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QUIET CLEAN SHORT-HAUL EXPERIMENTAL ENGINE (QCSEE)

Preliminary Under the Wing Flight Propulsion System Analysis Report

February 1976

by

Advanced Engineering & Technology Programs Department

General Electric Company

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16. Abstract

The Quiet Clean Short-Haul Experimental Engine (QCSEE) Program includes the preliminary design and installation of high bypass, geared turbofan engine with a composite nacelle forming the propulsion system for a short-haul passenger aircraft. These flight systems contain the technology required for externally blown flap type aircraft with under-the-wing (UTW) propulsion system installations for introduction into passenger service in the mid 1980's.

Based on the flight designs the program provides for the design, fabrication and testing of this UTW experimental engine containing the required technology items for low noise, fuel economy, with composite structure for reduced weight and digital engine control.

The report deals with the QCSEE UTW Flight Propulsion System installation and nacelle component and systems design features on a short-haul, powered lift aircraft. A substantial portion of this report was produced at Douglas Aircraft Company and covers their efforts in support of the QCSEE program.

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FOREWORD

The Quiet Clean Short-Haul Experimental Engine (QCSEE) Program is currently being conducted by the General Electric Company, Aircraft Engine Group in accordance with NASA Contract NAS3-18021, under the direction of Mr. C.C. Ciepluch, NASA Project Manager. The Program includes the design, manufacture and test of an under-the-wing (UTW) and an over-the-wing (OTW) experimental engine. Both engines are intended to develop the technology needed for externally blown flaps, short takeoff and landing, commercial, short-haul aircraft.

To ensure the selection of appropriate flight system parameters and characteristics, and to provide design guidance, subcontracted study support was obtained from aircraft manufacturers and operators. General Electric selected The Boeing Company to provide support for the OTW system, Douglas Aircraft Company to provide support for the UTW system, and American Airlines to evaluate both installations and provide an appropriate operational scenario. Specific subcontracted effort consisted of guidance in selection of the engine cycles, installation design, propulsive-lift interactions, control interfaces, acoustics, performance, and economic analyses.

Although earlier studies had indicated a need to operate from a 609.6 m (2000 ft) runway, it was concluded by all contributors that the Flight studies of a commercial short-haul transport should be conducted based on a 914.4 m (3000 ft) runway, typical of existing close-in airports.

The experimental system retained the 609.6 m (2000 ft) runway requirement to assure technology margin for the aircraft ready to enter airline service in the mid 1980's. In either case, the propulsion system would be designed to meet a noise requirement of 95 EPNdB at 152.4 m (500 ft) sideline during approach and takeoff. Since final system requirements will not be definitized for some time, the experimental engine objectives [including 609.6 m (2000 ft) runway], being the more stringent, were not changed. Thus the technology margin that is developed in the QCSEE Program will be adequate for any foreseeable system requirement.

This report covers the subcontracted analyses of the UTW aircraft system. The propulsion systems used in the study were projected "flight" systems based on the technology being developed in the experimental program. Propulsion system weight, performance, and installation features have been projected in a rational basis from the experimental propulsion system design.

Design studies were performed to develop concepts for integrating the flight design version of the QCSEE propulsion system with the UTW, externally blown flap (EFP), powered lift aircraft addressed to the requirements for an airliner designed for short-haul service. The airplane concept was based on technology consistent with providing a reliable vehicle that is both durable and economical to operate and that could be ready to enter airline service in the mid 1980's. The preliminary design studies produced definitions of the airplane configuration, characteristics, performance, and operating economics, in addition to developing the concept of the engine pod/pylon/airframe accessories arrangement.

Attention was devoted to identifying the geometry of a suitable engine air inlet including the needed ice protection features. The airplane accessories requirements were identified and the concept of an accessories pack installed remote from the top-mounted engine accessories gearbox was developed. The space envelope, weight, and significant features and characteristics of the various components were identified as were the estimates of the shaft power demands this system placed on the engine power takeoff (PTO) drive. Studies were made to develop the concepts for the cabin environment control and lifting surface ice protection systems needed for the short-haul airplane. The air bleed demands imposed on the engines were estimated and concepts for matching the bleed capacity to the demands were developed.

In addition, studies were made to identify the significance of two variants of the QCSEE; one with bottom-mounted accessories and the other with reduced acoustic treatment. These studies indicated the impact of the concepts on airplane characteristics, performance, and economics.

The conclusions that could be drawn from the results of the work have been documented.

Significant contributions to this report were made by the Douglas Aircraft Engineering Department in the areas of aerodynamics, advanced design, avionics, environmental control, power plant, structures, and weights. American Airlines provided the short-haul aircraft requirements and conducted various installation reviews of the propulsion system.

This report deals exclusively with the QCSEE UTW Flight Propulsion System design and analysis based on the aircraft use of a 914.4 m (3000 ft) runaway. With the longer runway the aircraft will attain a higher take-off velocity permitting reduced aircraft flap angles and reduced engine thrust. On approach, the longer runway permits higher aircraft approach velocity also reducing flap angle and thrust requirements. This results in a reduction in the amount for noise suppression panel treatment required to meet the acoustic program objective.

TABLE OF CONTENTS

TABLE OF CONTENTS (Continued)

TABLE OF CONTENTS (Concluded)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Concluded)

LIST OF TABLES

LIST OF TABLES (Concluded)

1.0 SUMMARY

The Quiet Clean Short-Haul Experimental Engine (QCSEE) Program includes the preliminary design and installation of a high bypass, geared turbofan engine with a composite nacelle forming the propulsion system for a short-haul, passenger aircraft. These flight systems contain the technology required for externally blown flap type aircraft with under-the-wing (UTW) propulsion system installations for introduction into passenger service in the mid 1980's.

Based on the flight designs the program provides for the design, fabrication and testing of a UTW experimental engine containing the required technology items for low noise, fuel economy, with composite structure for reduced weight and digital engine control.

This report summarizes the preliminary design of the QCSEE UTW Flight Propulsion System installation and nacelle component and systems design features on a short-haul, powered lift aircraft. A substantial portion of this report was produced at Douglas Aircraft Company and covers their efforts in support of the QCSEE program.

The major purpose of the QCSEE Program is to develop and demonstrate the technology required for propulsion systems for quiet, clean, and economically viable commercial short-haul aircraft. This comprehensive program includes the following objectives:

- To develop the propulsion system technology which will permit a short-haul aircraft to achieve the system noise goal of 95 EPNdB along a 152-m (500-ft) sideline when the engines are scaled to a total installed thrust of 400,300 N (90,000 lb). The design shall also minimize the ground area (footprint) exposed to objectionable noise levels.
- To demonstrate a propulsion system which will meet advanced pollution goals under all operating conditions.
- To develop the technology for very-high-bypass ratio engines with quiet low-pressure-ratio geared variable-pitch fans.
- To develop the technology required to meet propulsion system performance, control, weight, and operational characteristics.
- To develop the material, design, and fabrication technology for quiet propulsion systems which will yield engine designs which have an uninstalled thrust-to-weight ratio greater than 6 to 1 and installed thrust-to-weight ratios greater than 3.5 to 1.
- To develop the technology which will yield engine thrust response characteristics required for powered lift operations.

- To provide the technology which will permit the design of quiet, efficient, lightweight thrust reversing systems for powered lift aircraft.

- To provide the technology to permit the design of integrated engine and nacelle installations which will be tolerant to aerodynamic distortions expected with operating flight conditions (such as high crosswinds, large angles of attack, and side slip) and still provide good cruise performance.

- To provide the digital electronic engine control technology required to improve engine and fan pitch control, thrust response, operational monitoring, and relief of some of the pilot's workload especially during powered lift flight operations in the terminal area.

2.0 INTRODUCTION

This report presents the preliminary results of activity conducted under the Supporting System Design and Economics Studies task of the Quiet Clean Short-Haul Experimental Engine (QCSEE) program. The primary objective of contract tasks 1.1.2, 1.1.3, 10.1, and 10.3 is to provide design guidance to the experimental engine design based on studies of flight installation systems and to evaluate and update a conceptual flight propulsion system design based on QCSEE test results. This report covers the QCSEE Preliminary UTW Flight Propulsion System Analysis during the period of the detail design of the UTW experimental engine.

The aircraft economics and engine installation studies were conducted by Douglas Aircraft as a subcontractor to General Electric. The representative short-haul airliner was evolved from past Douglas studies including "Study of Quiet Turbofan STOL Aircraft for Short-Haul Transportation," conducted under NASA contract NAS-26994, and "Analysis of Commercial and Military STOL Design Criteria," conducted under Air Force contract F33615-72-C-1534. These studies showed that the high bypass, low pressure ratio, variable-pitch turbofan engine has the potential of providing an economical propulsion system for achieving the very quiet aircraft noise levels of 95 EPNdB on a 152.4 m (500 ft.) sideline.

For powered - lift aircraft, the flap-engine exhaust interaction noise becomes dominant, and the minimum noise is highly dependent upon the engine exhaust velocity. Both the jet noise and flap/exhaust interaction noise can be reduced by the lower exhaust velocities associated with higher bypass ratio engines. For current engines, the thrust reverser is the single heaviest propulsion installation component, other than the basic engine. At high bypass ratios, the weight penalty of the thrust reverser becomes more severe. The variable-pitch fan allows thrust reversal with less weight penalty than a conventional reverser in a high bypass fan installation. The variable-pitch fan can also provide a more rapid thrust response by keeping the low pressure spool rpm up at low power by using a lower fan pitch. Numerous other advanced technology items are in the QCSEE program such as electronic controls, airflow control with a modulating nozzle, integrated engine/nacelle structure, and near-sonic inlet for noise reduction. These items require a higher degree of engine-to-airframe integration than is provided by current design approaches.

The specific study aircraft was designed to meet the requirements specified by American Airlines in the document, "Operational Scenario and General Requirements for Multi-Engine STOL Passenger Transport Airplane for Introduction in 1980-1982," dated February 13, 1974 (Revised 4/10/75), see Appendix A. The resultant airplane carries 162 passengers, has a design field length of 914.4 m (3000 ft) and a design range of 926 km (500 NMi). It requires four variable-pitch fan engines with an uninstalled takeoff thrust of 81,400 N (18,300 lb) per engine.

A concept for integrating a flight version of QCSEE with this airplane has been evaluated. Completed activities include those that address structural interfaces, such as pylon/engine lines and parting surfaces, accessory locations and volumes, and basic nacelle internal and external aerodynamic lines.

The process of developing the concept of integrating the production derivative of QCSEE with an airliner having an EBF powered lift system embraced these tasks, which are documented herein:

Airplane configuration studies
Airplane weight estimates
Airplane performance
Airplane economics
Airplane subsystem airbleed requirements
Engine air inlet design parameters
Engine inlet ice protection airbleed needs
Engine inlet ice protection system concept
Airbleed subsystem concept
Airplane shaft-driven accessories concept
Disposition of heat from accessories
Idle thrust effects on airplane ground handling
Thrust reverser effectiveness requirements
QCSEE installation design concept
Installation concept for bottom accessories (trade study)
Installation concept for reduced acoustic suppression (trade study)
Digital control integration
Making wing flaps compatible with turbine exhaust
Pylon and engine mount concepts
Propulsion system fire protection concept
Propulsion system cooling and ventilation concept
Engine and accessories drains concepts
Core cowl and pylon protection against overpressure
Engine durability requirements for short-haul application
Engine performance and installation requirements for short haul
Impact of bottom accessories version on airplane
Impact of reduced acoustic suppression version on airplane
Pressure and inertia forces on inlet duct
Airplane takeoff and landing profiles for noise estimates
Maintainability assessment
Access provisions concepts
Maintenance workstand concept

3.0 AIRCRAFT REQUIREMENTS AND CHARACTERISTICS

3.1 AIRCRAFT REQUIREMENTS, GROUND RULES, AND SIZING METHODS

The principle design requirements for the baseline UTW aircraft are:

Payload - 150 to 170 passengers
Range - 926 km (500 NMi)
Field Length - 914.4 m (3000 ft) for SL, 32.2° C (90° F) conditions
Noise - No greater than 95 EPNdB on 152.4 m (500 ft) sideline

A range capability of 1389 km (750 NMi) with a full passenger load was also desired when operating from runways longer than 914.4 m (3000 ft). These design goals are based on requirements specified by American Airlines (Appendix A).

The aircraft sizing process is illustrated in Figure 1. Takeoff and landing performance is calculated to determine combinations of wing loading (W/S) and aircraft thrust-to-weight ratio (T/W) that satisfy the 914.4 m (3000 ft) field length requirement. Parametric aircraft are sized for each of these combinations and the design point is selected on the basis of minimum direct operating cost.

3.1.1 Takeoff

The takeoff and landing performance ground rules are based on FAR Part 25 and Part XX (Reference 1). Takeoff field length is defined as the greater of:

- 1.15 x all engine takeoff distance to 10.7 m (35 ft) height.
- Distance to 10.7 m (35 ft) height with critical engine failure at V_1.
- Distance to accelerate to V_1 and then decelerate to a stop.

The following constraints were used in calculating takeoff field length.

- Rolling friction, $\mu = 0.025$
- Fuselage angle of attack $\leq$ ground limit = 15°
- Rotation rate, $\leq$ 5°/sec

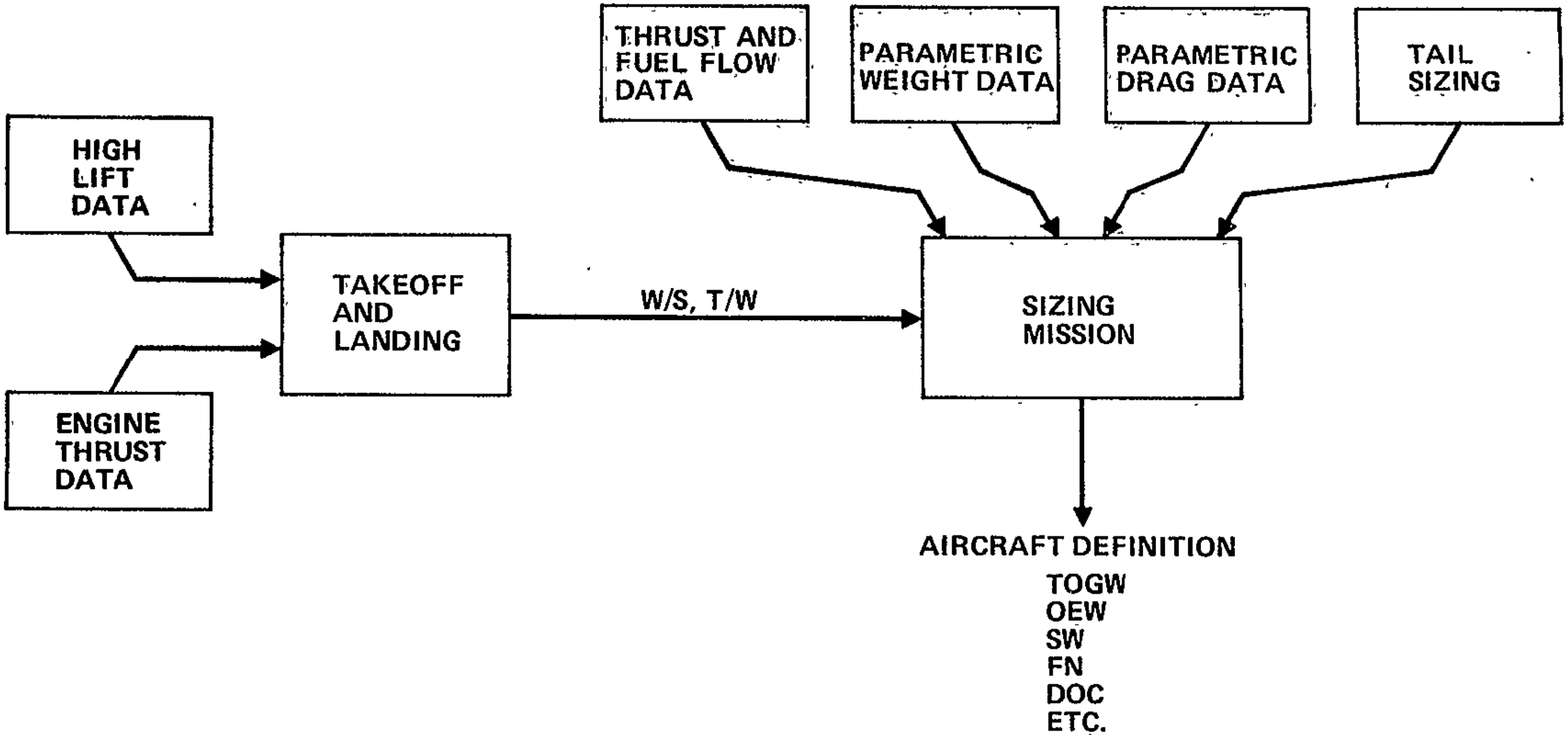


Figure 1. Aircraft Sizing Process.

- $C_L \leq 0.9\ C_{L_{max}}$
- No deceleration during air run to 10.7 m (35 ft)
- 2.6 m/sec (5 knot) early rotation may not give greater one-engine-failed takeoff distance
- Accelerate - Stop distance based on three second delay after reaching V_1 followed by a deceleration of 0.4g to a stop.
- Engine-failure recognition time = 1 sec

Takeoff performance was estimated by calculating the time history of the takeoff flight path. This method allows for recognition of changes in aerodynamic characteristics and flight limitations which occur during the maneuver. The calculations are governed by the following assumptions:

1. The aircraft is assumed to be a point mass, i.e., second-order rotational dynamics have been ignored and the analysis is essentially two dimensional.

2. The forces acting on the aircraft are summed in the longitudinal and normal directions and are a function of true airspeed, flight path angle, angle of attack, and height above the ground.

3. Any restriction on speed, acceleration, attitude, etc. may be imposed as desired.

4. The path is generated by numerical integration of the forces acting on the aircraft over small increments in time using a digital computer.

Powered-lift ground effects and flap angle optimization are included in the calculations.

3.1.2 Landing

The methods and assumptions used in calculating landing field length are essentially the same as those used for takeoff performance. The landing maneuver consists of three segments: approach, flare, and ground roll, as shown in Figure 2. Landing field length is defined as the landing distance over a 10.7 m (35 ft) obstacle divided by a 0.6 factor, i.e., a 914.4 m (3000 ft) field length requires a landing distance of 548.6 m (1800 ft). The following constraints were used to establish the equilibrium approach conditions:

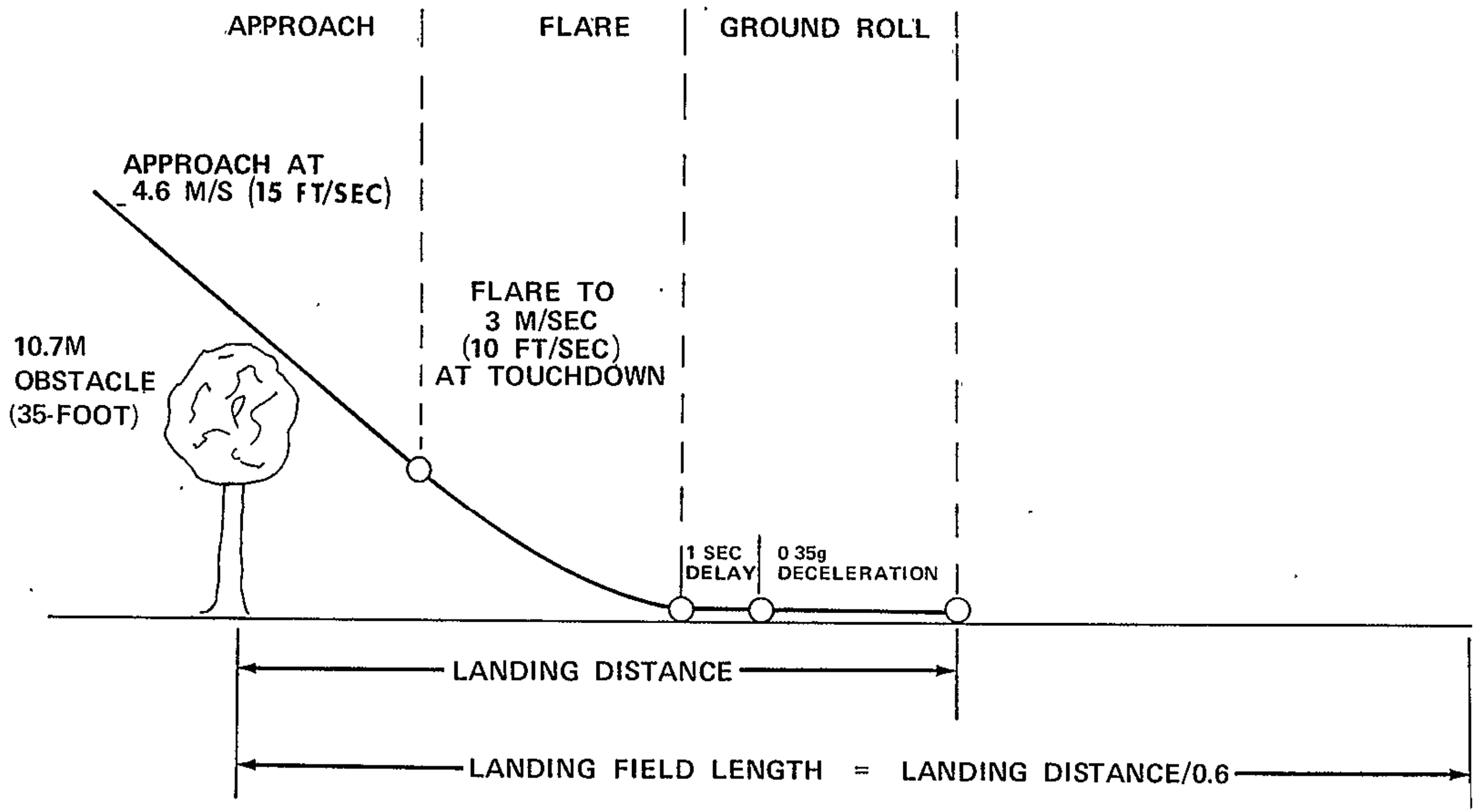


Figure 2. Landing Field Length Definition.

- Approach thrust $\leq$ 65% of takeoff gross thrust. This provides sufficient thrust margin to permit continuation of the approach after failure of an outboard engine.

- Sink rate = 4.6 m/sec (900 fpm). For a 914.4 m (3000 ft) landing field length, this corresponds to an approach path angle of approximately 5.2 degrees.

- $V_{approach}/V_{stall} \geq 1.3$ with approach thrust

- Available load factor $\geq$ 1.2 at 90% of $C_{L_{max}}$

- Angle of attack margin $\geq$ 10 degrees from stall

The flare maneuver is governed by the following constraints:

- Fuselage angle of attack $\leq$ ground limit + 15°

- Rotation rate $\leq$ 5°/sec

- $C_L \leq$ 100% of $C_{L_{max}}$ in ground effect

The flare maneuver was accomplished by retracting the DLC spoilers at the flare height and rotating the aircraft at 5°/sec. As the aircraft approaches the ground, C_L and C_D tend to drop off due to ground effect. Retraction of the DLC spoilers provides an incremental load factor of approximately 0.1g.

The ground roll consists of one second at constant speed from touchdown to deceleration device effectiveness, followed by a constant deceleration of 0.35 g to a stop. Landing, like takeoff, was calculated for sea level, 32.2° C (90° F) conditions.

3.1.3 Sizing Mission

The thrust-to-weight ratio and wing loading combinations which satisfy the takeoff and landing field length requirements together with parametric weight data [OEW = f(TOGW, W/S, T/W)], installed thrust and fuel flow maps, and drag and tail sizing information are used as inputs to a computer program which performs the aircraft sizing calculations. This program was developed by Douglas Aircraft Company over the last six years specifically for the sizing of aircraft in the advanced-design concept stage. The methods used are essentially those of classical, airplane performance. The mission profile used in sizing the baseline aircraft is shown in Figure 3. The computer program was used to calculate 2-degree-of-freedom mission time histories, iterating on the weight, thrust, drag and tail sizing data to determine the characteristics such as TOGW, wing area, engine size, OEW, fuel burned, etc. of an aircraft which satisfies the requirements of the mission profile with the desired payload. When a solution has been found, the program calculates a direct operating cost (DOC) breakdown.

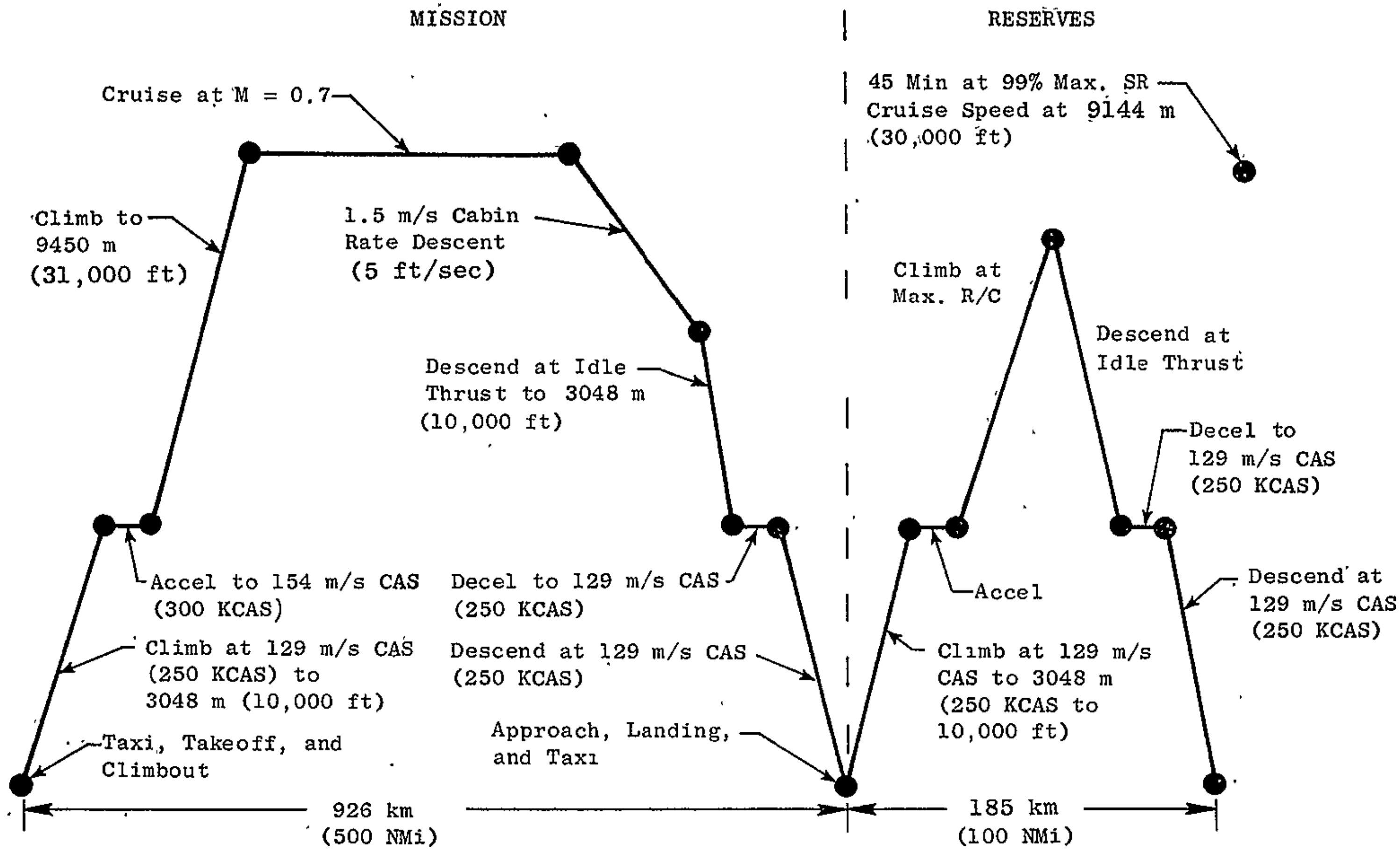


Figure 3. Mission Profile, Sizing.

A constant cruise Mach number of 0.70 at 9450 m (31,000 ft) was selected on the basis of a trade off between minimizing block time and minimizing fuel consumption. The profile accounts for the 129 m/sec (250 knot) speed limit below 3048 m (10,000 ft) altitude and a cabin pressurization rate-limited descent. Fuel reserves include provision for a 185 km (100 NMi) diversion to an alternate destination plus 45 minutes hold at long-range cruise speed. Mission performance was calculated for standard day conditions.

3.2 AERODYNAMIC CHARACTERISTICS

The aerodynamic high lift characteristics used for the takeoff and landing calculations are based on Douglas wind tunnel data for the externally-blown-flap. These data were adjusted to reflect an increase in wing aspect ratio from 7 to 9 using the Elementary Vortex Distribution (EVD) powered-lifting surface theory. Trimmed lift and drag characteristics for the flap settings used for the baseline aircraft are shown in Figures 4 through 7. Figures 4 and 5 are for a takeoff flap angle of 20 degrees with all engines operating and with one outboard engine failed respectively. Figures 6 and 7 are for a landing flap angle of 48 degrees with DLC spoilers extended (used during landing approach) and with DLC spoilers retracted (used during flare maneuver), respectively. These data are modified by empirical equations to represent in-ground-effect data. These powered-lfit ground-effect equations are presented in Appendix B.2 of the NASA Short-Haul System Study (Reference 2).

The cruise drag characteristics have been estimated by the well-established Douglas drag prediction procedure for jet transport aircraft. The cruise drag consists of the zero-lift parasite drag and the drag-due-to-lift at Mach numbers below those at which compressiblity effects exist, plus the drag-due-to-compressibility. The zero-lift parasite drag and the drag-due-to-lift are evaluated at 0.5 Mach number, but at the Reynolds number corresponding to the design cruise points; in this way, the compressibility drag, which accounts for any drag increase at Mach numbers above 0.5, does not include a Reynolds number variation with Mach number. A breakdown of the estimated zero-lift parasite drag for the baseline aircraft is shown in Table I. The nacelle drag is discussed in Section 3.3-5. The trimmed cruise configuration estimated total drag characteristics (zero-lift parasite, lift dependent, and compressibility drag) are presented in Figure 8 for a range of lift coefficients and Mach numbers.

3.3 AIRCRAFT DESCRIPTION

The external characteristics of the baseline aircraft are shown in Figure 9. Four variable-pitch (QCSEE) engines of 81,400 N (18,300 lb) rated thrust are mounted on the 143 m^2 (1541 sq ft) wing. The aspect ratio 9 wing utilizes a supercritical airfoil section and has approximately 5 degrees of sweep at the quarter chord. This small amount of wing-sweep

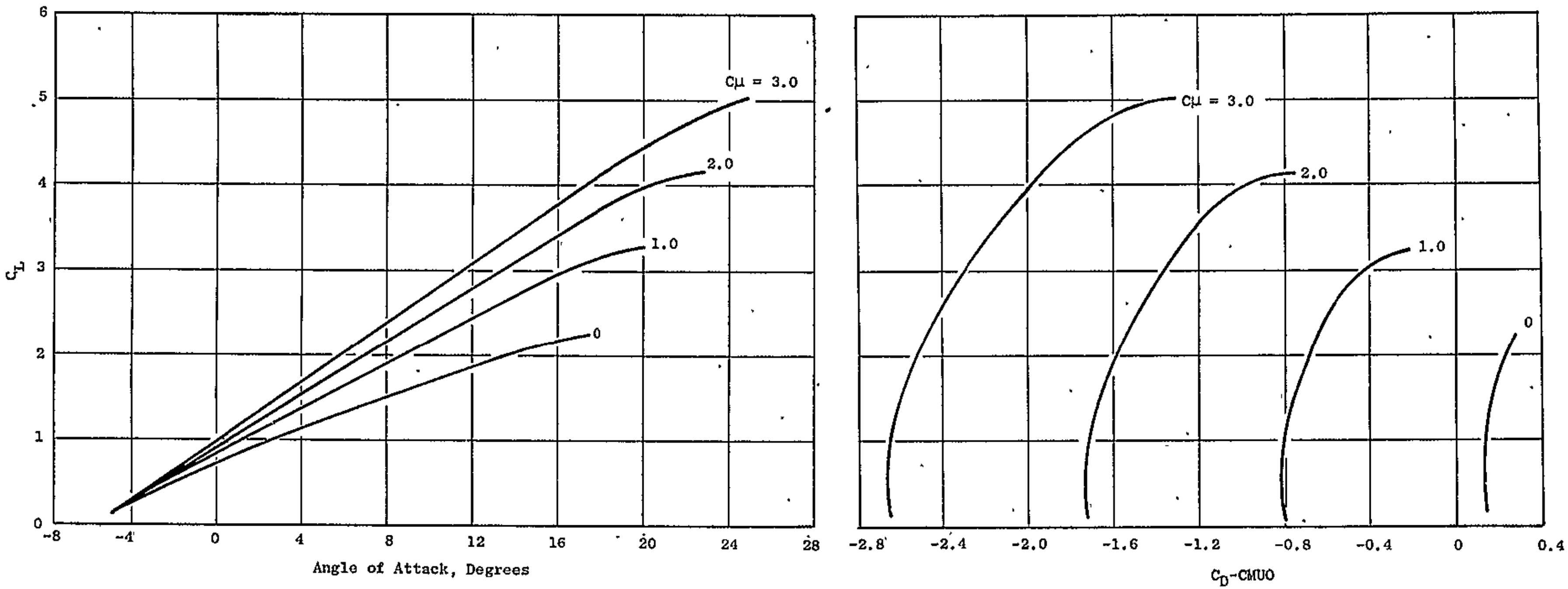


Figure 4. Aircraft Lift and Drag, T/O (All Engines).

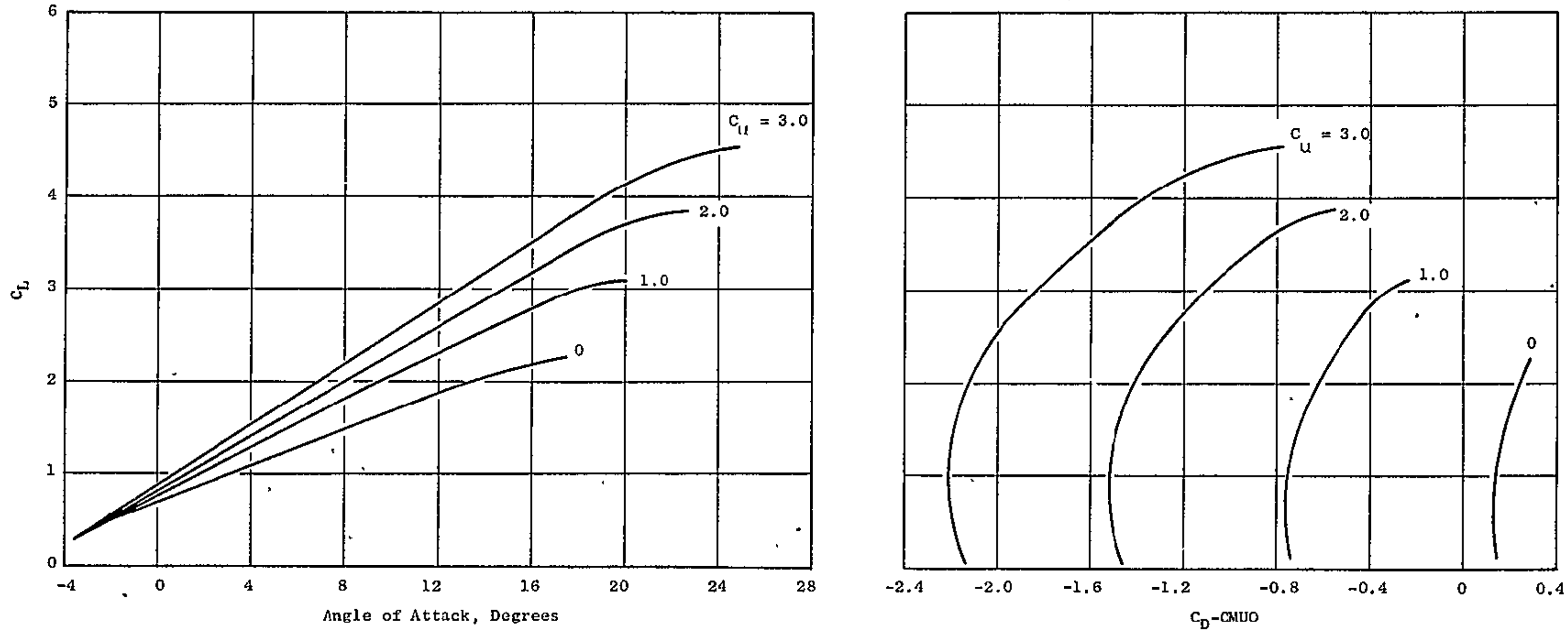


Figure 5. Aircraft Lift and Drag, T/O (One Engine Out).

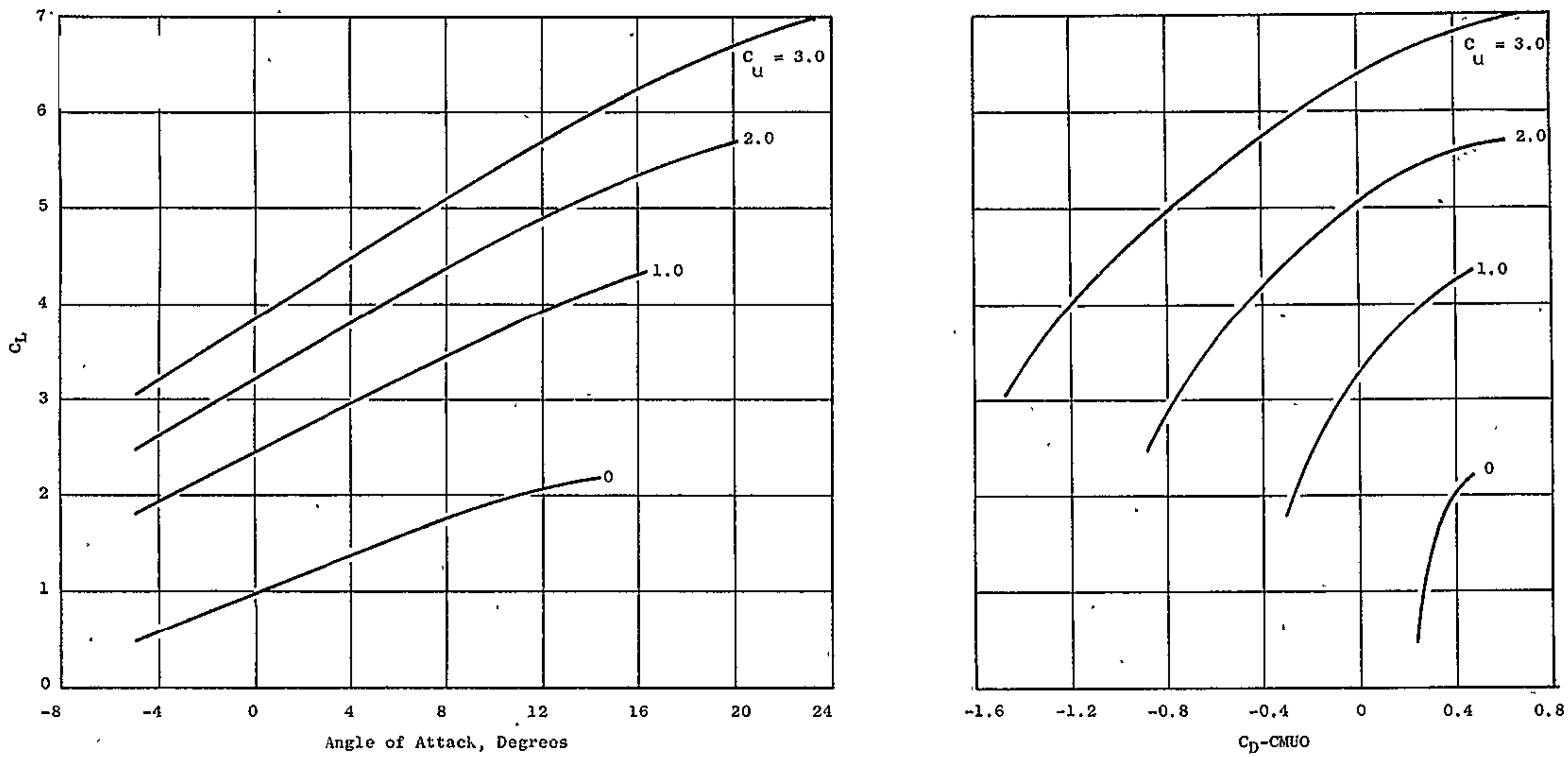


Figure 6. Aircraft Lift and Drag, Landing (All Engines - Spoilers Extended).

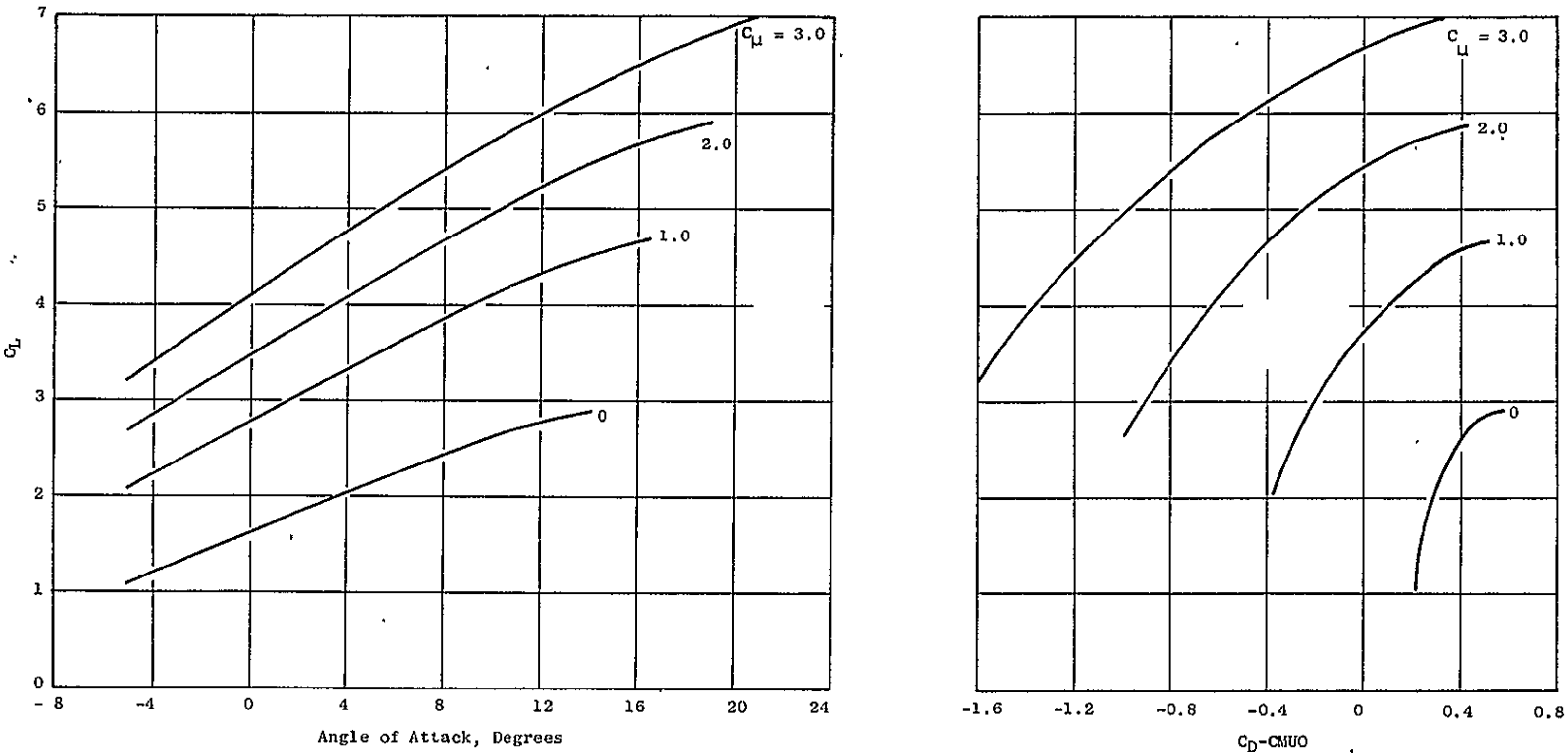


Figure 7. Aircraft Lift and Drag, Landing (All Engines - Spoilers Retracted).

Table I. Low Speed Drag Breakdown.

Component	Equivalent Parasite Drag Area	
	Sq m	Sq Ft
Fuselage		
Friction, Form, Roughness*	0.922	9.93
Aft - Fuselage Upsweep	0.124	1.34
Canopy	0.005	0.05
Gear Pods	0.159	1.71
Wing		
Friction, Form, Roughness	0.839	9.03
Flap Hinge Fairings	0.139	1.50
Horizontal Tail		
Friction, Form, Roughness	0.248	2.67
Vertical Tail		
Friction, Form, Roughness	0.164	1.77
Nacelles & Pylons		
Friction, Form, Roughness	0.405	4.36
Subtotal	3.006	32.36
Miscellaneous Items		
Excresences (7.1% of subtotal)	0.214	2.30
Airconditioning (0.7% of subtotal)	0.021	0.23
Control Surface Gaps	0.028	0.30
Total	3.269	35.19
Induced Drag Efficiency Factor	0.765	

*Wing, vertical tail and gear pod footprints removed from fuselage wetted area.

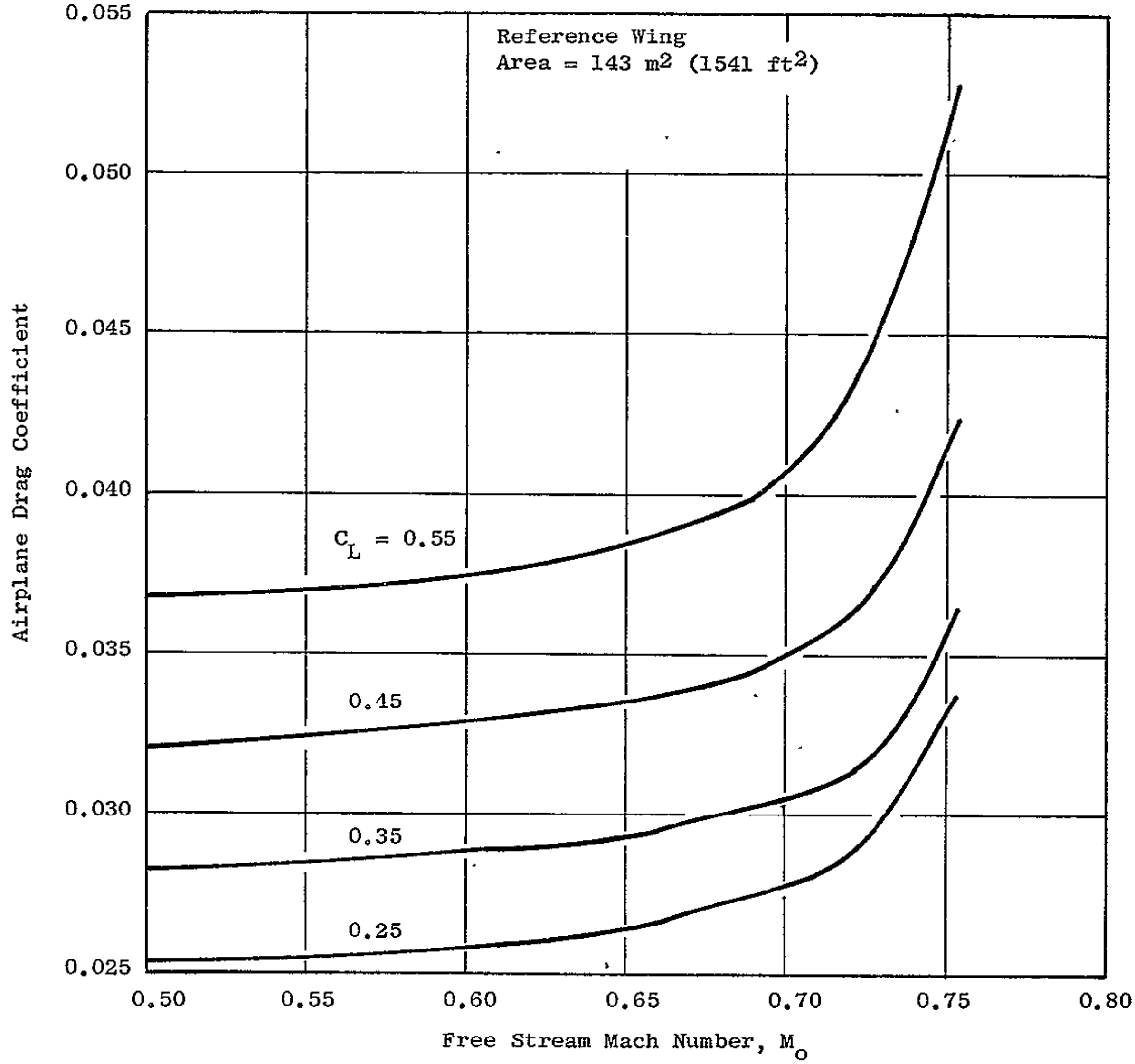


Figure 8. Drag, Cruise Configuration.

