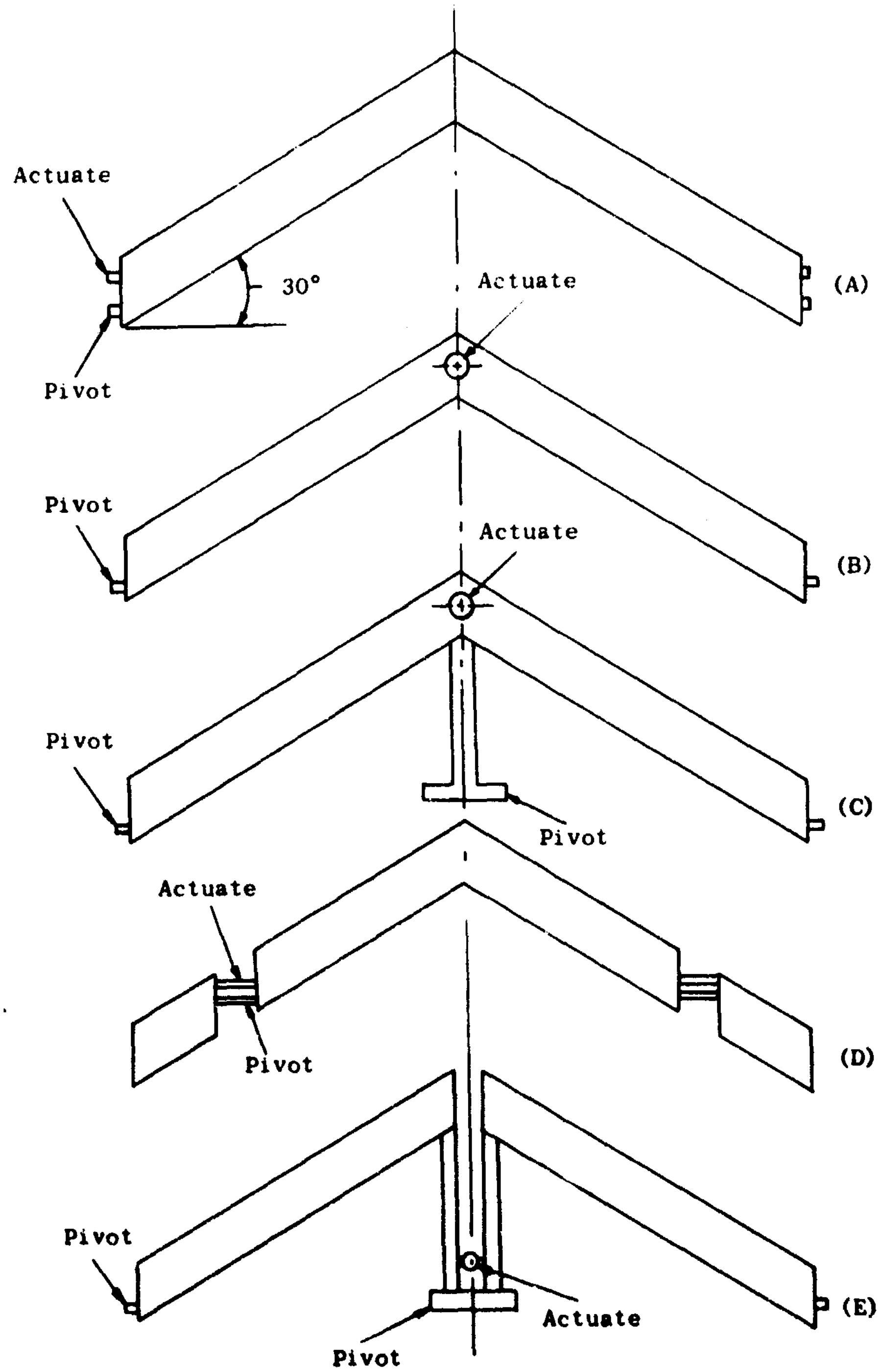


Figure 83. Swept Louver Actuation Schemes.

Since a swept louver actuation system has not been designed or tested, there is no precedent from which to draw experience. Several methods may have to be designed and tested from among those systems currently under consideration. Another area of resolution is the interface problem of whether the louver system is to be mounted on the fan or the airframe. This will determine the size and weight of the rear frame and scroll.

ALTERNATE FRAME DESIGNS

Rear Frame Loads

The present rear frame design makes it desirable to transmit exit louver loads directly to the airframe. If, because of interface requirements, the exit louver loads are assigned to the rear frame, then the present rear frame mounting system must be redefined for the added loads.

Door Latching Requirements

The present front frame design consists of four struts. The proposed door latching system requires at least six latching points. This interface problem remains unresolved.

Aft Main Frame Design

In the present design, the front frame is the main frame. A design has been studied in which the rear frame becomes the main frame (see Figure 84). Locating the main frame aft of the rotor creates the following changes in design philosophy:

1. The bellmouth and butterfly door support become part of the airframe construction. It is not believed feasible to mount the bellmouth from the scroll, because the forward honeycomb fan seal must be mounted to a structure that doesn't expand radially.
2. The bulletnose becomes a spinner which is attached to the rotor disc. This will cause slight increases in the rotor weight and gyroscopic moment.
3. The load-carrying struts must pass through the hot (1050-degrees Fahrenheit) turbine gas stream. This will necessitate fabricated steel construction, probably of Inconel 718 material. Air cooling could be incorporated to prevent the reduction of material properties due to temperature.
4. The pushrod and exit louver linkage will be mounted inside the main strut.
5. The rear frame will be connected directly to the major and minor struts. This will result in torque loads being transferred to

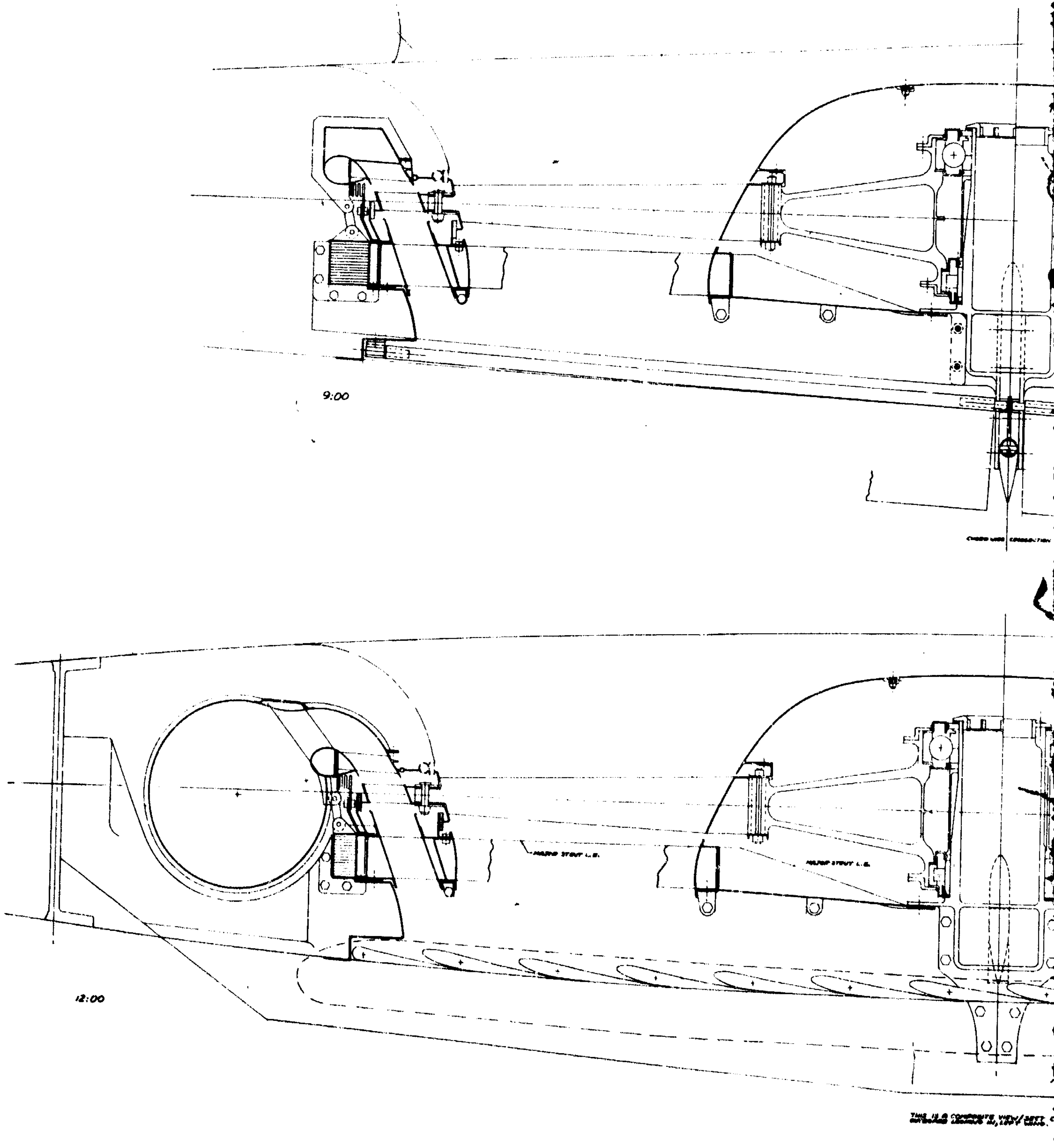


Figure 84. Aft Main Fr

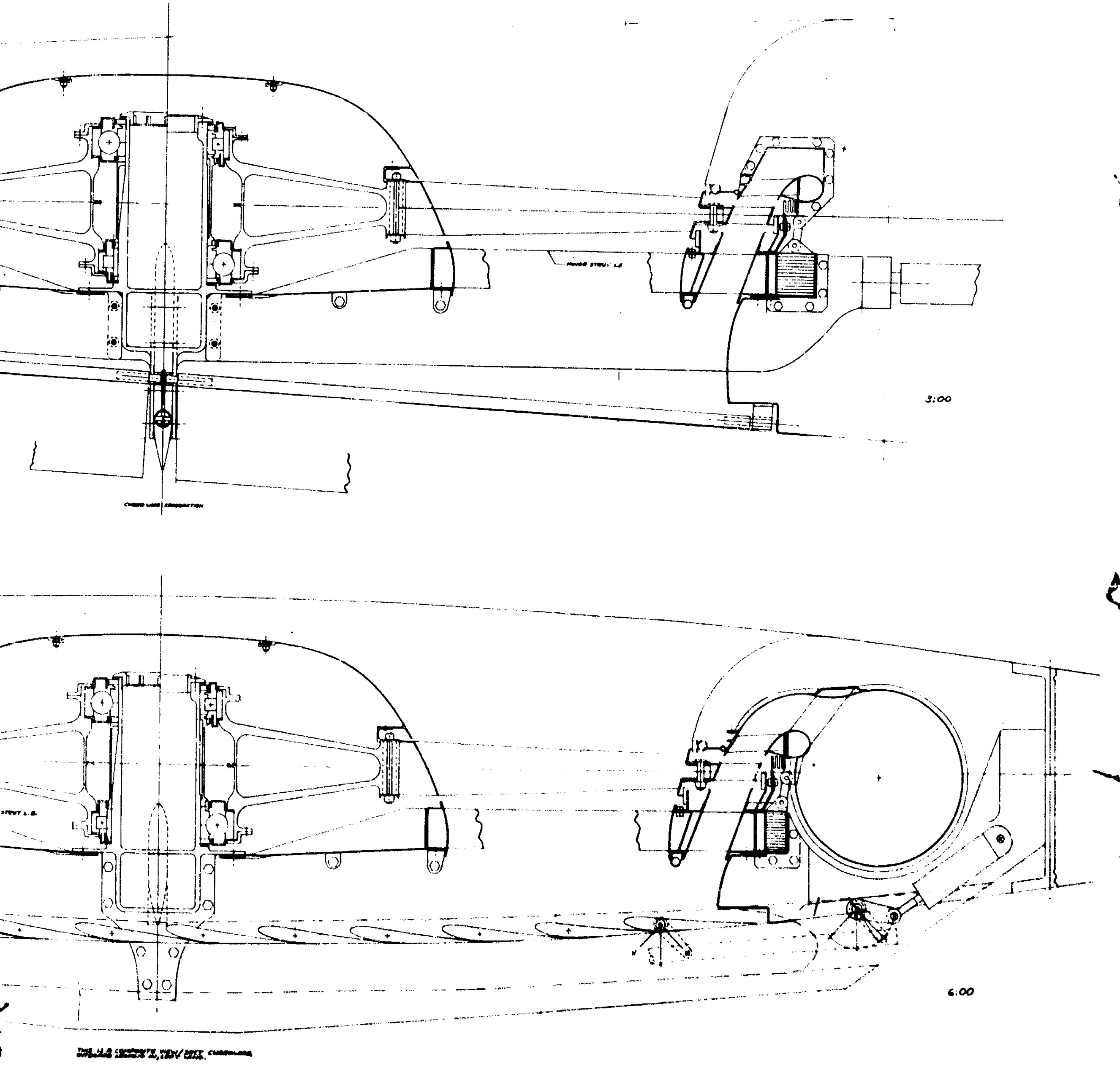


Figure 84. Aft Main Frame Design.

the frame at the dishpan flange, outer turbine box, and inner ring flange, if advantageous. This will help lighten the dishpan construction somewhat.

6. The scroll torque will be transferred to the frame mount which supports the scroll. The scroll and rear frame torque are almost equal and opposing.

7. The minor strut requires that the space between the fan and the exit louver operating envelope be at least 6 inches.

The savings in weight for this type of construction may be as much as the bellmouth structure weight on the normal design. Further design studies are required for a more accurate estimate.

DOOR ACTUATION SYSTEMS

The airframe manufacturer's proposed system for door actuation is undesirable because it requires a large hole in the highly-loaded fan hub. An alternate design was suggested on page 101.

The installation of doors and actuating mechanism requires that the rotor center line be lowered approximately 2 inches from its present position in the wing. This directly affects the use of balanced louvers.

These interface problems have not been resolved.

WING CAVITY COOLING TECHNIQUES

The XV-5A uses engine-driven squirrel-cage blowers to cool the wing and fuselage cavities. The XV-5A lift fans provide limited cooling capability. If the lift fans could be used to provide cooling air, then the requirements for engine horsepower extraction for cooling could be reduced.