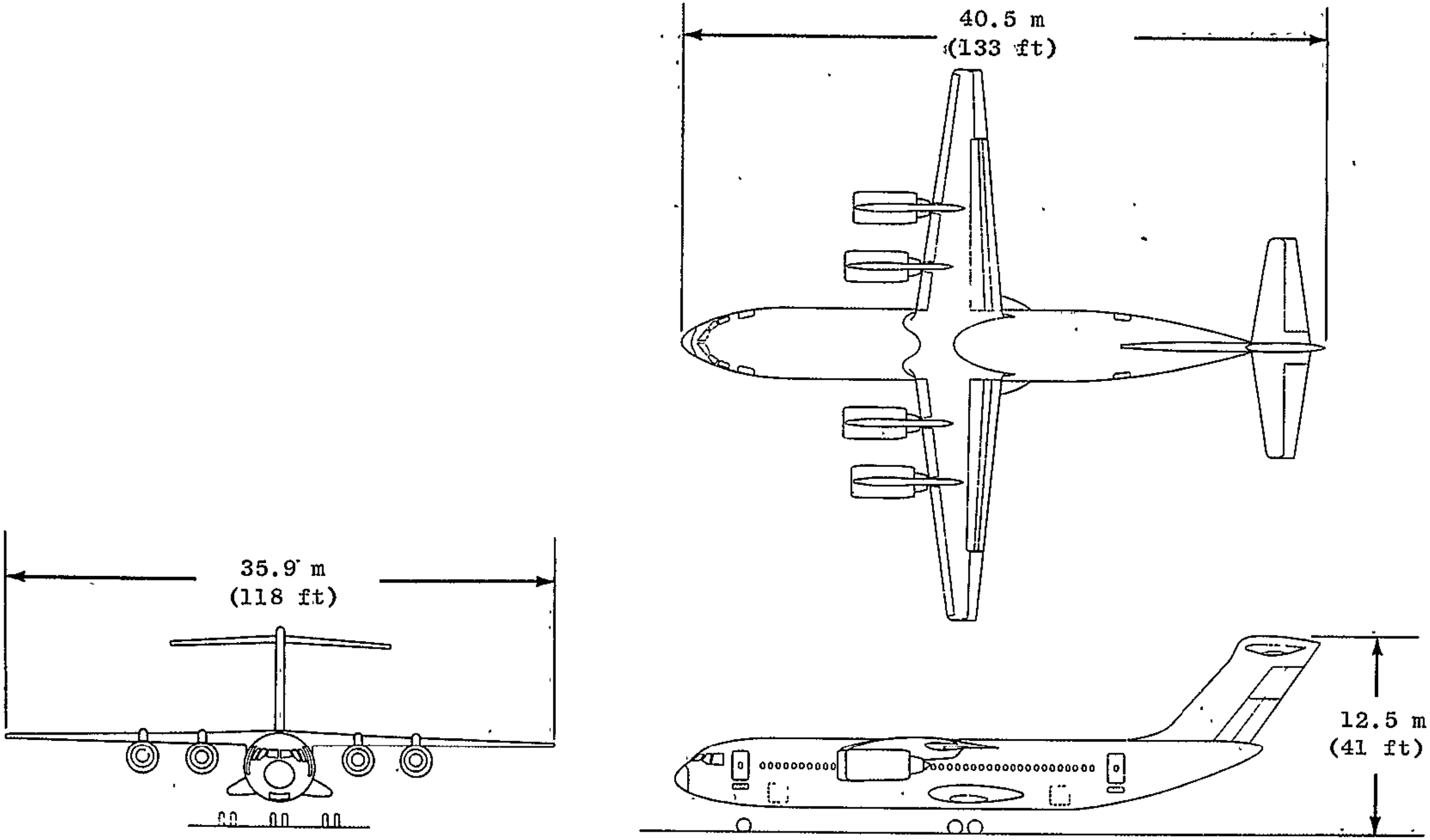


Figure 9. QCSEE Baseline EBF Aircraft.

limits cruise Mach number to about 0.74, which is ample for the short stage lengths on which the aircraft will be operated. A 35-percent chord, double-slotted flap in conjunction with full-span leading-edge devices, provide outstanding high-lift performance. Spoilers are used for direct lift control in the approach mode. The high-lift system does not require any engine bleed.

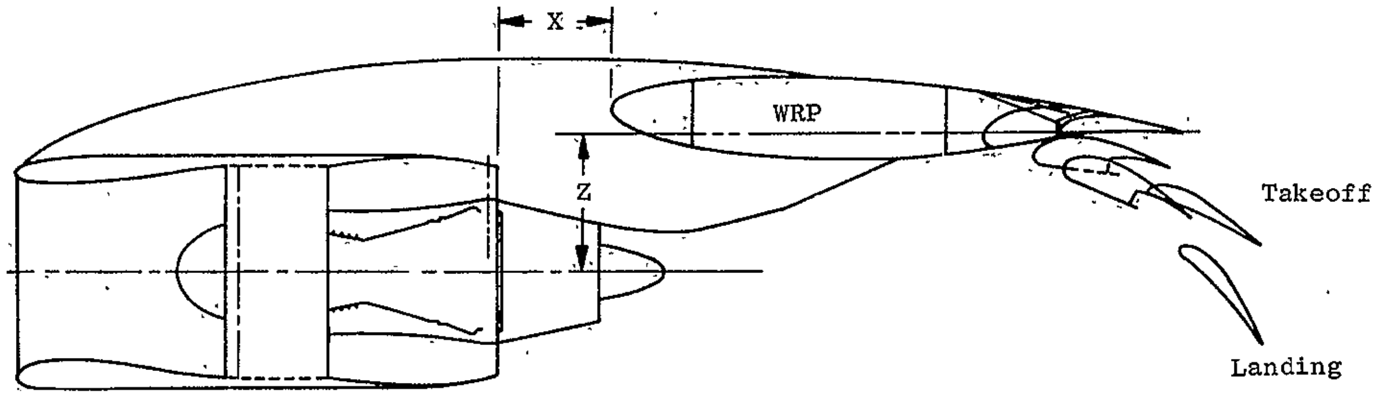
The engines are located well inboard to reduce engine-out asymmetric effects. The location of the outboard engine at 50 percent of the wing semispan allows sufficient spacing to avoid significant interference drag penalties. Figure 10 defines the location of the engine relative to the wing-flap at the inboard pylon station. The takeoff and landing flap positions and supercritical airfoil section are clearly visible. The placement of the engine relative to the wing leading edge and wing reference plane are based on extensive Douglas low and high speed wind tunnel test experience.

In order to reduce the dependence of the aircraft on ground support equipment, airstairs have been included in the design. They are located at the left side entrance doors at each end of the passenger compartment. More than 57 m^3 (2000 ft^3) of cargo volume is available in forward and aft under-floor compartments.

The 4.6 m (15 ft) diameter wide-body fuselage is designed to accomodate 162 passengers in a double aisle, six abreast, single-class seating arrangement with 86.4 cm (34 in) pitch. Seat size is similar to that in a DC-9, and approximately 2.5 cm (one inch) wider per passenger than DC-8 or B707/727/737 seats. An aisle width of 50.8 cm (20 in) was chosen, consistent with current wide-body airplanes, to minimize passenger loading and unloading times. The arrangement of the passenger compartment is shown in Figure 11. Overhead baggage compartments are like those in current wide-body aircraft, and coat racks are located near the exits. Due to the short stage-lengths over which the aircraft will be operated, there are no provisions for hot meal service galleys: however, buffet-coffee bars are located at each end of the passenger compartment.

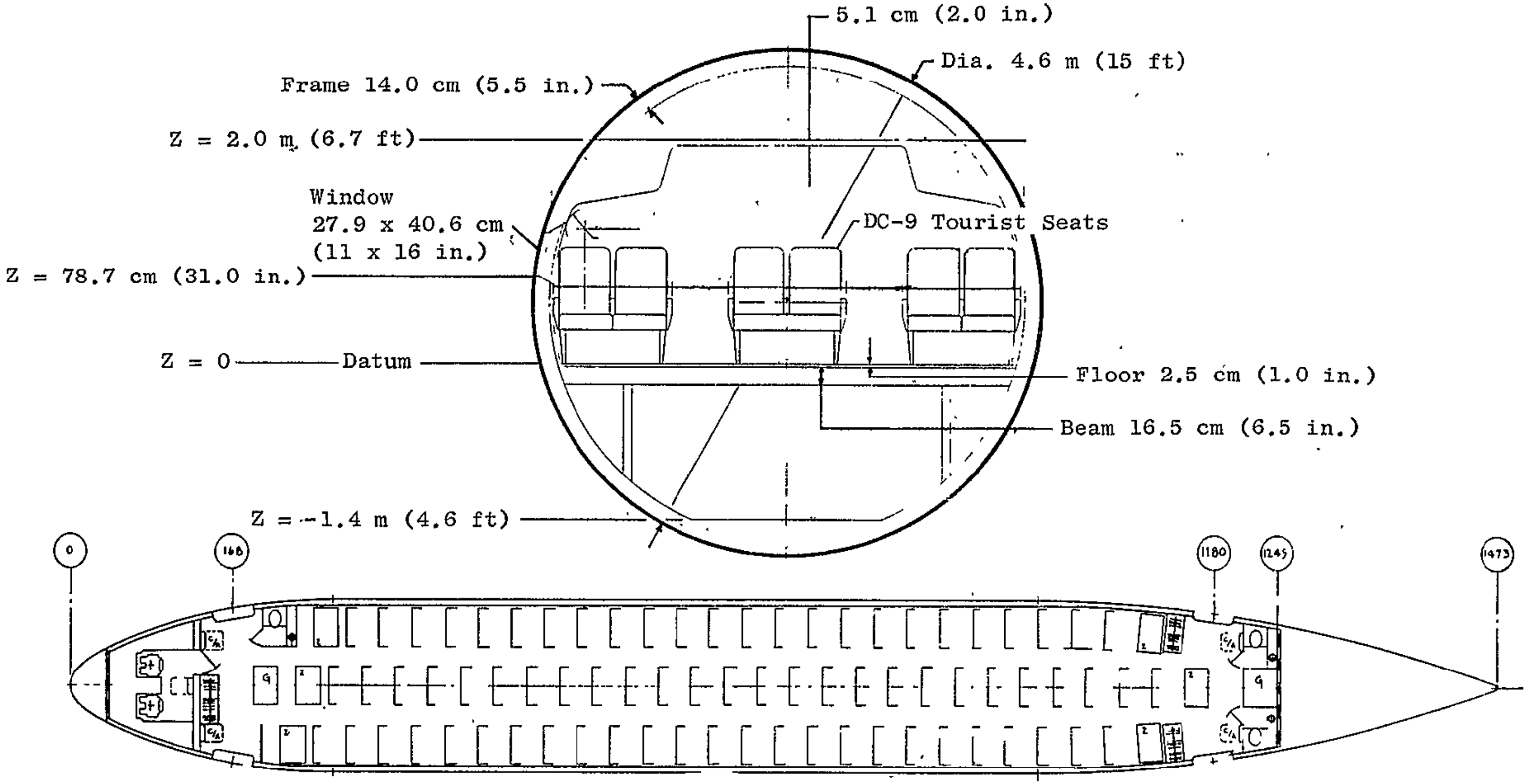
The cockpit area is arranged for operation with two crew members, but jump seats are provided for an optional third crewman and for an observer. The third crewman's seat is positioned midway between the pilots' seats behind the center pedestal so that he can reach all center pedestal controls including throttles and can monitor systems on the overhead panel. In addition, he could assist in check list procedures and provide a third pair of eyes for avoidance of mid-air collisions. The observer's seat position is suitable for observing all crew procedures as would be required by a check pilot.

The sizing chart for the baseline aircraft, shown in Figure 12, consists of plots of direct operating cost at 926 km (500 NMi), gross weight, rated thrust per engine, and uninstalled thrust-to-weight ratio (T/W)



- $X/C = 0.20$
- $Z/C = 0.24$
- No Engine Tilt Relative to WRP

Figure 10. Engine/Flap Relationship.



All - Tourist Class

6 Abreast (2 - 2 - 2) @ 86.4 cm (34 in.) Pitch = 162 Seats
All Seats are DC-9 Size, 106.7 cm (42 in.) Width Double

Aisle Width, 50.8 cm (20 in.)

Figure 11. 162-Passenger Interior.

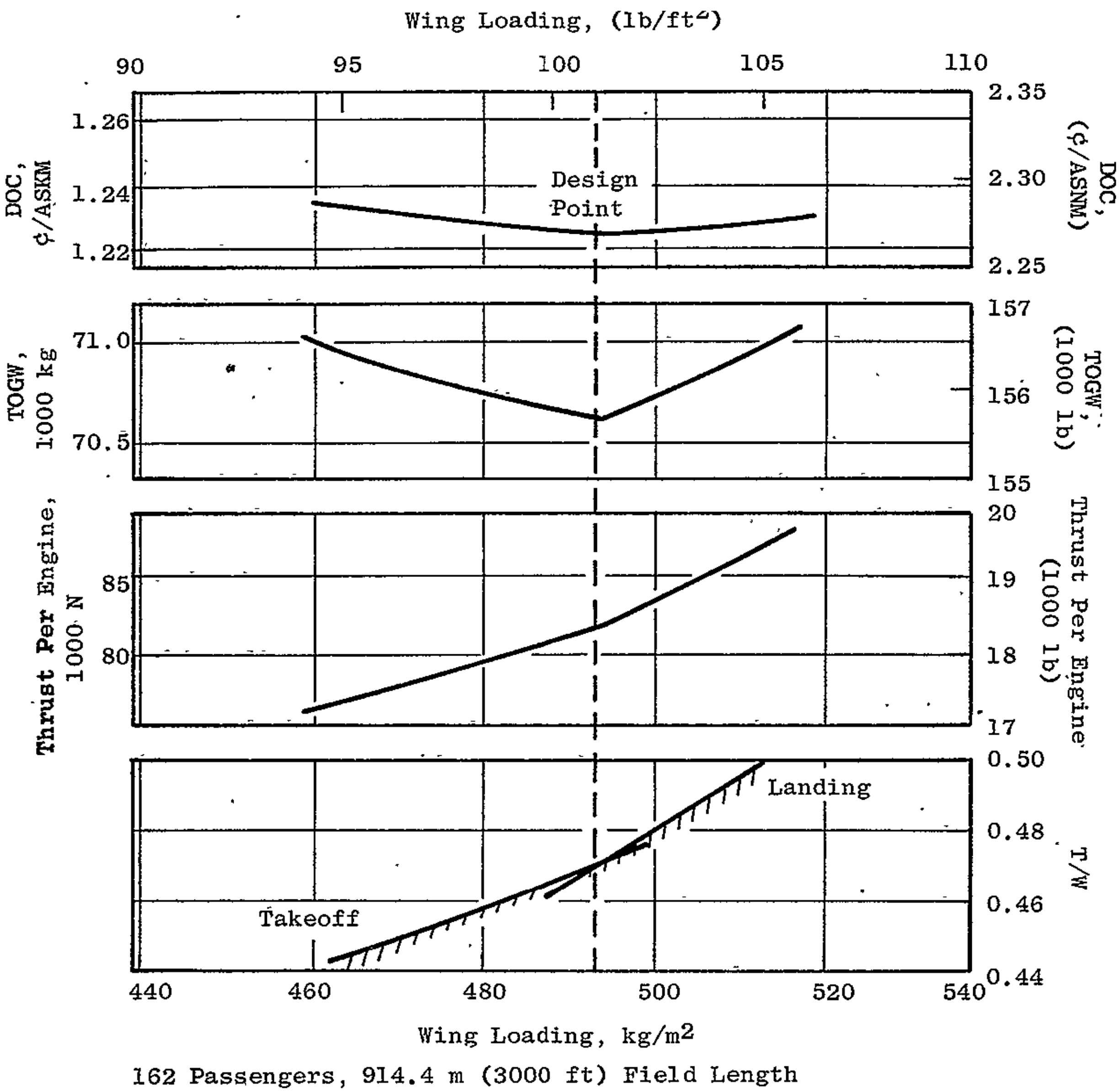


Figure 12. Aircraft Sizing.

as a function of wing loading (W/S). The plot of T/W vs W/S shows lines of constant 914.4 m (3000 ft) takeoff and 914.4 m (3000 ft) landing field length. The minimum DOC and gross weight occur at the W/S and T/W where the takeoff and landing field length are both equal to 914.4 m (3000 ft); W/S = 494 kg/m^2 (101.2 psf) and T/W = 0.471. The baseline aircraft was sized essentially at this point [W/S = 493 kg/m^2 (101.0 psf), T/W = 0.470] which results in a near-optimum design at an uninstalled thrust level of 81,400 N (18,300 lb) per engine. Direct operating cost is relatively insensitive to design wing loading. Changing wing loading by 24 kg/m^2 (5 psf) will change DOC by only 0.1 percent. Any engine size between 75,600 and 89,000 N (17,000 and 20,000 lb) could be used and still meet all design requirements without any significant increase in DOC. There is no measureable difference in DOC between sizing at a W/S of 494 and 493 ks/m^2 (101.2 and 101.0 psf).

A summary of the basic aircraft characteristics is shown in Table II. The primary aircraft design requirement is to carry 162 passengers, 926 km (500 NMi) from a 914.4 m (3000 ft) length field. The alternate mission is to carry the same payload 1389 km (750 NMi) from a longer runway. The 1389 km (750 NMi) mission requires the fuel load be increased by 2268 kg (5000 lb), which increases the required field length to 975 m (3200 ft). Both missions are flown at a cruise Mach number of 0.70 which represents a compromise between the high speed (M = 0.74) and long range (M = 0.62) cruise speeds.

3.4 WING FLAP ENVIRONMENT

The close coupling of the propulsion system and aircraft wing required for an EBF installation results in a higher temperature wing flap operating environment. Figures 13 through 15 depict the superposition of the QCSEE exhaust isotherms on the baseline airplane flaps. Shown are the environment the flaps are in for takeoff; for go-around and for thrust reversing. From these figures the summary presented in Table III is made of flap heating from the turbine exhaust stream.

For operation at these temperatures, Inconel 718 is proposed for the rear flap box beam section and for the leading and trailing edge sections covering approximately 58% of the flap span. Aluminum alloy is used for the balance of the rear flap. Titanium alloy is used for the forward flap box beam section and for the leading and trailing edge sections, also for approximately 58% of the flap span. Aluminum alloy is used for the balance of the flap.

This construction concept has been accounted for in estimating the weight of the baseline airplane.

Table II. Aircraft Characteristics.

Design Field Length	914 m	(3,000 ft)
Payload [162 PSGR at 91 kg (200 lb)]	14,700 kg	(32,400 lb)
Design Range	926 km	(500 N Mi)
Max Range with Design Payload	1,389 km	(750 N. M.)
Design Takeoff Gross Weight	70,620 kg	(155,700 lb)
Max Takeoff Gross Weight	72,890 kg	160,700 lb)
Max Landing Weight	70,620 kg	(155,700 lb)
Wing Area	143 m^2	(1,541 ft^2)
Rated Thrust Per Engine	81,400 n	(18,300 lb)
W/S	493 kg/m^2	(100 lb/ft^2)
T/W Uninstalled		0.47
Cruise Mach Number		0.70
DOC at 926 km (500 N Mi)	1.23 (¢/ASKm)	(2.27 ¢/ASNN)

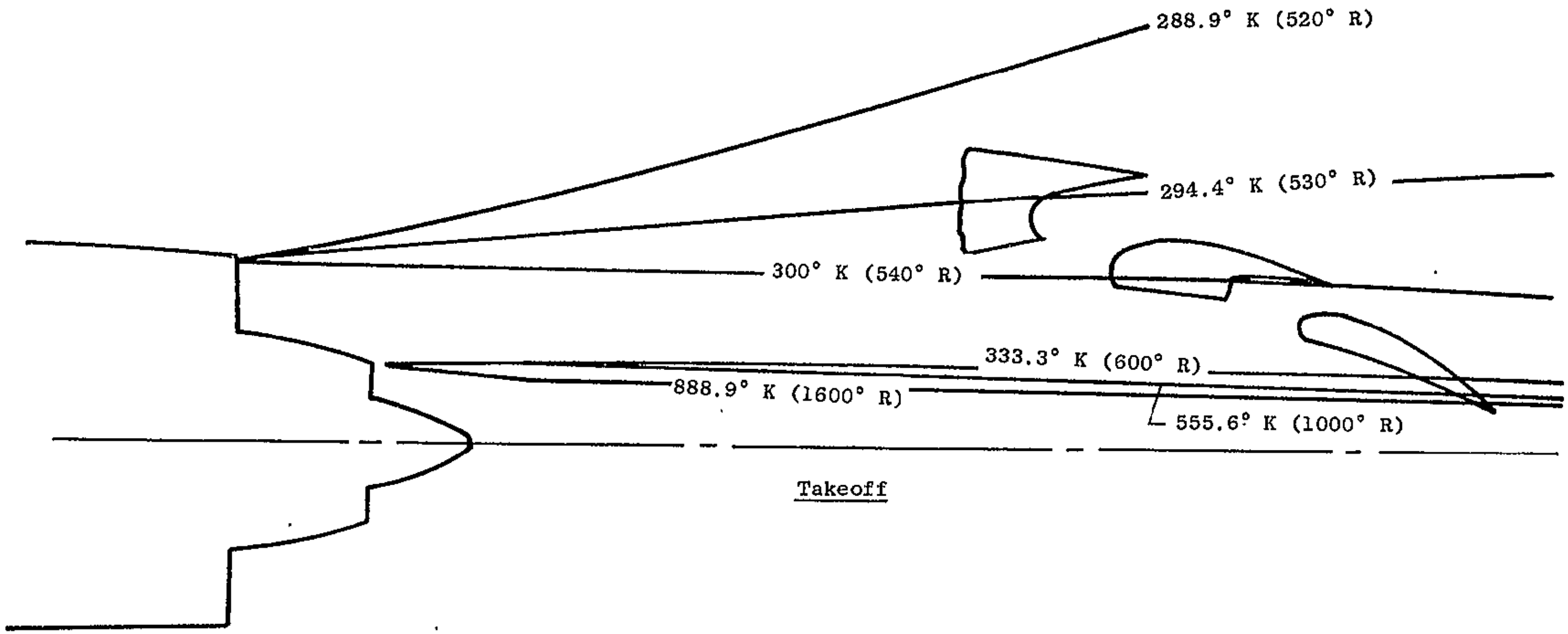


Figure 13. Flap Heating, Takeoff.

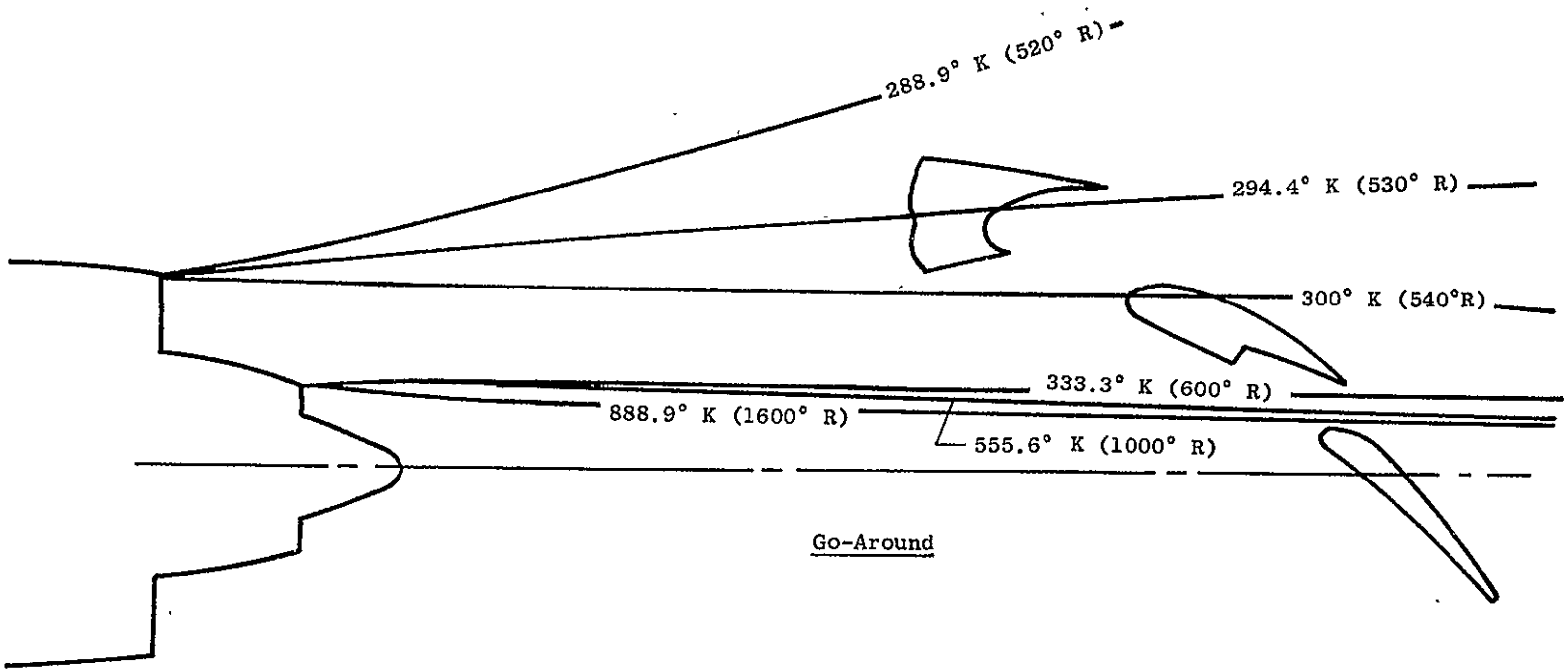


Figure 14. Flap Heating, Go-Around.

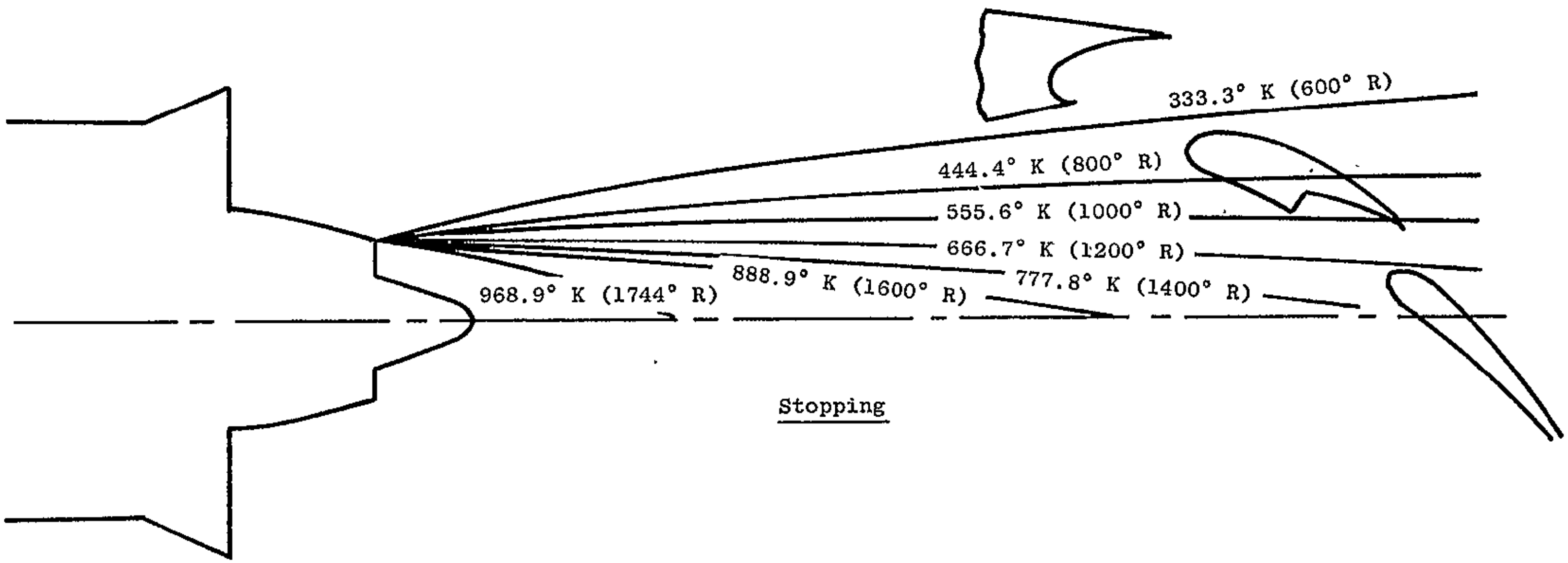


Figure 15. Flap Heating, Reverse Thrust.

3.5 AIRCRAFT WEIGHTS

The group weight statement for the airplane is shown in Table IV and a breakdown of propulsion system weights in Table V. Advanced construction techniques and materials are incorporated in the aircraft as follows:

- Composite Materials: Control surfaces and a limited amount of secondary structure.
- Advanced Metallic Structural Concepts: Integrally machined stiffeners, advanced alloys and honeycomb sandwich used in wing and tail boxes.
- Carbon Brakes.

Table III. Flap Peak Temperature.

Flap Member	Operating Condition	Flap Peak Temperature	
		° K	° F
Rear Flap	Go-Around-Flaps in Approach Configuration Takeoff Thrust	888.9	1140
Rear Flap	Takeoff - Flaps in TO Configuration Takeoff Thrust	838.9	1050
Rear Flap	Reversing - Flaps in Approach Configuration Max. Reverse Thrust	800	980
Forward Flap	Reversing - Flaps in Approach Configuration Max. Reverse Thrust	555.6	540

Table IV. Group Weight Statement.

	Weight	
	kg	lb
Wing	9,531	21,012
Tail	1,864	4,109
Fuselage	10,051	22,158
Landing Gear	2,815	6,206
Surface Controls	1,546	3,409
Propulsion	9,029	19,906
Auxiliary Power Plant	410	903
Instruments and Navigational Equipment	506	1,116
Hydraulic	562	1,240
Pneumatic	443	977
Eletrical	1,139	2,511
Electronics	758	1,672
Furnishings	6,439	14,196
Air Conditioning and Anti-Icing Equipment	855	1,886
Auxiliary Gear	14	30
Manufacturer's Weight	45,964	101,331
Operational Items	1,324	2,919
Operational Empty Weight	47,288	104,250
Payload	14,697	32,400
Fuel	8,641	19,050
Design Takeoff Gross Weight	70,626	155,700

Table V. Propulsion System Weight.

	Weight	
	kg	lb
Flight Design QCSEE Installation	479	1057
Less - Fire Detection and Extinguishing (Charged by Furnishings) Per MIL-STD-1374)	-25	-56
Less - Bleed and Anti-Ice (Charged to Air Conditioning & Anti-Icing Per MIL-STD-1374)	-13	-28
Net Installation	441	973
Additional Engine Systems$^{(1)}$	28	62
Pylon	298	658
Airplane Accessores Gear Box, PTO Shaft, Supports	20	45
Single Propulsion System Less Engine And Fuel System	788	1738
Engine	1384	3051
Single Propulsion System Less Fuel System	2712	4789
Total Propulsion System Less Fuel System	8689	19,156
Fuel System	340	750
Total Propulsion System	9029	19,906

$^{(1)}$Engine controls, auto throttles, cooling system, start system, and ignition system (portions of cooling, start, and ignition systems are included in GE engine and nacelle).

4.0 PROPULSION SYSTEM/AIRPLANE INTEGRATION

4.1 REQUIREMENTS

The baseline short-haul airplane concept described in this report is based on a propulsion system designed to meet the following requirements. These requirements are based on QCSEE Program objectives as modified by DAC to meet their projected aircraft needs.

4.1.1 Takeoff Thrust

The engine should provide takeoff thrust of 81,400 N (18,300 lb) (uninstalled) flat rated to 32.2° C (90° F) at sea level.

4.1.2 Thrust-to-Weight Ratio

The static sea level thrust-to-weight design goals for the EBF uninstalled engine and the total propulsion system (including the engine and nacelle but without aircraft accessories and pylon) are as follows:

	N/kg	lb/lb
Uninstalled Engine	60.8	6.2/1
Propulsion System	42.2	4.3/1

4.1.3 Idle Thrust

Ground idle thrust should not exceed 3492 N (785 lb) on installed basis (4.5% of TO thrust) to limit the maximum ratio of total-idle-thrust-to-airplane-weight to 0.03 to ensure satisfactory airplane ground handling qualities. Idle thrust values closer to 3% of TO thrust, providing an upper limit of thrust-to-airplane-weight of 0.02, are preferred.

If the above ground handling limitations cause in-flight compressor discharge bleed air pressures to be lower than required for air conditioning, namely 13.8 N/cm^2 (20 psi) gage pressure, it will be necessary to provide a "flight idle" limit, independent of the "ground idle" limit, that satisfies the air conditioning requirements.

4.1.4 Reverse Thrust

Reverse thrust equivalent to 35% of TO thrust is required to enable the baseline airplane to stop on the runway without brakes. Capability to back the airplane with reverse thrust is also desired.

4.1.5 Cruise Envelope

The engine should have capability to cruise at pressure altitudes to 9450 m (31,000 ft) at Mach numbers up to 0.74.

4.1.6 Inlet Distortion

The system shall be stall free and otherwise satisfactory when operating at inlet angles of attack (α_i) up to and including 44° with approach M_o of 0.18 [61.7 m/sec (120 knots)] and up to 51° at M_o of 0.12 [41.2 m/sec (80 knots)] and with cross winds at 90° to inlet axis of up to 18 m/sec (35 knots).* This enables the airplane to perform the flight demonstrations needed to certificate the type for airline service.

4.1.7 Thrust Response

The system shall be capable of reaching 95% TO thrust from flight idle thrust within 5.0 seconds. The system shall be capable of reaching TO thrust from approach thrust (0.65 TO thrust) within 1.0 second for purposes of executing a go-around maneuver with the airplane.

The system shall be capable of reaching maximum reverse thrust from approach thrust within 1.5 seconds for purposes of arresting the landing roll on a 914.4 m (3000/ft) long runway without assist from the wheel brakes.

4.1.8 Emissions

The propulsion system shall incorporate the best available exhaust pollution reduction technology to comply with QCSEE Program emission goals.

4.1.9 Durability

The system durability shall be consistent with short-haul operations wherein the utilization rates are 3000 flight hours per year and there are 700 - 1000 takeoff and reverse cycles per 1000 flight hours. Propul-

*Differences from Experimental Engine Requirements are explained in Section 3.3.1 on pages 73 through 76.

sion system condition indicators shall be consistent with the needs of performing maintenance on an on-condition basis rather than on a calendar or cumulative flight hour basis. This combined with confidence in component reliability will permit maintenance costs per flight hour to be guaranteed.

4.1.10 Mounting

The propulsion system needs to be suited to mounting below a pylon beam that cantilvers it below and ahead of the wing such that the fan nozzle is ahead of the leading edge and low enough that the fan stream just grazes the underside of the wing.

Provisions for attaching thrust reacting links at the core/fan frame interface at approximately the 10 and 2 o'clock positions and for inserting a side and vertical load reacting pin at the 12 o'clock position are desired to accommodate the whiffle-tree mounting concept. This concept is preferred because it equalizes the reactions at the engine attach points and minimizes the bending moment imposed on the engine casing.

Provisions are desired for attaching three links to the top section of the turbine case, as is typical of current practice in turbine engine installations, to react the side and vertical forces.

The engine mount attachments and the related carry-through structure must be consistent with the airplane design concept of providing load paths that will handle the limit flight forces in the event any single element of the mount system fails.

4.1.11 Lube Oil System

The lube oil capacity should be compatible with utilizing the engines at least 10 hours per day without need to service the tank. Provisions are needed for indicating oil quantity to the flight crew, and for filling the lube oil tank from a remote location. The heat load from engine rotor bearings, accessory drive power train, and fan drive reduction gear should be transferred to the fuel system.

4.1.12 Bleed Air Requirements

Engine bleed air requirements are established by the Environmental Control and Ice Protection System and by the Engine Starting System. Flow rate, pressure, temperature, and cleanliness/purity of the bleed air are significant parameters.

4.1.12.1 Bleed Flow Requirements

The air flows presented in Table VI will satisfy aircraft requirements. The bleed flows actually extracted from the engines will depend on the bleed system controls, described in Section 3.2.17, and on the operating conditions of the airplane and engine. For the purposes of this report, it has been assumed that engine extractions essentially correspond to aircraft requirements as shown in Table VI.

Table VI. Aircraft Bleed Flow Requirements.

1.	Minimum cabin flow to provide pressurization:	0.68 kg/sec	(1.5 lb/sec)
2.	Desired cabin flow for normal air conditioning:	2.04 kg/sec	(4.5 lb/sec)
3.	Engine inlet ice protection flow, per engine:	0.27 kg/sec	(0.6 lb/sec)
4.	Left wing ice protection flow:	0.41 kg/sec	(0.9 lb/sec)
5.	Right wing ice protection flow:	0.41 kg/sec	(0.9 lb/sec)
6.	Tail ice protection flow:	0.68 kg/sec	(1.5 lb/sec)

4.1.12.2 Bleed Pressure Requirements

The minimum gauge pressure at which rated cabin pressurization and air conditioning can be provided is approximately 13.8 N/cm^2 (20 psi).

It should be possible to deliver the above pressure from the interstage bleed port (for economy reasons) during normal power cruise, and from the compressor discharge bleed port during idle descent without impacting on crew workload by requiring engine power adjustments.

4.1.12.3 Bleed Temperature Requirements

The minimum temperature at which ice protection can be provided at the bleed flow rates given in Paragraph 3.1.12.1 is 204.4° C (400° F). The aircraft is not provided with any means of increasing the temperature of the bleed air extracted from the engines.

The maximum allowable temperature for air which is to be routed through the aircraft is 232.2° C (450° F). It is recognized that engine bleed temperature will sometimes exceed this value. The aircraft bleed system will therefore include a heat exchanger in each pylon to limit the maximum air temperature, as described in Paragraph 3.2.17.

4.1.12.4 Bleed Air Cleanliness/Purity Requirements

Airborne Dirt: The inherent, centrifugal particle-separating characteristics of the engine fan and compressor should be exploited to the greatest practical extent so as to minimize the dirt content of bleed air.

Engine Generated Impurities: Bleed air shall be free of engine-generated impurities during normal operation.

Contamination Produced by Engine Failure: The inherent, centrifugal particle-separating characteristics of the engine compressor should be exploited to the greatest practical extent so as to minimize the contamination of bleed air in case oil enters the compressor.

4.1.13 Airplane Accessories Drive

The propulsion system should provide a shaft power takeoff (PTO) drive suitable for driving an airplane accessories pack mounted atop the pylon beam aft of the fan frame.

The airplane concept includes an accessories pack composed of a variable-displacement hydraulic pump having a maximum displacement of from 36 to 49 cm^3 (2.2 to 3.0 cubic inch) per revolution, a 60 KVA rated integrated drive generator (IDG), and an air turbine starter. These are all mounted on a gearbox designed to provide the appropriate reduction gear ratios between the input drive and the accessory drives.

A power takeoff drive shaft connects the input drive on the accessory gearbox to the power takeoff pad on the engine. The latter will be the interface between the quick engine change unit (QECU) and the airplane.

The shaft power demands for a typical mission profile are shown in Table VII.

The power takeoff drive train would have to be designed to accommodate the overloads related to fault clearing in the electric power distribution systems as well as the starter cranking loads.

4.1.14 Nacelle Ventilation and Cooling

Sufficient airflow should be provided through the core engine zone and the accessories section for purging the space of flammable vapors that might be present as the consequence of fuel or lube system component failure, and for cooling selected components, i.e., the lube oil tank. Airflow rates that provide for 6-10 air changes per minute in the affected spaces are typical of current practice and are estimated to be appropriate for this concept.

Table VII. Shaft Power Extraction.

Operating Mode	Load on PTO Drive[1]		Load on PTO Drive[2]	
	kw	HP	kw	HP
Idling at Terminal	35.0	47	3.7	5
Taxiing to TO Position	58.2 - 71.6	78 - 96	15.7 - 29.1	21 - 39
Takeoff Run	54.4 - 70.8	73 - 95	29.8 - 46.2	40 - 62
Climb	39.5 - 47.7	53 - 64	15.7 - 23.9	21 - 32
Cruise[3]	38.0 - 46.2	51 - 62	15.7 - 23.9	21 - 32
Descent	40.3 - 48.5	54 - 65	15.7 - 23.9	21 - 32
Holding	40.3 - 48.5	54 - 65	15.7 - 23.9	21 - 32
Landing Approach	43.3 - 66.4	58 - 89	15.7 - 38.8	21 - 52
Go-Around	39.5 - 66.4	53 - 89	15.7 - 38.8	21 - 52
Landing Roll Out	76.8	103	50.7	68

[1]The two driving on-line IDGs

[2]The two driving hyd pumps and standby IDG

[3]0.7 M, 9450 m (31,000 ft)

For the core zone, it is envisioned that air from the fan discharge duct would be introduced through inlets near the front of the core cowl, sweep the length of the compressor, combustor and turbine casings, and discharge to the atmosphere from exits on the core cowl afterbody, downstream of the fan nozzle.

For the accessories zone, a ram scoop or scoops would in stream air at the front of the zone, directing the flow at the engine lube oil tank and other selected accessory components. The air would flow rearward through the engine and airplane accessories sections, and exhaust to the atmosphere through an exit in the pylon upper fairing near the wing leading edge.

4.1.15 Fire Protection

To implement the baseline airplane fire protection concept, the propulsion system must embody certain features to prevent the outbreak of fire and to contain an outbreak, in the event the prevention system is defeated, until the suppression system is brought into play.

The required features are, in addition to the flammable vapor purging function noted in discussion related to ventilation and cooling, the enclosing of all the fuel-carrying tubes in the core zone with fireproof shrouds that are vented to the atmosphere and fireproof construction at the boundaries of the core zone and accessories section.

The shrouded tube concept serves to provide double-wall containment of the fuel. The vent feature affords a practical means of verifying the fuel-tight integrity of the primary (inner) tube and also of safely disposing of leakage from the primary, should that occur. The vent exit should be positioned to assure that leakage cannot possibly enter the engine air intake passages even, for instance, in the reverse pitch mode where the fan nozzle is in the flare position, functioning as the air inlet.

The fireproof construction is vital to the containment of a fire until the flight crew has become aware of the situation and taken action to isolate the affected zone by shutting off the various supplies of flammable fluids (fuel and hydraulic oil) and to suppress the fire by discharging the fire extinguisher system.

4.1.16 Drains

Provisions are required to facilitate collecting leakage that may develop with the accumulation of operating time, from accessory shaft seals, etc., and from various fluid leaks and spills that may occur in the accessory section as a result of component failure or maintenance actions, so that they can be safely discharged to the atmosphere. It is planned that this would be accomplished by routing the drain to an exit mast protruding from the trailing edge of the pylon fairing where emissions would either fall clear of any part of the pod or be swept away in the turbine exhaust stream.

Provisions are needed to prevent fluid that spills into the accessories section from either leaking to the core zone, where it could represent a potential fire hazard or running down the sides of the pod where it would detract from the appearance of the airplane and also constitute a maintenance burden. This will be accomplished by providing a "gutter" at the interface between the accessory cover/fairing and the fan casing or pod outer diameter to contain a spill until it drains away through the drain system that exits through the pylon trailing edge or through the fan frame vanes to the fan casing or pod outer diameter at the bottom center lines.

4.1.17 Safety

The propulsion system should be capable of accommodating large compressed air leaks into the core engine space, as would be the case if a bleed duct ruptured or a duct joint separated, without sustaining overpressure distress to the core cowl. This accommodation could be made by providing hinged blow-out doors on the core cowl afterbody (downstream of the fan nozzle) that limit the pressure rise in the core zone. The airplane design concept affords protection to the pylon by means of a presure limiting blow-out in the pylon upper fairing.

4.1.18 Inlet Loads

The provisions for attaching the inlet duct to the fan frame must satisfy all of these loading conditions:

- Axial load stemming from the pressure distribution on diffuser, lip, and exterior cowl surfaces. The loads prevailing when the fan is operating in reverse pitch must be assessed, as well as those occurring in the normal forward thrust mode.
- The moments, shears, and axial loads related to the pressure distributions and inertia forces acting on the inlet during airplane maneuvers, gust encounters, and high sink-rate landings.
- The inertia forces associated with the forced vibration accompanying the loss of a group of adjacent fan blades at high fan rotor speeds.

Design criteria that identify a significant portion of the above design requirements are discussed in detail in Section 3.3 Nacelle Aerodynamics.

4.1.19 Maintainability

Since the time required to replace engine components constitutes an important portion of the propulsion system maintenance load, the engine configuration should address minimizing this burden, thus reducing this component of airplane DOC.

The propulsion system should therefore be designed to meet the following goals:

- The engine should be capable of being trimmed on a test stand and then not require additional trimming when installed on an aircraft.
- Access for inspection (including borescoping operations) and adjustments shall be accomplished without disturbing components or systems.
- Quick-opening doors shall be provided for access to the core to expedite turnaround and through-flight servicing actions.
- Borescope ports for inspecting the internal condition of the compressor, combustor, and turbines shall be suited to the performance of this task while the engine is in place on the wing.
- The engine shall be compatible with other nondestructive inspection/test techniques such as X-ray, zyglo, and radioisotope.
- The engine design should accomodate the forces related to maintenance personnel grasping and/or standing on those parts of the engine which could serve as hand holds or steps.
- Accessories and components requiring routing servicing actions must be accessible without need to remove or disassemble unrelated parts or systems.
- Fan blades must be replaceable and pitch change mechanism serviceable without removal of the inlet duct. Fan duct surfaces must be durable enough to be compatible with meeting this goal.
- The tools required for line maintenance at way stations are limited to those reasonably expected to be part of an aircraft mechanics' kit.
- Engine hoist and ground handling equipment attach points must be accessible for the use intended without disturbance to the demountable engine assembly.

- The engine assembly must be replaceable (on the wing) without disturbing the rigging of the throttle and fuel shutoff controls in the airplane.

For guidance in establishing goals, the accepted replacement times for selected (significant to QCSEE concept) engine components in current generation turbofan installations are noted below.

The time represents elapsed time needed to replace the designated component (see Table VIII), while the engine is in place on the aircraft exclusive of the time required to gain access to the component (positioning ladders or work stands, gathering tools, obtaining the needed replacement part, opening cowls and access doors, the performing of functional testing, etc.) The tasks are performed by no more than three men having appropriate skills of average skill level.

Table VIII. Component Replacement Time.

Subsystem	Component	Minutes
Pneumatic	Intermediate bleed check valve	12
	CDP bleed control valve	15
	Inlet ice protection valve	10
Power Plant	Inlet duct	45
	Fan cowl half	20
	Core cowl half	20
Engine	Fan spinner	15
	Single fan blade	75
	Fan rotor assembly	250
	Combustor	65
	H.P. turbine	210
	L.P. turbine	210
	Fuel pump	60
	Fuel control	50
Exhaust	Core nozzle	80
	Nozzle plug	15
Lube Oil	Engine oil tank	20
	Lube pump - pressure	35
	Lube pump - scavenge	35
	Fuel/oil heat exchanger	25

On the basis of experience with current turbofan-powered airliners, it appears reasonable to use 240 minutes as a goal for the replacement time for the complete baseline engine change unit by a five-man crew.

4.1.20 Engine Control System

An engine control system based on an engine-dedicated, full-authority digital electronic controller which used digital data paths to communicate with the engine and with an aircraft thrust control system, air data computers, and engine-mounted systems, is required for each engine. Each engine control system will provide selected automatic operating modes in response to thrust level and mode selection commands from the aircraft thrust control system. In addition, each engine control system will provide engine status data for crew display, recording, and onboard diagnostic purposes, over the entire airplane operating envelope. This engine status data will include selected failure indications and corrective action advisories generated within the engine control system.

4.2 INSTALLATION CONCEPT

Figure 16 shows the orientation of the propulsion system with airplane wing needed to integrate the two to achieve a high performance externally blown flap (EBF) powered lift system. The relationship of the double-slotted wing flap is shown for the cruise (fully retracted) and the takeoff and landing approach (maximum extension) operating conditions. Figure 17 depicts the propulsion system installation concept.

4.2.1 Inlet Duct

The pod features a long, axisymmetric inlet duct with a relatively small throat section designed to induce a high inlet Mach number which serves to suppress the radiation of noise forward from the engine. The inlet highlight to throat diameter ratio of 1.21 adequately accommodates the expected inlet angle of attack that will be encountered in airline operations. The low wing loading of the baseline airplane coupled with the inlet location far-ahead of the wing, results in such modest upwash angles that there is no need to incorporate droop in the inlet. The leading edge section is all metal and is fitted with a hot air ice protection system.