The LFX rear frame design incorporates provisions for aspiration of the air in the wing cavity by providing a flow path through the turbine stators into the fan stream. The available area of the stators limits the amount of aspiration. It is estimated that this aspiration technique will reduce the squirrel-cage blower requirements about 12 horsepower.

Another possibility for wing cavity cooling is to use the front frame strut as an air passage. It is possible that this method would expose the rotor blades to a two-per-revolution excitation and cause some blockage to the flow of air through the fan. If this method of cooling is studied further, these two possible effects should be considered.

A third technique is to cut the bellmouth to allow a flow of air from the wing cavity. This would require that the blade tips be designed commensurate with this added flow, to maintain the blade tip efficiency.

THREE-ENGINE AIRCRAFT CONFIGURATIONS

The LFX propulsion system, using two GE1/J1B core engines, meets the requirements of the LFX primary mission. The LFX secondary mission, however, requires a three-engine aircraft, the third engine being used for fan mode operation only.

Figures 85 and 86 illustrate two suggested propulsion system arrangements using three engines. In Figure 85 the third engine faces forward, while in Figure 86 the third engine faces aft. The two cruise engines are shown mounted in external pods on the aft section of the fuselage.

The third engine does not require a diverter valve because it is not used for cruise. It is shown ducted directly to the fans.

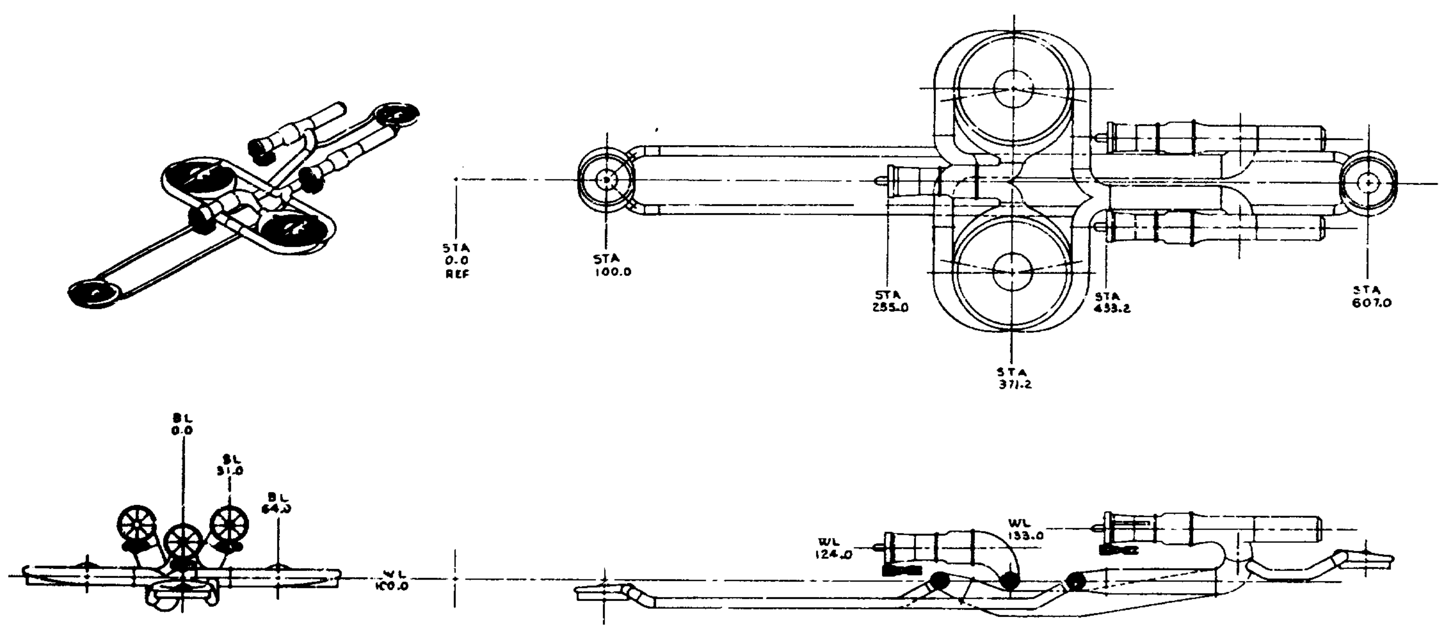


Figure 85. Propulsion System Arrangement Using Three Core Engines.

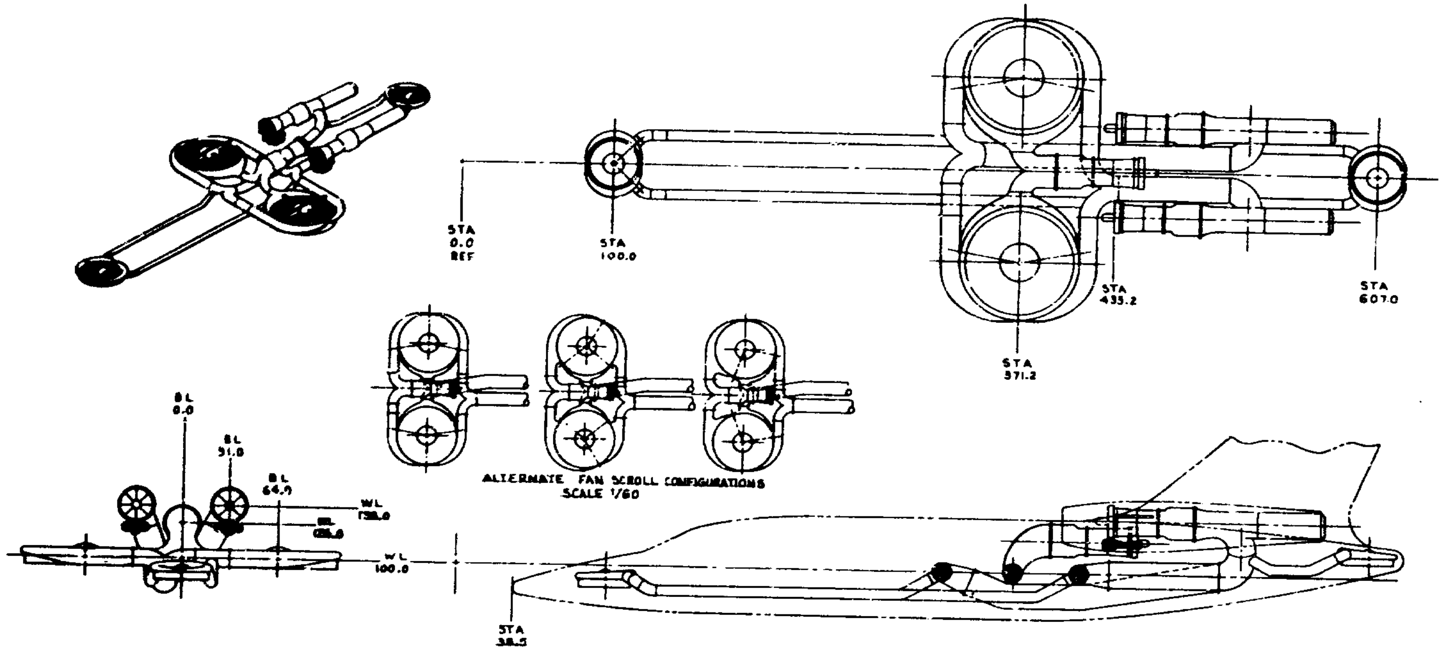


Figure 86. Alternate Propulsion System Arrangement Using Three Core Engines.

(U) EXPLORATORY DEVELOPMENT EFFORT

The advanced lift fan propulsion system study conducted under U.S. Army Contract DA 44-177-AMC-341(T), completed on 31 January 1966, has produced significant results leading toward a fan component technology demonstration program. Objectives were established, defining a fan lift/weight ratio in excess of 20, a 1.20-to-1.30 fan pressure ratio and an initial 1400-degrees-Fahrenheit tip turbine inlet temperature. The aircraft mission analysis indicated that a 57-inch-to-60-inch fan diameter would include the optimum size.