The inlet duct construction is a composite sandwich composed of Kevlar face sheets, epoxy bonded to aluminum alloy flexcore honeycomb.

4.2.2 Fan Module

The fan module is composed of a single-stage fan assembly featuring 18 variable-pitch, wide-chord fan blades of all composite construction, supported in an all-composite fan frame which includes 33 integral stator vanes; a planetary reduction gear to drive the fan at part power turbine shaft speed; the fan thrust bearing; the fan pitch change mechanism

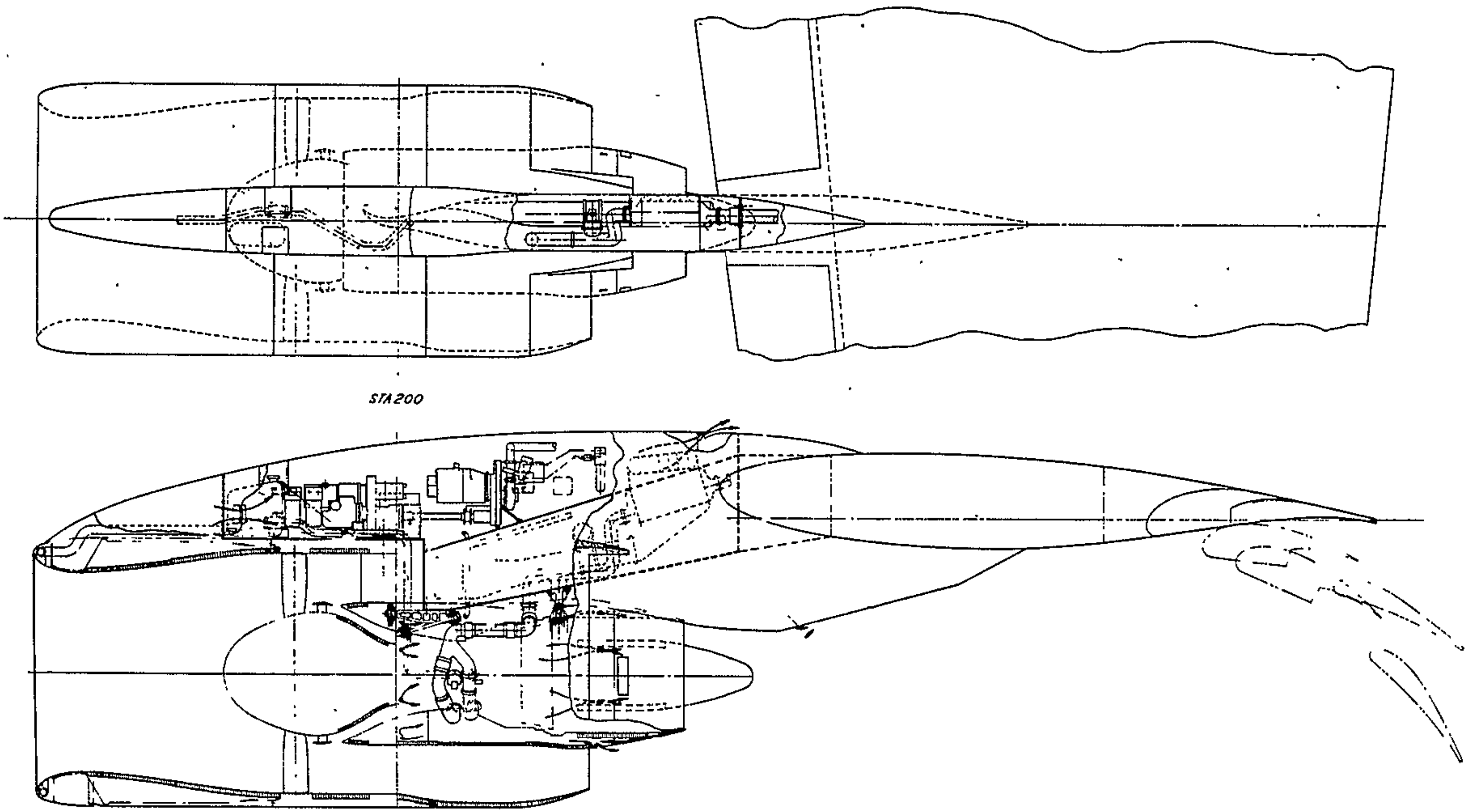


Figure 16. Baseline Propulsion Unit.

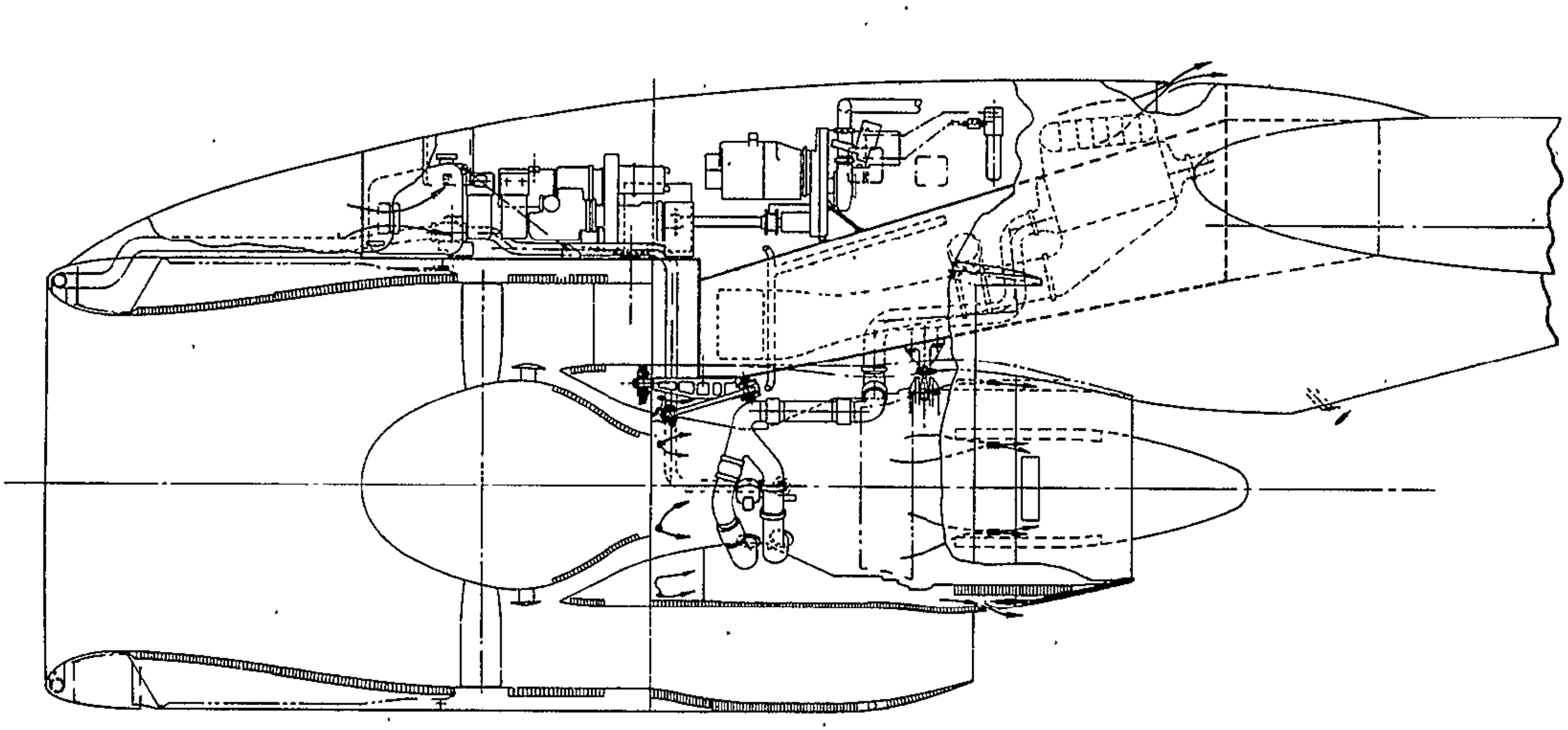


Figure 17. Baseline Installation Concept.

enclosed by the fan spinner; and the power takeoff shaft (tower shaft) which drives the engine accessories located atop the fan frame. The tower shaft is carried in the 12 o'clock position fan stator, which serves also as the leading edge section of the portion of the pylon that lies within the fan passage.

A belt of Kevlar felt is embedded in the fan frame on the exterior of the fan tip treatment to serve as a containment ring for the fan.

No provisions are made to heat the spinner or fan stators for ice protection. Acoustic treatment is incorporated in the passages downstream of the fan including the compressor inlet. A socket designed to react side and vertical forces is incorporated at the rear of the fan frame (directly back of the tower shaft) for engine mounting purposes.

The forward end of the fan frame is fitted with 16 rotary latches that are used to attach the inlet duct.

4.2.3 Gas Generator

The gas generator or core engine module is composed of a nine-stage, axial flow compressor featuring variable-geometry inlet guide vanes and stators on the first three stages; a very short, high heat release, annular combustor and a single-stage, air-cooled drive turbine.

Lugs are provided at the 10 and 2 o'clock positions at the fan frame interface for attaching thrust reacting links for mounting the engine. Provisions are incorporated in the turbine frame to attach links to react the vertical and side loads and rolling moments for engine mounting purposes. Ports are provided on the compressor casing for bleeding air from the 5th stage and from the compressor discharge.

4.2.4 Power Turbine

The low pressure element of the engine's turbine system drives the fan. The two-stage uncooled turbine is directly coupled to the sun gear in the planetary reduction gear assembly, and in the rear, it is supported by the turbine frame.

The reduction gear system effectively isolates the axial loads in the fan from those in the power turbine. As a result, the thrust load on the fan cannot offset the axial load on the power turbine and must be reacted solely by the fan thrust bearing. To compensate for the absence of the offsetting thrust from the fan, the power turbine is fitted with an air balance piston. This system, located at the rear of the turbine, utilizes air from the compressor discharge to apply a forward thrust force on the low pressure turbine rotor, offsetting the normal rearward load. This minimizes the thrust bearing load and permits the use of a smaller bearing. The gases exiting the power turbine are exhausted to the atmosphere through a fixed-area, plug-type nozzle.

4.2.5 Fan Exhaust

The fan exhaust passage is a modified annulus formed by the core engine cowl and the fan duct cowl panels. The pylon effectively partitions the upper half of the annulus into separate ducts and is used to provide a secondary support for both the fan cowl/duct and core cowl. Hinges are used to attach the cowls to the pylon and latches are fitted to join the cowl halves which are designed to butt at the bottom. This arrangement provides quick and convenient access to the core engine. The core cowl afterbody, which is an integral part of the turbine exhaust duct, serves as located and provides support for the rear of the core cowl panels. The leading edge of the cowl is supported in a tongue and groove joint on the fan frame.

The fan duct/cowl is similarly supported at the frame. These cowl halves are fitted with hinged trailing edge flaps to provide capability to vary the nozzle area and to flare the nozzle to make it function as the engine inlet when the fan is operated in reverse pitch. Oil-operated actuators are built into the cowl assemblies to position the nozzle flaps. Since neither the core cowl nor the fan duct cowl panels would normally need to be removed from the pylon when an engine change is made, the concept envisions that the airplane hydraulic pump (located on the airplane accessories pack atop the pylon) supplies the power needed to operate the nozzle flaps. This avoids disturbing the nozzle hydraulic system during engine changes.

4.2.6 Engine Accessories

The engine accessories are grouped atop the fan frame and are enclosed by a nonstructural, streamlined shape that constitutes an extension of the pylon beam upper fairing.

Included in the accessory group are the various shaft-driven accessories (fuel pump and hydromechanical fuel control, lube oil pump, "hydraulic" pump for the fan pitch change system, ignition generator) fitted to the accessory gearbox, which in addition features a power-takeoff drive for a separately mounted airplane accessories pack; the lube oil tank; fuel and lube oil filter modules; and the engine digital control. The bundle of fuel and oil tubes and wires comprising the umbilical needed to link the accessories with the gas generator is carried across the fan passage in a cavity at the rear of the top fan stator.

4.2.7 Engine Support Concept

The engine is attached to the underside of a box beam cantilivered from the pylon stub, which is designed to distribute the pylon reactions to the wing box and also to provide an interface that accommodates the use of the same pylon beam for all four engine positions. The stub concept effectively overcomes the variations in geometry caused by wing sweep and taper. This feature obviously enhances the producibility of the airplane and is expected to have significant economic impact. The box beam construction provides multiple paths for loads so that a fail-safe structure is assured. The bottom spar and the bulkhead at the pylon stub interface are burnthrough resistant to provide fire barriers between the core and accessories zones and the wing. The mounting concept shown on Figure 18 is predicated on achieving a fail-safe system that accommodates moving the engine in a vertical direction only in getting it on and off the wing. Also, this mounting does not require the installation of fasteners at the mount/pylon interface that are loaded in shear under static conditions.

Additionally, it is the objective to minimize the bending moments induced in the engine casing to limit the stresses and the case deflections, and thus to minimize the compressor blade tip clearance allowance needed to accommodate the bending deflection.

The whiffletree configuration of the engine front mount satisfies these objectives. It is composed of a two-piece central assembly designed to engage a socket in the fan frame and transfer the engine reaction to the toe of the pylon beam, which is set back from the frame in diference to the need to accommodate the space envelope of the engine umbilical. Vertically oriented bolts attach the fitting at the pylon interface. The thrust reaction is shared by a pair of links attached to lugs at the 10 and 2 o'clock positions on the fan frame/core engine interface and to the ends of a beam that is joined by a pivot pin through its middle to aforementioned fitting. This arrangement serves to equalize the reactions on the engine frame, and to keep the thrust load line close to the engine axis so that the bending moment imposed on the engine case is minimized. Implicit in the fail-safe concept is the capacity of the fan frame socket to transfer the thrust reaction in the event of the loss of the structural integrity of an element of the whiffletree. This mount arrangement also provides the space envelope requirements of the compressor air bleed system.

The rear mount is composed of a two-piece fitting and three links designed to attach the turbine frame to the underside of the pylon in a fashion that reacts side and vertical forces on the engine and the related moments. Accommodation is made for engine thermal expansion. As with the front mount, vertically oriented bolts are used to make the attachment at the pylon interface. Laminated links provide a fail-safe design in that a fracture in one element is unlikely to propagate across the interface.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

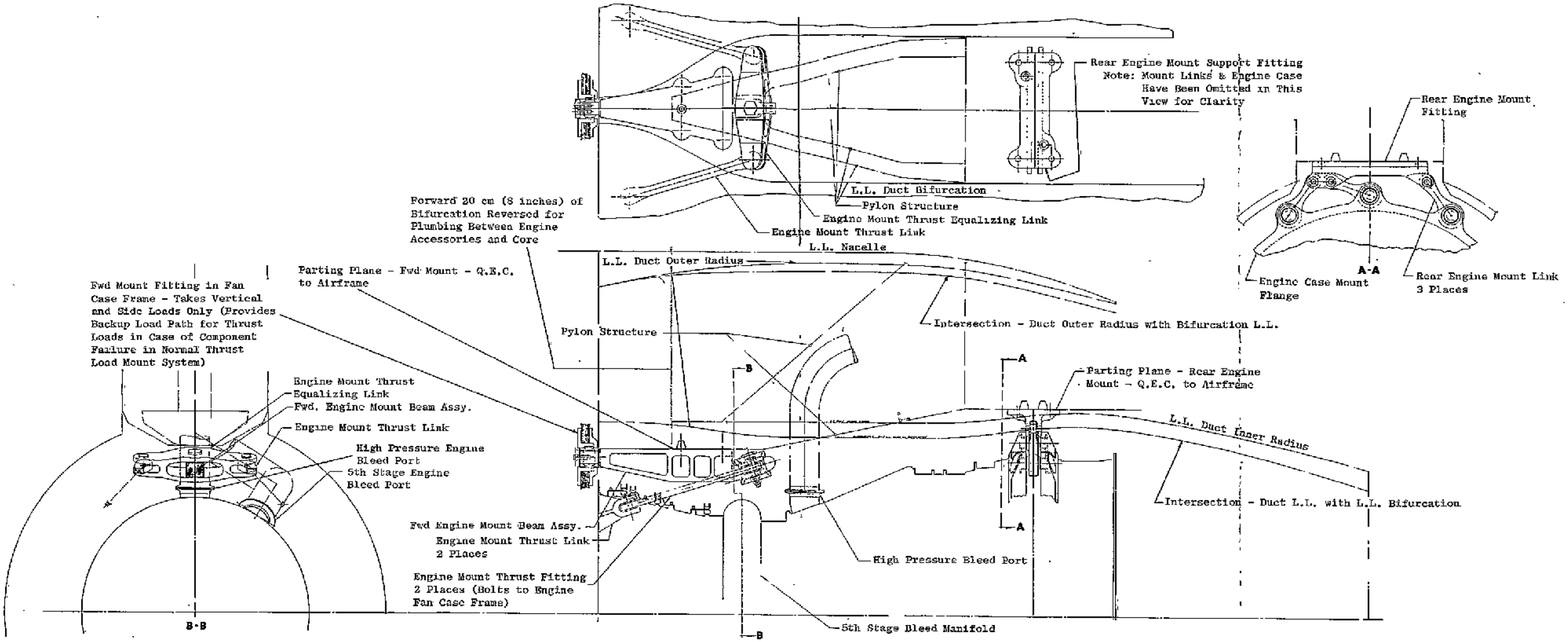


Figure 18. Engine Mounts.

4.2.8 Pylon Beam Auxiliary Functions

In addition to primary function of supporting the engine pod and reacting the propulsive and reverse thrust, the pylon beam also:

- House the air bleed precooler subsystem
- Supports the airplane accessories pack
- Supports the fire detector systems that survey the core zone and accessories section.
- Supports the fire extinguisher agent distributor tube and outlets that direct the agent to the core zone and accessories section.

4.2.9 Fire Protection

The concept of fire prevention - containment - suppression is realized by:

- Encapsulating all the fuel tubes in the core zone with steel shrouds, and by ventilating the core zone and accessories sections to purge them of flammable vapors.
- Providing fireproof enclosures for the core zone and accessories section to prevent the spread of fire beyond the space of origin.
- Providing reliable fire detectors to alert the flight crew to the existence of a fire.
- Providing a fire extinguishing system capable of discharging two shots of fire suppressing agent to selected spaces.

4.2.10 Fire Detectors

Continuous element fire detectors are mounted on the pylon beam to survey the core zone and accessories section. An element mounted on the pylon upper spar surveys the accessories section. A separate detector element mounted on the pylon lower spar surveys the thermal enviroment in the core zone. This arrangement avoids the service troubles and attendant reliability degradation that have been experienced with detector installations mounted directly on engines or on nacelle panels where the need to traverse hinged joints, exposure to engine vibration and flexing, and the presence of disconnects that must be separated for engine removals have proved to be troublesome.

4.2.11 Fuel Isolation

The fire containment concept is backed up by providing the capability to cut off the supply of fuel to the fire zone. This is done by providing the flight crew with the controls needed to shut off the supply of fuel and hydraulic oil to the affected space.

4.2.12 Fire Extinguisher

The fire extinguisher system for the propulsion system is composed of four fire extinguisher agent (bromotrifluoromethane) containers and the related agent distributor tubes and cockpit controls. Two agent containers are located in the wing leading edge between each pair of engines. Each of the containers is fitted with two squib-operated discharge valves. The distribution system for each pair of containers is so configured that either or both containers can be discharged to either of the adjacent engines. The distributor arrangement causes both the core and accessories spaces to be simultaneously flooded with extinguishing agent.

4.2.13 Airplane Accessories Pack

The shaft driven airplane accessories are not carried on the engine, as is typical of contemporary turbine engine installations, but rather are carried, along with the air turbine starter, on a separate accessory gearbox located atop the pylon beam. The concept shown in Figure 19 envisions a gearbox designed to carry the air turbine starter power turbine module; a 60 KVA integrated drive generator (IDG); a variable-displacement hydraulic pump and incorporates a fuel/oil cooler that carries away the heat generated by gear meshing and bearings in the gearbox and in the oil-cooled IDG.

The gearbox concept features oil-lubricated splines for the accessory drives and integration of the reduction gear needed to adapt the starter power turbine output to the starting requirements so that only the air turbine module portion of the starter is exterior to the gearbox.

The lubricating oil (MIL-1-7808 or MIL-L-23699 oil would be used as in the engine) system for the gearbox and IDG would be isolated from the engine lube system to avoid the possibility of a failure in one system resulting in contamination of the other. Filters would be fitted to trap contaminates to prevent clogging critical oil passages or scoring bearings or gears.

The IDG, composed of a hydraulic drive and an oil-cooled, high speed generator, represents a highly weight efficient power conversion module. A shaft disconnect that can be operated by the flight crew provides the capability to decouple the IDG from the accessory drive to prevent its destruction when sensors indicate impending failure. A quick attach-detach

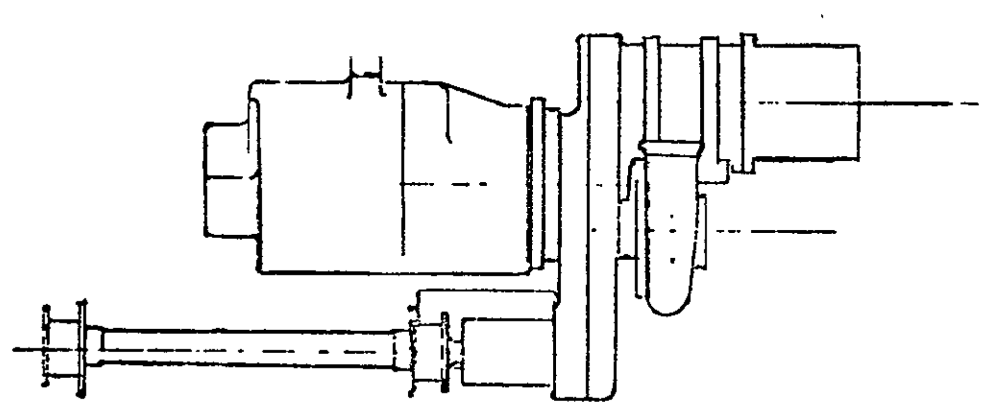


Figure 19. Airplane Accessories Pack.

mount (QAD) is provided to accommodate easy and rapaid replacement of the unit. The hydraulic pump is composed of two modules: a manifold module, which is a static assembly attaching to the accessory gearbox pad and to which the pump pressure, suction and case drain lines are connected; a plug-in pump module containing the pistons, cylinder block, wobble plate, and drive, and featuring an electric depressurizing valve (EDV) which enables the pump to be zero stroked for unloading the engine for starting, or for deactivating when failure is impending, to avoid contaminating the system. The plug-in concept enables the pumping module to be removed without disturbing the hydraulic plumbing. Self-closing valves in the manifold prevent the loss of fluid or the entry of air, thus purging and topping-off activities are not required when pump changes are made unless contaminants are present in the affected system. The pump displacement can be varied from zero up to a maximum of from 36 - 49 cm^3 (2.2 to 3.0 cubic inches) per revolution. A 3.0 CPR pump of this type is now in production for a military aircraft application.

4.2.14 Accessories Pack Oil Cooling

Fuel bypassed from the QCSEE fuel control is used as the heat sink for the heat generated in the accessories pack. The cooling conept is depicted schematically in Figure 20, which shows the fuel flow from the fuel tanks to the engine, through the heat exchangers, and back to the fuel tanks. Fuel from the wing tanks is delivered to the engine positive displacement pump. From the pump, fuel at high pressure flows to the engine lube oil heat exchanger and also to a diverter valve (shown in Section 4.5 Figure 88) in the engine fuel control. After passing through the engine heat exchanger, which cools the oil for the engine lube system and fan reduction gearbox; the fuel is routed to the fuel control where it flows to the metering valve and to the diverter valve. Fuel passing through the metering valve is consumed by the engine combustion cycle. Fuel passing through the diverter valve flows to the control bypass valve and then back to the wing tanks. The diverter valve receives fuel from both upstream and downstream of the engine lube oil heat exchanger. The flow split from these two sources is a function of fuel pressure level (pump discharge) and consequently a function of engine thrust level. At low pressure level corresponding to ground idle and high pressure level corresponding to takeoff and climb, the diverter valve permits fuel flow to the wing only from upstream of the engine heat exchanger, hence there is no engine lube heat returned to the wing tanks at these conditions. At intermediate pressure levels corresponding to cruise and descent, a regulated amount of fuel from downstream of the engine heat exchanger is permitted to enter the diverter valve where it mixes with upstream fuel and then flows to the bypass valve and back to the wing tank. The diverter valve permits a portion of the engine lube oil heat to be returned to the wing tank during those conditions where the metered (engine combustor) fuel heat sink is insufficient to provide the necessary engine oil cooling.

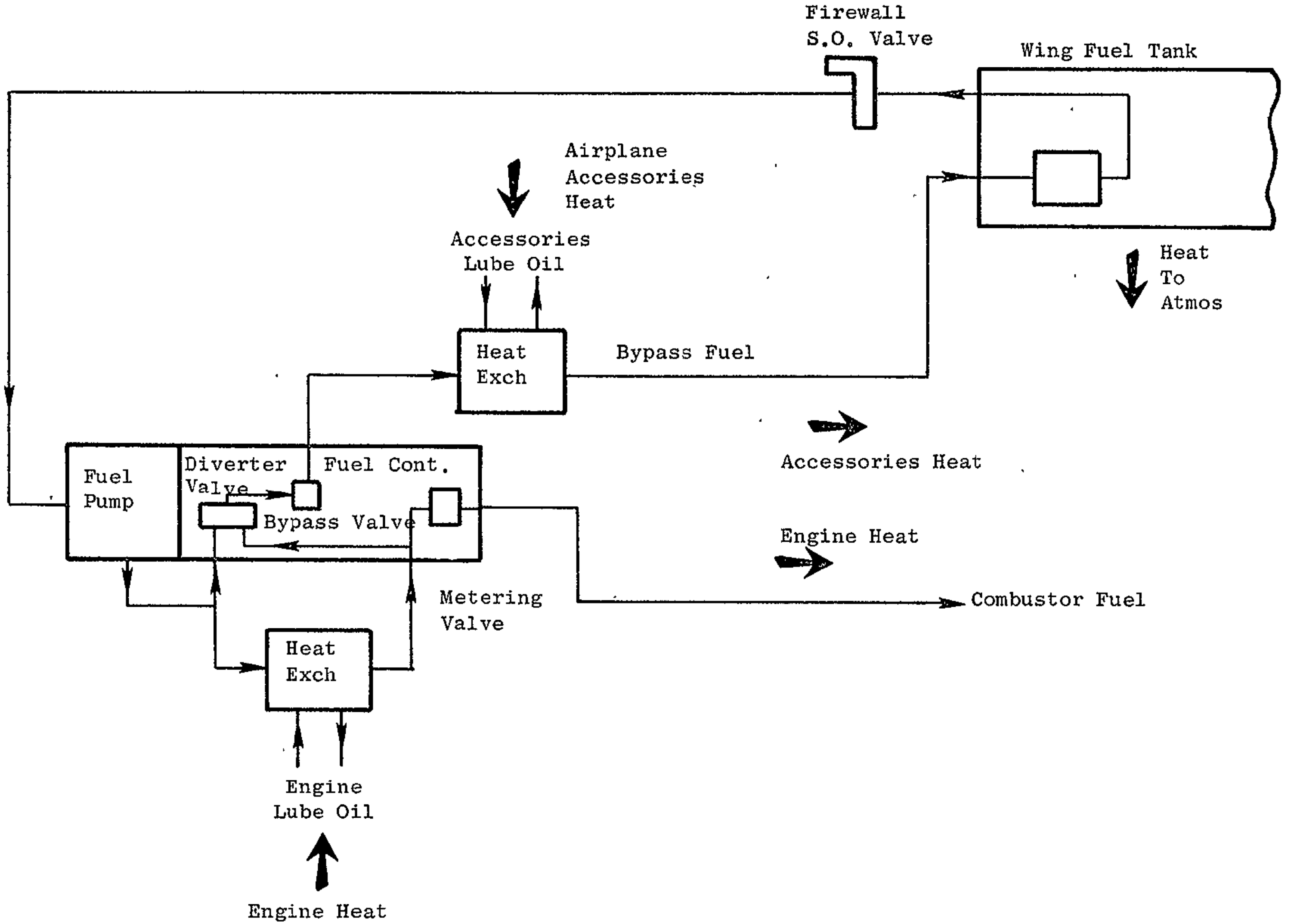


Figure 20. Airplane Accessories Cooling.

The heat exchangers for the aircraft accessories pack receive fuel from the engine control bypass which is the quantity of flow represented by the difference between fuel flow to the engine pump and fuel flow to the engine combustor. It is estimated, on the basis of current predictions of engine and aircraft heat loads, that the bypass fuel is of sufficient quantity and low enough temperature to adequately cool the accessory pack. Further, and most importantly, it is also estimated that the rate of heat addition to the wing tank, including both engine and aircraft heat input at extremes of fuel temperature, is less than the rate of heat loss from the wing tanks to the atmosphere. These considerations and assumptions include the effect of hot fuel in the tanks [up to 50° C (122° F)], low fuel reserves (10 percent), and both ground static and in-flight heat transfer from the wing surfaces during hot atmospheric conditions. The interference of the accessory pack cooling system and the aircraft fuel system is shown in Appendix C.

4.2.15 Airplane Accessory Pack

The accessories pack concept is addressed to meeting the needs of the baseline airplane's electric and hydraulic power subsystems where the electric system is envisioned as composed of two isolated channels supplied by separate engine driven generators and the hydraulic system of four isolated systems supplied by separate variable-displacement hydraulic pumps.

Generators are fitted to three of the four accessories pack gearboxes. Two of the three are sharing the airplane electric power load and the third is on standby, available to pick up the load if one of the on-line generators fails. A fourth generator is fitted to an APU to provide electric power when the airplane is parked with the propulsion engines shut down. Figures 21 through 25 and Tables IX through XIV provide the space envelope, weights, and performance requirements applicable to the accessories pack components.

4.2.16 Bleed Air System

Historically the flow requirements of the Environmental Control and Ice Protection System have equalled or exceeded those of the Engine Starting System. No change in this relationship is foreseen for a QCSEE-powered short-haul airplane. Since the demands of these two systems are not concurrent, the following discussion is limited to Environmental Control and Ice Protection System.

An effort has been made to configure the Environmental Control and Ice Protection System to be compatible with current QCSEE engine characteristics without impacting unreasonably on aircraft performance, or requiring technology advancements. The resulting baseline system is shown in Figure 26. The system is designed to provide cabin pressurization, air conditioning, engine inlet ice protection, wing ice protection, and tail ice protection.

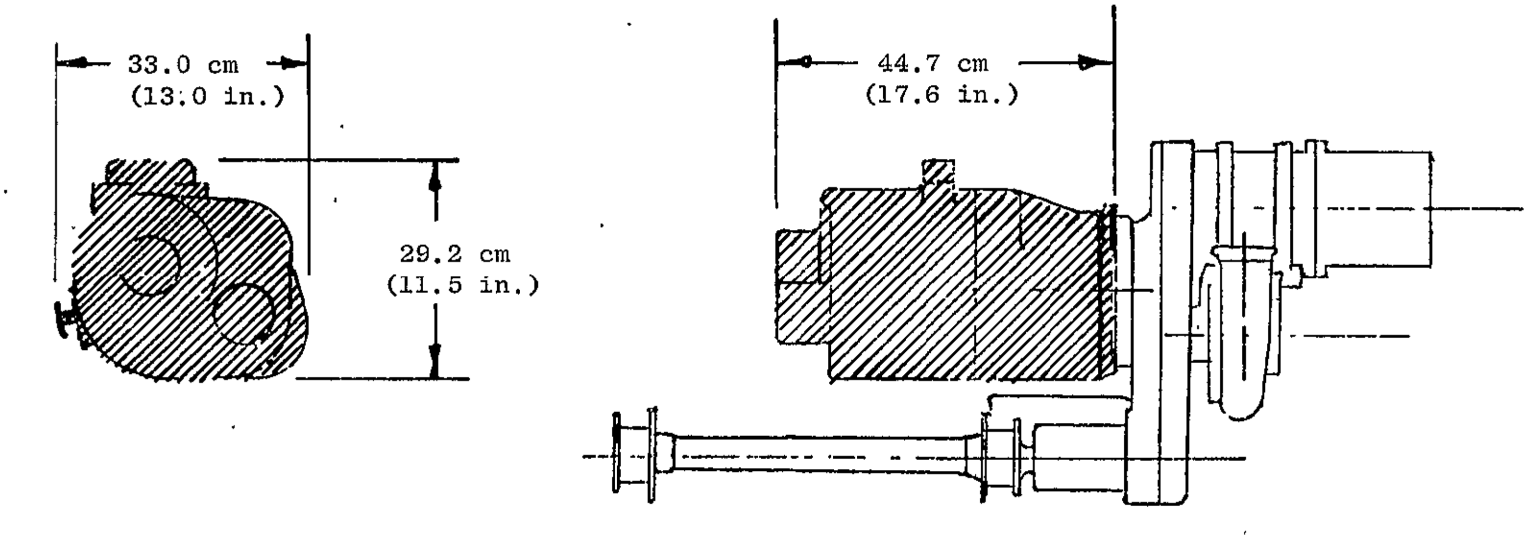


Figure 21. 60 KVA Integrated Drive Generator, Size.

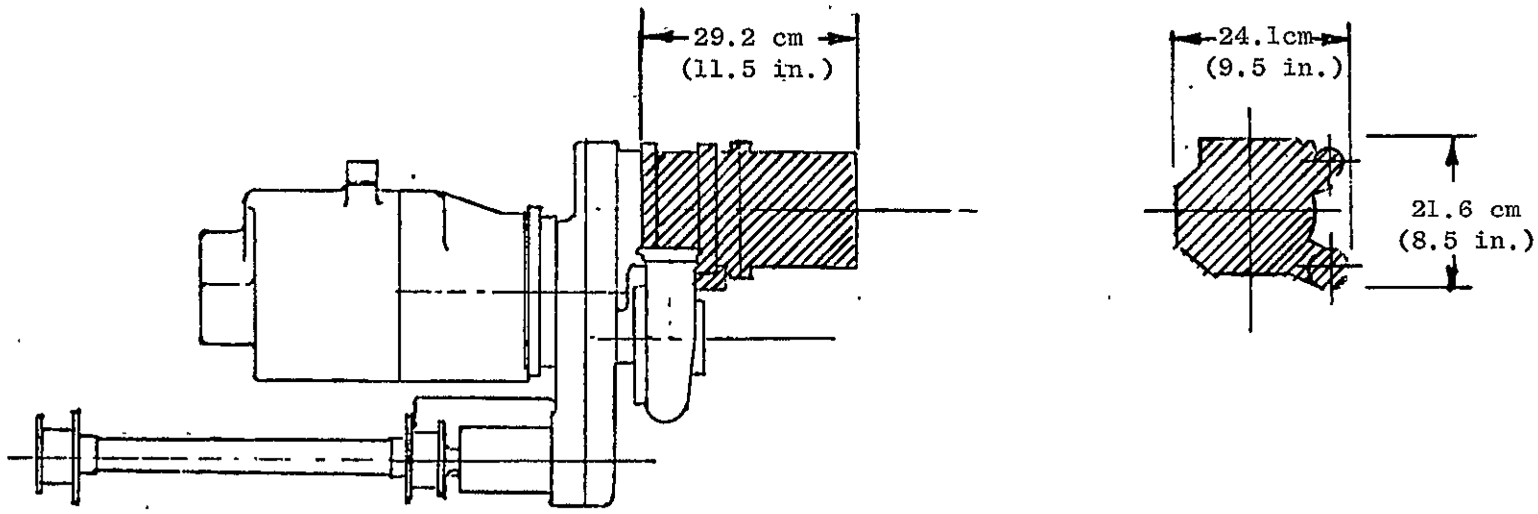


Figure 22. Hydraulic Pump, Size.

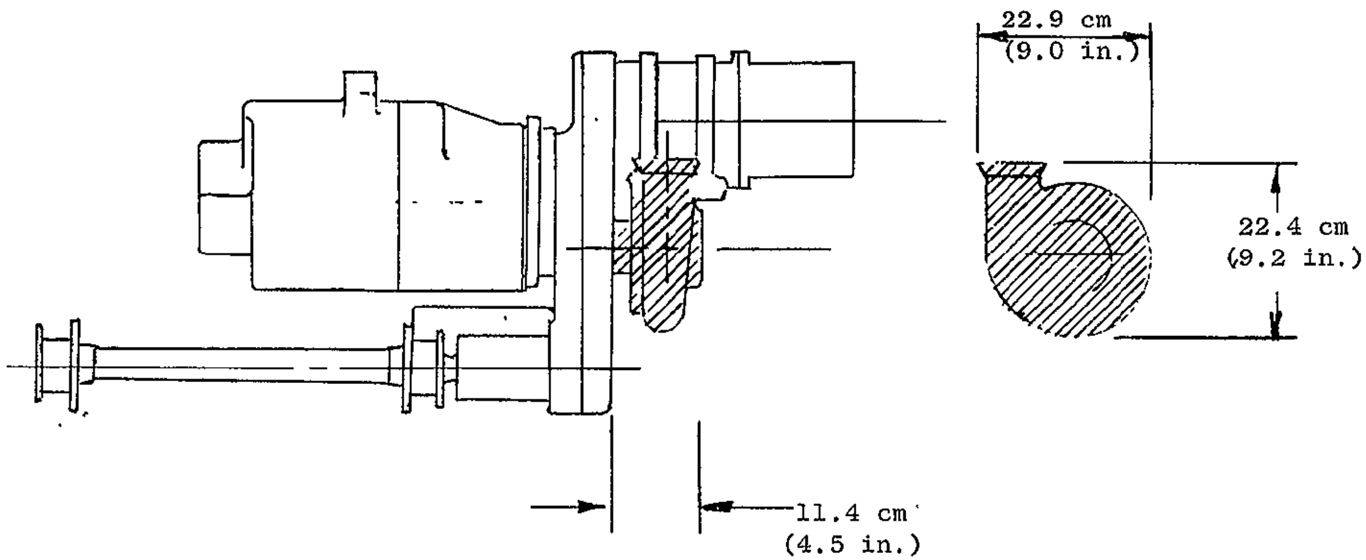


Figure 23. Air Turbine Starter, Size.

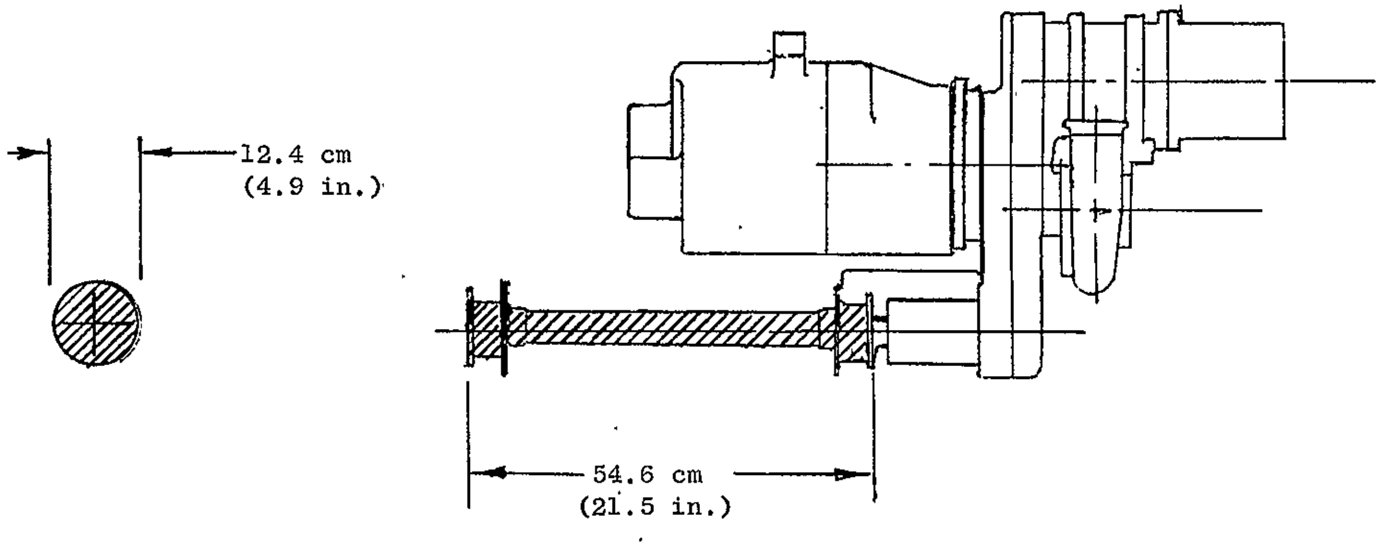


Figure 24. Power Takeoff Shaft, Size.

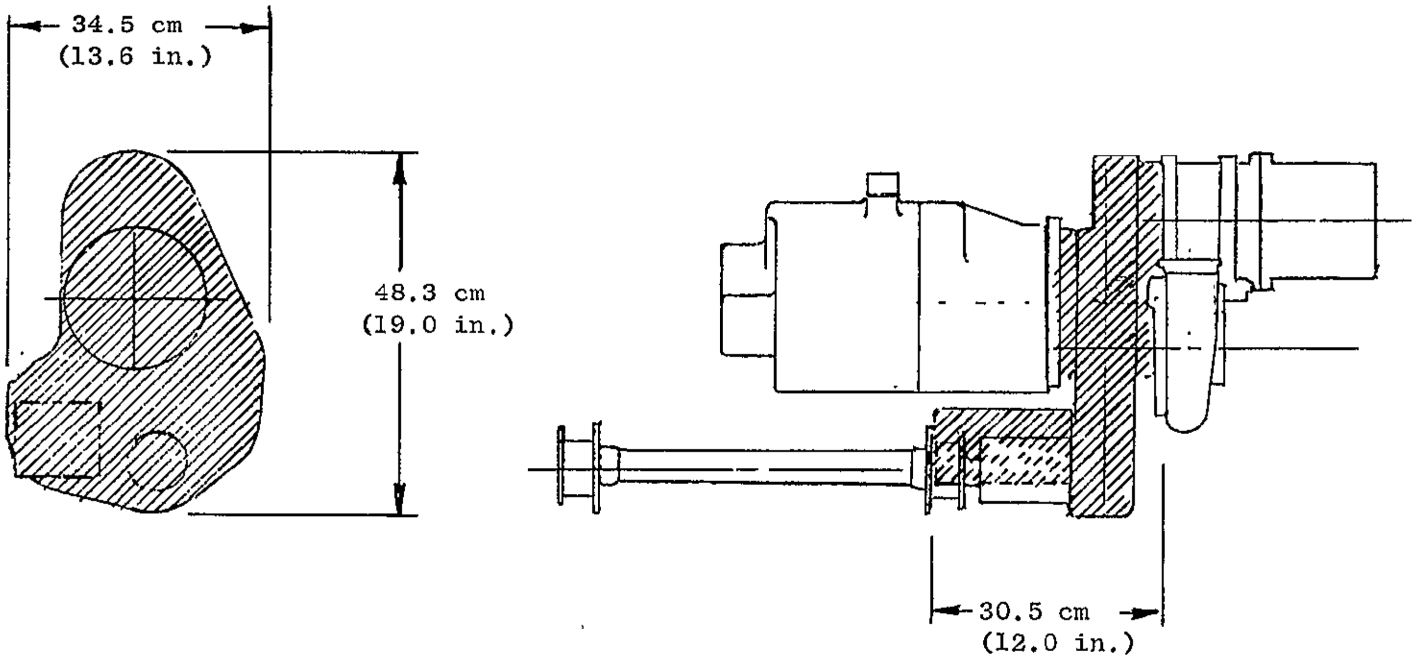


Figure 25. Accessory Gearbox, Size.

Table IX. Integrated Drive Generator Characteristics.

Rating	60 kva
Generator Speed	12,000 rpm
Output Power Frequency Control (400 Hz) ± 1% Within Idle, T/O Thrust Drive rpm	
Controls Over Input Speed Ratio	1.9:1
Generator Oil Cooled	
Unit QAD Mounted	
Weight	40.6 kg (89.4 lbs)
Requirements	3 per Airplane

Table X. Hydraulic Pump Characteristics.

Variable Displacement Type	
Maximum Displacement	49.2 cm^3 PR (3 CIPR)
Elect. Depressurizing Valve (EDV) to Ease Starting Engine	
Pumping Module Replaceable without Disturbing Hydraulic Plumbing	
Pump Module QAD Mounted	
Weight, (Incl. EDV)	16.1 kg (35.5 lbs)
Requirements	4 per Airplane

Table XI. Air Turbine Starter Characteristics.

Power Turbine Module QAD Mounted on Accessory Gearbox	
Reduction Gear Integrated with Accessory Gearbox	
Cutoff Speed	6400 PTO Shaft rpm
Weight, (w/o Control Valve)	3.8 kg (8.4 lbs)
Design Parameters	
Limit Torque at PTO Drive on Engine	400 m N (295 ft-lbs)
QCSEE Ignition Occurs at	2500 PTO Shaft rpm
Starter Cutoff at	6400 PTO Shaft rpm
Requirements	4 per Airplane

Table XII. Power Takeoff Shaft Characteristics.

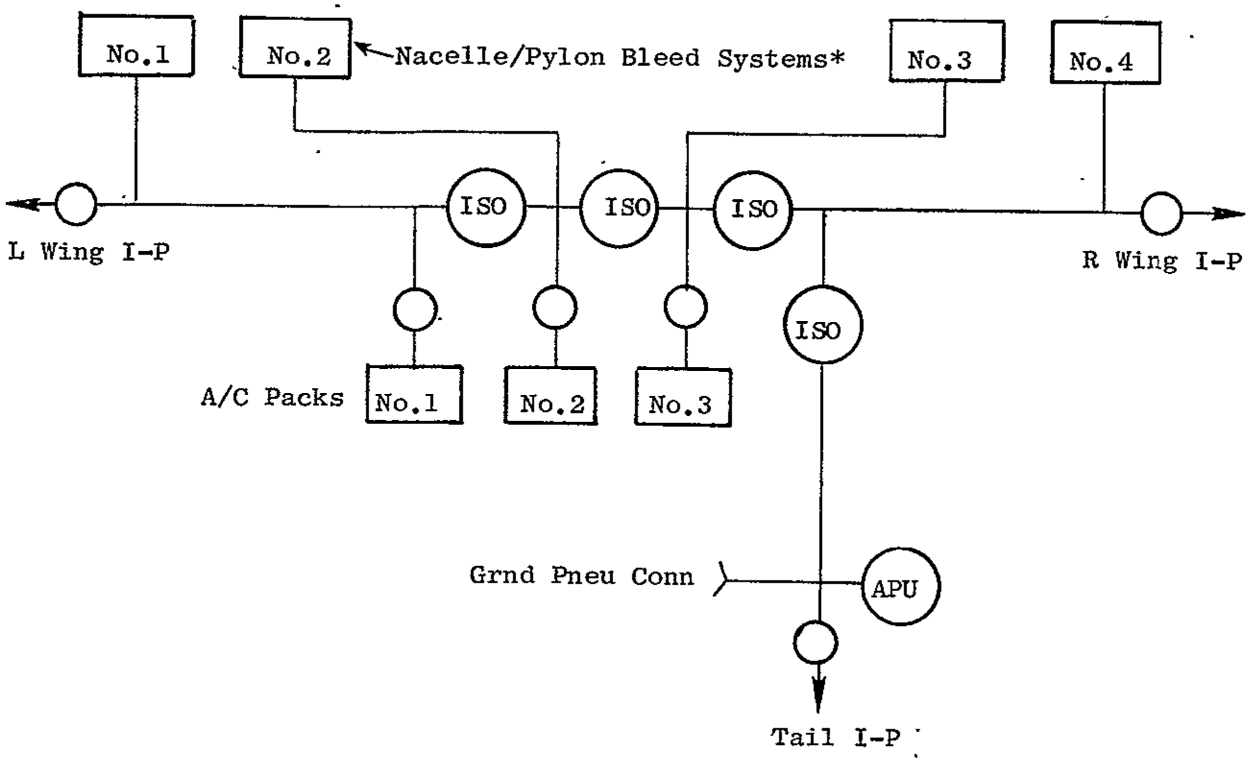
Connects Accessory Gearbox to PTO Drive	
Transmits Shaft Power to IDG and Hydraulic Pump	
Transmits Starting Torque to Engine PTO Drive	
Nominal Length	50.8 cm (20 in.)
Weight	1.95 kg (4.3 lbs)
Design Parameters	
Limit Speed	14460 rpm
Handle Starter Torque	
Handle IDG Elect. Fault Clearing Torque	
Handle Hydraulic Pump Driving Torque	
Accommodate Relative Motions	
Shaft Telescopes to Accommodate Engine Removal	
Requirements	4 per Airplane

Table XIII. Accessory Gearbox Characteristics.