Several design problems identified in the above study will require clarification. Additional "design-in-depth" studies are required to provide additional trade-off data for evaluation of possible fan design paths leading to the complete development of an advanced fan.

In addition to the design investigations needed to identify the LFX system for demonstration, design verification (component) tests are required. These component tests, in conjunction with advanced design studies, will support the orderly technical progression toward an LFX demonstrator fan. This demonstrator fan, which would be tested under the last phase of this program, will represent a prototype for proving all necessary design concepts. An actual demonstration fan is considered necessary to achieve the technical advancement and to maintain technical momentum in lift fan technology. Verification and attainment of original objectives are the strongest of arguments in favor of continued technological advancement and hardware procurement.

PLAN FOR PERFORMANCE

This plan for performance will include the following items considered of primary importance to the further advancement of LFX technology, and leads to fulfillment of the stated LFX technical objectives.

Overall Plan for Performance

a. Phase 1 - Advanced Lift Fan System Studies

   1.1 Original LFX Contract
   1.2 Continuation of Original LFX Contract (Design in Depth)

b. Phase 2 - Final Design, Testing and Manufacture

   2.1 Detailed Design
   2.2 Component Testing
   2.3 Manufacture of Fan Demonstrator Hardware

c. Phase 3 - Demonstrator Testing

   3.1 Factory Testing
   3.2 Wind Tunnel Testing

Specific Plan for Performance

a. Phase 1 - Advanced Fan System Study (LFX)

1.1 The LFX study was initiated in July, 1965, and completed in January 1966. Preliminary designs were conceived; the first step was taken toward defining an advanced fan system under U.S. Army Contract DA 44-177-AMC-341(T).

1.2 A proposal was submitted to continue the work started in July 1965 through August 1966 (Contractor's Proposal P65-174, January 1966, "LFX - A Proposal for Continuation of Advanced Lift Fan System Studies"). The purpose of this proposal is to continue the technical momentum imparted to the advanced fan program and to define better an advanced fan. The results of this proposed study would be consolidation and solidification of the configuration of the main component of this LFX system, along with integration of previously accomplished work. Overall objectives were focused on the work necessary to make a future hardware program, or demonstrator, realistically attainable.

b. Phase 2 - Final Design, Testing and Procurement

2.1 Detailed Design - Carrying through the basic technical groundwork laid in Phase 1, Phase 2.1 would complete the detailed designs to the degree necessary for introduction of these designs into hardware. The detailed design work would be closely coordinated with the component testing being conducted in order that critical testing information could be integrated into detailed designs before drawings are issued for procurement. A final design review with the contracting agency will be held prior to release of drawings to manufacture.

2.2 Component Testing - A significant amount of component testing has already been identified as necessary to the orderly, technical progression of the LFX from paper designs to hardware. These testing requirements would probably be amended as Phase 1.2 work progresses, but at present are defined in the following areas:

a) Compressor Aerodynamic Performance - Effects of cross-flow velocities associated with necessary aircraft transition speeds vary widely in their anticipated effects on fan compressor performance. This is particularly true in the 1.25-1.30 pressure ratio single-stage fans such as the ones designed for the LFX system. Tests are recommended using scale model high pressure ratio fans, with capabilities of accepting various inlet and exit test configurations. (These tests and their results figure prominently in the programs of other interested agencies, as well.)

b) Turbine Aerodynamics - Improvement of turbine aerodynamics and efficiencies will result in increased net lift for specific core engine energy levels. Cascade or rotating rig tests should result in identification of the effects of seal clearances and scroll/turbine nozzle mismatches.

c) Power Transfer System - The power transfer system embodies many areas which require empirical verification through testing. These tests will serve to validate both performance and mechanical design approaches. Aerodynamic tests of power transfer scrolls with modulation capability up to about 50 percent (plus or minus) of nominal thrust should be performed along with testing of other identifiable types of power modulation. Unresolved problems of partial admission scrolls and their effect on turbine size and efficiency would also be determined in this testing.

d) Diverter Valve Development - The diverter valve in the lift fan system must operate continuously in the hot gas stream of the core engine. Seal leakage at the diverter valve doors is doubly harmful in that it causes a performance loss and also causes heating problems in non-active duct areas. Little diverter valve experience exists in either the temperature range or pressure range of advanced core engines. Tests of various seal concepts, various methods of cooling the valve bearings, and actuation system testing should be performed in pressures, flows and temperatures similar to those seen in LFX operating conditions.

e) Decrease Overall Diameter - The LFX design features a 360-degree active scroll arc. Design work and previous investigations indicated that this design concept would yield the minimum rotor diameter for specific core engine energy levels. However, using a low Mach number (0.25) for gas flow in the scroll and designing the scroll as a pressure vessel gives an installed fore and aft dimension as much as 37 percent greater than the turbine tip diameter.

   There are three possible methods of solving this problem: increasing duct Mach number; reducing the active arc of admission; and reshaping the scroll to fill more completely the space under the bellmouth and over the turbine rotor. Cascade tests and scale model ducting tests will serve to point out the optimum method of decreasing size while keeping the duct Mach number at a level consistent with reasonable internal aerodynamic losses.

f) Composite Structure Design and Development - The lift fan concept with its tip-mounted turbine exhausting concentrically around a column of cool fan air generates a requirement for materials capable of withstanding high and low temperatures in close proximity with each other. Materials with widely variant characteristics need to be joined together. Specific examples would be aluminum and steel for the turbine discharge area. Tests need to be made of bonding processes, mechanical ties and slip joints.

g) Fan Door Closure - The LFX design shows mounting provisions for a split butterfly door similar to that of the XV-5A. Actuation and mounting of the doors through the fan structure is costly in terms of weight. Other types of door closures should be evaluated to define better the potential weights and the effects on aerodynamic performance, such as splitting doors at the fan diameter and hinging ourboard, raising door in one piece above fan inlet, using inlet louvers, "roll-up" designs, etc.