Mounts Atop Pylon Beam	
Supports and Drives	60 kva IDG
Supports and Drives Variable Displacement Hydraulic Pump	
Supports Starter Turbine Module	
Incorporates Starter Turbine Reduction Gear	
Incorporates Fuel/Oil Cooler for Cooling IDG and Gearbox Lube Oil	
Incorporates Reduction Gearing for IDG and Hydraulic Pump Drives	
Weight, (Incl. Oil Cooler)	18.4 kg (40.5 lbs)

Table XIV. Accessory Gearbox Design Parameters.

PTO Shaft Speed Range at Idle	6700-7200 rpm
PTO Shaft Speed Range at Max Thrust	13495-14460 rpm
Transmit Starter Torque	
Transmit Power to IDG and Hydraulic Pump	
Transfer Heat from Lube Oil to Fuel Control By-Pass Fuel Returning to Wing Tank	
Requirements	4 per Airplane



* Blocks No.1 through 4 represent the nacelle/pylon bleed systems for the respective engines including engine inlet ice protection. Details of these subsystems are shown by Figure 27.

Figure 26. Environmental Control and Ice Protection System.

It has been an airline-imposed requirement for the most recent generation of commercial aircraft to provide unlimited aircraft dispatch capability with one bleed source inoperative and mission completion capability despite failure of a second bleed source in flight. It is believed that airline operating experience will justify the continuation of this requirement for QCSEE-powered aircraft. This means that all essential services must be provided with only two bleed sources operating.

It is a design objective to maximize engine bleed systems isolation, so as to prevent more than one engine feeding a duct rupture, to prevent the loss of cabin pressure due to a single bleed duct failure, to prevent interaction between bleed control systems, and to prevent engine overbleeding during operation with nonuniform engine power. The system shown in Figure 26 provides complete isolation except in some very rarely encountered situations such as the case of two inoperative bleeds on the same side of the aircraft at the same time that ice protection is required.

The use of three air conditioning packs instead of two originates from an airline requirement to provide unlimited dispatch capability with an inoperative pack, plus mission completion capability, despite another failure. Three packs are more favorable than two from the standpoint of peak bleed flow minimization in an emergency, since cabin pressurization for life support can be provided with two packs shut off. Three packs are considered more desirable than four because of maintenance and weight considerations.

When bleed system isolation is provided, each engine supplies a particular demand, rather than an average share of the total demand. In order to minimize the peak requirements per engine, the following modes of operation will be adopted when ice protection is needed and all engine bleed sources are operative:

1. The No. 1 air conditioning pack will be shut off.

2. Tail anti-icing will be provided by the APU, if operative. Otherwise the tail will be de-iced periodically with bleed from No. 4 engine. During the relatively short de-icing interval, both left and right wing anti-icing will be interrupted if the bleed limit of engine No. 4 would otherwise be exceeded.

In reviewing bleed requirements versus engine system capabilities, based on experience with prior aircraft, the following could be problem areas:

- Bleed Flow: The total allowable compressor bleed extraction from the QCSEE experimental engine is 13% of core flow; however, no more than 7% can be extracted from either interstage or compressor discharge. At some operating conditions, interstage bleed will not satisfy pressure and/or temperature requirements, thus (at these conditions) the total allowable extraction becomes limited to 7% of core flow. At low power conditions this flow, based on present core flow capability, will not satisfy aircraft requirements, as given in paragraph 3.1.12.2. Planned development of the core for commercial application, however, should provide adequate flow for these conditions.

- Bleed Pressure: The minimum acceptable bleed pressure for air conditioning and pressurization is believed to be available from compressor discharge at all conditions assuming that idle descent will be made at a flight idle power setting somewhat higher than the required ground idle setting.

  It appears that the minimum acceptable bleed pressure may not be available from the present UTW engine interstage ports at all cruise conditions. This conflicts with economy conditions, therefore, the use of a higher pressure bleed stage may prove desirable in a flight engine.

- Bleed Temperature: The minimum acceptable bleed temperature for ice protection is believed to be available from compressor discharge throughout the icing envelope at holding power conditions, or higher. However, flow demands exceed present UTW core engine compressor discharge bleed limitations at some conditions. A bleed system design approach that copes with this limitation is discussed in this section.

- Bleed Cleanliness/Purity: QCSEE, in addition to many engines in commercial service today, extracts interstage bleed air from the outer diameter of the compressor rather than the inner diameter and thus does not make favorable use of the particle separation characteristics of the compressor. However, provision for flow extraction across the compressor annulus is not practical in such small compressors. The compressor discharge bleed, also extracted from the outside, does involve a sudden change in direction which is judged to provide an effective separator.

Interfacing with the Baseline Airplane Environment Control and Ice Protection System, shown schematically in Figure 26, is the bleed air system in each of the four engine nacelles. The nacelle or propulsion system bleed schematic is shown in Figure 27. Its purpose is to condition and supply bleed air for cabin pressurization; for air conditioning; for ice protection of the propulsion system inlet, wing, and tail; and for main engine starting.

A separate engine port will be needed to supply bleed air at compressor discharge pressure (CDP) exclusively for inlet ice protection. Flow rate is limited by a pressure regulator and the fixed flow resistance of the ice protection system. This system is designed to accept full CDP bleed temperatures.

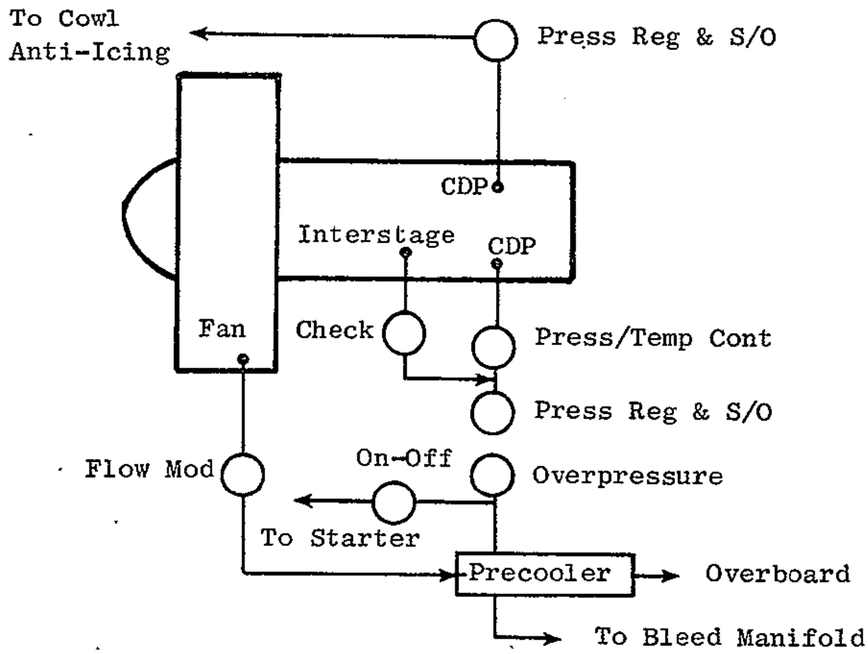


Figure 27. Nacelle/Pylon Bleed Air System.

Bleed air interstage pressure is used for the remainder of the Environmental Control and Ice Protection System whenever the flow pressure and temperature capabilities of this stage satisfy system demands. Otherwise, bleed air at CDP is used, either exclusively or mixed with interstage air, as described below.

Compressor discharge bleed is controlled by a special valve, having various pressure and temperature control modes. During normal operation, when ice protection is not required, this valve functions as a simple on-off valve and opens fully whenever interstage pressure is too low for air conditioning purposes. A check valve in the interstage bleed line prevents back-flow into the engine.

When ice protection is demanded, the CDP valve opens in a controlled manner whenever interstage bleed temperature is below the minimum value required for ice protection, namely 204.4° C (400° F). Somewhat similar performance is currently available from the "temperature augmentation valve" provided on the DC-9. It gradually increases the amount of CDP bleed in the mixture with interstage bleed. The QCSEE system requires another degree of complexity due to the lower interstage bleed temperatures. The mixed flow ranges from 100% interstage bleed to 100% CDP bleed when operating at a holding condition with a relatively low engine power setting.

An alternative approach would be to override the normal engine fan pitch controls in such a manner that higher bleed pressures and temperatures become available at any given engine thrust. This adversely affects fuel consumption, however, such an operating mode would occur only infrequently at a low power holding condition, mentioned above.

Downstream of the junction of interstage and compressor discharge bleed ducts is a regulator/shutoff valve. This valve limits the maximum pressure to which the bleed system is normally exposed. It also provides a way of shutting off interstage bleed.

Downstream of this valve is an "overpressure valve" which is essentially a backup pressure regulator, set to regulate to a somewhat higher pressure than the primary regulator. Existence of a higher-than-normal pressure in the system provides a warning in case the primary regulator malfunctions. The presence of this backup valve makes it possible to design the downstream duct system for the overpressure valve setting, rather than maximum bleed pressure, thus reducing bleed system weight.

A heat exchanger is located downstream of the overpressure valve. It functions as a precooler to limit the temperature of the bleed air delivered into the aircraft to a maximum of 232.2° C (450° F), for reasons of fire safety, and to limit the adverse impact on the strength of affected structure, in case of a bleed duct rupture. The cooling medium is fan discharge air supplied from a scoop in the side of the pylon. The flow rate of the cooling medium is controlled by means of an automatic flow modulation and shutoff valve that economizes on the use of fan air.

The precooler installation is shown in phantom in Figure 17. The bleed system valves are all located in the nacelle. This installation concept represents a significant departure from the QCSEE experimental engine configuration which features concentric interstage/CDP bleed ports.

4.2.17 Inlet Ice Protection

The engine inlet ice protection concept envisions a compressor discharge bleed port dedicated to engine ice protection exclusively. This system is isolated from the rest of the air bleed system. The concept provides maximum simplicity and reliability for this vital system. (See Figure 27).

The configuration of the inlet leading edge section is shown in Figure 28. The distribution manifold is constructed of Inconel, the bulkheads compartment are titanium and the other components including the lip are 2219 aluminum alloy.

The concept provides maximum ice protection capability by heating the entire water droplet impingement area. The hot airflow is controlled by a pressure regulator and shutoff valve discharging through a fixed-resistance distribution system.

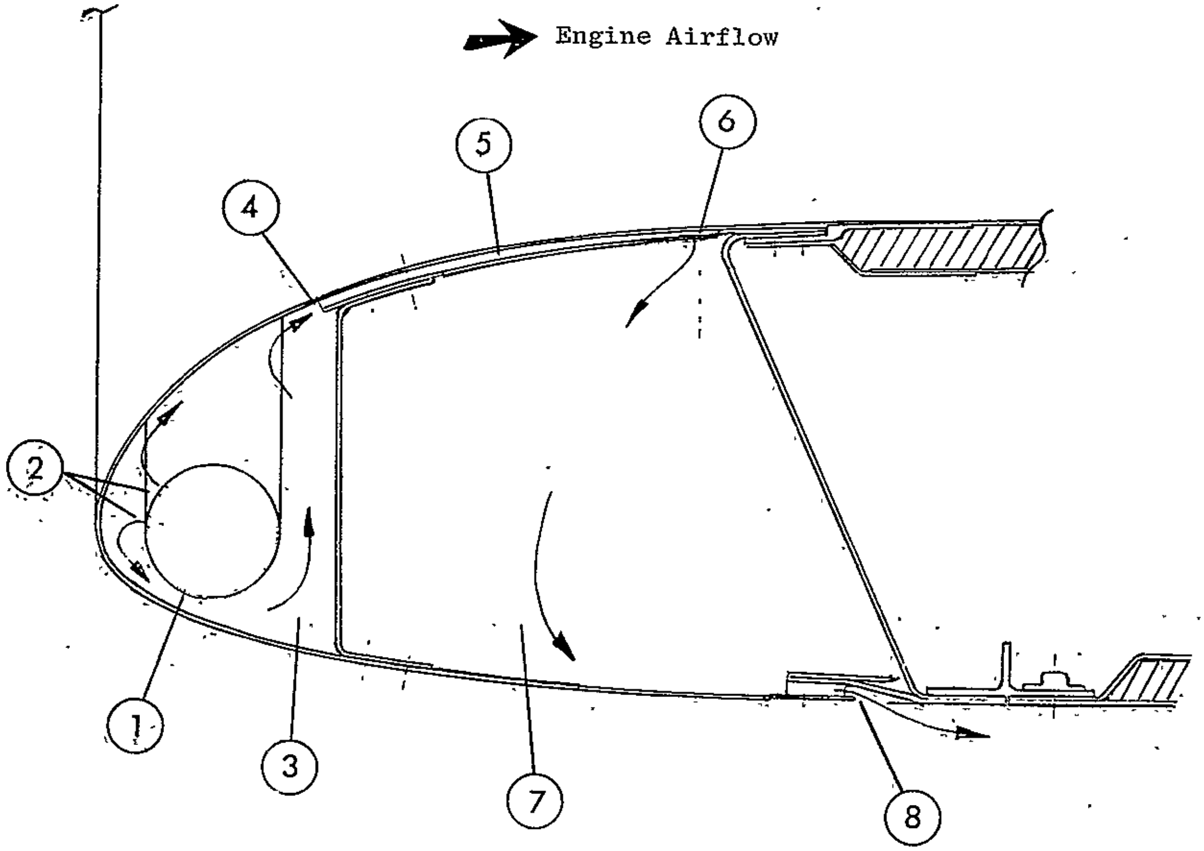
Design features contributing to low manufacturing cost and long life are as follows:

- No temperature-sensitive composites are used in the heated area.
- The distribution manifold, which is the only element in Figure 28 carrying appreciable pressure, is of homogeneous, tubular construction, (much superior to the built-up "D" sections commonly used on earlier aircraft).
- Double-skin passages begin well aft of the leading edge, where the compound curvature is relatively mild, and where attachment does not pose a significant concern.

4.2.18 Maintainability Access Provisions

The access provisions available for maintenance and inspection of engine and accessories are shown in Figures 29, 30, and 31.

Item 1 on Figure 29. These are the fan duct and core cowl doors hinged to the pylon and capable of opening wide, as shown in Figure 30 to fully expose the core engine. This provides ready access to borescope ports, variable inlet guide vane and stator mechanism, fan pitch change drive, engine mounts, the air bleed system valves and ducts, fuel manifold,



1 CIRCULAR HOT AIR DISTRIBUTION MANIFOLD

2 ORIFICES DIRECTING HOT AIR ON LIP

3 FORWARD COMPARTMENT

4 ENTRY TO DOUBLE-SKIN PASSAGE

5 DOUBLE-SKIN PASSAGE

6 DISCHARGE TO AFT COMPARTMENT

7 AFT COMPARTMENT

8 EXHAUST TO ATMOSPHERE

Figure 28. Engine Inlet Ice Protection.

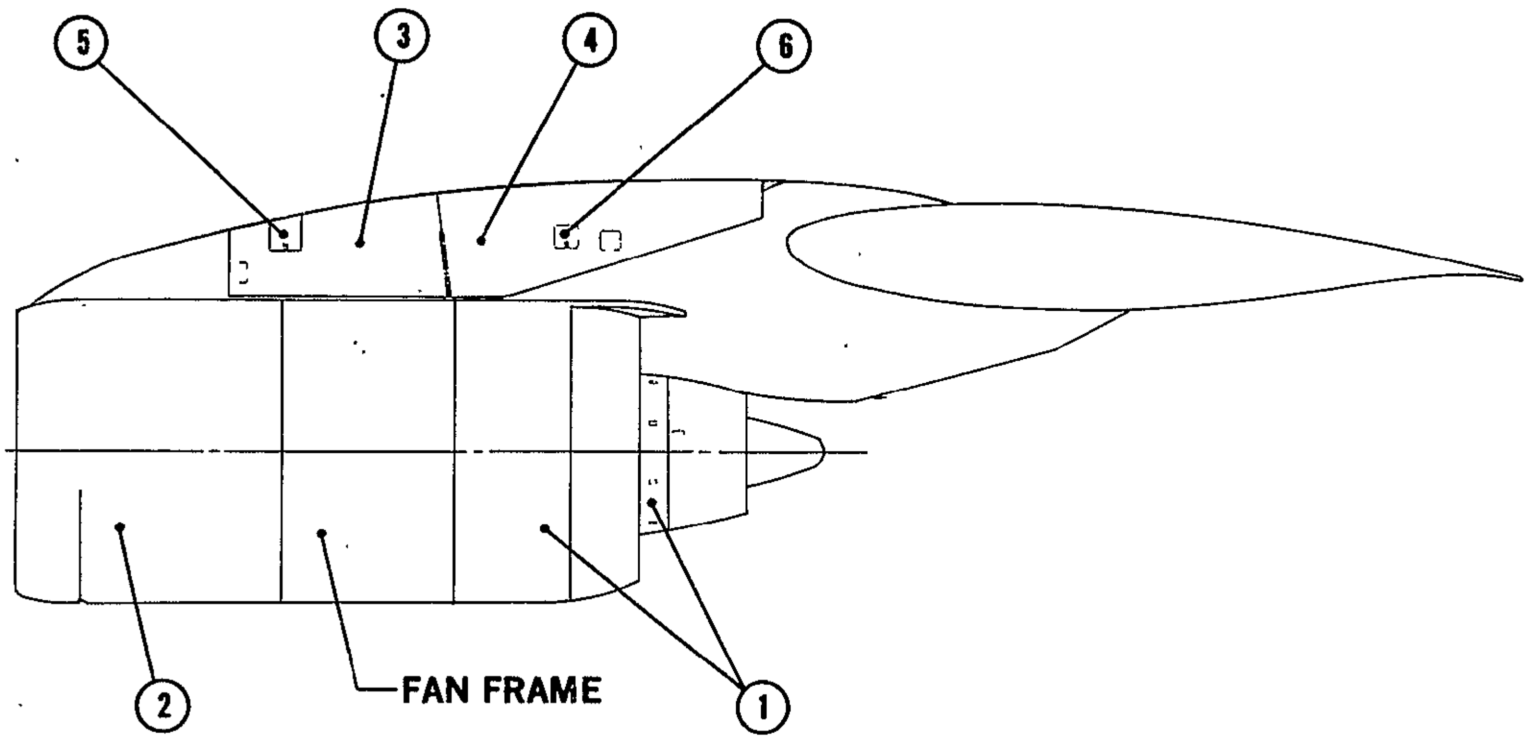


Figure 29. Access Provisions, Nacelle.

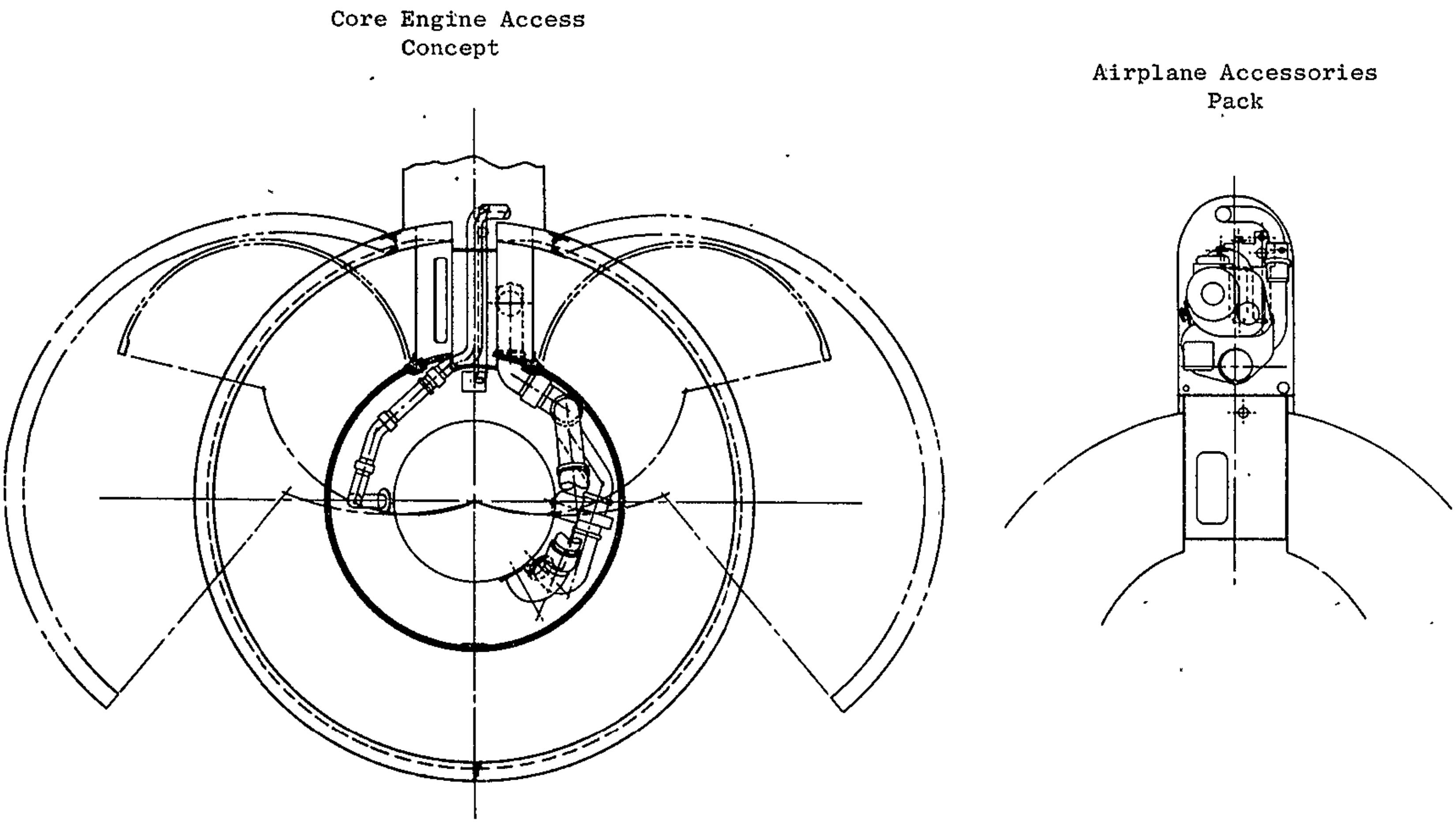


Figure 30. Engine Access, Fan and Core Cowl Doors.

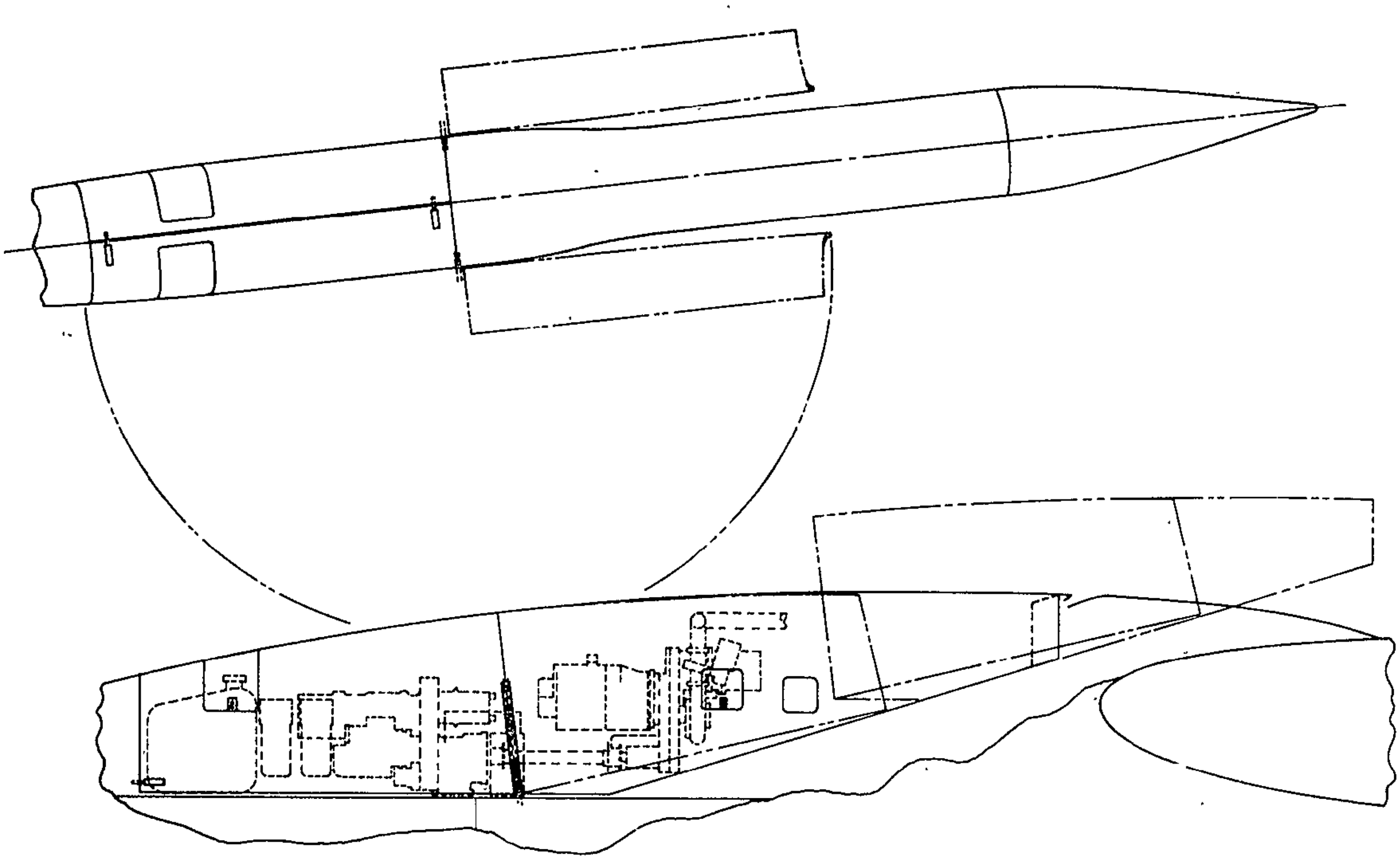


Figure 31. Accessory Access Provisions.

ignitors, lube oil scavenge pump, gas path thermocouples, fan nozzle actuators, etc.

Item 2. Inlet duct with removable leading edge section that accommodates easy access for inspection and repair of ice protection system components and of distress resulting from strikes on the leading edge by such objects as birds, hail, stones, airplane hardware items, and tire parts encountered on airport runways.

Items 3 and 4. These are hinged petal panels comprising part of the pylon upper fairing. The front section, Item 3, consists of a pair of fairing halves split along top centerline and joined to Item 4 with hinges so that two halves can be swung open and back against Item 4 to expose the QCSEE accessories group. The concept is depicted in Figure 31.

This provides access for maintenance and inspection actions to the fuel pump and control, the digital control, the lube oil tank, fuel and lube oil filters, the lube pump, the variable-geometry hydraulic pump, the accessory power takeoff drive pad, fuel supply connection, throttle linkage, etc. The rear section (Item 4) is arranged so that it can be translated rearward (carrying the hinged fairing halves along) as necessary to gain access to the airplane accessories pack (60KVA IDG, variable-displacement hydraulic pump, air turbine starter, starter control valve, hydraulic oil filter module, accessory gearbox, and related fuel/oil cooler) and the air bleed system components including the precooler located within the pylon box section. This concept permits operating the engines with the accessories section open for purposes of leak checking and making adjustments without need to detach fairings from the airplane or having parts overhanging the inlet, thus minimizing the risk of damage to the fairings or the engine.

Items 5 and 6. These are hinged inset doors that permit access for such functions as servicing the lube oil tank, and manually operating the starter and inlet leading edge ice protection control valves.

4.2.19 Maintenance Workstand Concepts

The high-wing arrangement needed with the EBF-powered lift system results in the engines being located high above the ground. This makes it necessary to use workstands in performing on wing propulsion system maintenance.

Figures 32 and 33 show the concept of a two-level workstand addressed to performing heavy maintenance at the operator's main base. The stand is arranged to accommodate work going on simultaneously in the accessories area at the top of the pod and pylon beam and also in the core engine zone. The lower level extends far enough beyond the extremities of the QCSEE pod to permit access to the inlet via a stepladder for tasks such as inspecting the fan, replacing fan blades, servicing the fan pitch control mechanism, etc; and, also, to the turbine exhaust duct for related inspecting and maintaining actions.

The upper level platforms are arranged to accommodate access to both sides of the accessories on the fan frame and pylon beam. The platforms inside the workstand columns are hinged to enable them to be moved clear of the space below the engine to permit lowering the QCSEE to a transport dolly and hoisting the replacement engine to position on the pylon beam. The end of the workstand facing the rear of the airplane is free of cross-bracing to enable the engine transport dolly to be spotted below the pylon. Figures 34 and 35 show how the kinds of portable workstands currently used to service airliners could be employed to perform light maintenance and for such ramp activities as inspecting the inlet duct and fan, servicing the oil tank, inspecting fuel and oil filters, etc. These stands feature work platforms that can be adjusted to convenient working height with built-in, hand-pumped, hydraulic systems.

4.3 NACELLE AERODYNAMICS

4.3.1 Inlet Operating Conditions

The QCSEE inlet was designed for a throat Mach Number of 0.79 and to operate at high angles of attack with sufficiently low engine-face distortion so that adverse engine operation would not occur. An alternate method of meeting the QCSEE aerodynamic and acoustic requirements was investigated, see Appendix D.

The relatively high one-dimensional inlet throat Mach number was selected to achieve forward-radiating fan noise attenuation, as discussed in Section 6 on Flight System Acoustics, but not too high, to provide sufficient margin from the inlet choking point for high internal performance, considering transient engine operational requirements, engine control tolerances, throat corrected flow variations due to aircraft operational effects, and inlet manufacturing tolerances. The high angles of attack resulted from anticipated STOL airplane angle-of-attack and crosswind conditions. The angle-of-attack condition defined by NASA requires satisfactory engine operation to 50 degrees at 41.2 m/sec (80 knots) air speed approaching the inlet (M_o = 0.12). The NASA-defined crosswind requirement was for satisfactory engine operation with 18.0 m/sec (35 knots) wind speed from the side of the inlet (β = 90 degrees). To verify these requirements, maximum STOL airplane inlet angles of attack were estimated using the characteristics of the Douglas YC-15 airplane powered by four QCSEE engines. The calculations were made for conditions corresponding to unaccelerated airplane flight at maximum lift coefficient for a range of power settings and flap settings. For a commercial STOL airplane, flight at these conditions would be accomplished during tests to determine the minimum airplane speeds to be used in service. The Douglas EVD Program (Reference 5) was used to estimate velocity vectors ahead of the wind at the inlet station. The inlet angle of attack was defined as the angular

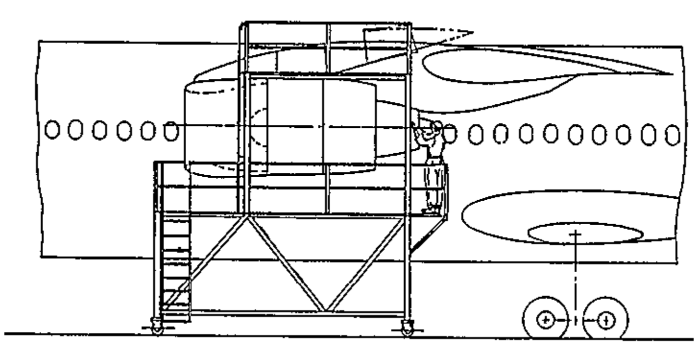


Figure 32. Maintenance Stand, Heavy (Side View).

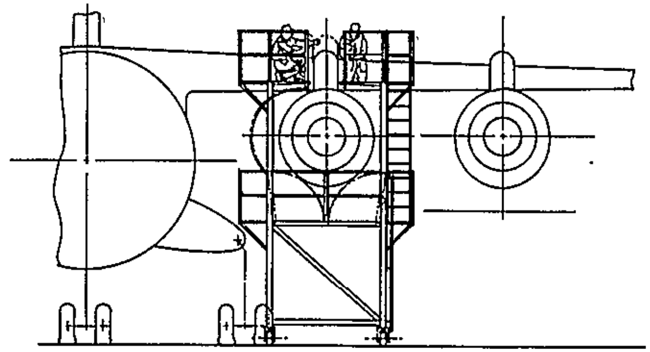


Figure 33. Maintenance Stand, Heavy (Front View).

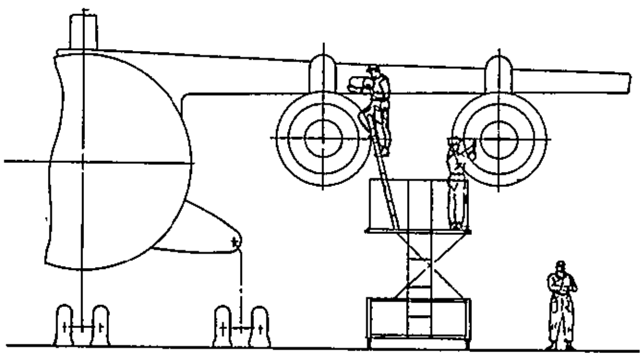


Figure 34. Maintenance Stand, Light.

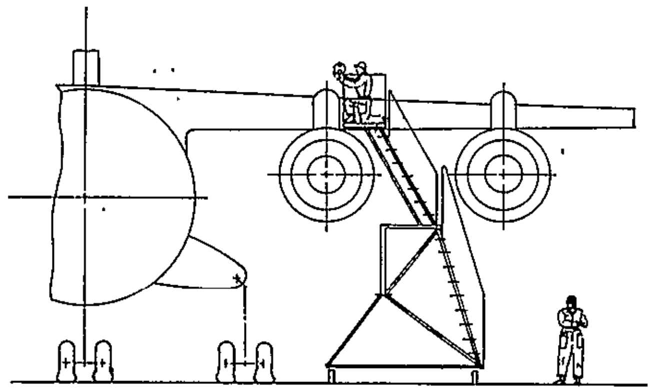


Figure 35. Maintenance Stand, Light (Alternate).

difference between the inlet centerline and the inlet approach velocity vector. The inlet approach Mach number was calculated from the magnitude of the approach velocity vector and the local speed of sound. The approach velocity vector was calculated at the point where the inlet centerline intersected the inlet leading edge plane. The fuselage, wind, flap, and power effects were simulated in the EVD program calculation. The inlet and nacelle geometry were not simulated. By not simulating the inlet and the nacelle geometry, the inlet angle of attack and approach Mach number were defined in the same manner with the inlet installed on the airplane as for an isolated inlet test. (For an isolated inlet test, the inlet angle of attack is defined as the angular difference between the inlet centerline and the test section velocity vector without the inlet installed. Similarly, the approach Mach number that would exist in the test section without the inlet installed.)

The final values of inlet angle of attack were determined by adding 13 degrees to the calculated value. The 13-degree increment was derived from DC-10 minimum-flight-speed tests. The maximum inlet angle of attack that was encountered during any of the tests exceeded the calculated inlet angle of attack corresponding to CL_{max} by this value. The results of the angle-of-attack calculations are shown in Table XV.

Table XV. Inlet Angle of Attack.

Airplane Configuration		Power Setting	Mach No. Approaching Inlet	Inlet Angle of Attack	
Flap Position	Slat Position			Calculated	Calculated Plus 13°
Landing	Full Extension	Takeoff	0.12	38°	51°
		Approach	0.14	32°	45°
Takeoff	Partial Extension	Takeoff	0.14	32°	45°
		Approach	0.16	28°	41°
Retracted (Used for takeoff or prior to flaps down approach)	Partial Extension	Takeoff	0.18	31°	44°
		Approach	0.19	29°	42°
NASA QCSEE Requirement		Any	0.12	50°	

The table indicates the inlet conditions at the maximum estimate angle-of-attack condition are practically the same as the NASA QCSEE requirement (51° at M_o = 0.12 compared to 50° at M_o = 0.12).

The most adverse inlet operating condition shown in Table XV is the flaps-retracted/slats-extended case with inlet angle of attack of 44° and approach Mach number of 0.18. This is shown by Figures 36 through 38 where the tabulated inlet operating conditions are compared with lines of inlet leading-edge-boundary-layer separation or high values of total-pressure distortion at the end of the inlet. The separation or high distortion conditions were estimated from Douglas YC-15 inlet model data. The solid lines in Figures 36 through 38 were estimated from the data and the dashed lines represent extrapolation of the data. For an inlet with internal lip-thickness radius ratio of R_{HL}/R_i = 1.22, higher distortion would be expected at takeoff power at α = 44° with M_o = 0.18 (stall with flaps retracted) than at takeoff power at α = 50° with M_o = 0.12 (stall with full flaps or NASA QCSEE requirement), or at α = 45°, M_o = 0.14 (stall with partial flap extensions). The reason is that the beneficial effect of the lower angle of attack is offset by the adverse effect of increased Mach number approaching the inlet.

4.3.2 Configurations for 30.5 cm (12-Inch) Model Tests

The aerodynamic performance of four axisymmetric inlet models with various leading-edge shapes was measured in the Lewis 2.7 x 4.6 m (9 x 15 ft) wind tunnel at angle-of-attack and crosswind conditions. The model geometries are summarized in Table XVI. Results of the test program are summarized in Reference 6.

Table XVI. Model Test, Inlet Data.

Internal Lip		External Cowl			Diffuser			
R_{HL}/R_i	Shape	R_{HL}/R_{max}	X/D_{max}	Shape	L/D	$(x/L)\,\theta_{max}$	θ_{max}	θ_{eg}
1.17	2:1 ellipse	0.905	0.20	DAC-1	0.83	0.50	8.7°	6.4°
1.21		0.905	0.20	DAC-1				
1.21		0.935	0.18	NACA-1				
1.25		0.905	0.20	DAC-1				

The diameter of the inlet duct at the simulated engine face was 30.5 cm (12 inches) which correspond to 16.9 percent of the full-scale inlet size.

The inlet lip geometries that were tested were selected after review of NASA, GE, and Douglas available inlet data. The internal lip thicknesses tested corresponded to thickness ratios of R_{HL}/R_i = 1.17, 1.21, and 1.25. The internal lip shape was a 2:1 ellipse. Based on the available Douglas data, this range of lip thicknesses was sufficient to include that which would result in good inlet performance at angle-of-attack and

Altitude: 3,048 m (10,000 ft)
Airplane Weight: 68,040 kg (150,000 lbs)
QCSEE Engine with
88,960 N(20,000 lbs) Thrust Rating

Symbol	Inlet Approach Mach Number	Power Setting
○	0.12	Take-Off
□	0.14	Approach
△	0.15	Idle

Notes: Symbols Represent Operating Conditions for QCSEE Inlet on YC-15, Lines Represent Conditions of High Inlet Distortion or Inlet Leading-Edge-Boundary Layer Separation Based on Douglas YC-15 Inlet Model Data.

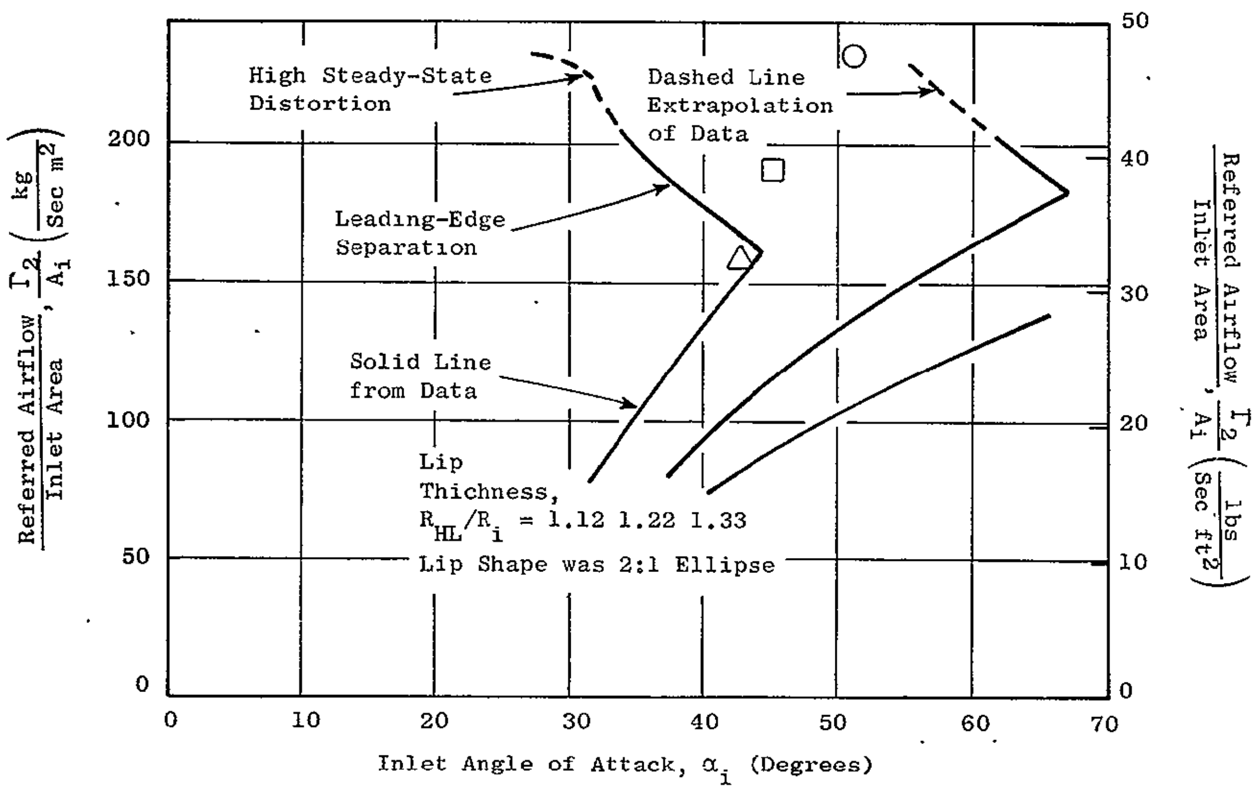


Figure 36. Inlet Requirements, Landing (Stall).

Altitude: 3,048 m (10,000 ft)
Airplane Weight: 68,040 kg (150,000 lbs)
88,960 N (20,000 lbs) Thrust Rating

Symbol	Inlet Approach Mach Number	Power Setting
○	0.14	Take-Off
□	0.16	Approach
△	0.17	Idle

Notes: Symbols Represent Operating Conditions for QCSEE Inlet on YC-15, Lines Represent Conditions of High Inlet Distortion or Inlet Leading-Edge-Boundary Layer Separation Based on Douglas YC-15 Inlet Model Data.

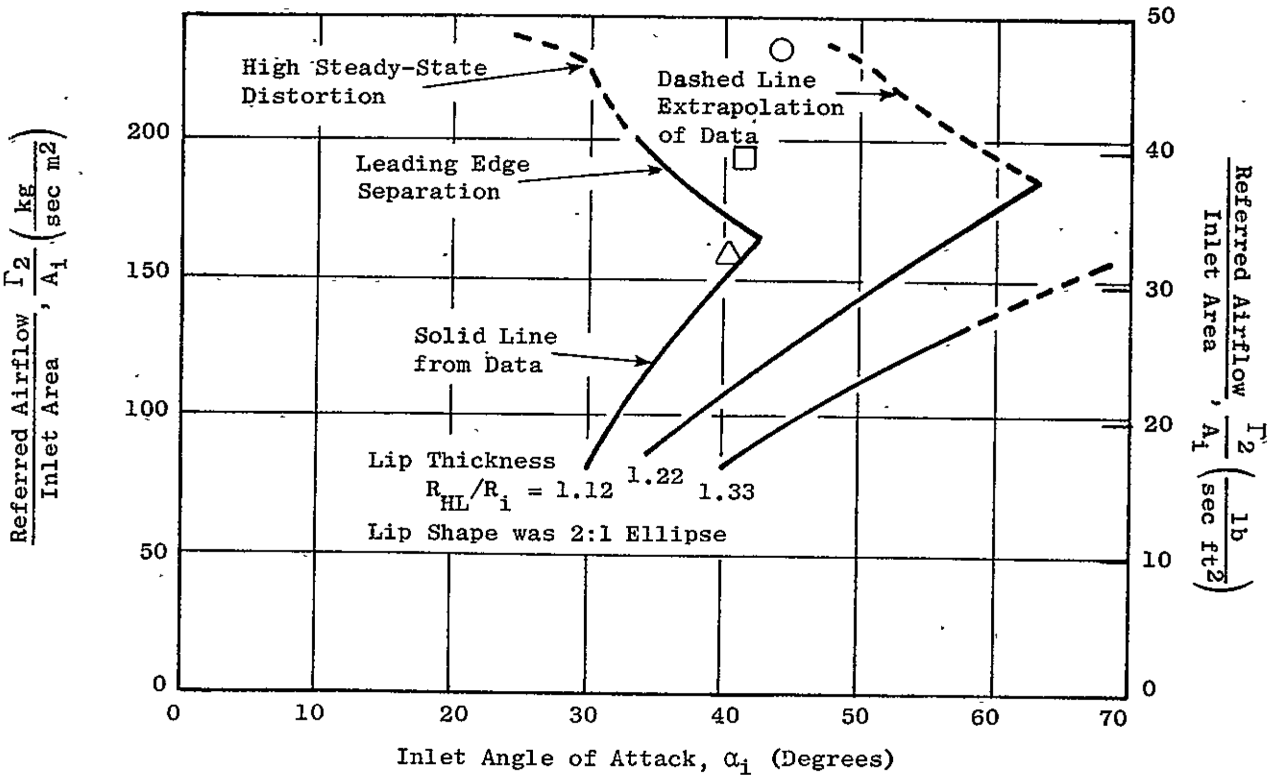


Figure 37. Inlet Requirements, Takeoff Flap Setting (Stall).

Altitude: 3,048 m (10,000 ft)
Airplane Weight: 68,040 kg (150,000 lbs)
QCSEE Engine with
88,960 N (20,000 lbs) Thrust Rating

Symbol	Inlet Approach Mach Number	Power Setting
○	0.18	Take-Off
□	0.19	Approach
△	0.20	Idle

Notes: Symbols Represent Operating Conditions for QCSEE Inlet on YC-15, Lines Represent Conditions of High Inlet Distortion or Inlet Leading-Edge-Boundary Layer Separation Based on Douglas YC-15 Inlet Model Data.

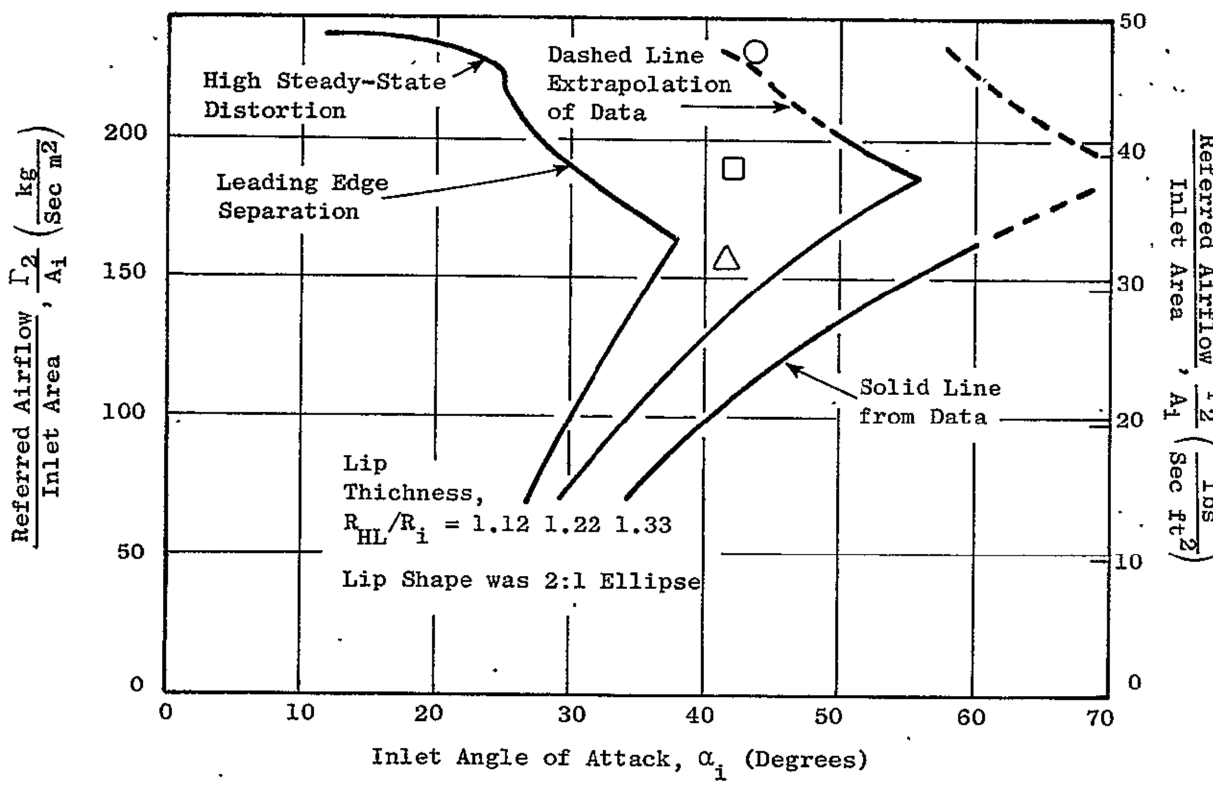


Figure 38. Inlet Requirements, Takeoff (Stall) (Flaps Retracted).

crosswind conditions. Figures 36-38 show that for good angle-of-attack performance at takeoff power at $\alpha = 50°$, M_o, = 0.12 (NASA QCSEE requirement) or = 44°, M_o = 0.18 (most adverse angle-of-attack condition) a lip-thickness ratio of about 1.20 to 1.23 would be required. Figure 39, summary of Douglas crosswind data, for R_{HL}/R_i = 1.16, shows that a lip-thickness ratio greater than R_{HL}/R_i = 1.16 would be required for good crosswind performance at takeoff power at the NASA QCSEE crosswind requirement of 18.0 m/sec (35 knots).

Two cowl-forebody geometries were tested with an internal lip thickness ratio of 1.21. One geometry had a relatively blunt nose shape and was defined by Douglas 1 (DAC-1) nondimensional cowl-shape coordinates, a cowl radius ratio of R_{HL}/R_{max} = 0.905, and a cowl-length ratio of X/D_{max} = 0.20. This geometry was also tested with internal-lip radius ratios of 1.17 and 1.25. The other geometry had a sharper nose shape defined by NACA-1 nondimensional cowl-shape coordinates, a cowl radius of R_{HL}/R_{max} = 0.935 and a cowl length ratio of X/D_{max} = 0.18.

Two external-cowl geometries were tested to investigate the possibility that the internal flow might be significantly affected by the external-cowl shape at high angle-of-attack conditions. Figure 40 shows the results of potential-flow calculations for the internal-flow pressure-coefficient distribution for an inlet with two external lip shapes, at angle of attach conditions corresponding to $\alpha = 50°$, M_o = 0.12, takeoff power. For the sharper shape, there was a significant high velocity peak in the internal flow, just outside of the inlet leading edge. This high velocity peak could adversely affect the internal inlet performance. This peak was reduced by a blunter external cowl shape as shown in Figure 40. The blunter shape of Figure 40 was approximately equivalent to the blunter shape of the 30.5 cm (12-inch) models.

One diffuser shape was used for all four inlet models. The diffuser length-to-diameter ratio of L/D = 0.83 was sufficient for the required amount of full-scale-inlet acoustic treatment and also resulted in low aerodynamic-performance risk based on NASA theoretical studies (Reference 6) and comparison with Douglas subsonic transport diffuser geometries. The diffuser inflection point at x/L = 0.5 was based on NASA theoretical studies (Reference 6).

4.3.3 Internal-Lip Design

Based on the 30.5 cm (12-inch) model test results, summarized in Reference 6, the smallest internal lip that had the required angle-of-attack and crosswind capability was the R_{HL}/R_i = 1.21 lip (inlet number 2). This lip thickness was required for relatively low values of steady-state distortion $[P_{T_{max}} - P_{T_{max}}/P_{T_{avg}} < 0.10]$ for any value of airflow at the NASA angle-of-attack and crosswind requirements. This is shown by Figure 41 and 42. The lowest lip thickness tested, with R_{HL}/R_i = 1.17, showed significantly higher distortion. Use of the larger lip thickness

Douglas Model Data for $R_{HL}/R_i = 1.16$
Confirmed by Full Scale Tests to Γ_2/A_i

■ DAC/GE Full Scale Data Without Separation, $R_{HL}/R_i = 1.16$

▲ DAC/GE Full Scale Flow Attachment During Engine Acceleration, $R_{HL}/R_i = 1.16$

NASA QCSEE Crosswind Requirement is for 18.0 m/s (35 knots)

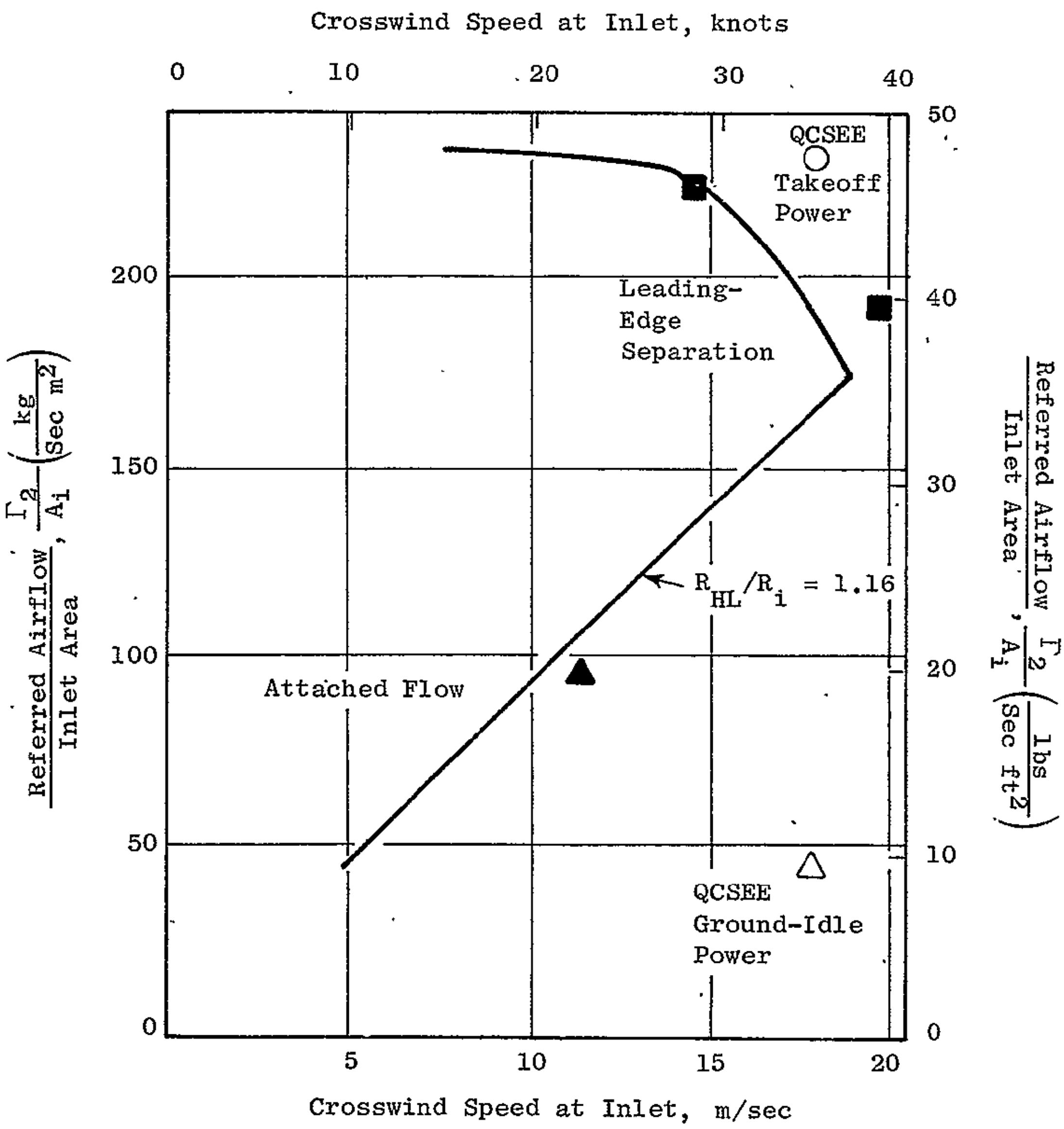


Figure 39. Inlet Crosswind Characteristics.

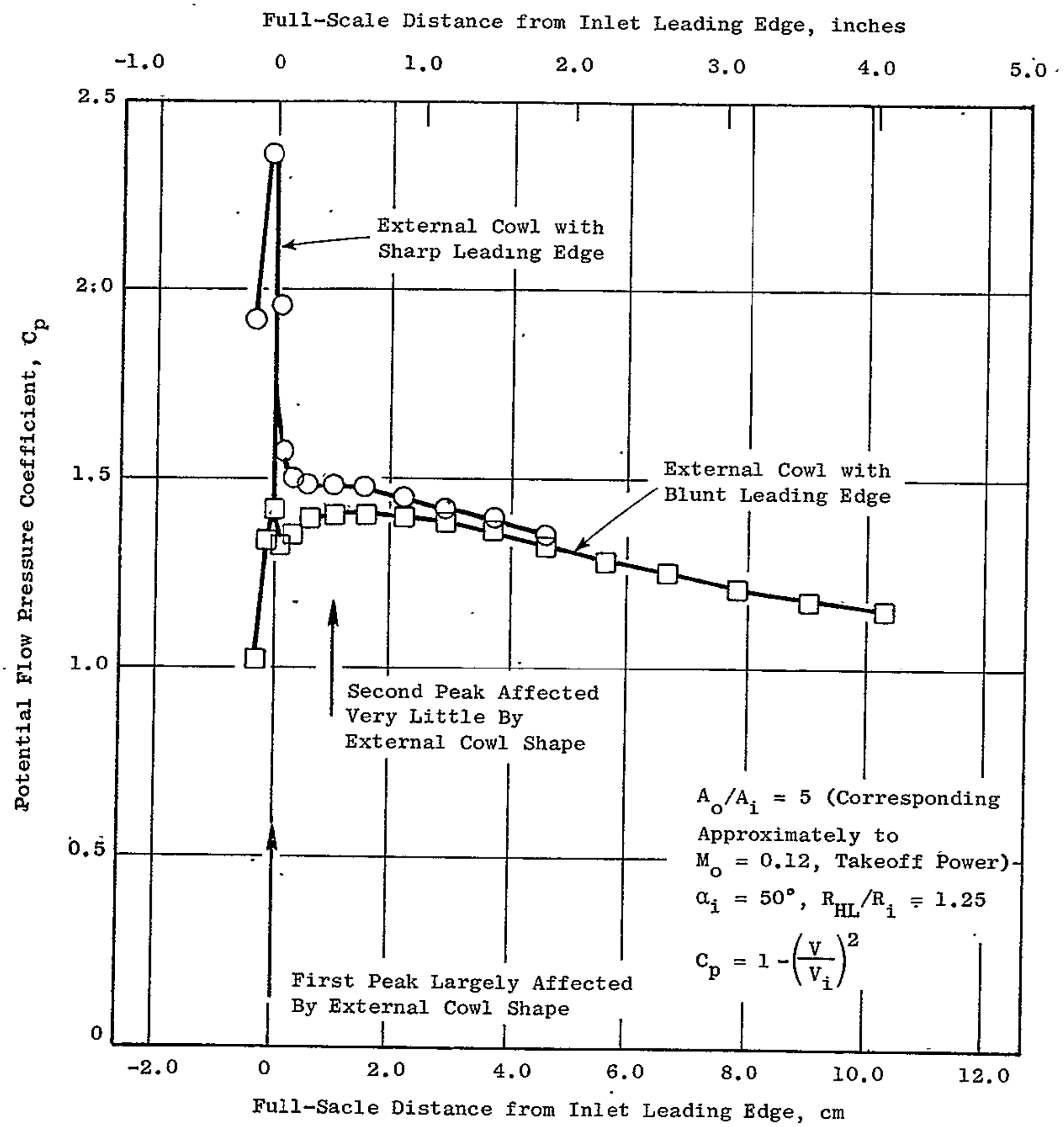


Figure 40. Inlet, Lip Shape Effect on Internal-Flow Pressure Distributions.

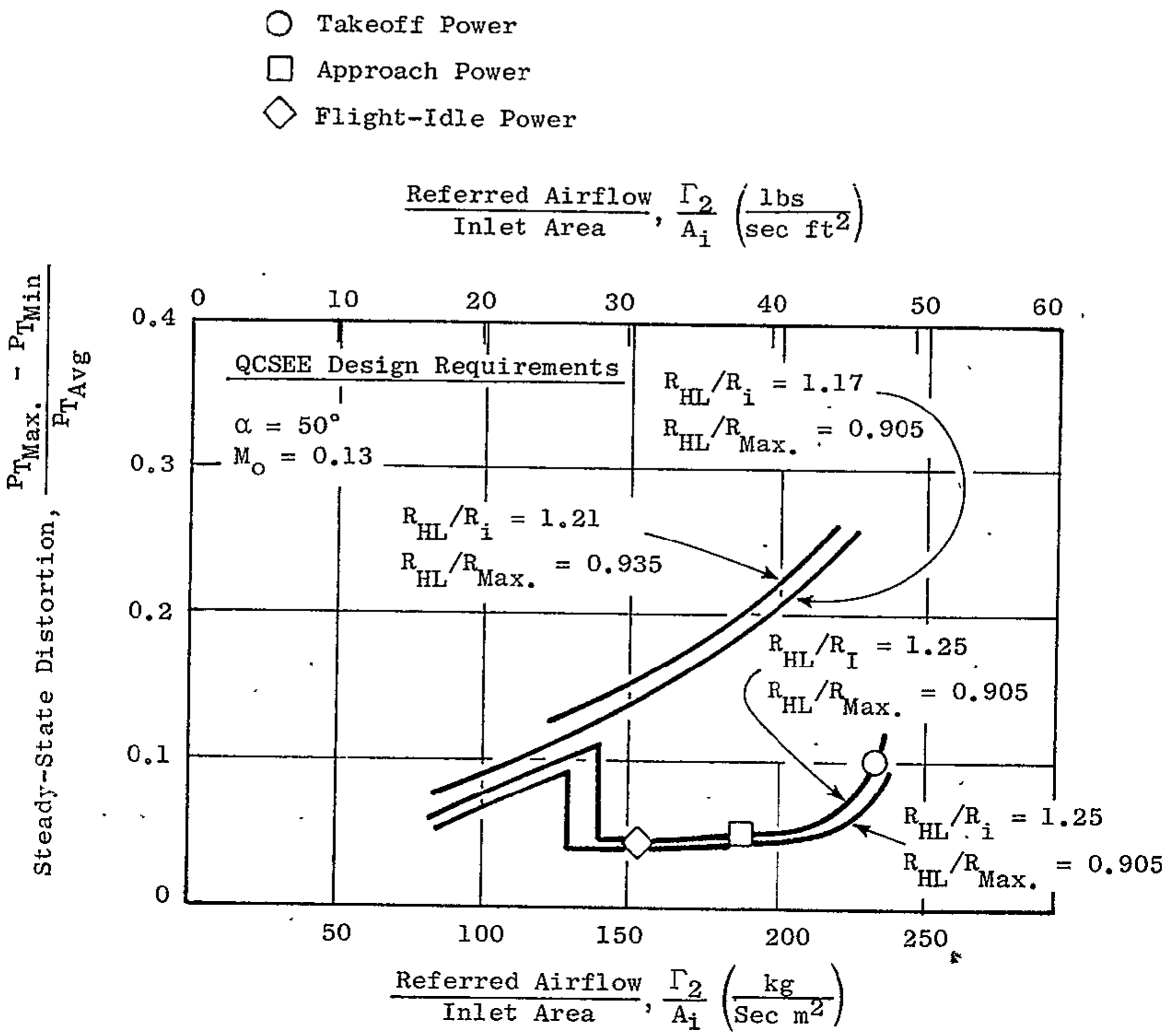


Figure 41. Inlet, Angle of Attack (Test Data α = 50°, M_o = 0.13).

C-2

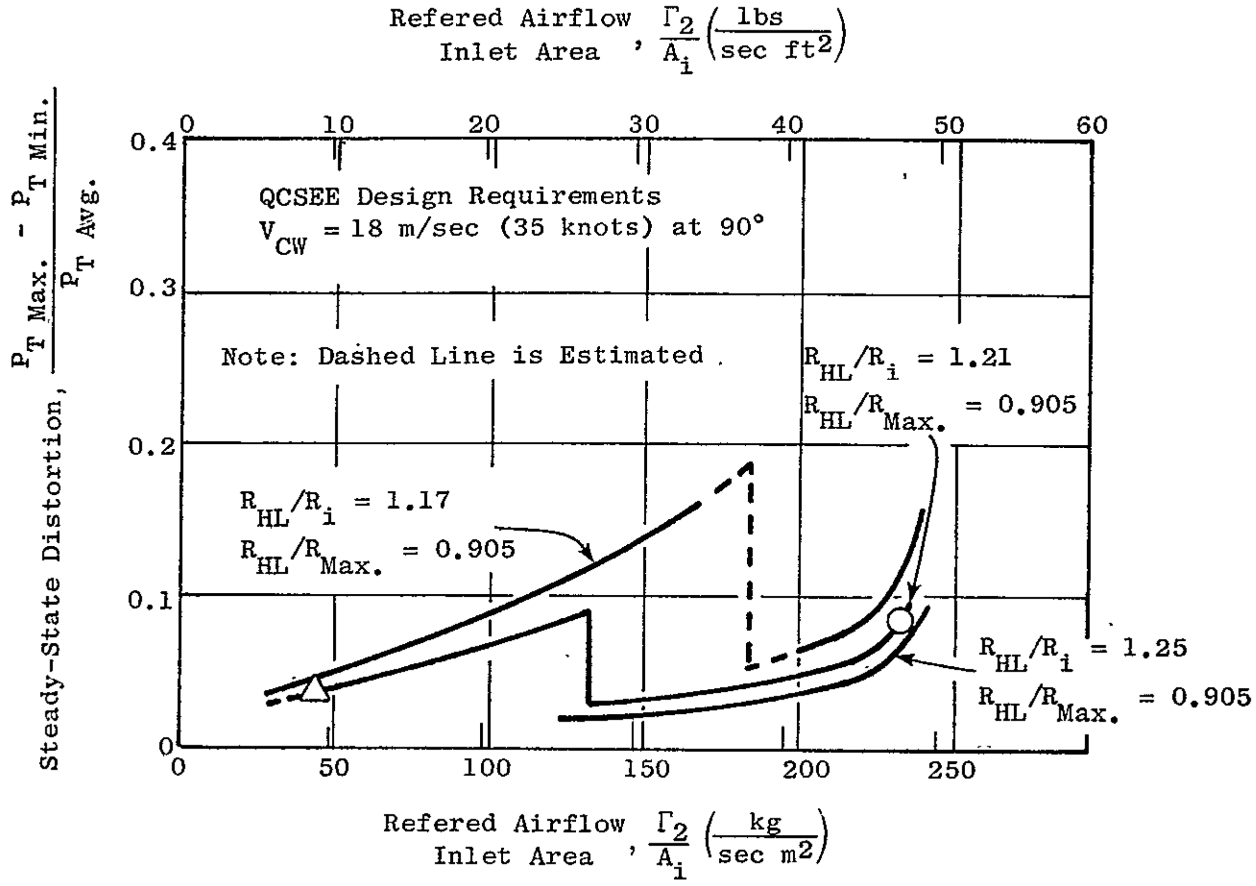


Figure 42. Inlet, Crosswind Performance (Test Data).

of R_{HL}/R_i = 1.25 would not have resulted in significantly better performance at high angle of attack conditions, but would have resulted in a larger external cowl maximum radius and therefore was not selected.

Figure 43 shows that for the most adverse YC-15 airplane stall condition, the steady-state distortion is higher than for the stated angle-of-attack and crosswind requirements. GE examined both the steady-state and dynamic distortion data for this operating condition and indicated acceptable engine operation, especially considering that this condition would exist only during demonstration flight tests and would not be an operational requirement.

4.3.4 External Cowl Design

The external cowl with the sharpest nose shape showed significantly higher distortion levels than did the configuration with the blunter shape (Figure 41). Therefore the external-cowl nose shape was made consistent with that used for the wind-tunnel tests. The variation of cowl-length ratio with radius ratio for DAC-1 cowl shapes for approximately constant cowl nose curvature is shown in Figure 44. Also shown in Figure 44 is the external cowl design point of R_{HL}/R_{max} = 0.90, X/D_{max} = 0.22. With the inlet area sized for 0.79 inlet throat Mach number at static takeoff-power conditions and internal lip-thickness ratio of R_{HL}/R_i = 1.21, an external cowl radius ratio of R_{HL}/R_{max} = 0.90 resulted from an R_{max} = 100.1 cm (39.4 in.). This value of R_{max} was the lowest possible as set by the engine cowling size requirements and therefore resulted in the minimum inlet weight. Inlet and cowl lines for the design described above are shown in Figure 45.

The external cowl drag characteristics at cruise conditions are shown in Figure 46. The design-point cowl had a small amount of shock drag at cruise that could be eliminated by increasing R_{max}. This was not done however, since the potential drag reduction would only be about 0.3 percent of airplane drag, equivalent to about 0.05 percent reduction in airplane direct operating costs. Also, some of this potential drag reduction would be offset by increased inlet weight resulting from the larger size. External cowl drag was estimated from Douglas tests of isolated cowlings in the Calspan wind tunnel where the drag was determined from wake survey measurements.

Proper design of the external cowl shape was necessary for the prevention of boundary-layer separation from the leading edge of the top part of the nacelle during engine-out takeoff climb. The relatively low mass-flow ratio for the windmilling engine combined with the inlet angle of attack at climb cause the cowl of the windmilling engine to be the most critical. Separation-free flow results in lower airplane drag during climb, higher climb gradient and, therefore, less distance between the lift-off point and the 10.7 m (35-foot) obstacle height. The engine-out takeoff field length (defined as ground run distance plus air distance to the obstacle)

Most Adverse Airplane Stall Condition

$\alpha = 44°$

$M_o = 0.18$

$R_{HL}/R_{Max.} = 0.905$

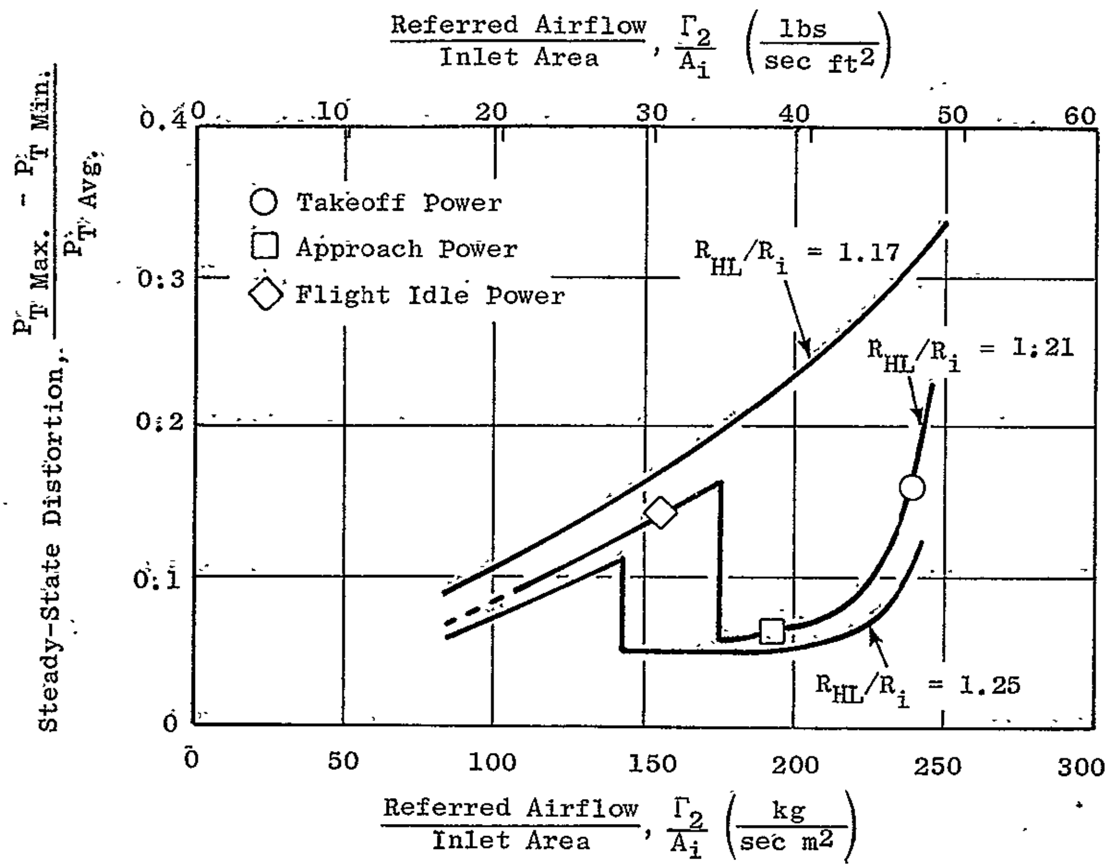


Figure 43. Inlet, Angle of Attack (Test Data $\alpha = 44°$, $M_o = 0.18$).

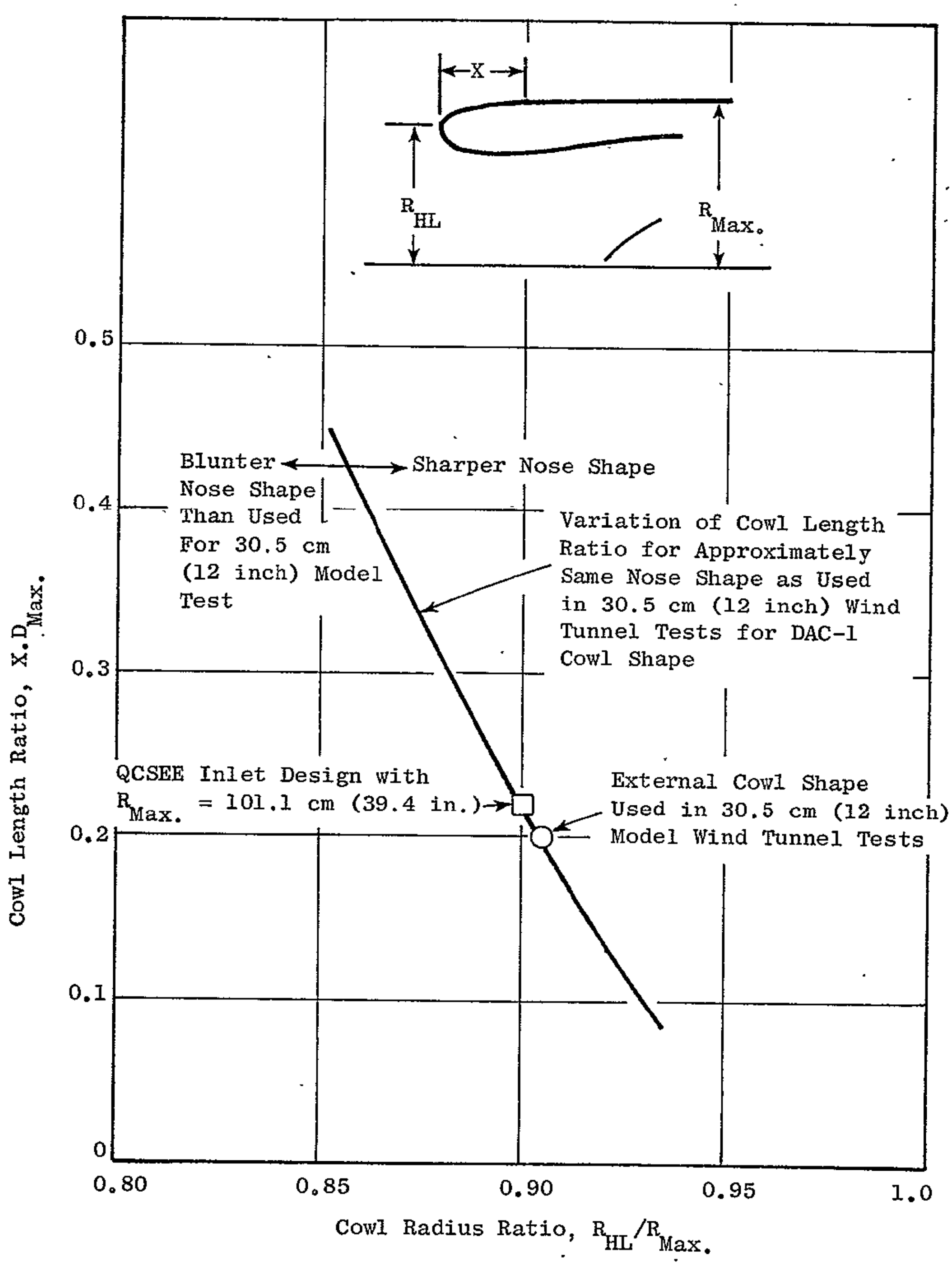


Figure 44. Inlet, Cowl Length to Radius Ratio ($R_{HL}/R_{Max.}$).

Internal Lip is 2:1 Ellipse, $R_{HL}/R_i = 1.21$

External Cowl is DAC-1, $R_{HL}/R_{Max.} = 0.90$, $X/D_{Max.} = 0.22$

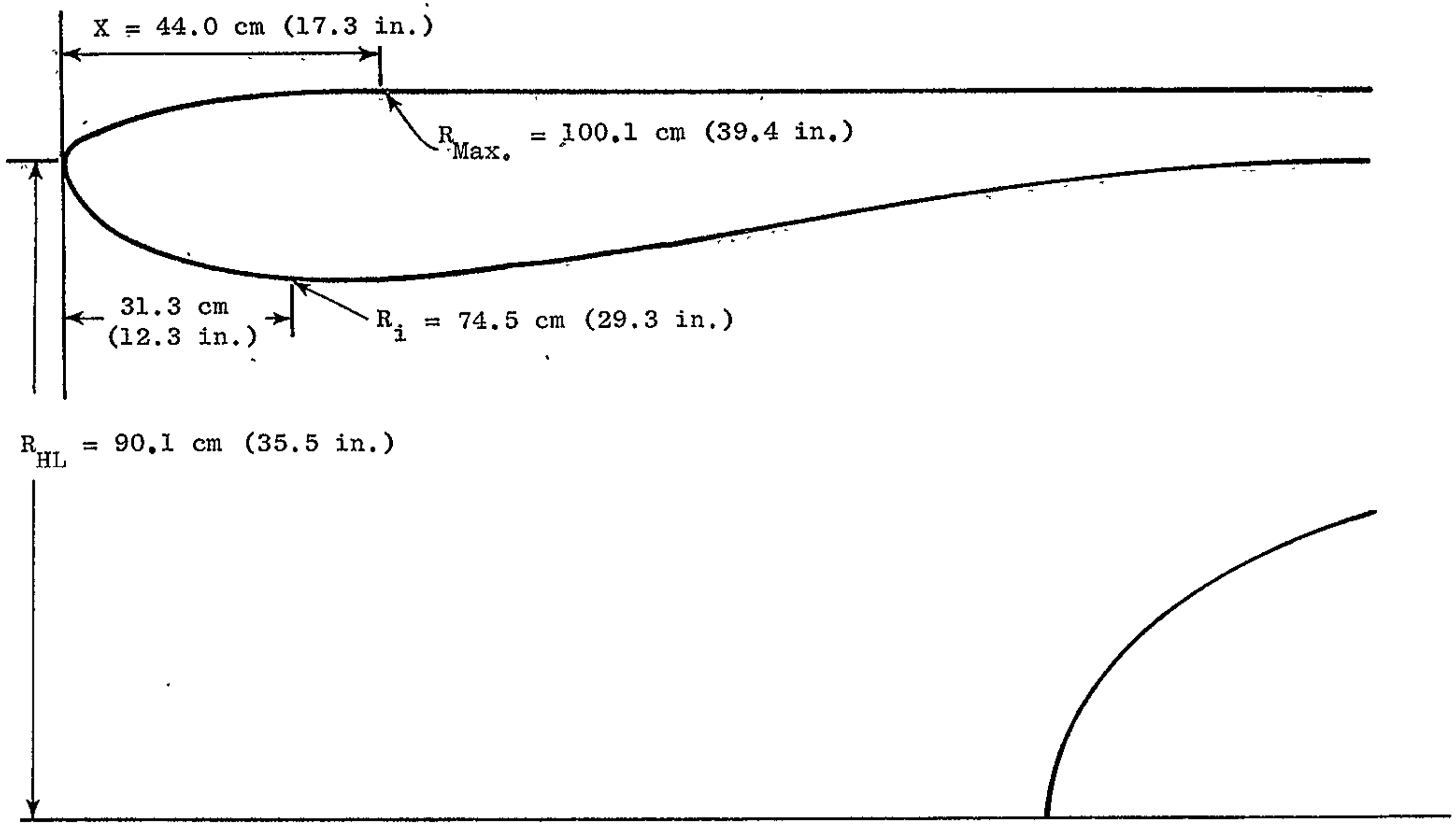


Figure 45. Inlet, Lines (High Mach).

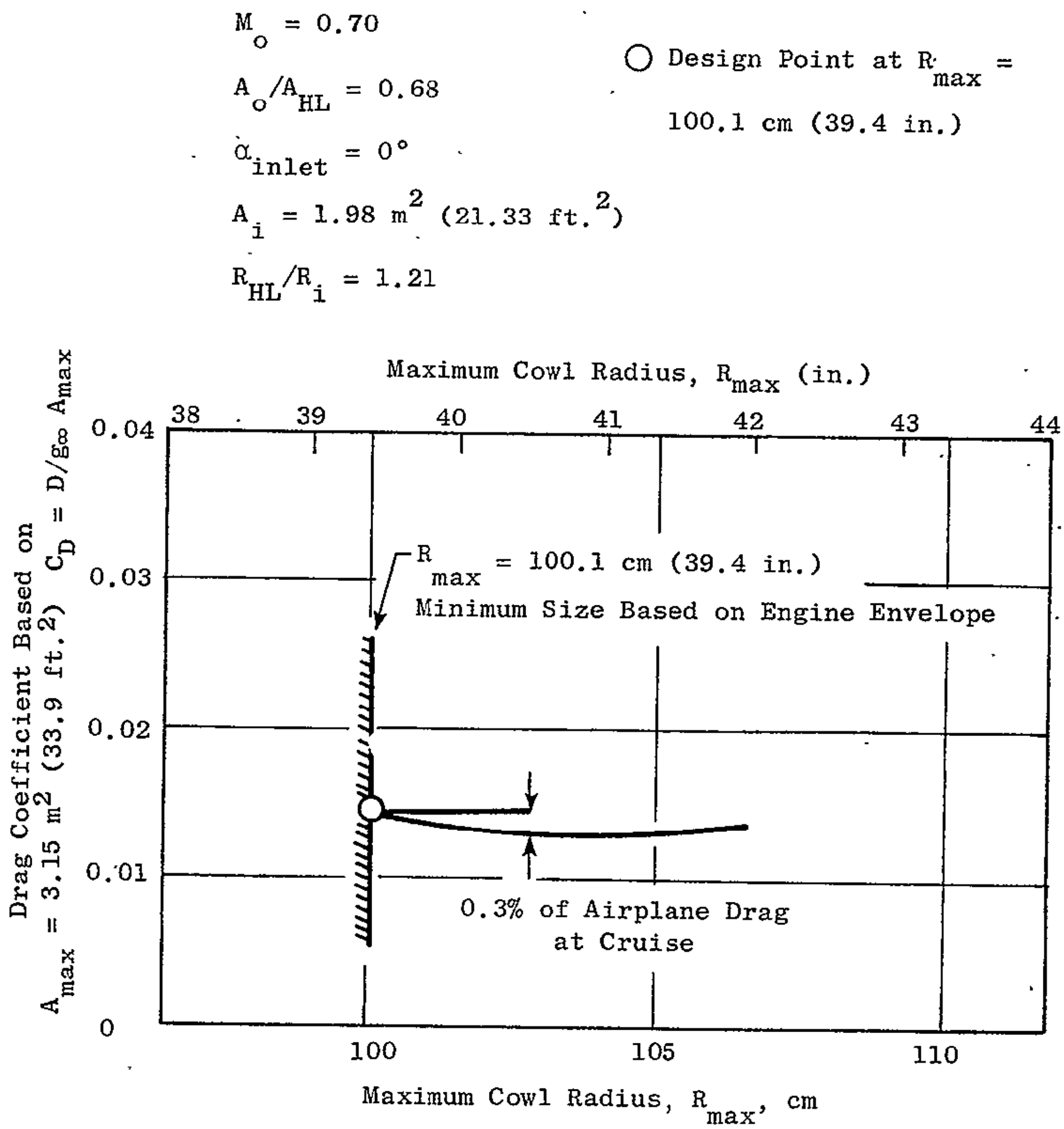


Figure 46. Inlet Drag, Cruise.

is therefore reduced if separation does not occur. Figure 47 shows estimated separation conditions compared to QCSEE operating conditions during engine-out climb. The QCSEE cowl should have attached flow at these conditions. Separation conditions were estimated by calculation of the potential-flow peak-pressure coefficient corresponding to cowl separation during Douglas DC-10 stall tests and the approximation that this critical value of peak pressure coefficient was independent of the cowl shape and inlet mass-flow ratio.

4.3.5 Nacelle Drag Estimates

Nacelle drag estimates were made so that airplane performance calculations would include all installation effects and also, to compare the installed performance of a nacelle with accessories mounted beneath the engine in the fan cowl with the drag of the baseline nacelle with the accessories mounted in the pylon.

The drag of all parts of the nacelle affected by the engine airflow was calculated by GE. Douglas calculated the external drag of the fan cowl and the drag of the part of the pylon that was scrubbed by the free-stream air. GE accounted for almost all of the pylon scrubbed by the engine airflow, but the Douglas-recommended pylon configuration had slightly more wetted area in the fan stream than did the GE pylon and Douglas accounted for this increment.

The fan-cowl drag was calculated as

$$(D/q_\infty)_{\text{fan cowl}} = (Kc_f S_w)_{\text{fan cowl}}$$

$$= (1.29)\ (.00239)\ (23.8\ \text{m}^2)$$

$$= 0.073\ \text{m}^2\ (0.79\ \text{ft}^2)\ \text{per engine}$$

S_w is the fan cowl wetted area. C_f is the average skin-friction coefficient on the fan cowl determined from Reference 8 at cruise conditions of $M_\infty = 0.7$ and 9450 m (31,000 ft) altitude. In Reference 8, average skin-friction coefficients have been determined for various amounts of surface roughness. The drag of small surface imperfections on the fan cowl was taken into account by assuming an equivalent sand-grain roughness of 0.0024 cm (0.00095 inch) which is representative of current technology jet transports.

The form factor, K, was determined as a function of the equivalent length-to-diameter ratio for the nacelle.

$$(L/D)_{eq} = \frac{\text{total nacelle length + inlet high-light diameter}}{\text{equivalent diameter}}$$

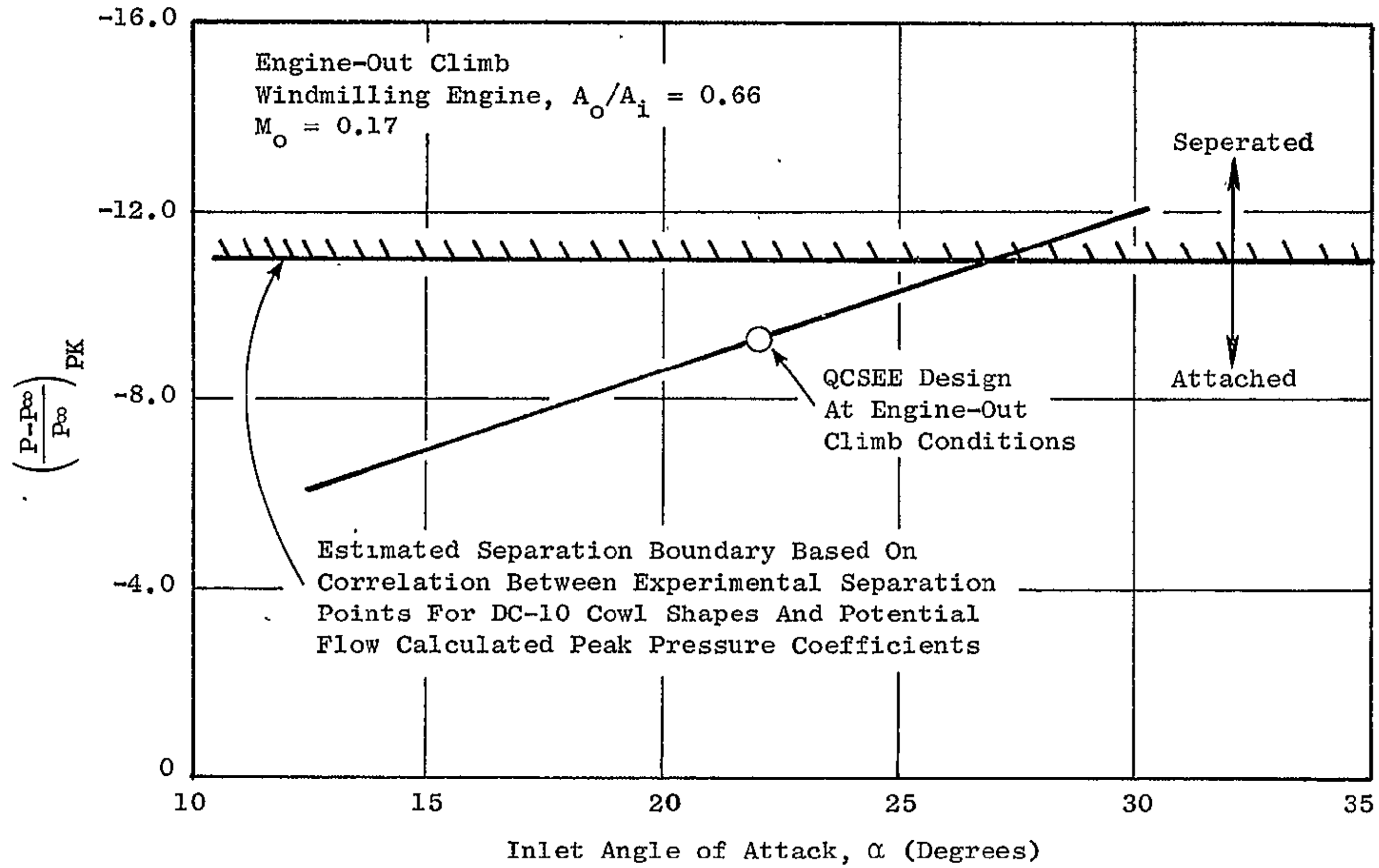


Figure 47. Inlet Flow Coefficient, Engine Out Climb.

The total nacelle length was the length from the inlet leading edge to the trailing edge of the core-nozzle plug. The equivalent diameter was based on the area difference between the maximum frontal area of the nacelle and average of the total engine exit area and the freestream area of the airflow that entered the inlet, or

$$\text{equivalent diameter} = 2 \left[\frac{A_{max} - 1/2\ (A_e + A_\infty)}{\pi} \right]^{1/2}$$

The variation of K with $(L/D)_{eq}$ was determined from full-scale and model-scale drag data for nacelles and various axisymmetric closed bodies.

The pylon drag was calculated as

$$(D/q_\infty)_{pylon} = (Kc_f S_w)_{pylon}$$

$$= (1.05)\ (0.00252)\ (9.4\ m^2)$$

$$= 0.025\ m^2\ (0.27\ ft^2)$$

The average skin-friction coefficient was determined from Reference 8 as described above. The pylon form factor was determined from Reference 9 using form factors for swept wings.

The incremental pylon drag for the part of the pylon scrubbed in fan-jet flow was

$$(\Delta D/q_F)_{pylon} = (c_f \Delta S_w)_{pylon}$$

$$= (0.00255)\ (0.79\ m^2)$$

$$= 0.002\ m^2\ (0.022\ ft^2)\ \text{per engine.}$$

$$(\Delta D/q_\infty)_{pylon} = 0.022\ q_F/q_\infty = 0.0037\ m^2\ (0.04\ ft^2)\ \text{per engine at cruise conditions}$$

4.3.6 Nacelle Placement

Figure 48 shows the nacelle position for the QCSEE-powered airplane. The engine axis was made parallel to the wing reference plane. The fan-cowl trailing edge was located forward of the wing leading edge by 20 percent of the wing chord at the inboard engine section. The nacelle was located below the wing so that the outer edge of the engine jet did not hit on the drooped leading edge of the wing but did graze on the wing lower surface ahead of the flap.

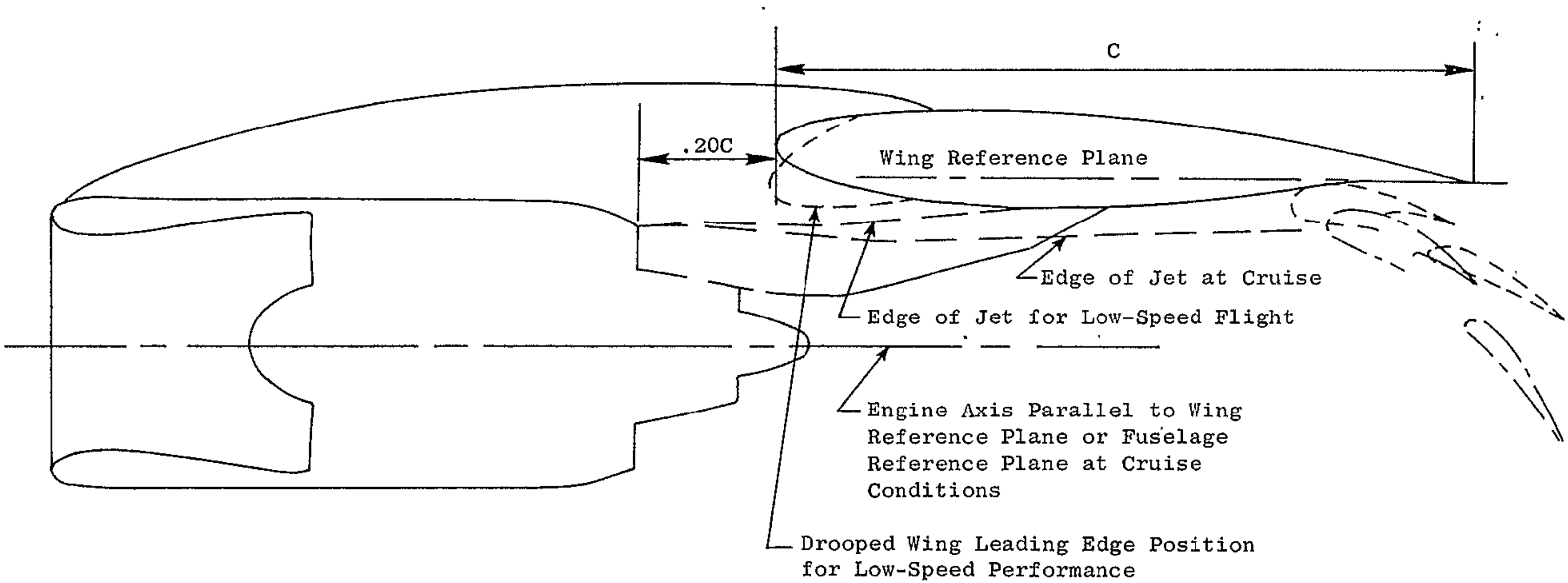


Figure 48. Wing/Propulsion System Location, Inboard.

In defining this engine location both low-speed and cruise performance considerations were taken into account, and the nozzle area at approach was assumed to be only slightly greater than the takeoff value. During the low-speed operation the flap system considerably influenced the engine position required for good engine exhaust turning characteristics and for high levels of powered lift. The DAC-developed STOL two-segment flap system used on the QCSEE configuration provided for significant downward flap displacement upon flap deflection. This enabled the flap system to effectively engage the exhaust flow of the high-bypass-ratio QCSEE engine without having to tilt the engine up toward the flap system. Extensive DAC static- and low-speed STOL wind tunnel testing utilizing ejector engine simulators has been conducted to investigate the aerodynamic effect of engine position relative to the wing and DAC two-segment flap (Reference 10 through 12). These tests were performed for both takeoff and landing flap settings and showed that good aerodynamic performance could be achieved without tilting the engine tailpipe up toward the flap system.

Upward tilt of the engine or nozzle would result in cruise performance penalties as shown by Figure 49. The estimated best alignment for the engine axis at cruise conditions was 0 to 1 degree (engine tailpipe down) with respect to the wing or fuselage reference plane. The thrust and drag increments shown in Figure 49 include wing induced drag changes, wing compressibility drag changes, thrust loss due to exhaust jet inclination, nacelle-induced drag, and cowl forebody compressibility drag. Wing-induced drag and compressibility drag changes resulted from changes in the wing lift coefficient required to offset changes in lift due to tilting of the engine thrust axis with respect to the freestream velocity vector, for cruise flight at constant total lift coefficient. The thrust loss due to exhaust jet inclination was estimated as

$$\frac{\Delta F_N}{F_N} = \frac{F_g}{F_N} [1 - \cos (\alpha_F + \theta)]$$

where ΔF_N is decrement of net thrust due to tilt. These increments were zero for $\theta = -\alpha_F$ since this engine-axis alignment would make the exhaust stream parallel to the freestream velocity vapor.