After evaluation through detailed conceptual design, the most promising of the alternate designs should be chosen and tested either in a scale model facility or, preferably, in full-scale static and cross-flow testing.

2.3 Manufacture of Fan Demonstrator Hardware - Based on results obtained in 2.1 and 2.2 above, final drawings will be released, suitable for procurement of fan demonstrator components. Long lead-time items would be released first, followed by the shorter lead-time items. These components would be planned for delivery in time to assemble and put to test by 1 April 1968.

c. Phase 3 - Demonstrator Testing

3.1 Factory Testing - It is planned to assemble two demonstrator fans for conducting this phase of the program. In the factory testing phase, two fans would give the program stability by making it possible to test a backup fan in the event that testing on the initial unit uncovered some hitherto unknown factors. The enhancement in timely program completion and reliability that would occur with two test units is considered to be a critically important item in this overall program.

It is expected that approximately 35 to 40 total hours of factory testing would be accomplished. Factory testing would consist of verification of sea level static performance levels at full and part speed points along with investigation and analysis of mechanical performance data. These hours would be logged on one test unit. The second unit would receive a

mechanical and aerodynamic performance checkout only, of 5 to 10 hours duration, assuming that no problems were encountered with the primary unit.

3.2 Wind Tunnel Testing - After successful completion of the factory testing program and demonstration of the sea level static performance objectives, the LFX demonstrator fans will be made available for full-scale wind tunnel testing at a suitable facility. Testing of the demonstrator in various wing or fuselage installations will be considered. Dual fan installations and analysis of their effects on each other can be investigated, since two fans will be available for testing. It is estimated that 135 to 150 hours of fan testing can be accomplished during these wind tunnel tests. Early indications of reliability can be obtained while ascertaining aerodynamic performance of an advanced fan system in a suitable test installation.

d. Advanced Fan Schedule

The schedule shown in Figure 87, together with the technical milestones and estimated costs which are shown in Figure 87, are suggested based on the Exploratory Development Plan outlined above and are intended to indicate a reasonable and orderly progression toward an early LFX demonstrator vehicle.

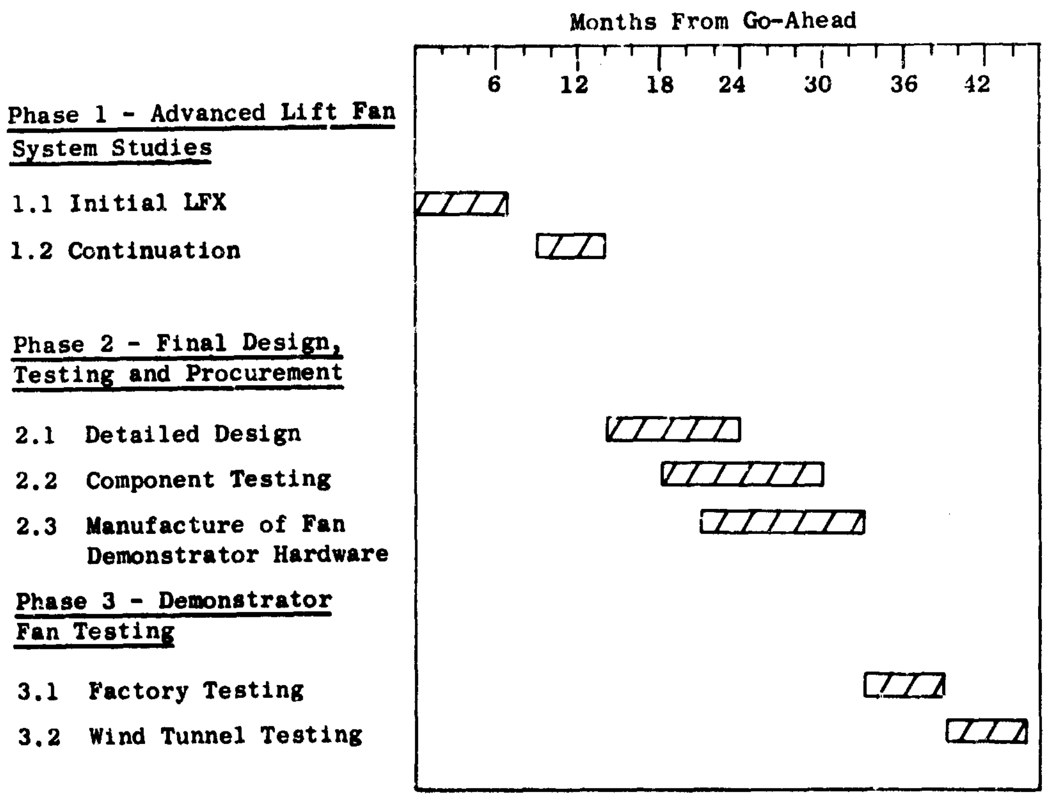
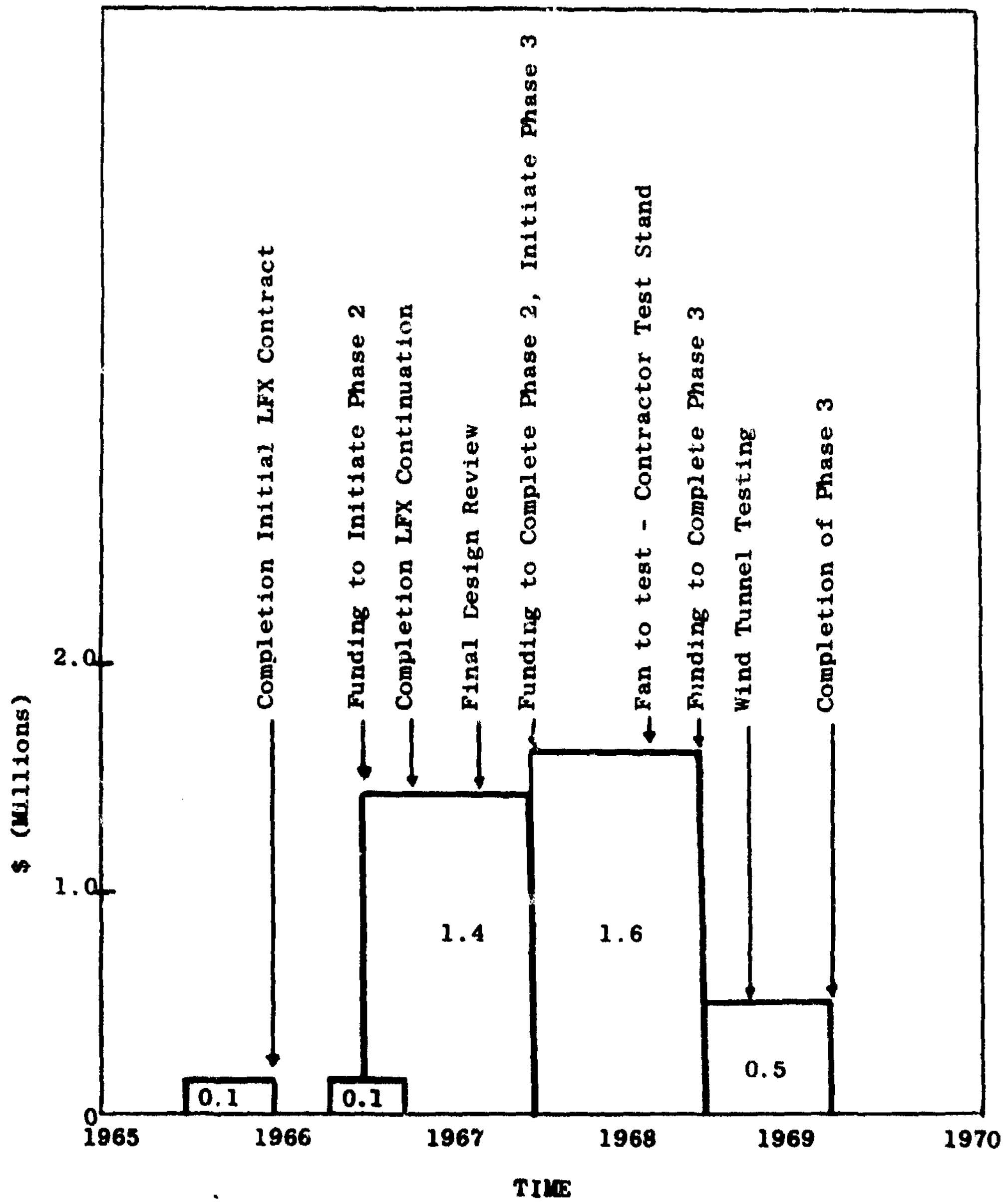