The nacelle-induced drag and cowl compressibility drag resulted from the nacelle not being aligned with the local flow. These increments were estimated to be zero at zero inlet angle of attack ($\theta = -\alpha$). The cowl compressibility drag, estimated to be about 0.25 percent of cruise net thrust at θ - 0 degree could be eliminated by bending the inlet down by 1.8 degrees with respect to the engine centerline to align the inlet with the local flow.

Cruise Conditions: $M_o = 0.7$

$C_l = 0.4$

Airplane Angle of Attack, $\alpha_F = 1.5°$

Inlet Angle of Attack, $\alpha_i = 1.8°$

<u>YC-15 Aerodynamic Characteristics</u>

Thrust Loss Due to Exhaust Jet Inclination, $\frac{\Delta F_N}{F_N} = \frac{F_g}{F_N}\left[1 - \cos(\alpha_F + \theta)\right]$

Nacelle Induced Drag

Cowl Forebody Compressability Drag (Could be Eliminated by Inlet Bend to Align Inlet with Approach Flow Direction)

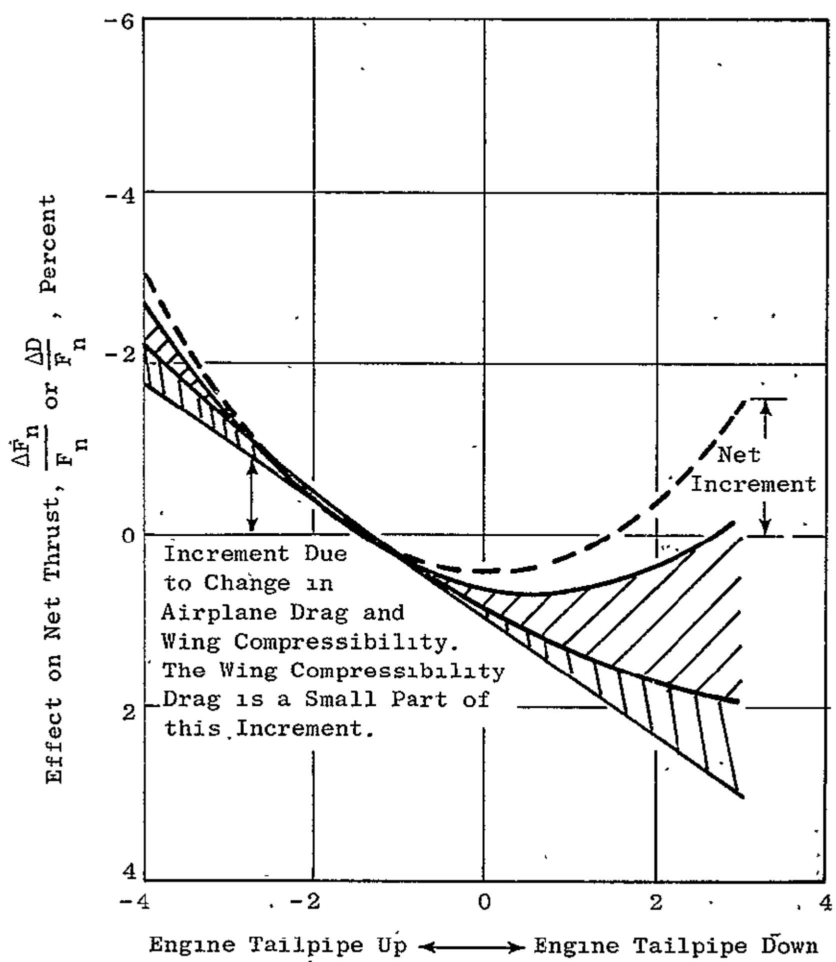


Figure 49. Engine Alignment, Net Thrust at Cruise.

The vertical engine location for achieving good zero-tilt low-speed performance required that the fan flow miss the deflected leading-edge device and graze the wing lower surface ahead of the first flap slot. This ensured that high-energy engine air entered the slot which resulted in efficient flow turning and good powered-lift characteristics. At cruise conditions, this engine position was estimated to have no adverse interference effects on the wing lower-surface flow. This estimate was based on Douglas YC-15 model tests where wing lower-surface static-pressure distributions were measured with and without the presence of an engine jet. No indication of large decreases in local pressures on the wing was noted. Increased skin-friction drag on the wing lower surface was estimated to be small since the size of the engine jet at cruise conditions was smaller than for low-speed operation and jet impingement would not occur at cruise conditions (Figure 48).

Wind-tunnel data also indicated that for high-bypass ratio engines (large fan diameter relative to wing chord) maximum lift was sensitive to fore and aft engine location. The fan exit should be about 20 percent of wing chord in front of the leading edge in order to minimize detrimental fan-flow/leading-edge interference on maximum lift. Cruise considerations also favored the 20-percent wing-chord forward position since this position was far enough forward to result in insignificant wing/nacelle interference drag penalties.

4.3.7 Aerodynamic Loads on Inlet Cowl

The aerodynamic loads analysis consisted of definition of the design criteria and attendant QCSEE inlet cowl loads for use in the strength analysis of the inlet-to-engine-frame interface.

The Douglas YC-15 aerodynamic characteristics, flight envelope, and maximum design speed were used to establish the design criteria and resulting loads conditions for analysis. The YC-15 maximum flight design speed and maneuvering envelope at a design gross weight of 68040 kg (150,000 lbs) are presented for reference as Figures 50 and 51, respectively. As a result of that definition the six flight conditions and two takeoff conditions were selected for structural analyses as shown in Table XVII.

A summary of the estimated aerodynamic loads on the inlet cowl of the NASA QCSEE type engine for the specified design conditions is presented in Table XVIII.

The inlet-cowl loads are based on estimated pressure distributions on an axisymmetric body, obtained by using the streamflow, static, and crossflow potential-flow solutions. The component potential flow are combined to satisfy the onset-flow angles, and the inlet mass flow and pressures are computed at 30-degree radials around the cowl. At each radial, summary of the force and moment functions is made from which the net radial forces and moments are found by summing the component radial values.

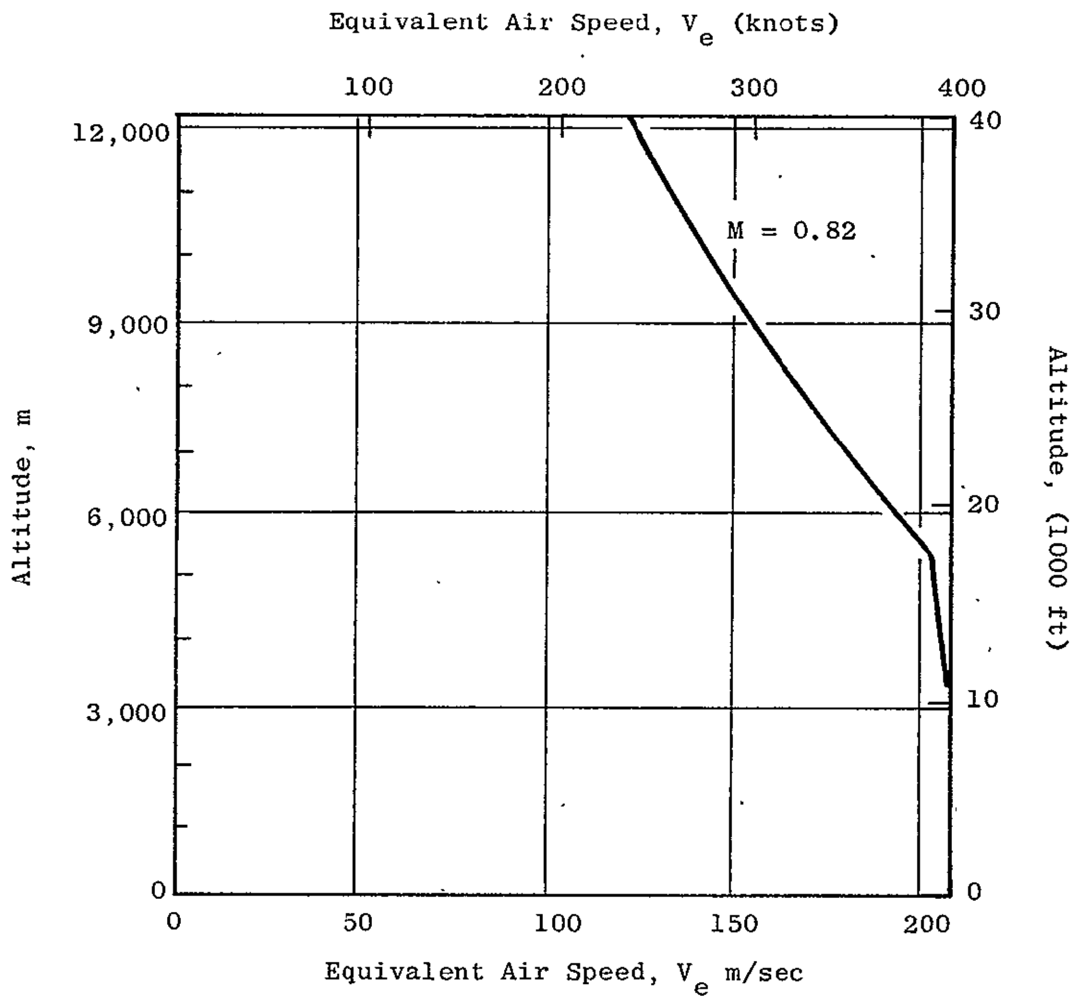


Figure 50. Maximum Flight Design Speed, DAC YC-15.

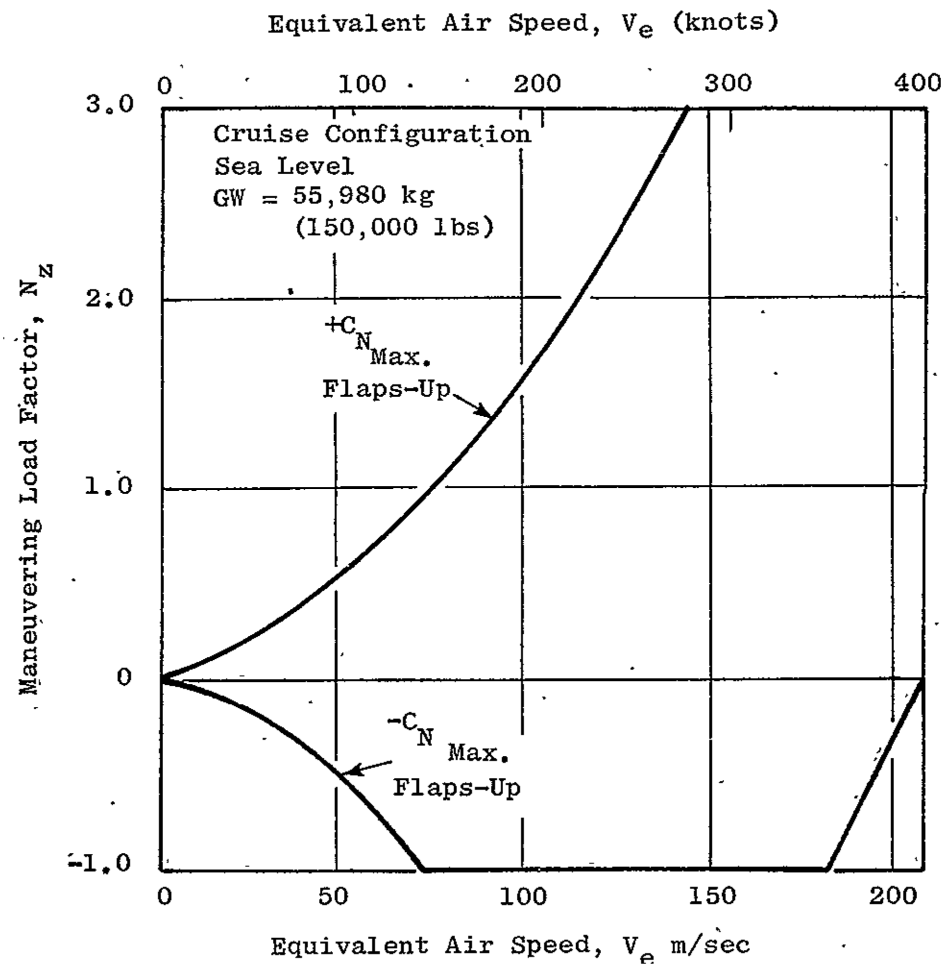


Figure 51. Maneuver Envelope, DAC YC-15.

Table XVII. Maximum Inlet Load Conditions.

Case	Flight Condition	M_o	Altitude		Velocity		αF	βF	n_z	n_y	p
			kft	km	Knots (Des)	m/sec	(Deg)	(Deg)	(g)	(g)	(Deg/sec)
1	Abrupt Roll	0.751	18	5.5	350	180	0.8	-3.9	2.4	2.45	85.6
2	Max. Speed	0.605	SL	SL	400	206	0	0	1.0	0	0
3	3 g Pull-Up @ Max. q	0.605	SL	SL	400	206	5.9	0	3.0	0	0
4	3 g Stall	0.40	SL	SL	265	136	12.5	0	3.0	0	0
5	-1 g @ Max.q	0.528	SL	SL	350	180	-3.9	0	-1.0	0	0
6	Max. β_q	0.559	SL	SL	370	190	0.5	6	1.0	-1.2	0
7	Takeoff Thrust 35 knot Crosswind	0	SL	SL	0	0	0	0	---	---	---

αF = angle of attack of FRP

βF = angle of sideslip of FRP (positive airplane nose left)

M_o = freestream Mach number

p = rate of roll

n_z = load factor, perpendicular to FRP, positive up

n_y = lateral load factor, perpendicular to plane of symmetry, positive our right wing

Table XVIII. Estimated Inlet Load Conditions.

Case	Altitude km (Ft)	Freestream M_o	α_F (Deg)	β_F (Deg)	Power Setting	Engine Inlet M_I	α_c (Deg)	β_c (Deg)	Forces N (Lbs) F_x	F_y	F_z	Moments, cmN (In-Lbs) M_y	M_z
1	5.5 (1810)	0.751	0.8	-3.9	MCT	0.789	1.5	-3.9	-6254 (-1406)	13804 (3103.5)	3029 (681)	172608 (15,275)	1928221 (170,639)
2	S.L.	0.605	0	0	MCT	0.789	0.4	0	-605 (-136)	0	-765 (-172)	-408145 (-36,119)	0
3	S.L.	0.605	5.9	0	MCT	0.789	9.0	0	-3990 (-897)	0	36767 (8266)	5216204 (461,611)	0
4	S.L.	0.40	12.5	0	MCT	0.789	18.4	0	-5129 (-1153)	0	37119 (8345)	6277014 (555,488)	0
5	S.L.	0.528	-3.9	0	MCT	0.789	-4.9	0	979 (220)	0	-18504 (-4160)	-3087996 (-273,274)	0
6	S.L.	0.559	0.5	6.0	MCT	0.789	1.2	6.0	-302 (-68)	-22578 (-5076)	2438 (548)	127961 (11,324)	-3451099 (-305,407)
7	S.L.	0	0	0	TO	0.789	0	0	-24064 (-5410)	0	632 (142)	114096 (10,097)	0
8	S.L.	0.053	0	90	TO	0.789	0	90	-34227 (-7695)	-6774 (-1523)	418 (94)	123419 (10,922)	-1818656 (-160,943)

Note: Forces and moments based on ambient pressures
Bulkhead forces and moments not included in above table

The nacelle inlet axis orientation, sign convention of forces and moments, and cowl reference data are presented on Figure 52.

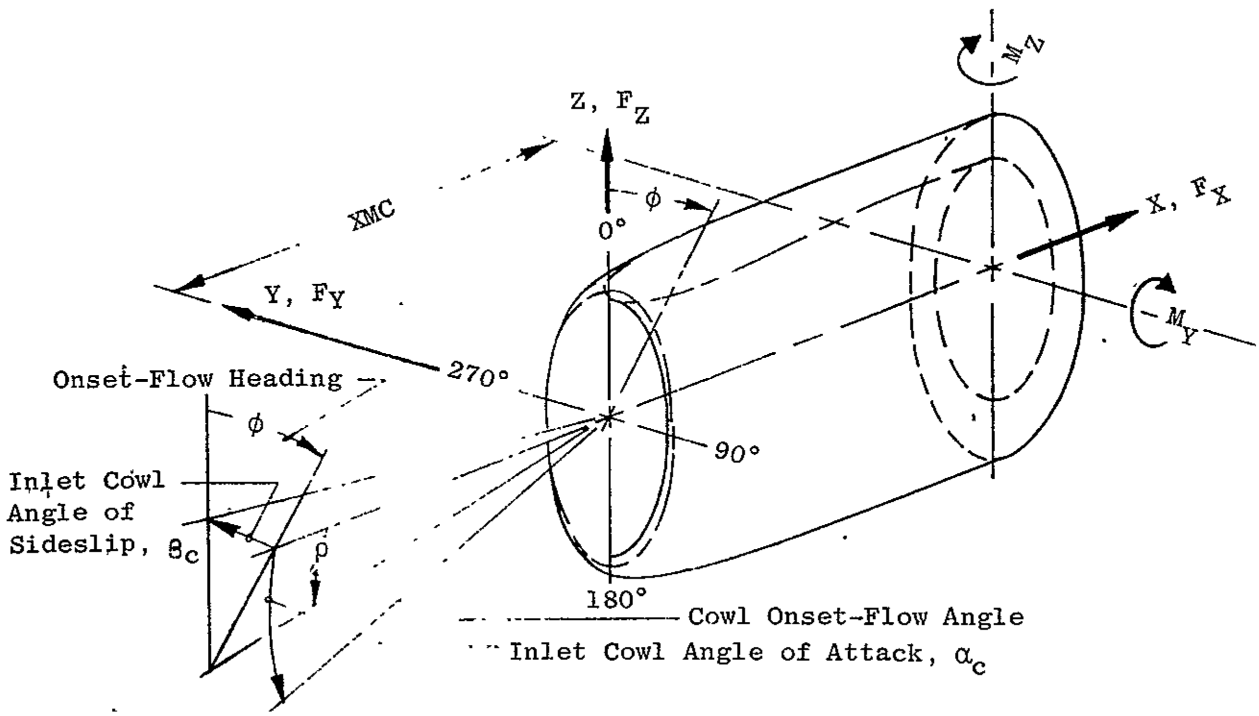
4.4 THRUST SETTINGS

4.4.1 Ground Idle Thrust

The selection of ground idle thrust level is based on a tradeoff between requirements for taxi thrust levels and exhaust emissions standards. A low ground idle thrust minimizes brake wear, reduces pilot work-load during ground handling, and reduces landing stopping distances. On the other hand, exhaust emission standards are hard to meet at low thrust settings due to poor combustion efficiency. Figure 53 compares the aircraft static thrust-to-weight ratio with ground idle thrust for the QCSEE aircraft with the DC-9 and DC-10. Thrust-to-weight ratios are shown for a weight range from the operators-empty-weight to maximum takeoff gross weight. The DC-9-30 and DC-10-30 are considered to have satisfactory idle-thrust levels from the ground-handling point of view. On QCSEE, a ground idle thrust level equal to 4-1/2 percent of the static takeoff thrust would ease the emissions problem and still provide a manageable amount of idle thrust.

4.4.2 Reverse Thrust

Although the use of reverse thrust may not be considered in determining landing field length, it is required for landing on icy runways or in the event of brake failure. Figure 54 shows distance to stop from touchdown plotted as a function of reversing effectiveness for a landing without brakes. Reverse thrust is assumed to be effective from 1 second after touchdown down to a speed of 5.1 m/sec (10 knots). For a 914.4 m (3000 ft) runway, approximately 121.9 m (400 ft) are consumed during the flare maneuver leaving 792.5 m (2600 ft) of runway to bring the aircraft to a stop. A thrust reverser effectiveness of 0.3 to 0.4 is required to prevent an overshoot. Accuracy in estimating stopping distances is limited by the uncertainty in airplane lift and drag in ground effect with reverse thrust. A design requirement for 35 percent reverses effectiveness should not be relaxed until data become available for more accurate calculations.



Left-Hand Inlet Cowl

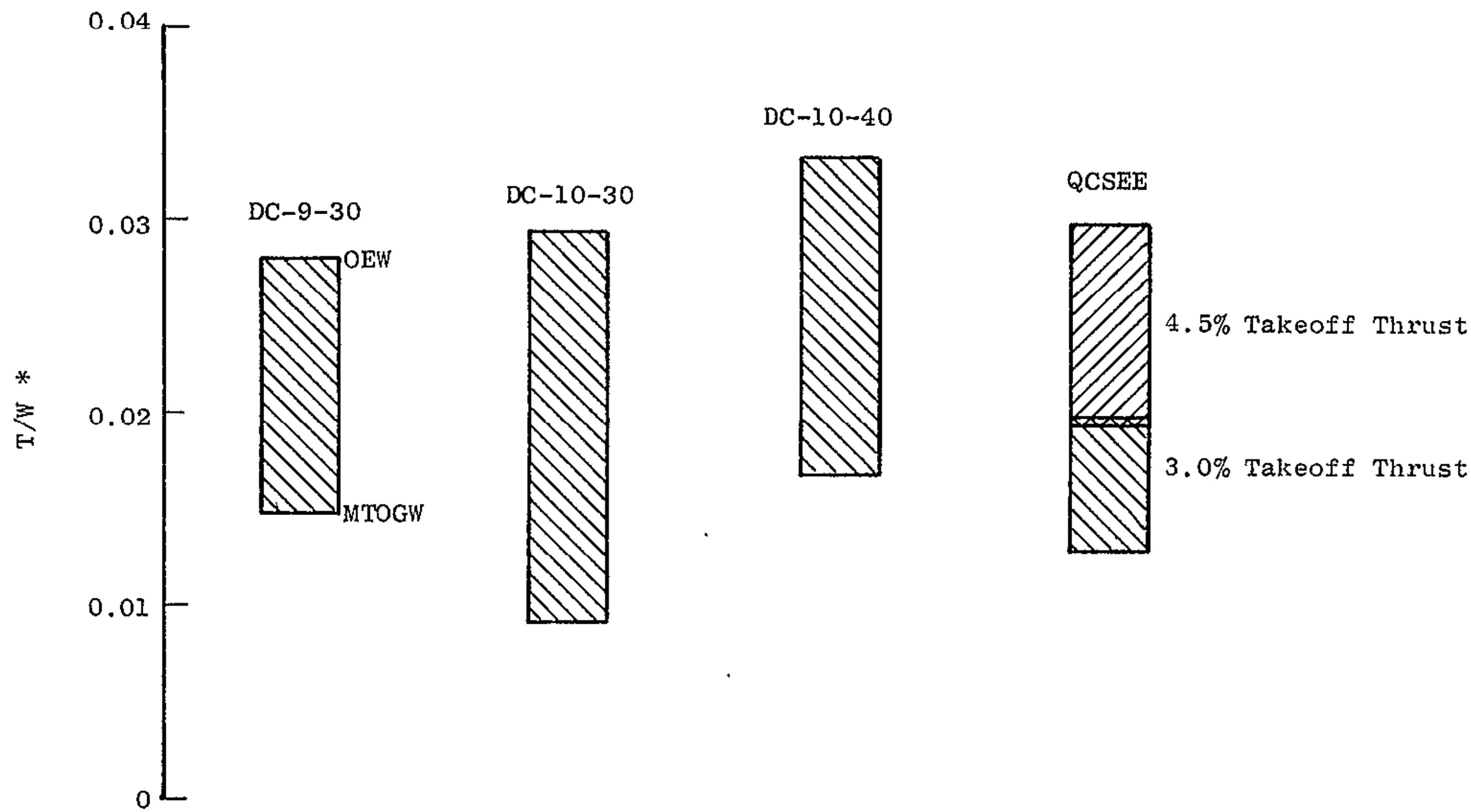
The body is a left-hand inlet cowl. The origin of coordinates is taken on the cowl centerline at the plane of the cowl leading edge. Positive forces correspond to positive coordinate directions. Moments are computed about the Y and Z axes as translated a distance XMC to the bulkhead station. Angles, forces, and moments are positive as shown. The inlet cowl angles α_c and β_c are determined by applying local-flow and orientation corrections to α_F and β_F

Inlet Cowl Reference Data

Inlet Area = 1.77 m^2 (19.03 sq ft)

Bulkhead Station @ 1.8 m (71.06 in.) Aft of Cowl Leading Edge

Figure 52. Axis Orientation and Sign Convention.



* Ground Idle, Sea Level, Static

Figure 53. Ground Idle Thrust.

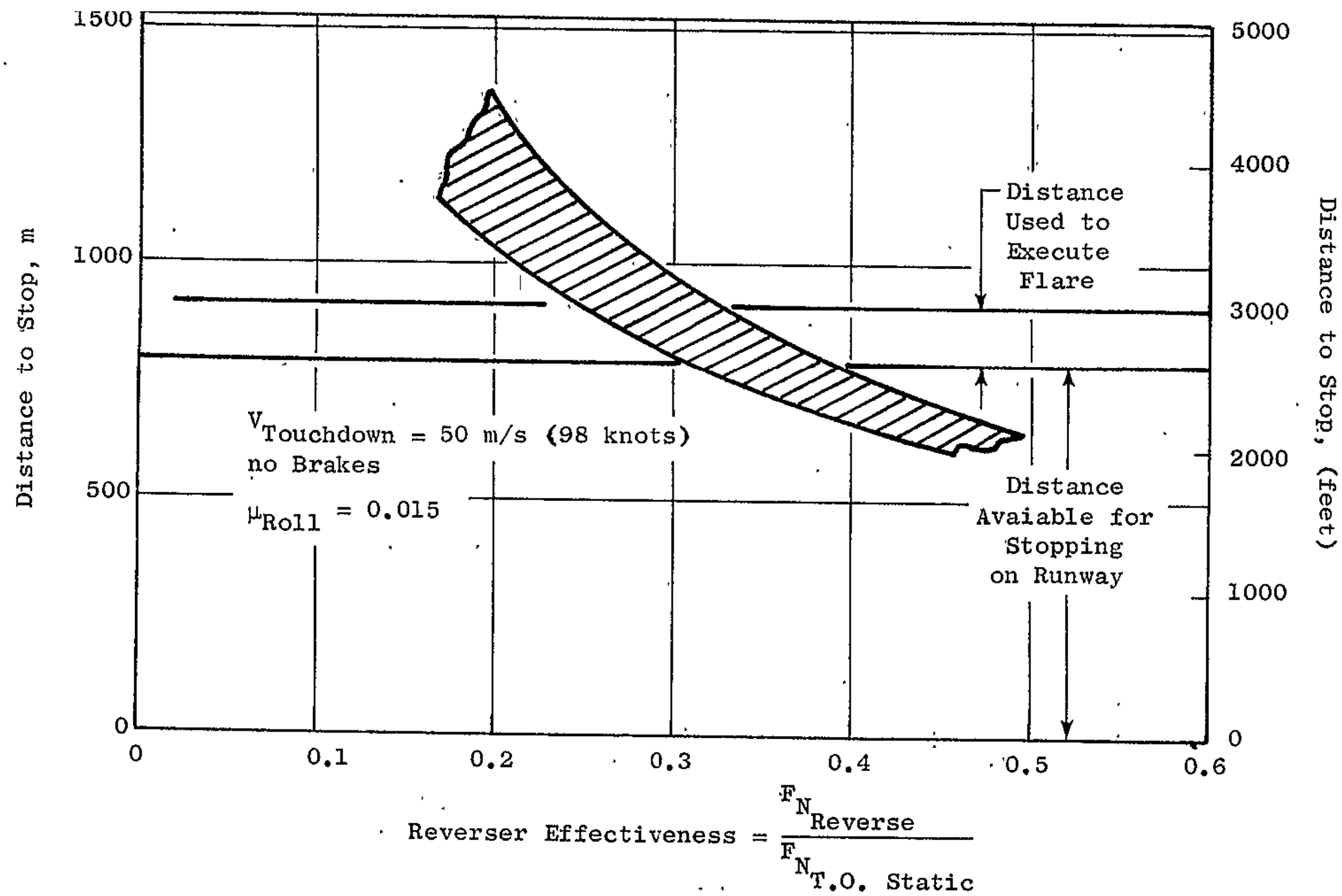


Figure 54. Reverse Thrust.

5.0 NACELLE COMPONENTS AND SYSTEMS

5.1 UTW FLIGHT PROPULSION SYSTEM

The "flight" under-the-wing propulsion system installation is shown in the Douglas scope drawing, Figure 17, and the preliminary outline/installation drawing. Figure 55. The major nacelle components consist of the inlet, including the extension of the accessory cover at the top; the fan cowl assembly (right and left halves), which includes the flare nozzle and actuation system; the core cowl; and the core exhaust nozzle and plug. All components are axisymmetric and have acoustic treatment as an integral part of the structural walls. The nacelle maximum diameter is 200.2 cm (78.8 in) with an overall length from the inlet "highlight" to the tip of the core exhaust nozzle (engine station 304.5) of 535 cm (210.7 in.) and 414 cm (163.0 in.) from the inlet to the fan exhaust plane (engine station 256.5).

The under-the-wing propulsion system installation on the high-wing, short-haul aircraft (see Figure 9) results in a height to the bottom of the nacelle of approximately 3.0 m (10 ft). To reach engine accessories, a maintenance stand is required whether the accessory compartment is on the top or bottom of the engine. This fact in addition to several significant advantages, presented below, in combination with the integrated nacelle concept lead to the selection of the top-mounted accessory gearbox.

- Shortens configuration hardware (tubes, ducts, cables, wires, etc.) - with top accessories there is the minimum distance from the engine to the engine accessories and then on to the aircraft interconnect points.

- Eliminates need for fan casing mounted installation hardware - on the DC-10/CF6, typical under-the-wing installation, all pipes, wires and cables are routed around the fan case to the bottom-mounted accessories resulting in brackets, additional connections, and other casing-mounted hardware.

- Integrated engine/nacelle structure - The high Mach inlet throat reduces required throat area, resulting in a smaller inlet diameter. Inlet throat to highlight diameter ratio required for good crosswind flow capability and highlight to maximum diameter ratio consistent with aircraft Mach number requirements permits nacelle thickness and nacelle maximum diameter to be kept to a minimum.

- Eliminates fan cowl over fan casing - The above changes permit integration of the fan cowl, a nacelle component, into the engine structure. One less nacelle component saves cost and weight. This also permits thinner nacelle walls - approximately 10.2 cm

(4 in.), compared to 25.4 cm (10 in.) at the top and sides and 50.8 cm (20 in.) thickness at the bottom of the nacelle on the DC-10/CF6 installation.

- Eliminates bottom pylon - With no bottom accessories, no bottom pylon is required. Both top and bottom accessory, under-the-wing installations require a top pylon.

- Reduced "projected" frontal area - The top accessory cover fits in the silhouette of the aircraft pylon and eliminates the bottom nacelle bulge, typical for systems with accessories mounted under the engine.

The propulsion system aerodynamic flowpath is shown in Figure 56. The cross-hatched sections indicate the acoustically treated areas. No fan exhaust splitter or core nozzle wall treatment for noise suppression is required to meet the acoustic objective of 95 EPNdB with the baseline aircraft for a 914.4 m (3000 ft) field length. Other features shown in Figure 56 are the aircraft pylon cover extended forward to form an aerodynamic-shaped housing for the engine gearbox and accessories; the separately mounted aircraft gearbox and its accessories; digital control; lube oil cooling and oil tank; and the inlet anti-icing system. Also indicated is the variable area flare nozzle shown in the cruise (minimum area), takeoff and reverse thrust positions. With the variable-pitch fan blade the flare nozzle becomes the system inlet for reverse thrust operation.

5.2 REQUIREMENTS

The flight propulsion system for the quiet, clean, short-haul aircraft will be designed to meet the aircraft requirements shown in Section 3.1 and in addition meet the following engine system requirements.

5.2.1 Noise

95 EPNdB - 152.4 m (500 ft) sideline, during takeoff and approach.
100 PNdB - 152.4 m (500 ft) sideline, during maximum reverse thrust.

The above objectives are based on a four-engined 400,320 N (90,000 lb) SLS thrust aircraft.

5.2.2 Oil Consumption

0.9 kg/hr (2 lb/hr) maximum.

5.2.3 Dumping

No fluids shall be dumped under normal engine operation. Dumping may occur in case of abnormal operation such as seal failure.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Dimensions in cm
( ) Dimensions in inches

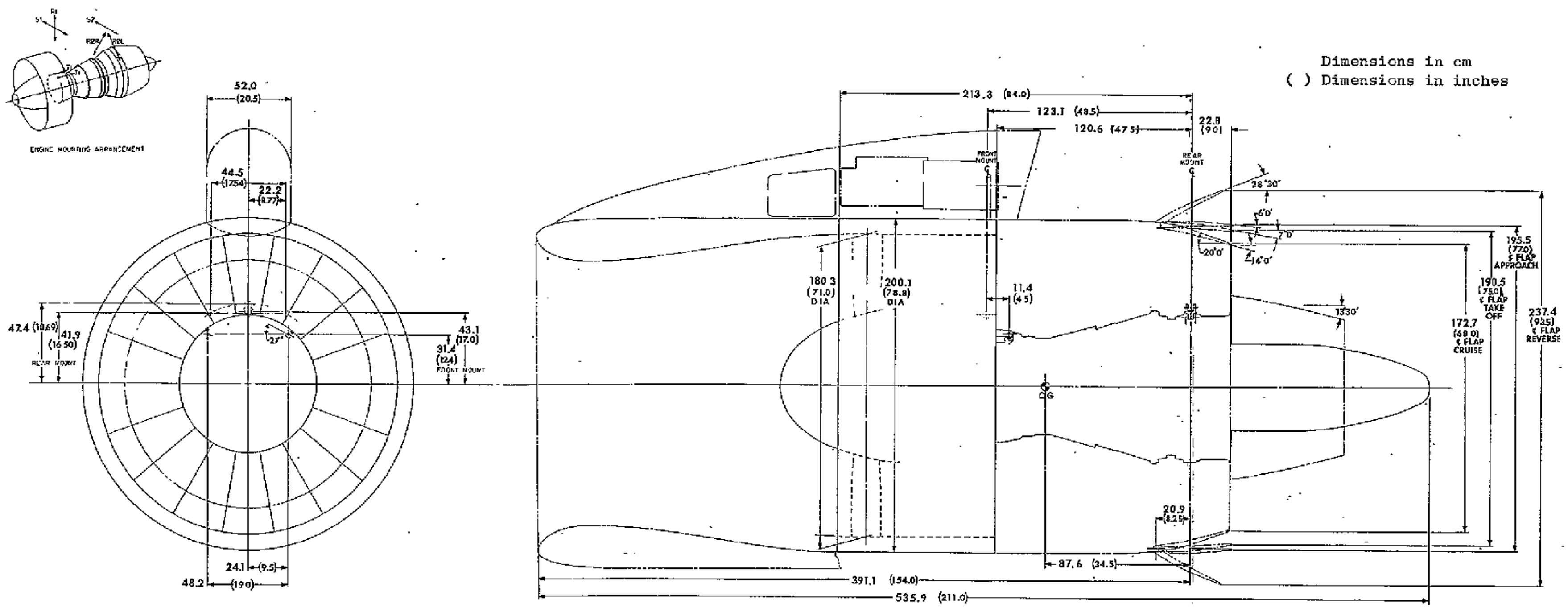


Figure 55. Preliminary Installation Outline.

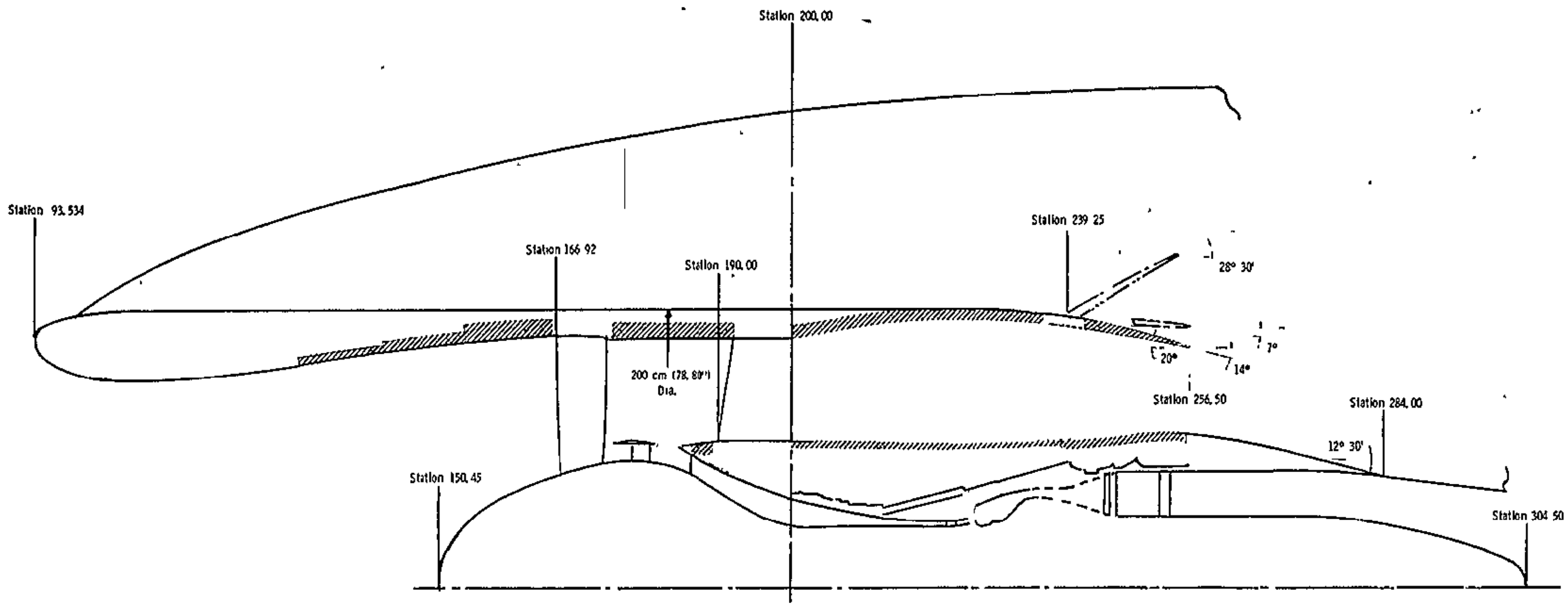


Figure 56. UTW Flight Flowpath.

5.2.4 Life and Duty Cycle

The engines shall be designed for a useful life of 36,000 hours over a 15 year period, based on the typical 402.3 km (250 mile) mission cycle shown in Table XIX.

Cycle life shall be based on 48,000 mission cycles plus 1000 ground checkout cycles to full power. An aborted takeoff rate of 0.1 per 1000 take offs and an aborted landing rate of 0.1 go-around per 1000 flights should be assumed.

The above missions shall occur over the normal sea level and altitude ambient temperature distributions shown in Figure 57 and 58. The engine shall be capable of operation throughout the flight envelope shown in Figure 59. The corresponding Pt2-Tt2 envelope for a - 7.8° C (+18° F) day is provided in Figure 60. Inlet characteristics are shown in Figure 61 at various flight Mach numbers for several inlet throat Mach numbers based on a total fan flow of 405.5 kg/sec (894 lb/sec).

5.2.5 Flight Maneuvers

- The engine and its supports shall withstand without permanent deformation the conditions specified on Figure 62 (MIL-E-5007C except for precession rates). The calculated weight of the engine and engine-mounted nacelle components shall be increased by the specified weight allowed for all engine-mounted accessories.
- The engine and its supports shall withstand without failure static loads equivalent to 1.5 times the flight limit specified above for metal parts and 3.0 times for composite parts.
- At maximum allowable engine speed, the engine shall withstand without permanent deformation a gyroscopic moment imposed by a steady angular velocity of 1.0 radian per second in yaw, combined with a vertical load factor of $\pm$ 1, for 15 seconds.
- The engine shall be capable of withstanding loads caused by seizure of either rotor with deceleration from maximum rpm to zero rpm in one second.
- Composite parts shall be capable of withstanding unbalance loads caused by the loss of five adjacent composite fan blade airfoils at rated rpm. Metal parts shall be capable of withstanding unbalance loads caused by the loss of 2-1/2 adjacent composite fan blade airfoils at rated rpm.

5.2.6 Flight Attitudes

The engine shall be capable of operating within the range of flight attitudes shown in Figure 63.

Table XIX. Mission Cycle, Design.

Segment	Altitude		Mach No.	% Thrust	Time, min	% Time
	Km	Kft				
Start	0	0	0		0.5	1.1
Idle-Taxi	0	0	0		3.1	6.9
Takeoff	0	0	0		1.22	2.7
Climb (1st Seg)	0 - 3.05	0 - 10	0.38	Max. Climb	5.0	11.1
(2nd Seg)	3.05 - 7.62	10 - 25	0.45	Max. Climb	5.0	11.1
Cruise	7.62	25	0.76	Max. Cruise	14.0	31.1
Descent	6.10	20	0.6 - 0.3	F.I.	11.7	26.0
Approach	0.30 - 0	1 - 0	0.12	65% Max. F_N	1.3	2.9
Reverse Thrust	0	0	0.12 - 0	Max. Rev	0.08	0.2
Idle-Taxi	0	0	0	3 - 20	3.1	6.9
					45.0	100.0

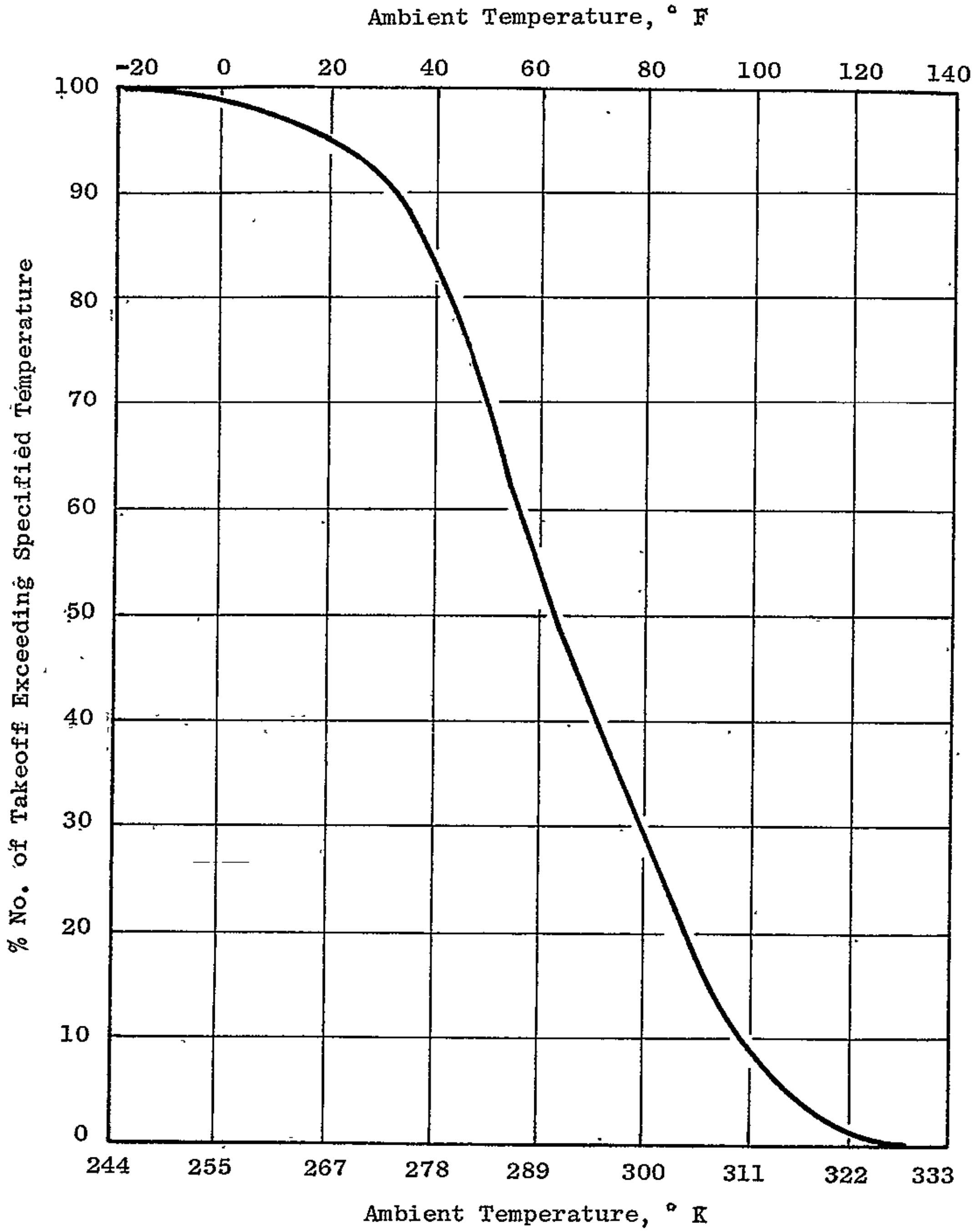


Figure 57. Takeoff, Temperature.

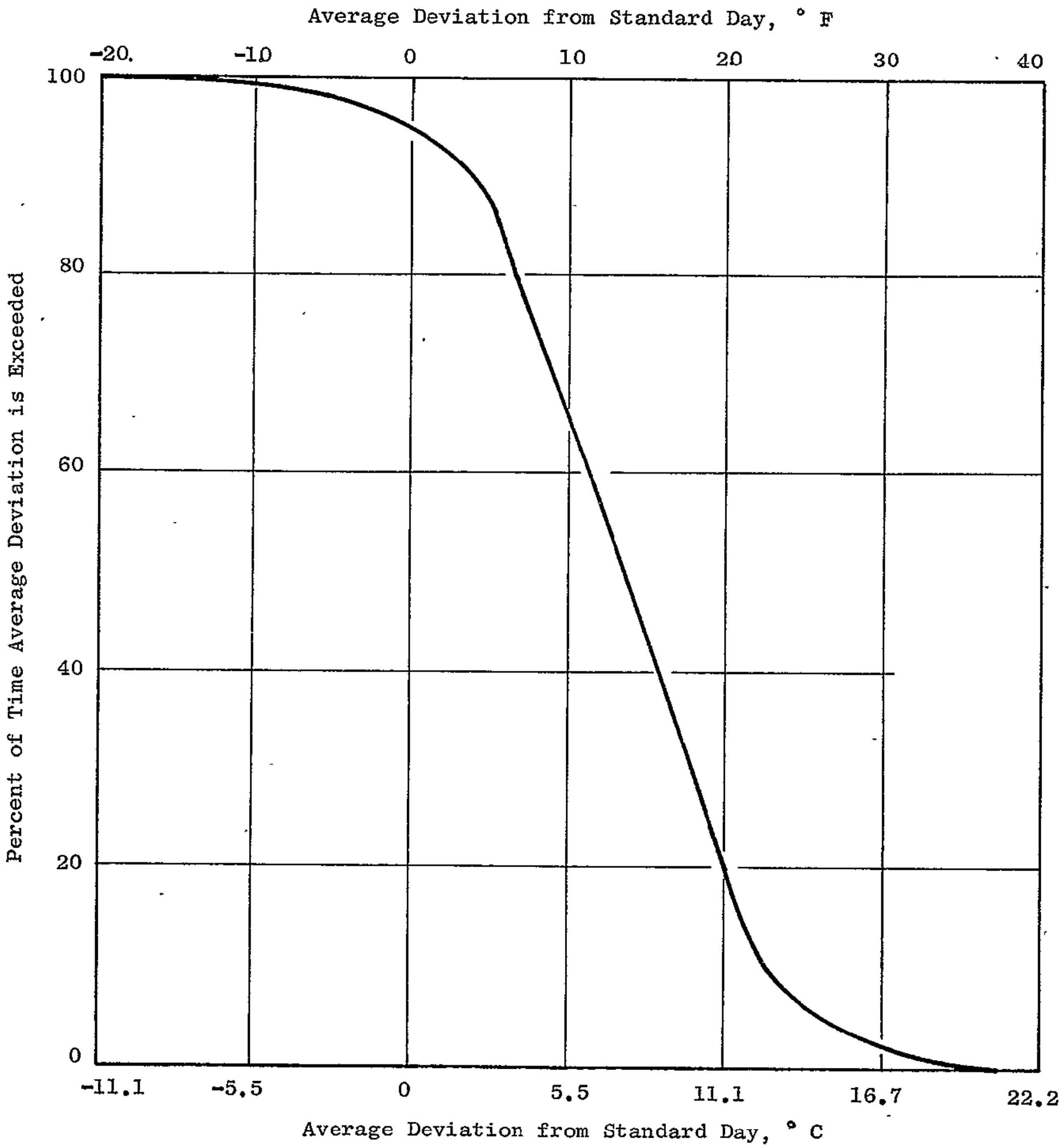


Figure 58. Flight, Temperature.

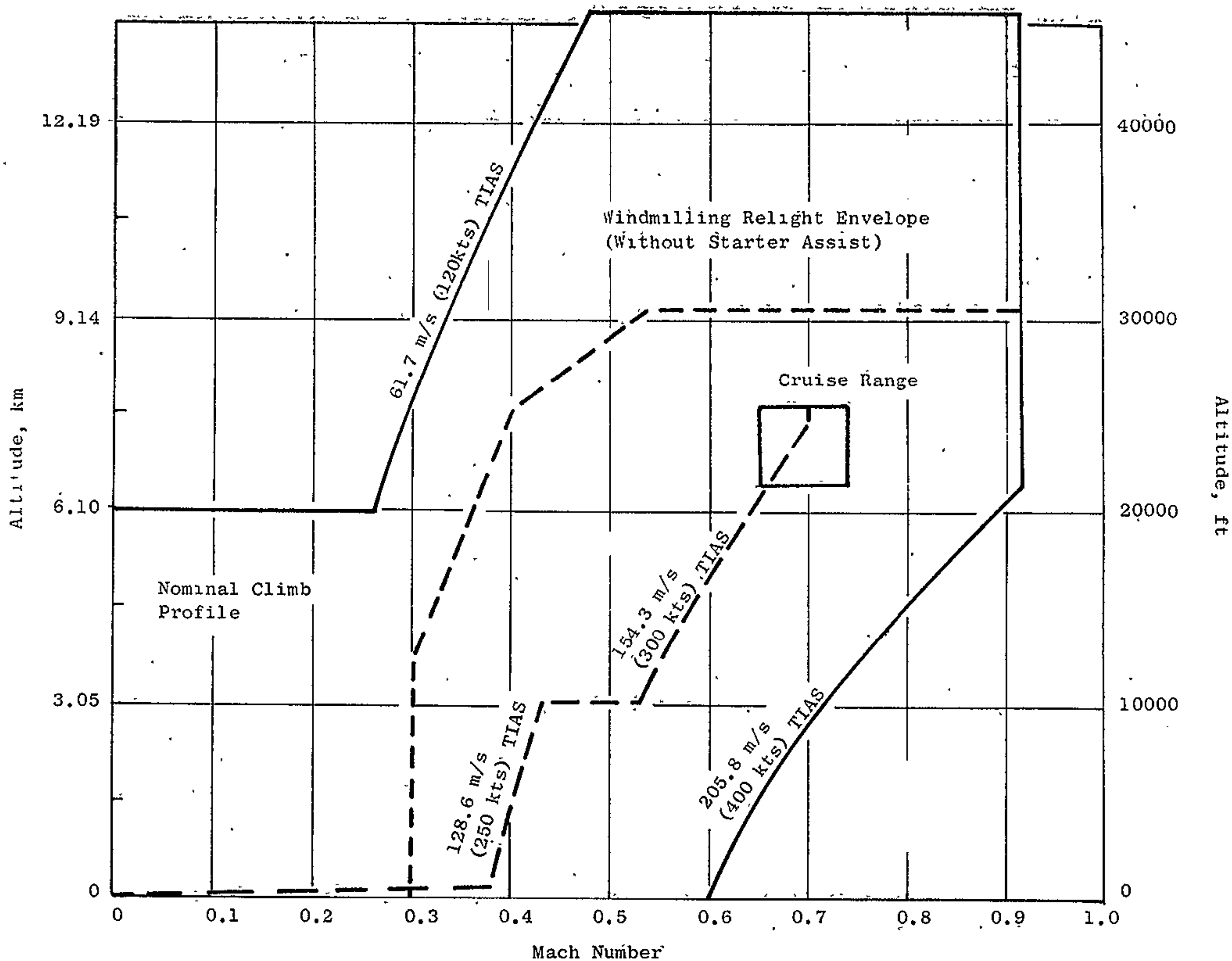


Figure 59. Operating Envelope.

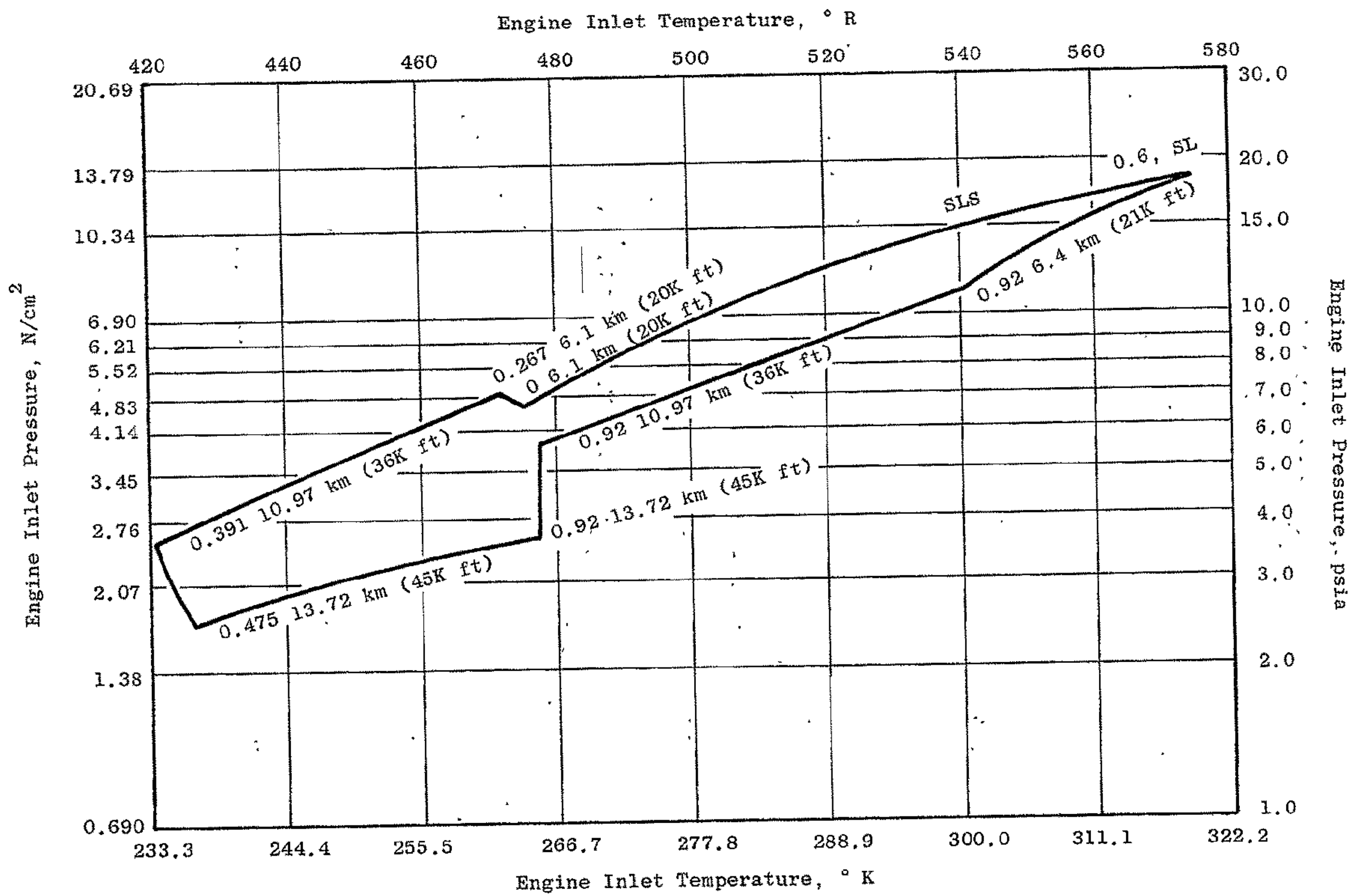


Figure 60. Inlet, Temperature/Pressure Envelope.

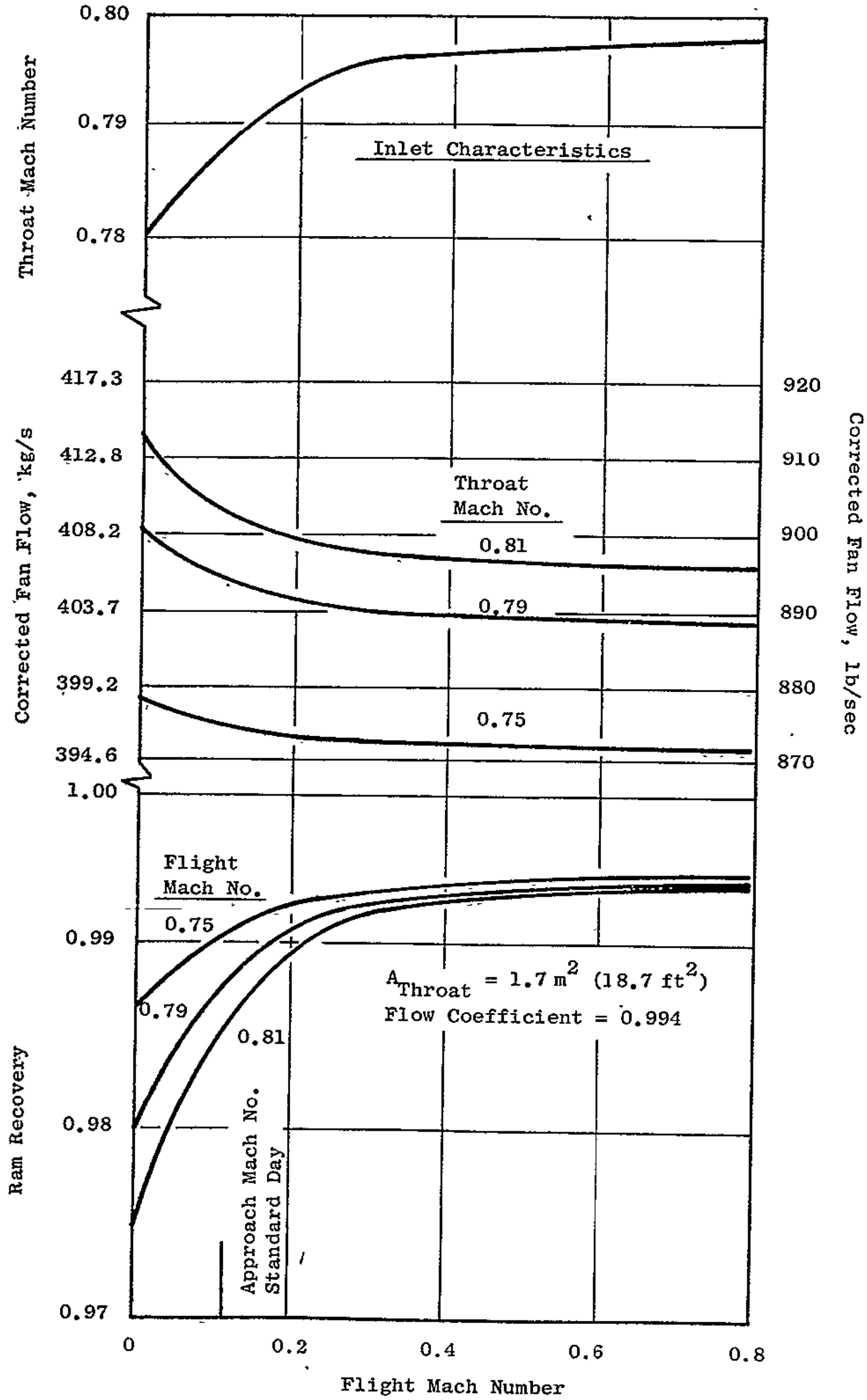


Figure 61. Inlet Characteristics.

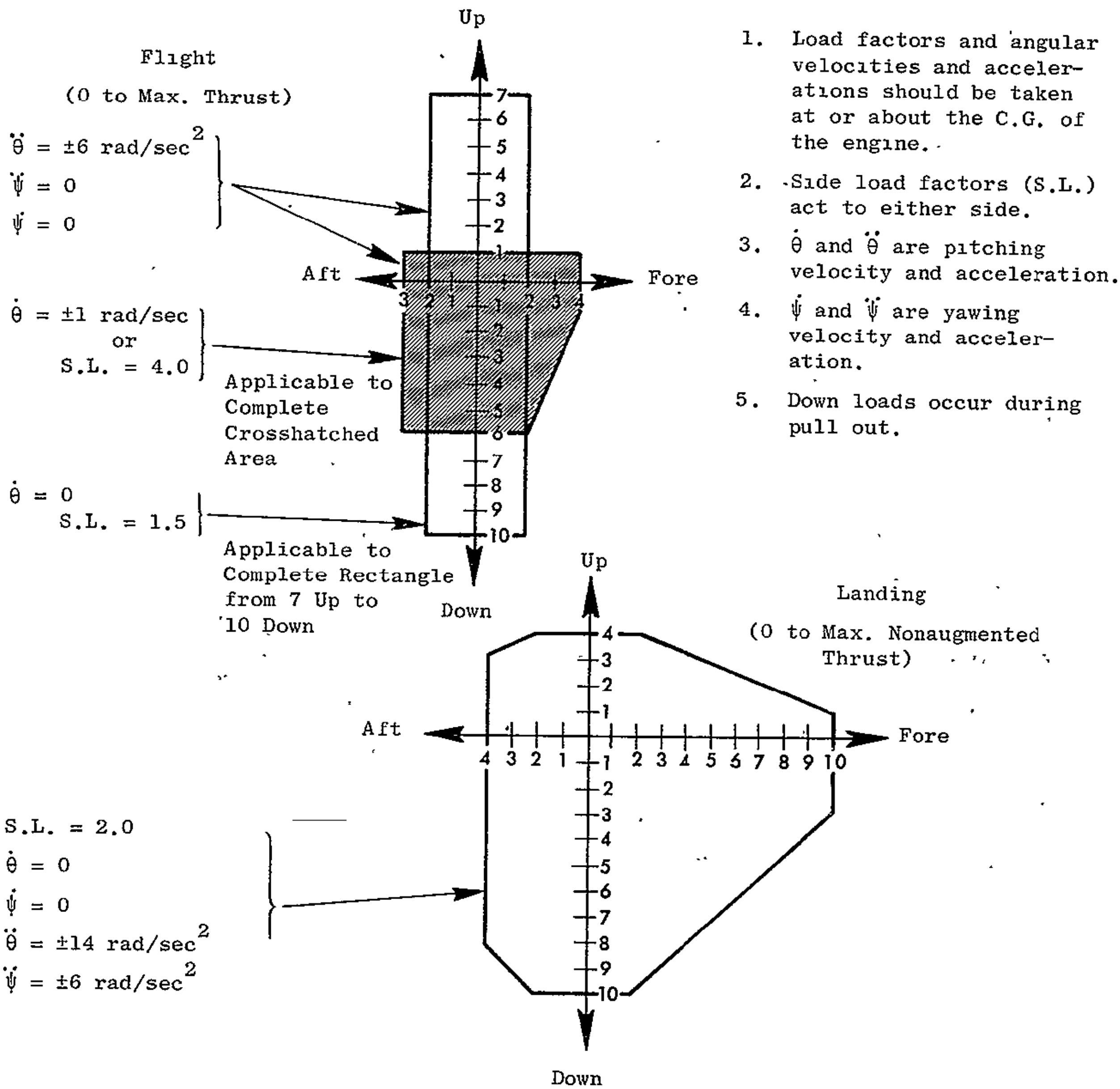


Figure 62. Maneuver Loads, Design.

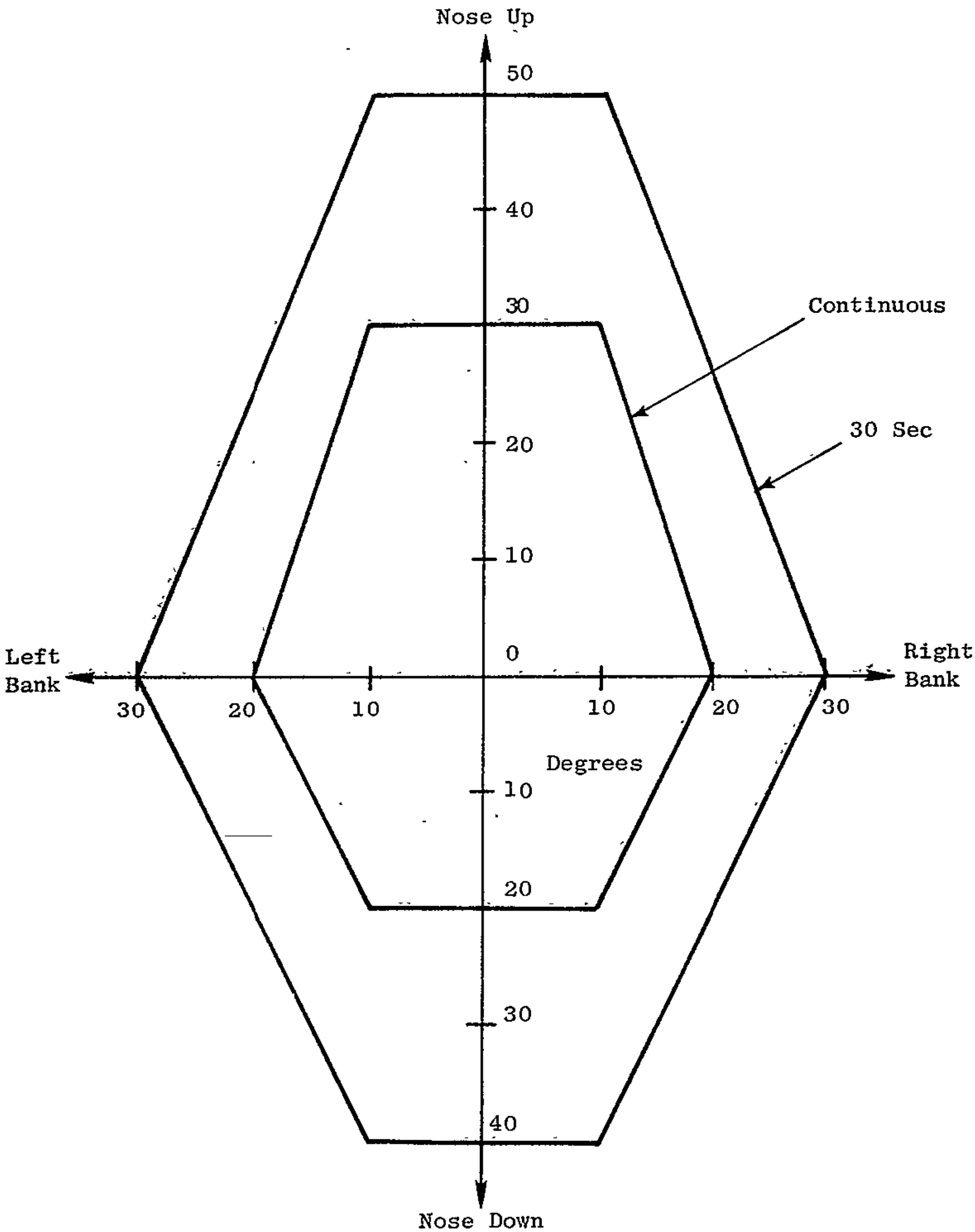


Figure 63. Attitudes, Design.

5.2.7 Main Engine Mounts

The rear mount reacts vertical, side, and torque loads in the plane of the turbine frame; thrust, vertical, and side loads shall be reacted in the plane of the fan frame. Mount locations and other mount system data including unit maneuver load mount reactions are shown in Figure 64.

5.2.8 Moments of Inertia

The polar moment of inertia of the rotors is estimated to be:

	UTW
Core	6.8 kg m^2 (162 lb ft^2)
Fan, fan shaft, and ring gear	23.8 kg m^2 (566 lb ft^2)
LP turbine, LP shaft, and sun gear	4.2 kg m^2 (100 lb ft^2)
Star gears and bearings	0.2 kg m^2 (5 lb ft^2)
Transverse moment	1561.3 kg m^2 (37,050 lb ft^2)

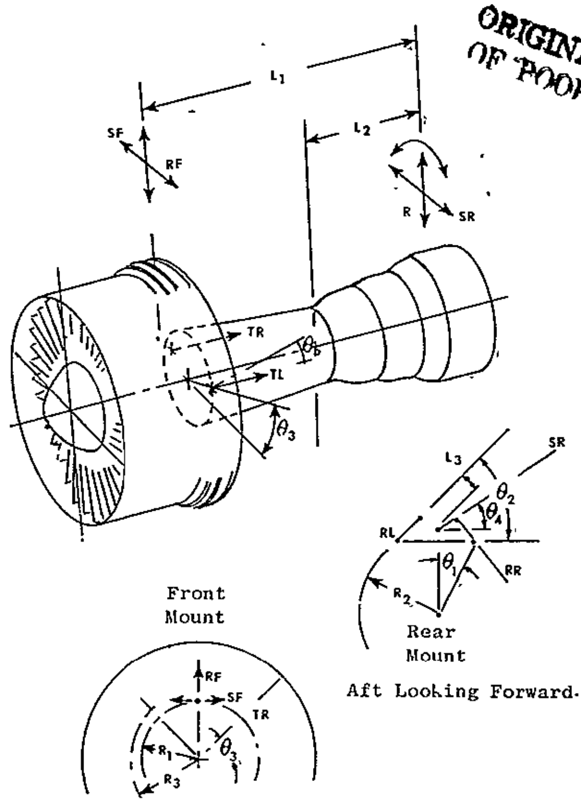
5.3 NACELLE - COMPOSITE COMPONENTS

The large amount of acoustically treated area required for noise reduction results in a relatively long nacelle which could become a very significant portion of the total propulsion system weight. To keep weight to a minimum, the conventional metal construction is replaced with a composite nacelle with integrated acoustic treatment. The flight nacelle components described in this section utilize these advanced composite materials, and their associated design concepts and are based wholly or in part on the designs developed for the UTW experimental engine.

The major portion of the nacelle, with exception of the core cowl, operates at very modest temperatures, less than 82.2° C (180° F), permitting use of a wide variety of composite materials. The primary composite material selected for these areas consists of a woven Kevlar 49 fabric impregnated with an epoxy resin system. This material exhibits light weight, good tensile strength, moderate stiffness, and excellent impact strength. Its major drawback is its poor compressive strength, therefore, in areas requiring higher compressive capabilities, woven glass cloth is substituted for the Kevlar. Where this is necessary, the standard 7781 weave "E" glass is used, impregnated with the same matrix system as the Kevlar.

For the core cowl, which must operate at elevated temperatures, a graphite/polyimide system is used allowing long term operation at 287.8° C (550° F). The honeycomb core material in the low temperature areas is

L_1 = 123.1 cm (48.5 in.)
L_2 = 85.8 cm (33.8 in.)
L_3 = 17.4 cm (6.8 in.)
θ_1 = 28°
θ_2 = 65°
θ_3 = 45°
θ_4 = 30°
R_1 = 43.2 cm (17.0 in.)
R_2 = 46.2 cm (18.2 in.)
Thrust @ Mounts (Installed)
77,395 N (17,400 lbs)
TR - TL



POUNDS

	TR	RF	SF	RR	RL	SR
Thrust	8700	-	-	-	-	-
1G Down	-	3200	-	775	775	-
1G Side	-	-	3200	7800	7800	7275
1 Rad/sec Pitch	-	-	3775	4575	4575	4675
1 Rad/sec Yaw	-	3775	-	2100	2100	-
1 Rad/sec^2 Pitch	-	285	-	175	175	-
1 Rad/sec^2 Yaw	-	-	285	350	350	375
Blade Out	-	23,900	23,900	24,325	24,325	19,900

NEWTONS

	TR	RF	SF	RR	RL	SR
Thrust	38,698	-	-	-	-	-
1G Down	-	14,234	-	3447	3447	-
1G Side	-	-	14,234	34,694	34,694	32,359
1 Rad/sec Pitch	-	-	16,791	20,350	20,350	20,794
1 Rad/sec Yaw	-	16,791	-	9341	9341	-
1 Rad/sec^2 Pitch	-	1268	-	778	778	-
1 Rad/sec^2 Yaw	-	-	1268	1557	1557	1668
Blade Out	-	106,307	106,307	108,198	108,198	88,515

Figure 64. Mount Load Summary.

Hexcel's corrosion resistant 5052 aluminum core. For the higher temperatures in the core cowl, HRH 327 glass/polyimide core is used. The honeycomb core in the acoustically treated panels is slotted to provide drainage.

The specific nacelle components utilizing composite materials are the inlet, outer cowl doors, fan variable nozzle flaps, and the inner cowl door. These are discussed individually below.

5.3.1 Inlet

The UTW QCSEE inlet is the largest single piece of the overall nacelle structure, being almost 177.8 cm (70 inches) long and nearly 200.7 cm (79 inches) in diameter, and has extensive acoustic treatment. In order to reduce the weight of this large structure as much as possible it will be constructed mainly of lightweight Kevlar/epoxy material and the acoustic treatment will be incorporated as part of the permanent structure.

The inlet consists primarily of inner and outer honeycomb sandwich walls separated and supported by circumferential stiffeners as shown in Figure 65. The face sheets of these sandwiches are all made from Kevlar/epoxy. The inner skin of the inner wall is perforated with hole configurations that suit the acoustical requirements of the inlet. The inner wall thickness (honeycomb depth) is also tailored to acoustical requirements. The outer wall thickness is sized to provide adequate stiffness. Honeycomb material is aluminum.

Aerodynamic loading of the inlet is far more significant than inertia loading. The primary cause for this is the large transverse load reaction on the inlet as it turns the entering engine airflow during any flight condition in which the direction of the free stream air is not parallel to the inlet axis. In contrast, the lightweight structure of the inlet produces relatively low inertia loads. The most severe aerodynamic loads occur during a 3 g stall, sea level, at a flight Mach Number of 0.4, and maximum continuous engine power, as shown in Table XX. For design analysis, the loads resulting from this condition were combined with the most severe additive inertia loads caused by dynamic landing. In addition, compressive hoop loads were considered for the sea level static takeoff power operating condition. The stress levels for these loads and this construction are shown in Table XXI. These are based on each facing consisting of three plies of woven Kevlar/epoxy material giving a total face sheet thickness of 0.084 cm (0.033 in.). Buckling allowables for this construction are shown in Table XXII. The sensitivity of this configuration to local loads is shown in Figure 66. Stiffeners are segmented and are constructed of aluminum sheet with flanged weight reduction cutouts and with composite (Kevlar) flanges to provide bonded attachment to the walls. The stiffener flanges were designed to prevent the bearing load between the stiffener flange and sandwich wall from exceeding 248.2 N/cm^2 (360 psi). This resulted in a flange width of 1.55 cm (0.61 inches). Using a 13,790 N/cm^2 (20,000 psi) flange bending stress allowable results in a flange thickness of 0.178 cm (0.07 inches).

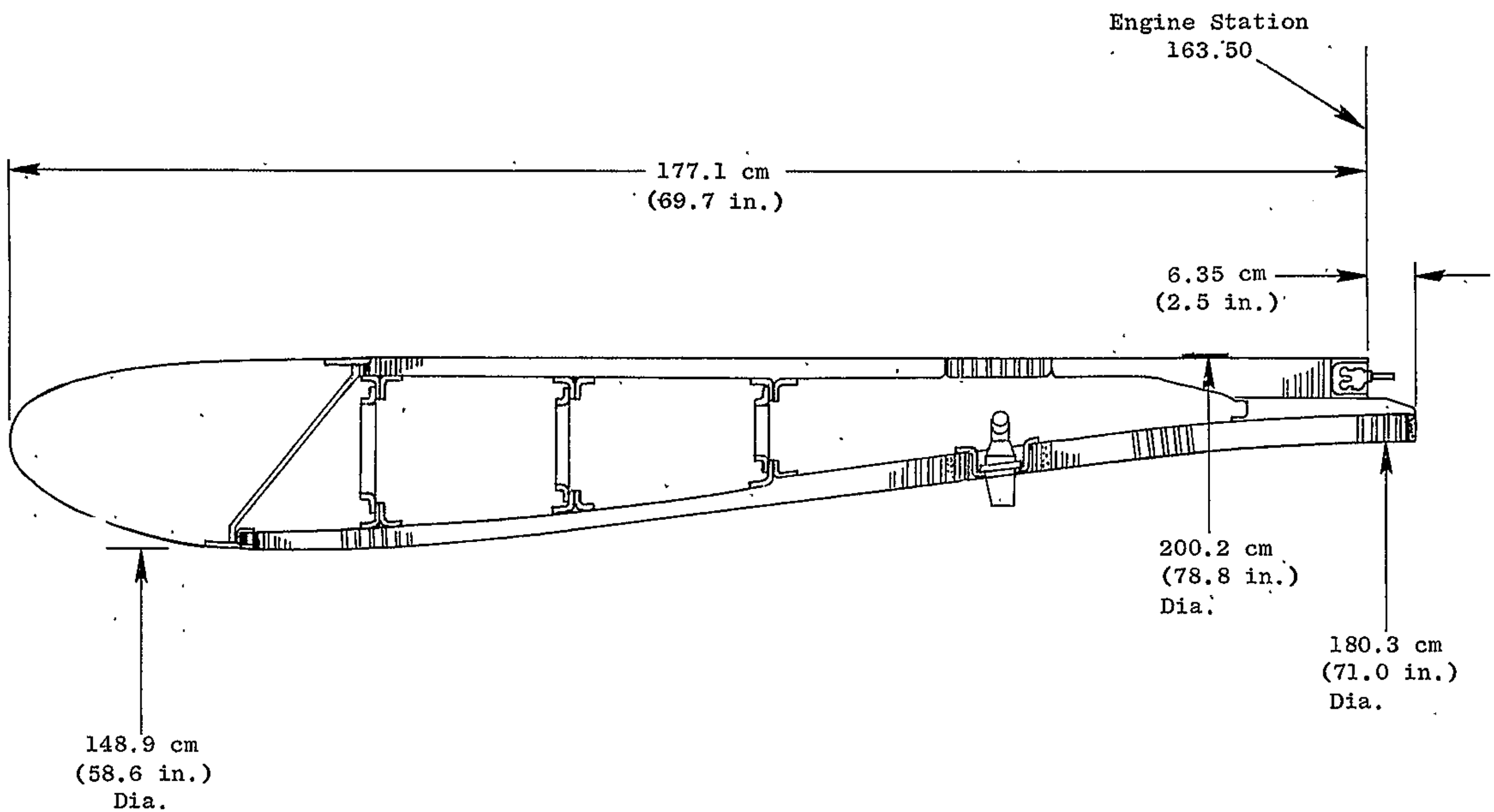


Figure 65. Inlet, Cross Section.

Table XX. Inlet Design Loads.

- Derived from DAC "Design - TO" Criteria Aerodynamic Loads from 3 g Stall Inertia Loads from Dynamic Stall
- Predominating Load Source:

  Aerodynamic Loads from 3 g Stall

  (98% of Latch Loads)
- Maximum Combined Loads

Bending	6,931,680 cm N (613,423 in.-lb)
Axial	8,358 N (1,879 lb)
Transverse	37,119 N (8,345 lb)

Table XXI. Inlet, Stress and Deflection.*

Type	Load Magnitude	Direction	Stress Type	Stress Calculated	Stress Allowable	F/S	Deflection
Bending	6,931,680 N (613,423 in.lb.)	-	Compression	1402 N/cm^2 (2034 psi)	6895 N/cm^2 (10,000 psi)	4.9	0.058 cm (0.023 in.)
Axial	8,358 N (1,879 lb.)	Forward	Tension	1583 N/cm^2 (2296 psi)	48265 N/cm^2 (70,000 psi)	30.5	0.058 cm (0.023 in.))
Transverse	37,119 N (8,345 lb.)	-	Shear	403 N/cm^2 (584 psi)	6895 N/cm^2 (10,000 psi)	17.1	0.206 cm (0.081 in.)
Hoop	2.8 N/cm^2 (4 psi)	Burst	Tension	1834 N/cm^2 (2660 psi)	48265 N/cm^2 (70,000 psi)	26.3	-
(Max Range Expected)	5.9 N/cm^2 (8.5 psi)	Crush	Compression	3897 N/cm^2 (5652 psi)	6895 N/cm^2 (10,000 psi)	1.8	-

*Aerodynamic loads from 3G stall (M_p = 0.4 at SL) combined with inertia loads from dynamic landing.

Table XXII. Inlet, Buckling Loads.

Buckling from Bending					
Load Conditions	Critical Moment		Actual Max. Moment		F/S
	cm N	in-lb	cm N	in-lb	
Moment on Outer Skins	455,667,000	40,329,928	6,930,700	613,423	65.7

Buckling from Compressive Hoop Load					
Load Conditions	Critical Pressure		Actual Pressure		F/S
	N/cm^2	psi	N/cm^2	psi	
Pressure Supported by Single Sandwich Wall (2 skins) Between Stiffeners 33.0 cm (13 in.) Apart	153.8	223	5.9	8.5	26.2
Pressure Supported by Overall Wall Structure	275.1	399	5.9	8.5	46.9
Pressure Supported by Single Outer Aluminum Nose Wall 40.6 cm (16 in.) Long with Stiffening Corrugations	351.0	509	5.9	8.5	59.9
Pressure Supported by Single Inner Aluminum Nose Wall Aft of Corrugation with Corrugation Sheet Extended to Form Doubler: 6.4 cm (2.5 in.) Long x 0.183 cm (0.072 in.) O/A Thk	16.2	23.5	5.9	8.5	2.76

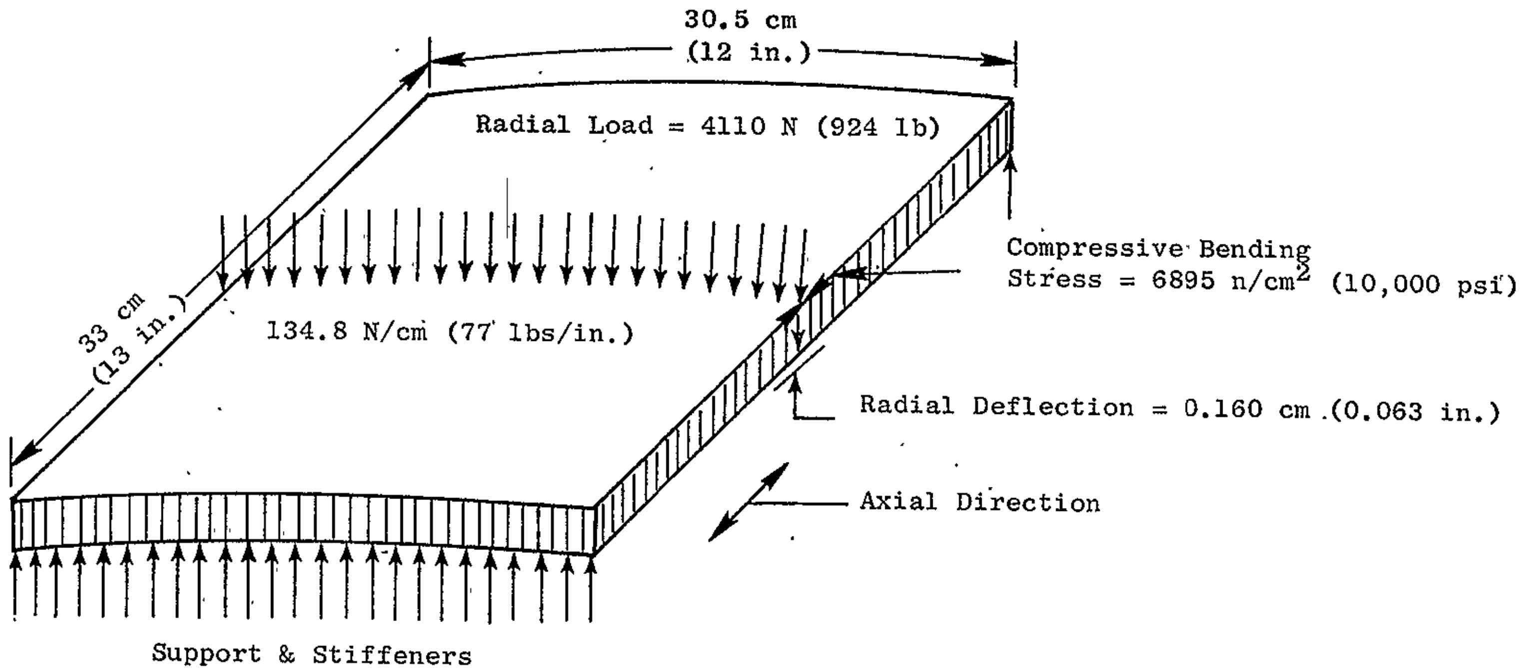


Figure 66. Inlet, Local Loading.

The leading edge of the flight inlet is all titanium for resistance to foreign object damage and erosion and for anti-icing provisions. This section is removable (by unbolting) from the main body. A corrugated backup sheet provides passages for anti-icing airflow. This arrangement has the advantages of isolating the anti-icing air from the composite materials and of containing the flow for effective heat transfer and minimum air usage. A sketch of this concept is shown in Figure 67.

The rear of the inlet is attached to the forward end of the fan frame by 16 rotary latches. Each of these latches is operated by turning a flush receptacle. A pressure and acoustical seal is achieved at this joint by means of a thick (in the radial direction) elastomer gasket. The latch loads for this installation are shown in Table XXIII.

As may be seen from the above analysis, the inlet is not highly stressed. The skin gages selected were estimated minimum gages required for local impact loads during handling.

The composite QCSEE inlet described above weighs 156 kg (345 lb) compared to the weight of a typical current technology metal inlet (scaled to the same size) of 217 kg (479 lb).

5.3.2 Fan Bypass Duct (Fan Cowl)

The fan bypass duct and fan nozzle constitute the outer nacelle section aft of the fan frame (see Figure 68). These components are designed to take full advantage of advanced-type composite materials in order to provide a lightweight, thin-profile nacelle suitable for advanced air transports.

The fan bypass duct is designed as right- and left-hand sections split on the vertical center line. A roll out view of one section is shown in Figure 69. Each section is attached at its upper edge to the aircraft pylon structure by means of a piano-type hinge and is sealed to the pylon as shown in Figure 70. The sections are fastened to each other along the bottom vertical centerline by a series of cowl latches. This permits the duct to be opened for accessibility to the core cowl, and thereby to the core engine. The duct forward "close out" ring contains a circumferential inward facing tongue which engages a corresponding groove in the fan frame when the duct sections are closed and latched (see Figure 71). The tongue and groove are tapered on the forward side (with corresponding axial free play in the piano hinge) to aid in the engagement when closing the duct. The aft surface of the joint is vertical for transmittal of the duct axial loads into the fan frame and hence through the engine mounts into the pylon.

The duct is of sandwich-type construction with Kevlar 49 face sheets and 5052 aluminum honeycomb core with stiffening rings and members where required. The inner face sheet provides a continuation of the outer wall of the fan exhaust duct and is perforated so that, along with the sized honeycomb core, it constitutes the sound suppression treatment in the duct. This treatment being integral with the duct, is also part of the load-

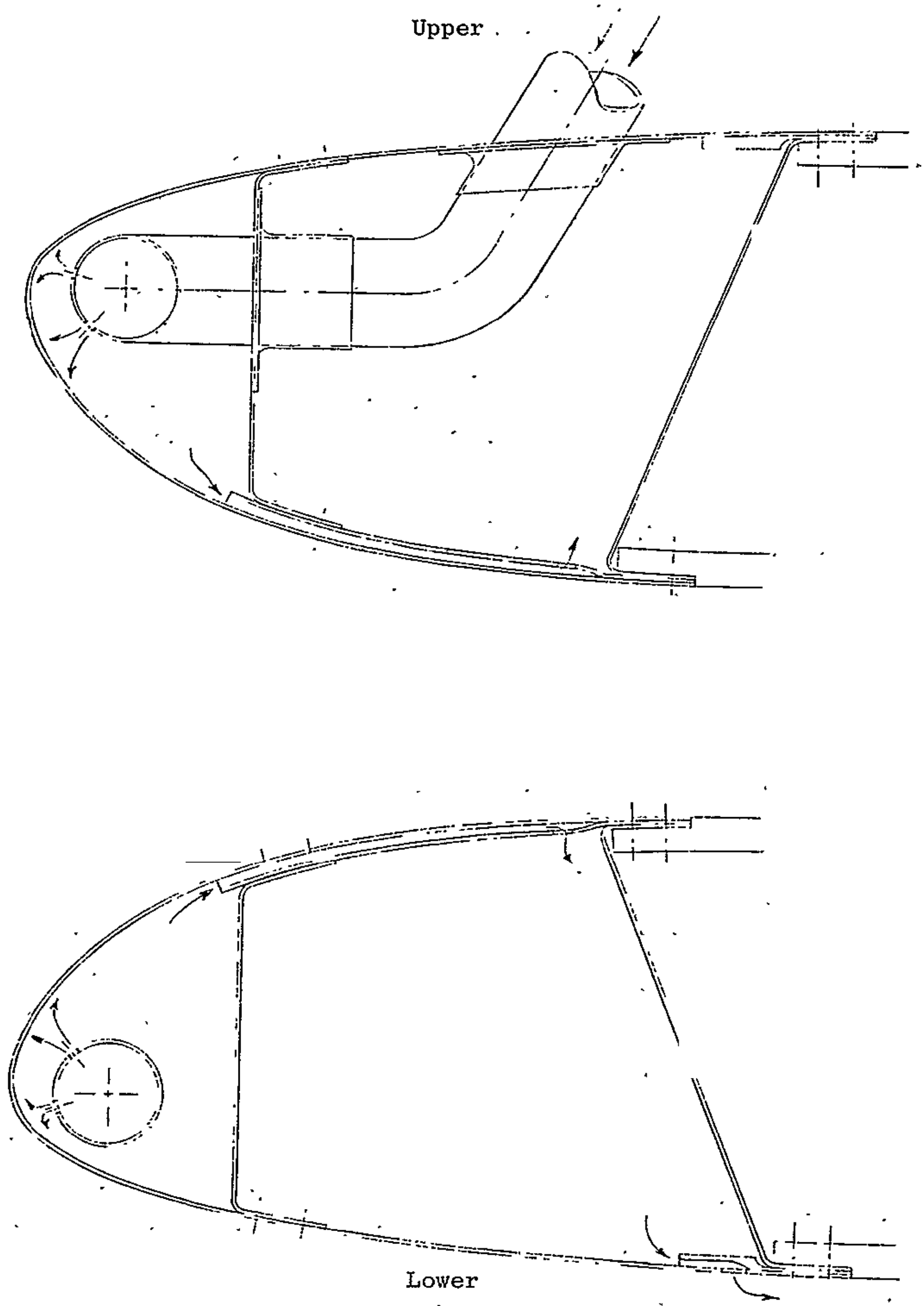


Figure 67. Inlet, Lip/Anti-Icing.

Table XXIII. Inlet, Latch Loads.

Latch Configuration	Maximum Latch Load		Ultimate Latch Strength		Latch F/S
	N	lb	N	lb	
All 16 Latched	9,617	2162	28,801	6475	3.00
One Latch	10,737	2414	28,801	6475	2.68
Two Latches Open	13,135	2953	28,801	6475	2.19
Six Latches Open	25,073	5637	28,801	6475	1.15

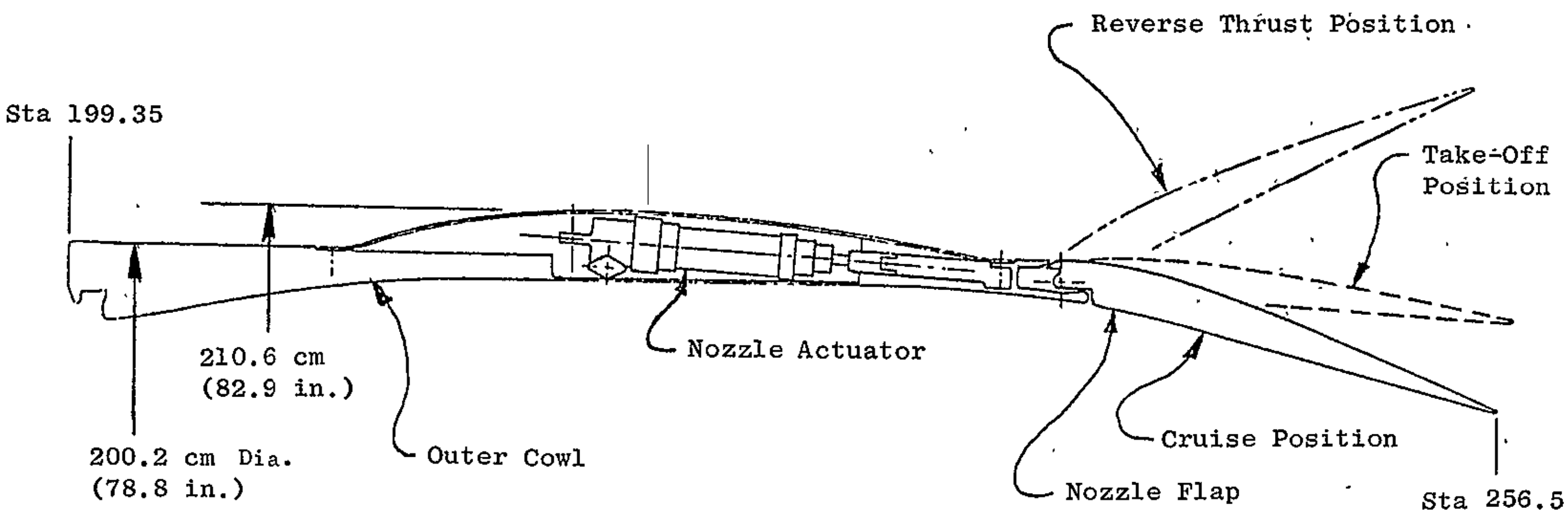


Figure 68. Fan Cowl and Flare Nozzle, Cross Section.

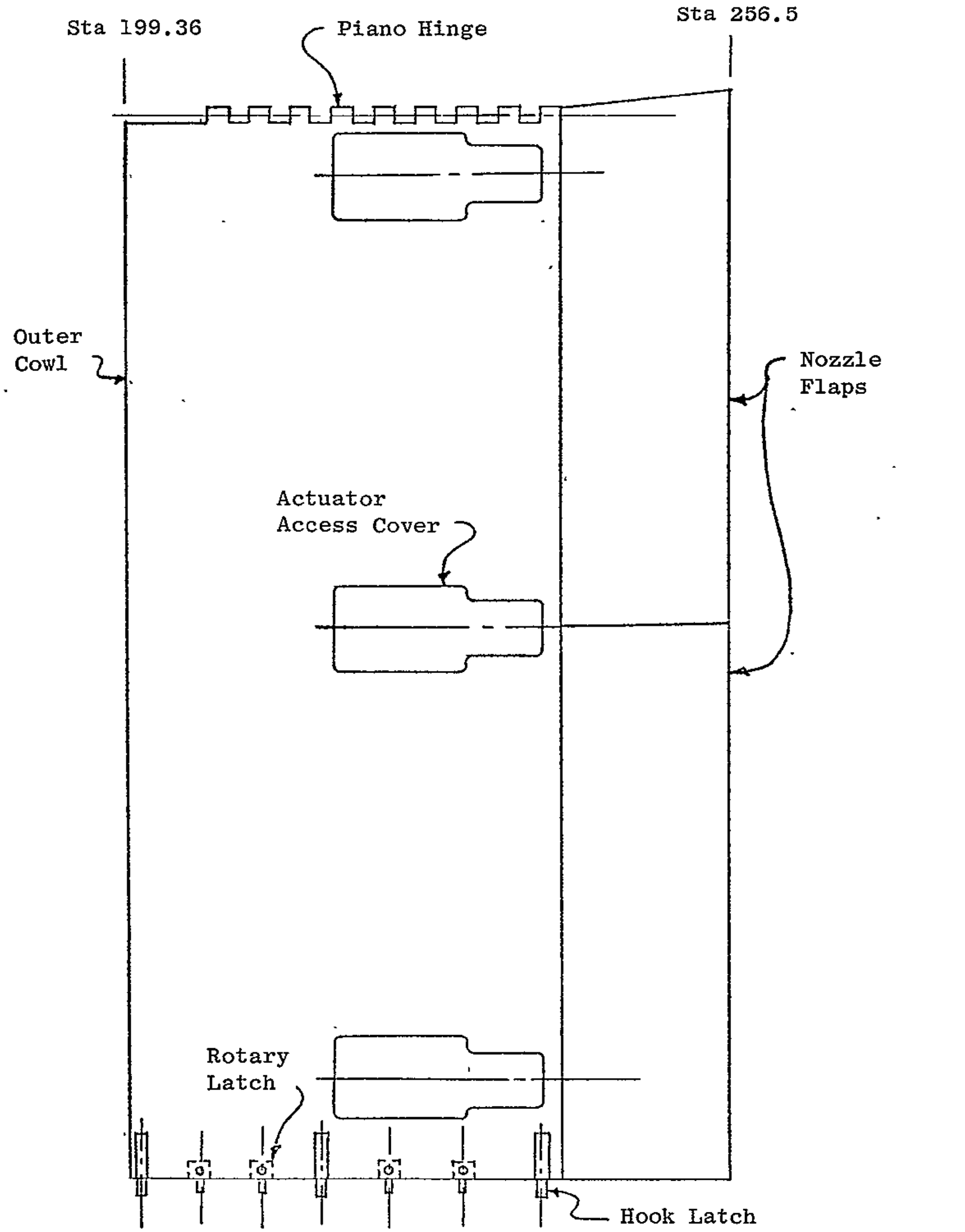


Figure 69. Fan Cowl and Flare Nozzle, Development.