Figure 87. Schedule for Advanced Lift Fan Demonstrator Program.



Total Expenditure - $3.7 Million, FY66 through FY69

Figure 88. Technical Milestones and Estimated Costs for Advanced Lift Fan Demonstrator Program.

(C) CONCLUSIONS (U)

(C) PRINCIPAL CONCLUSIONS (U)

1. (C) Two GE1/J1B turbojets can do the primary mission operating from a 90-degree-Fahrenheit/2500-foot vertical takeoff environment and cruising entirely at 0.9 Mach flight speed.

2. (C) Two LFX wing fans of 1.25 pressure ratio coupled with two advanced design engines provide the best system choice, considering all factors, at approximately 20,000 pounds takeoff gross weight.

3. (U) Choice of pressure ratio for pitch fan (or fans) has insignificant effects on aircraft gross weight or performance and can therefore be an aerodynamic scale of the wing fan design.

4. (U) Using LF-2 fan experience as the basis of mechanical design technology, conceptual design of each component verifies an attainable lift to weight ratio of 16 for a 10,000-pound-thrust lift fan, including a thrust vectoring system.

   Additional design study indicates that a lift to weight ratio of 19 is feasible through component design changes, structural material changes, and modest predicted improvements in properties of known, proven materials.

   It should be noted that the context of lift to weight ratio as used here is conservative in comparison to other V/STOL lift propulsion devices. For example, the fan described above as 19 lift to weight ratio is capable of continuous full admission operation at 23 lift to weight ratio without exceeding any design limits.

5. (U) Additional design studies and aerodynamic design verification tests are needed to provide a solid advanced technology foundation from which an advanced demonstrator lift fan could be produced to meet the objectives as defined in this study.

(C) DETAILED CONCLUSIONS (U)

(C) Mission Analysis - Primary Mission (U)

1. (C) The primary mission can be performed by twin GE1/J1B-LFX-powered aircraft of about 20,000 pounds.

2. (U) The J85 core engines are too small for this LFX application. The advanced GE1 core engine is too large for this application and can offer growth capability to an LFX aircraft.

3. (C) The minimum size of scaled engines which can meet the primary mission requirements in a two-engine GE1/J1B aircraft is 88 percent, using a wing fan pressure ratio of 1.2. The minimum engine size for an aircraft with a wing fan pressure ratio of 1.25 is 93.5 percent. The full-size GE1/J1B core engine is, therefore, matched to the LFX primary mission requirements.

4. (U) Significant increases in payload or range capability can be obtained by reducing the 0.9 cruise Mach number. At 0.7 cruise Mach number, payload could be increased about 40 percent.

5. (U) The wing loading for the LFX vehicle in the primary mission will be between 60 and 70 pounds per foot^2, depending on the wing geometry selected. Performance (payload capability, range, maximum flight speed capability) increases with increasing values of wing loading.

(C) Mission Analysis - Secondary (6000/95) Mission (U)

1. (C) The two-engine GE1/J1B-LFX-powered aircraft cannot perform the extended range mission at the required vertical takeoff environment of 95 degrees Fahrenheit/6000 feet. This aircraft can perform the extended range mission on a sea level standard day with 93 percent of the required 3640-pound payload.

2. (C) The three-engine GE1/J1B aircraft cannot perform the secondary mission at the required vertical takeoff environment of 95 degrees Fahrenheit/6000 feet. This aircraft can perform the mission at 90 degrees Fahrenheit/2500 feet.

3. (U) The two-engine advanced GE1 aircraft cannot perform the secondary mission at the required vertical takeoff environment of 95 degrees Fahrenheit/6000 feet. This aircraft can perform the mission on a sea level standard day.

4. (U) The three-engine advanced GE1 aircraft can perform the secondary mission at the required vertical takeoff environment of 95 degrees Fahrenheit/6000 feet with a vertical takeoff gross weight of 35,000 pounds, using a wing fan pressure ratio of 1.2. This represents about 70 percent higher gross weight than that required for the primary mission. At vertical takeoff environments less severe than 95 degrees Fahrenheit/6000 feet, the three-engine advanced GE1 aircraft are large and are beyond the range of interest for this study.

5. (C) To perform the secondary mission, two GE1/J1B core engines must be scaled to over twice their design size, and two advanced GE1 core engines must be scaled to 1.68 times their design size. In both cases, the resulting vertical takeoff gross weight is larger than the 35,000 pounds required by three full-size advanced GE1 core engines.

6. (U) There is a gain of only about 200 pounds of payload using flat-rated fans (fans designed to the vertical takeoff environment of 95 degrees Fahrenheit/6000 feet) compared to the fans designed for sea level conditions which are then run off-design at 95 degrees Fahrenheit/6000 feet. This gain in payload is not sufficient to permit either of the two-engine aircraft to meet the secondary mission requirements.

7. (C) There is about a 4500-pound difference in payload capability between the primary and secondary missions for a two-engine GE1/J1B aircraft. About 2000 pounds of this difference is due to the change in the vertical takeoff environment from 90 degrees Fahrenheit/2500 feet to 95 degrees Fahrenheit/6000 feet. The remaining 2500-pound difference is due primarily to the increased requirements for mission fuel and is partly due to minor changes in mission assumptions.

(U) Fan Design

1. Objective lift performance of 10,540 pounds has been met within a 57-inch fan tip diameter.

2. Objective fan weight without power transfer was established at 506 pounds yielding a lift to weight ratio of 20.8.

3. Initial preliminary design based on J85/LF2 mechanical design technology produced a fan weight of 674 pounds, including power transfer capability, and a lift of 10,800 pounds. Lift to weight ratio was 16.0.

4. Additional design study aimed at reducing static component weights indicates feasibility of a 570-pound fan, including power transfer capability, with no performance compromise. This produced a lift to weight ratio of 19.0. Changes were limited to selection of better materials, minor improvements in properties of titanium sheet, and a revision in the connection of front and rear frame struts.

5. Lift to weight ratios beyond 19 require radical departure from proven aerodynamic concepts and structural arrangements.

(U) Additional Effort

1. Critical technology requirements have been identified along with recommendations for exploratory development. These include the two basic categories of a design in depth to optimize structural arrangement and component design and component tests to verify aerodynamic performance.

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Unclassified
Security Classification

DOCUMENT CONTROL DATA - R&D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author) General Electric Company Flight Propulsion Division Evendale, Ohio	2a. REPORT SECURITY CLASSIFICATION Confidential 2b. GROUP 4

3. REPORT TITLE

Advanced Lift Fan System (LFX) Study (U)

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)

Final Report - July 1965 - February 1966

5. AUTHOR(S) (Last name, first name, initial)

True, Harris C.
Volk, Lawrence J.

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
July 1966	184	18

8a. CONTRACT OR GRANT NO. DA 44-177-AMC-341(T) b. PROJECT NO. Task 1M121401D14415 c. d.	9a. ORIGINATOR'S REPORT NUMBER(S) USAAVLABS Technical Report 66-51 9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report) GE Company Report No. R66FPD185

10. AVAILABILITY/LIMITATION NOTICES

In addition to security requirements which apply to this document and must be met, each transmittal outside the Department of Defense must have prior approval of US Army Aviation Materiel Laboratories, Fort Eustis, Virginia.

11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY
	US Army Aviation Materiel Laboratories Fort Eustis, Virginia

13. ABSTRACT

This report presents results of conceptual design and analytical studies leading to identification of an advanced (1968-1970) lift fan propulsion system applicable to a U. S. Army V/STOL surveillance and target acquisition mission.

The successful XV-5A flight research vehicle is used as the progenitor of a family of tip turbine fan-in-wing aircraft from which mission and aircraft design analyses point out an advanced turbojet as the logical core engine gas producer from a stable of engines either available or under active development. These analyses also define optimum lift fan objectives in terms of dimensions and performance.

Results of conceptual lift fan design studies are presented in a comparison with objectives.

Critical technology requirements are identified and recommendations for an exploratory development effort are defined.

DD FORM 1 JAN 64 1473

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14. KEY WORDS	LINK A		LINK B		LINK C	
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