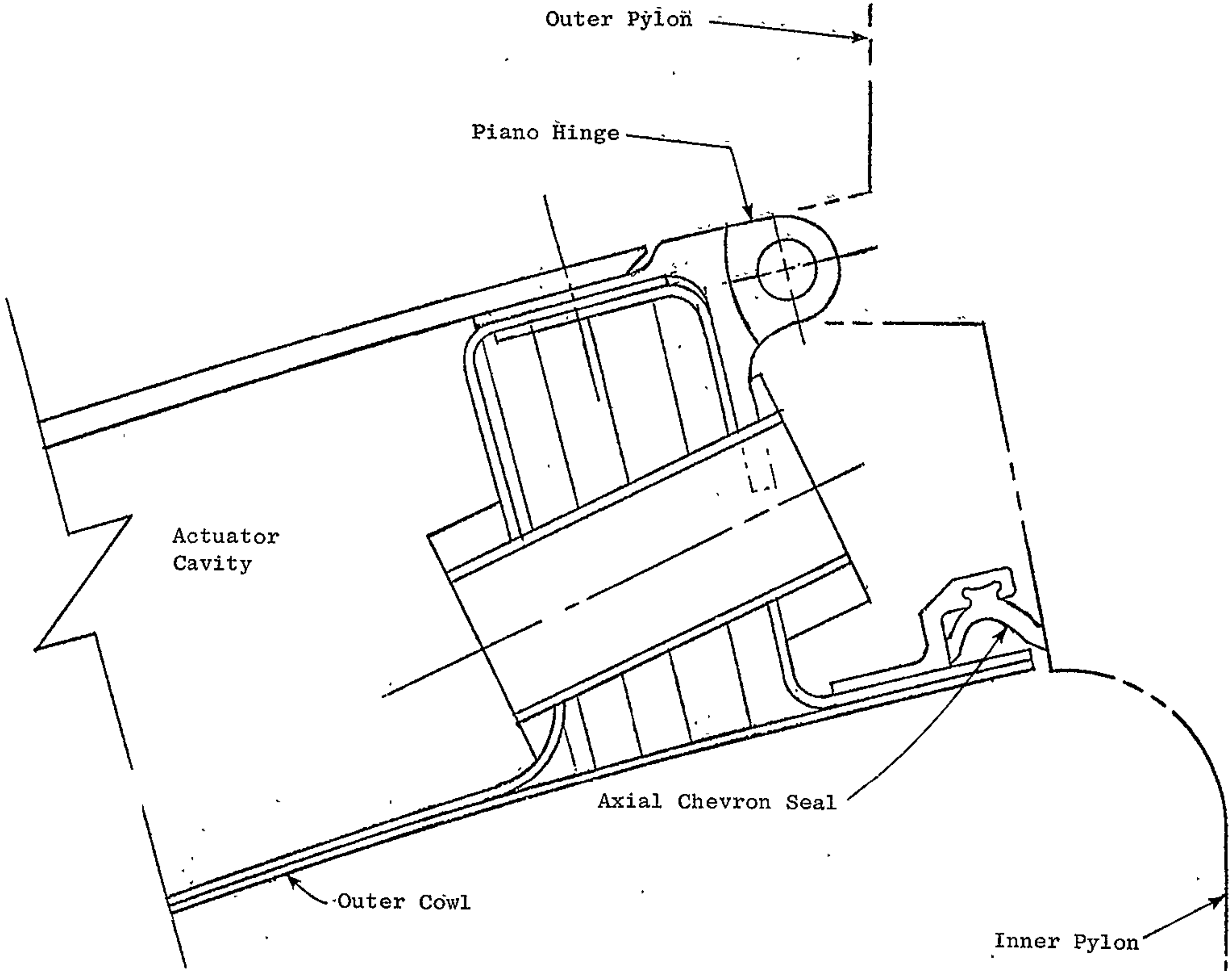


Figure 70. Fan Cowl, Hinge and Seal.

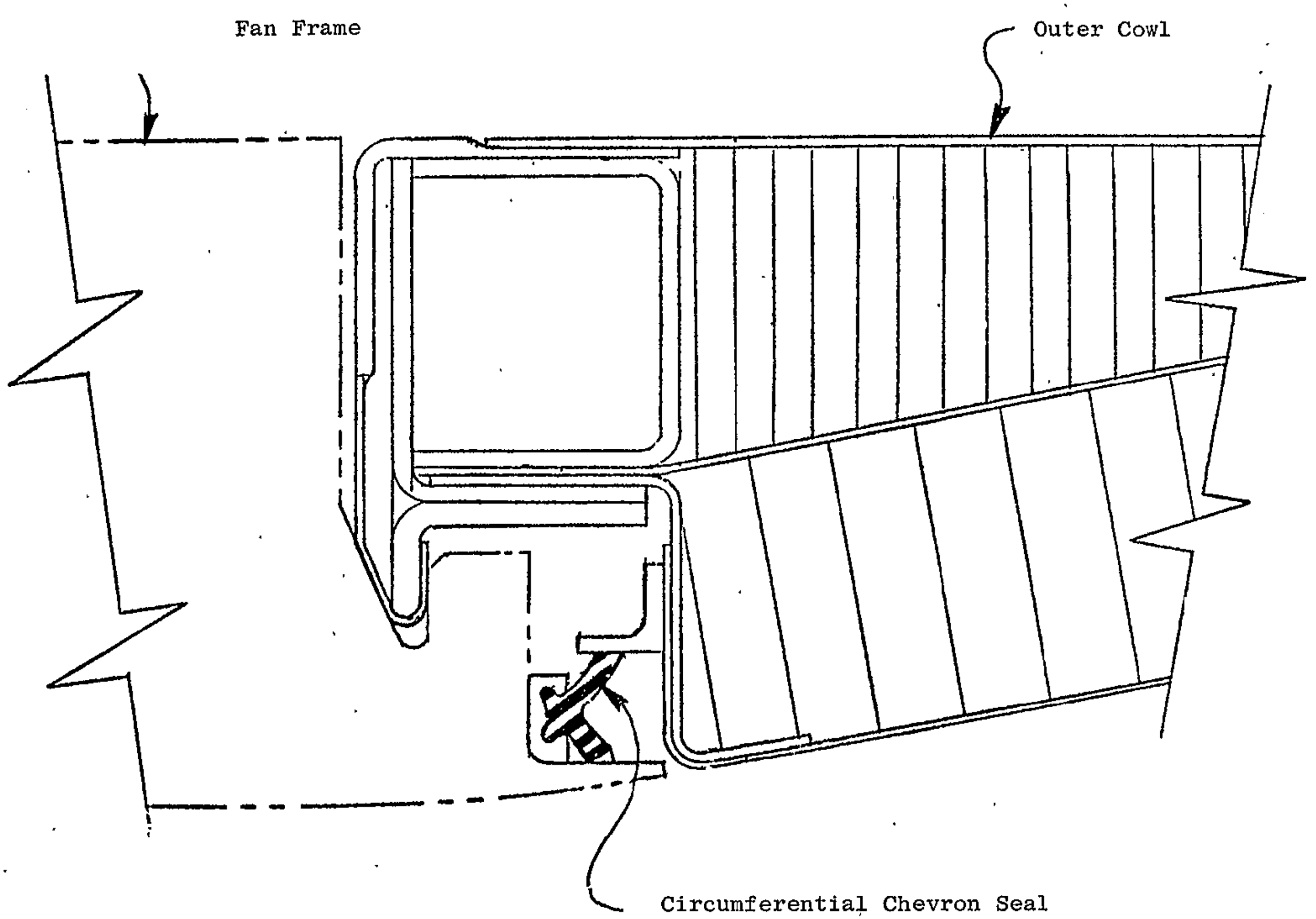


Figure 71. Fan Cowl/Fan Frame Interface.

carrying structure. The duct outer face sheet forms the external flow surface of the nacelle and is also structural. The minimum thickness (core height) of the duct is set by the acoustical design requirements. At the forward end of the duct, where additional thickness is required to mate with the fan frame flow path, a septum sheet is provided in the core area to maintain the proper honeycomb cell depth for acoustic treatment.

The fan nozzle actuation system will be contained within the envelope of the duct, except for external fairings locally over the actuators (three per duct half). These fairings will also be the access covers for the actuators and link system. The actuators will be mounted in cavities formed of molded parts bonded into the duct. A typical installation is shown in Figure 68. The aft end of the pan will contain the inner half of a track system for supporting and guiding the actuator rod end clevis during flap operation. The outer half of the track will be mounted in the actuator access cover. Clearance holes in the duct aft close out ring will allow free passage of the flap to actuator linkage during nozzle translation (see Figure 72). The forward actuator mount will be attached to an integral ring at the forward end of the actuator cavity, this ring serving to distribute the actuation loads. Tunnels formed of plastic tubing and built into the core will serve as passages for the actuation system synchronization cable, the hydraulic rod, and head end tubes and the seal drain tube. The hydraulic and drain tube passages will connect each actuator cavity and extend through both the upper and lower axial closeouts of the bondment. This will allow assembly of the tubes from either direction. The synchronization cable tunnel will extend from the upper axial bondment closeout to the center actuator cavity only.

A 2024 aluminum piano hinge will be mechanically fastened along the upper edge closeout and will be continuous from the aft close out ring to a point approximately 20.3 cm (8 in.) aft of the forward ring. Figure 70 shows a cross section of this hinge.

The bottom latch system will consist of seven Hartwell King latches of the hook type, one at each of the three circumferential rings and the other four equally spaced between these. Figure 73 shows a typical latch installation. With the duct closed and latched, the duct pressure loads will be resisted by hoop tension/compression stresses, the hinge forces being reacted by the pylon support structure. Table XXIV shows the results from preliminary load and stress calculations.

5.3.3 Fan Nozzle

The fan exhaust nozzle is a fully modulating, variable-flap type capable of not only providing various areas for forward thrust, but also of flaring outwards to provide increased area for intake flow to the variable-pitch fan in the reverse mode (see Figures 74 and 75).

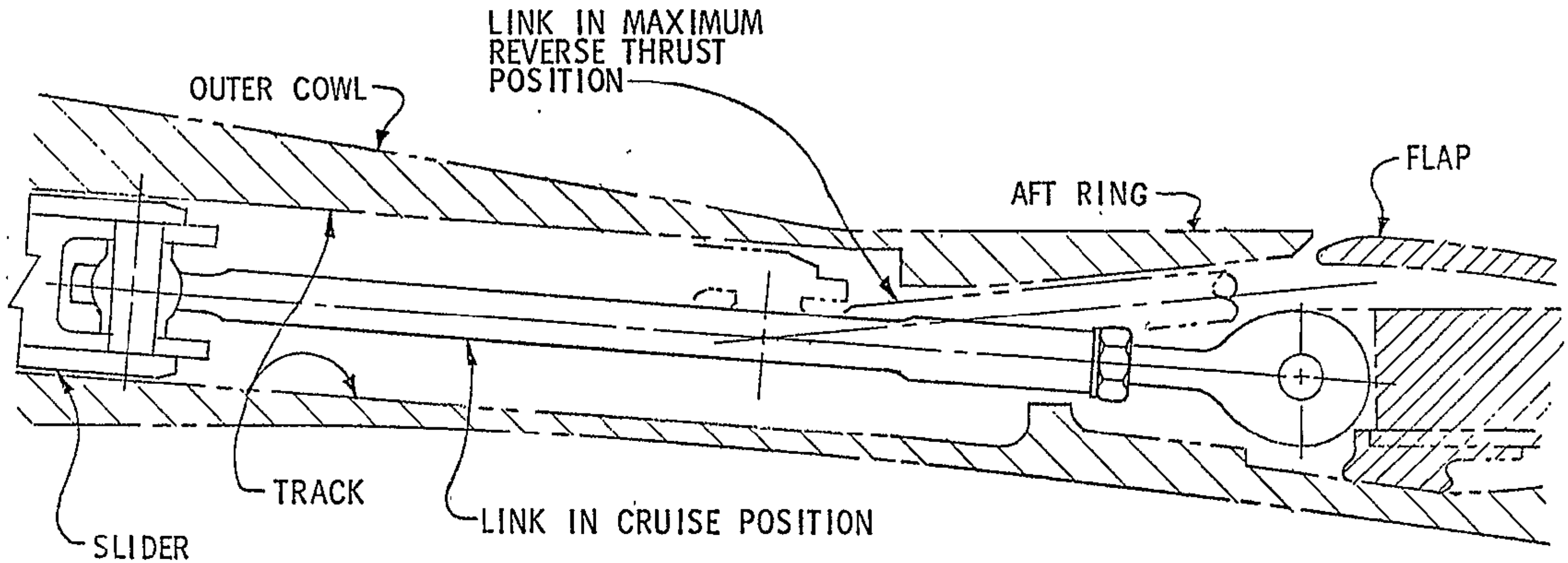


Figure 72. Flare Nozzle Actuation, Cross Section.

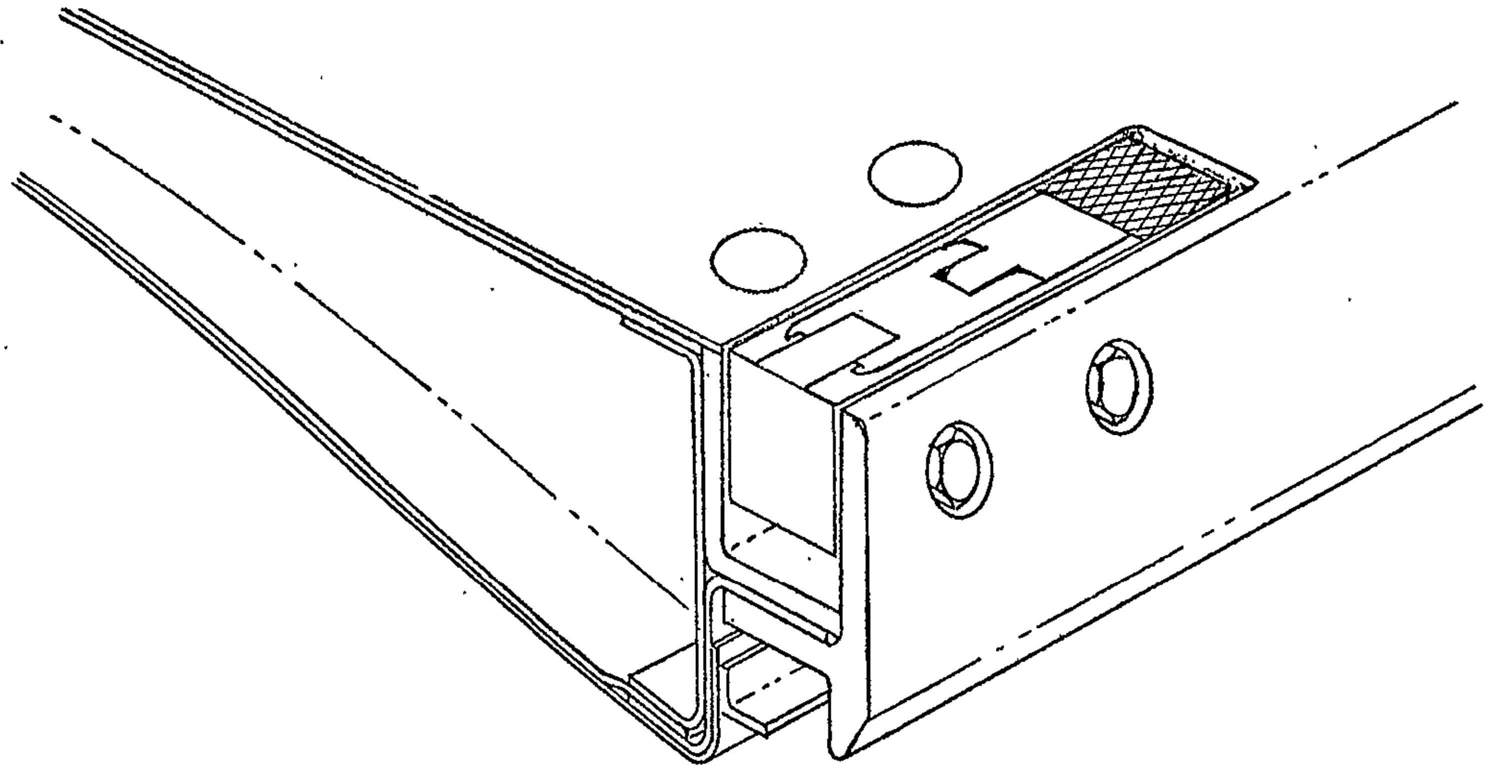


Figure 73. Fan Cowl, Latch Installation.

Table XXIV. Fan Cowl, Hinge and Skin Stress.

Flight Condition	Max ΔP	Hoop Load	Hinge Pin Bearing Stress	Hinge Pin Shear Stress	Skin Circumferential Stress	Skin Axial Stress
M = 0.3 @ SL 100%	1.1 N/cm^2 (1.60 psi)	109 N/cm (62 lb/in.)	214 N/cm^2 (310 psi)	386 N/cm^2 (560 psi)	948 N/cm^2 (1375 psi)(T)	476 N/cm^2 (690 psi)(T)
M = 0.8 @ 9.14 km (30 K ft)MCR	2.65 N/cm^2 (3.85 psi)	261 N/cm (149 lb/in.)	548 N/cm^2 (795 psi)	931 N/cm^2 (1350 psi)	2275 N/cm^2 (3300 psi)(T)	1138 N/cm^2 (1650 psi)(T)
M = 0.90 @ 7.1 km (20K ft)MRC	4.8 N/cm^2 (7.00 psi)	474 N/cm (271 lb/in.)	996 N/cm^2 (1445 psi)	1689 N/cm^2 (2450 psi)	4137 N/cm^2 (6000 psi)(T)	2069 N/cm^2 (3000 psi)(T)
M = 0.225 @ SL REVTH	-2.48 N/cm^2 (-3.60 psi)	-243 N/cm (-139 lb/in.)	510 N/cm^2 (740 psi)	869 N/cm^2 (1260 psi)	2137 N/cm^2 (3100 psi)(C)	1069 N/cm^2 (1550 psi)(C)

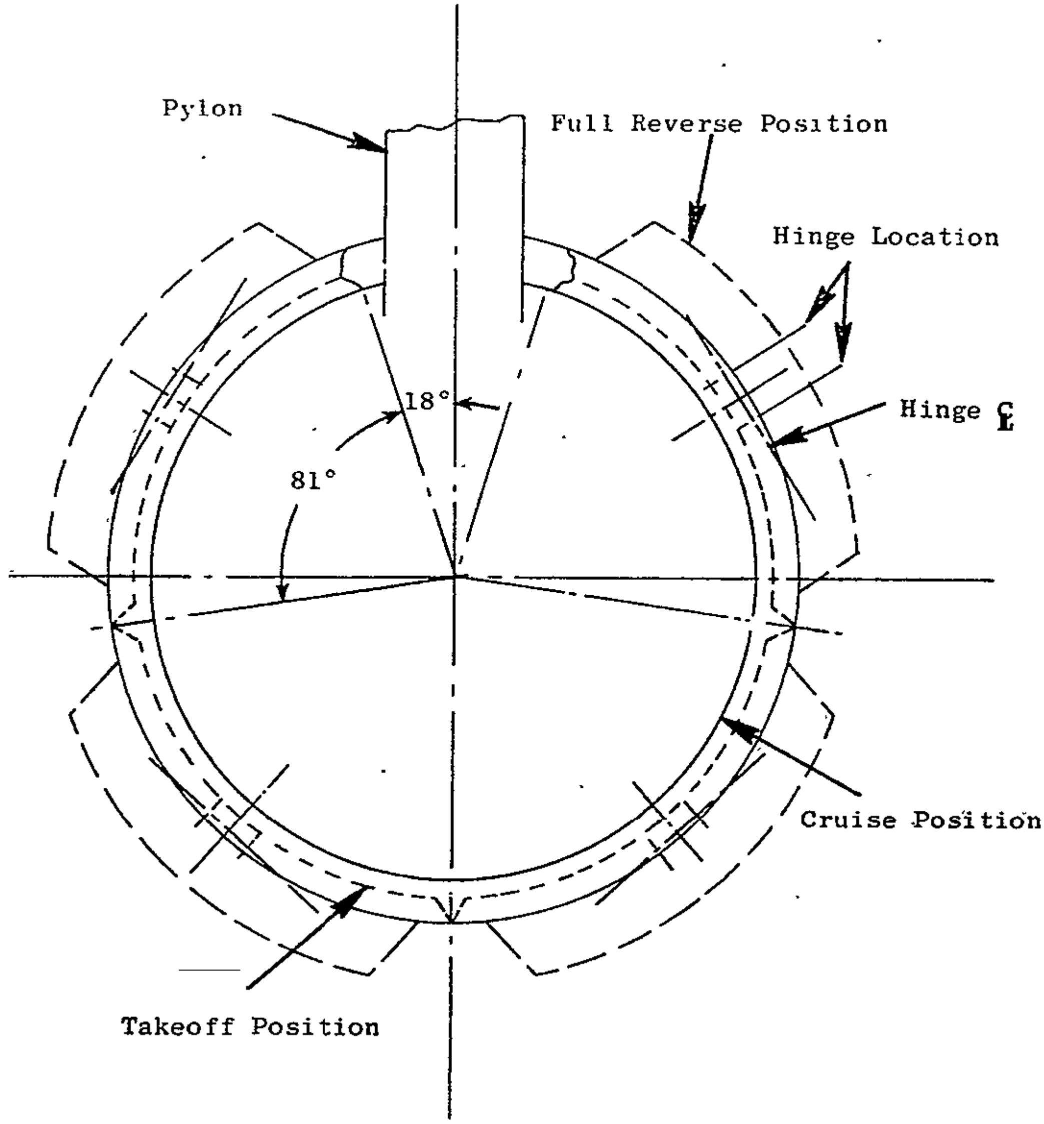


Figure 74. Flare Nozzle Flap Schematic.

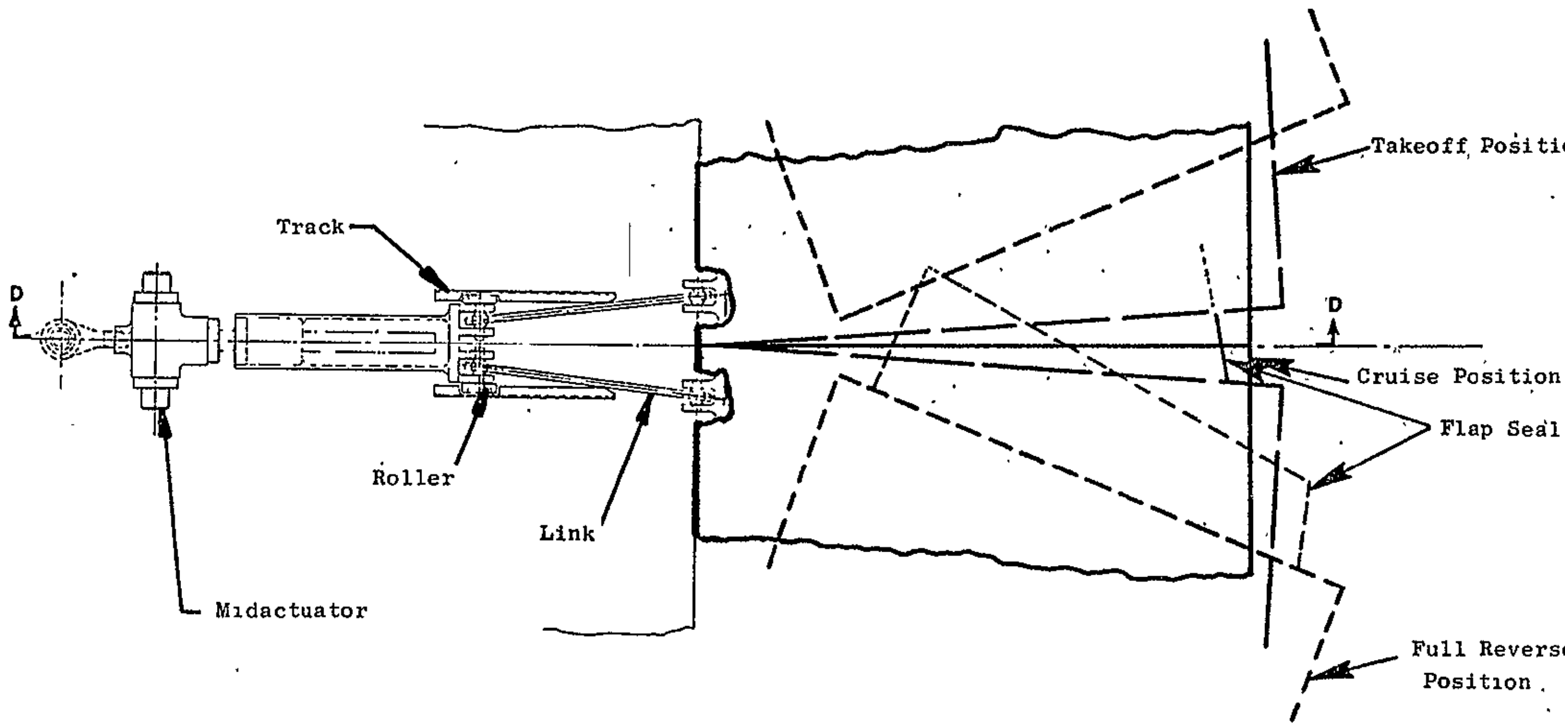


Figure 75. Flare Nozzle Actuation, Top View.

The nozzle consists of four hinged flaps each 44.96 cm (17.7 inches) axially from hinge centerline to trailing edge. The upper flaps are 83.7° wide and the lower flaps are 84.5° wide. A common hinge/actuator link clevis relationship is used for all four flaps, this relationship being determined by the location of the actuators in the fan outer duct. Two each of these flaps, an upper and lower, will be attached to the fan duct aft ring by means of the hinges, each flap having a pair of hinges 30.5 cm (12 inches) apart at the flap centerline along the flap forward edge closeout. The flaps will be connected to the actuation system in the fan duct by links and link clevises which are located 25.1° outboard of each hinge. The links will have spherical bearing rod ends to preclude any binding during flap operation. The flaps are designed such that they can be translated, with over travel, from an angle of about 13° in toward the engine centerline to an outward angle of approximately 29°. Located along the axial edges of each flap are seal assemblies, these seals are designed to give full sealing from a minimum nozzle area of 11,903.2 cm^2 (1845 $in.^2$) to a nozzle area of 16,780.6 cm^2 (2601 $in.^2$). Figure 76 shows the intraflap seal. From this point to the full reverse flap position the seals will be disengaged allowing a triangular-shaped void between the flaps. Sealing is not required while in the reverse mode. A circumferential seal is also provided along the forward inner edge of each flap, this seal contacting a contoured lip extension from the aft ring of the fan outer duct (see Figure 77). The seal will also provide full sealing over the same range as the axial intraflap seals.

Operation of the flaps will be by means of the previously mentioned actuation system and flap links. The midactuator in each duct half will be joined to both flaps mounted on that section (see Figure 74). This serves to synchronize the flap movement for that duct half, the synchronization between halves being done by the actuation system synchronization cables. The upper and lower actuators will be connected to the flaps by a single link only.

The construction of the flaps is similar to the rest of the nacelle (see Figures 78 through 81). They are of a sandwich type with Kevlar 49 face and 5052 aluminum flex core honeycomb. The outer face sheet is composed of three plys oriented 0°, +45°, and -45°. The inner face sheet is perforated from the forward closeout to within 12.7 cm (5 in.) of the trailing edge, at which point the sound suppression treatment becomes ineffective due to the low height of the core. This inner sheet is made up of four plys with a 0°, ±45°, 0° orientation in order to compensate for the loss of strength due to the perforations. The forward closeout is composed of two sections, each made up of 12 graphite/epoxy laminations and bonded together. The trailing edge is built up of Kevlar 49 laminations between the inner and outer face sheets and wrapped with two plys of fiberglass to prevent delamination. Deep channel-shaped axial closeouts are provided of three plys of Kevlar 49, the cavities being deep enough to accept the axial seal assemblies. The hinges and link clevises are made integral with the bondment, each one having legs which extend into the honeycomb, with the lugs protruding through the forward closeout. These are bonded in place and provide extensive interfaces with the

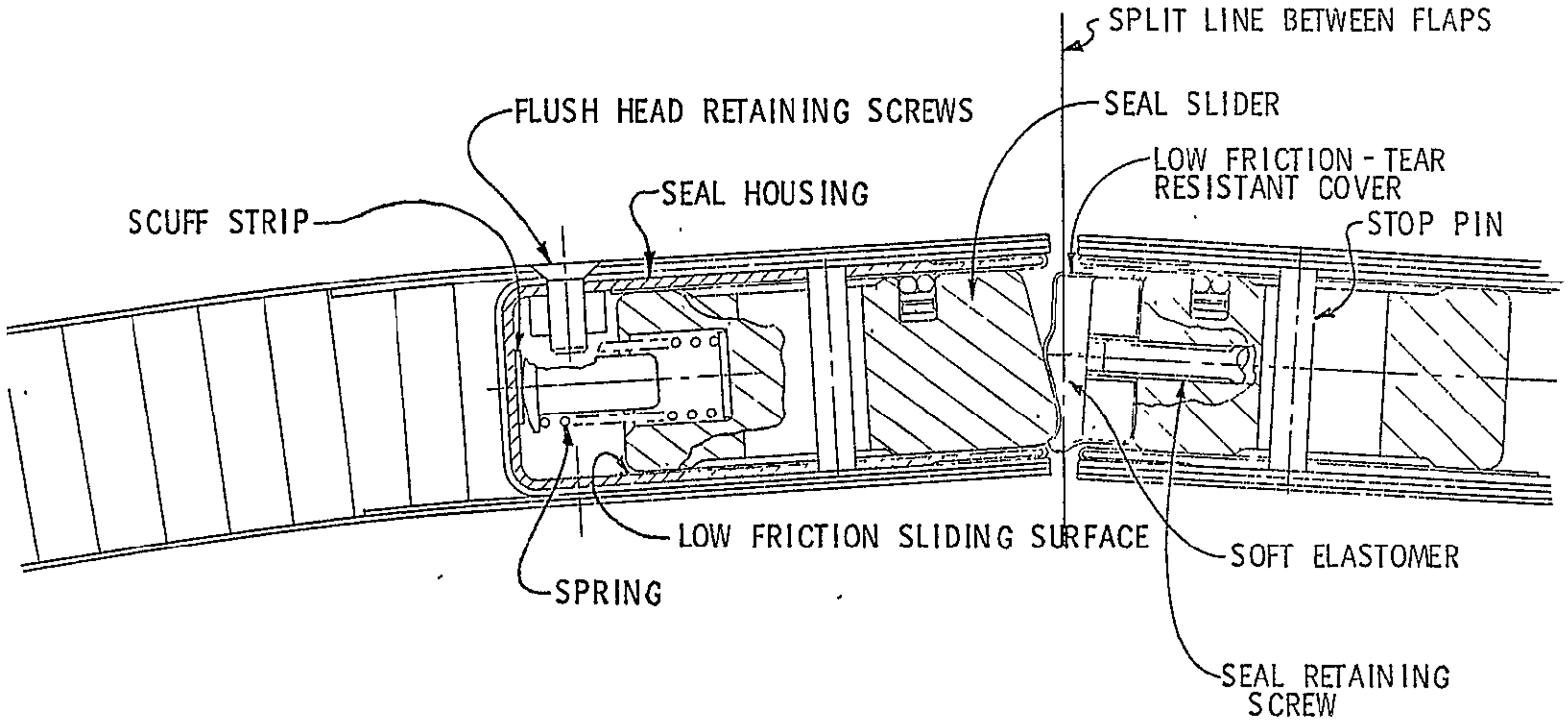


Figure 76. Flap Seal, Axial.

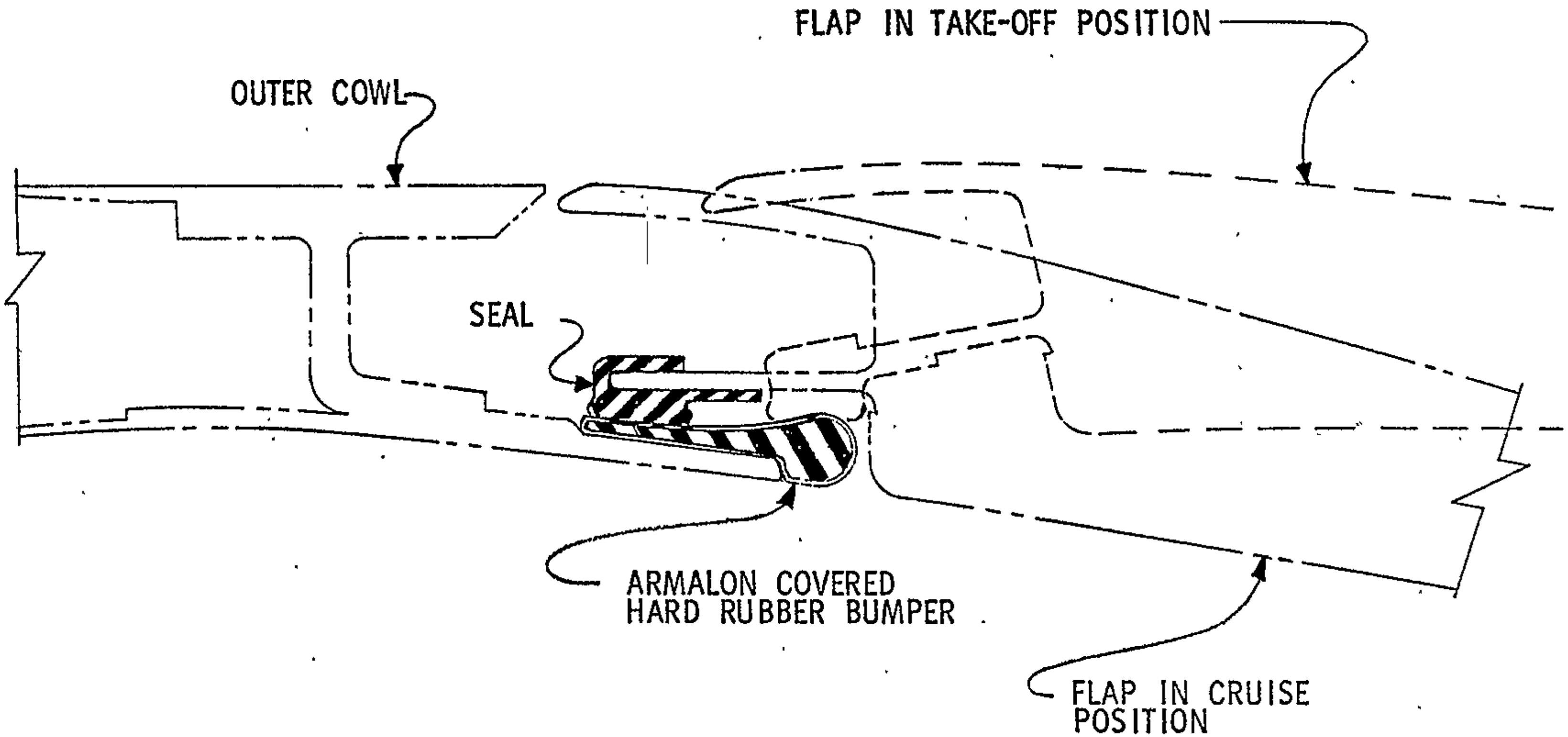


Figure 77. Flap, Circumferential.

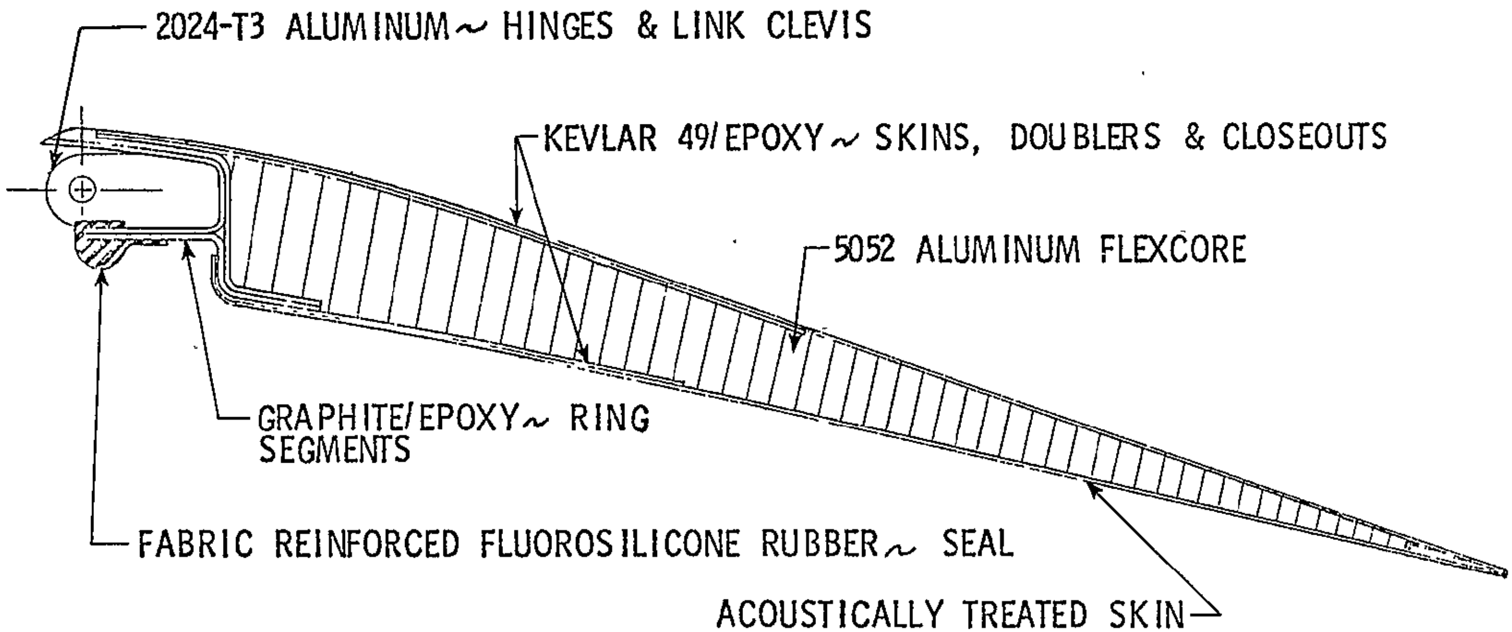


Figure 78. Flap, Cross Section.

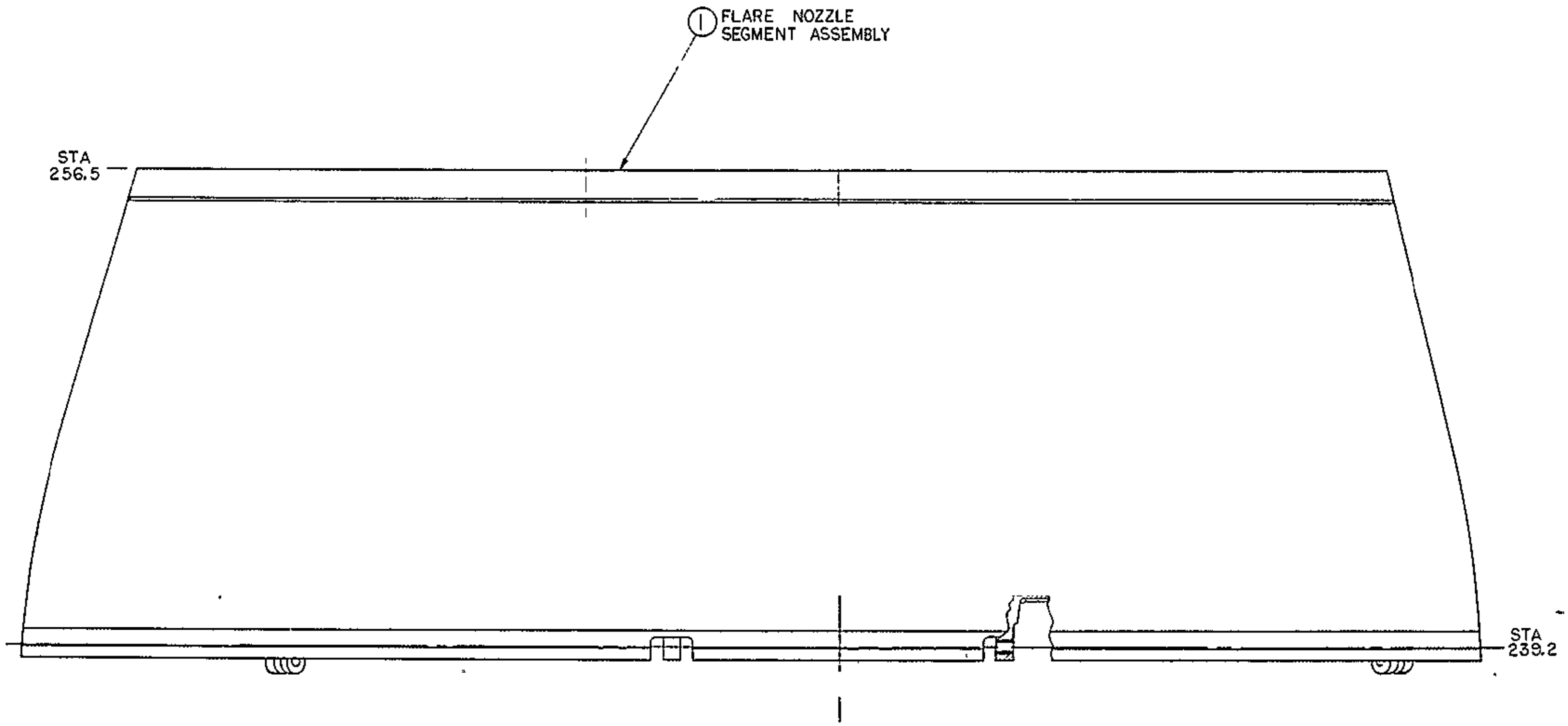


Figure 79. Flap, Construction, External View.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

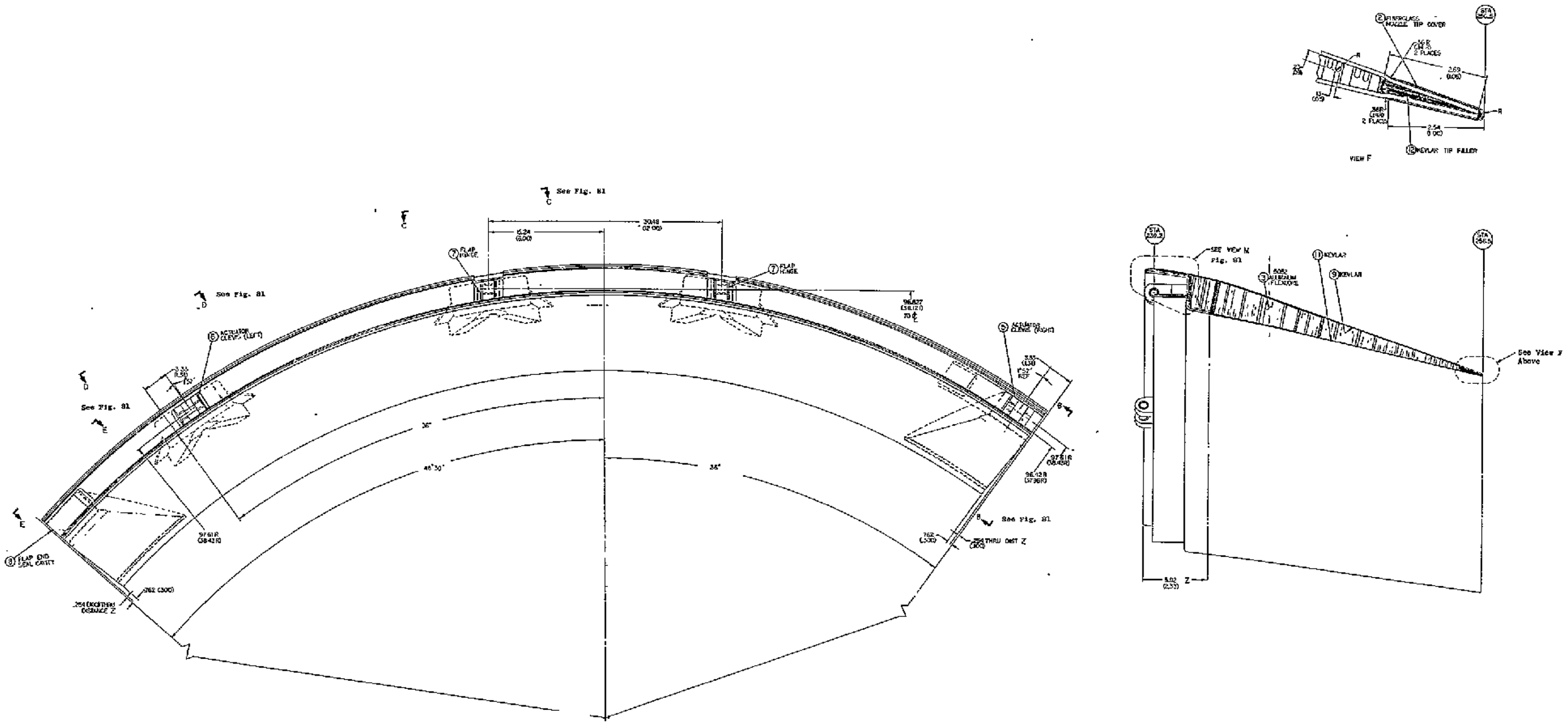


Figure 80. Flap, Construction, Cross Section.

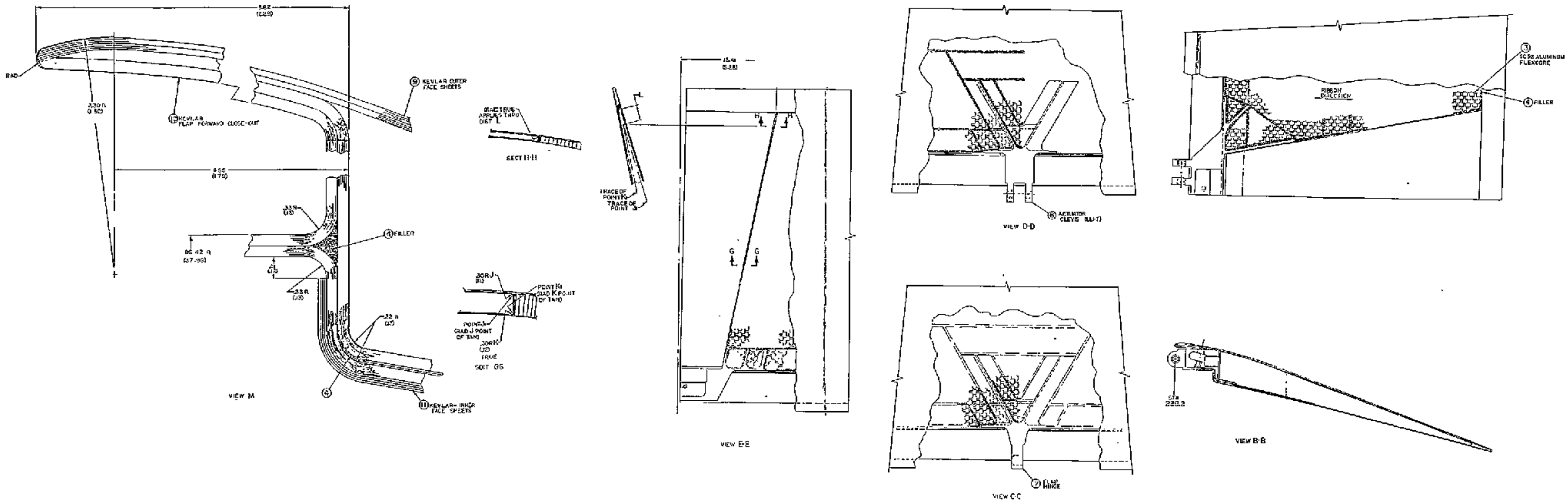


Figure 81. Flap, Construction, Attachment Details.

face sheets and closeouts for load transfer at low stresses. In addition, doublers are provided for both the inner and outer face sheets at the forward end of the flap, in order to reduce the bending stresses in this area. The outer doubler is three plys of Kevlar 49 and the inner doubler is two plys.

The intraflap seal assemblies are spring-loaded pivoted bumpers one of which in each pair has a soft face in order to provide good sealing (see Figure 76) otherwise, the seal components are identical between seal assemblies. The spring forces are such as to cause the seals to maintain contact with each other as the flaps move outward, except that when the nozzle reaches an area of 16,780.6 cm^2 (2601 $in.^2$), a slot in each seal slider bottoms out against a fixed pin mounted in the seal housing, preventing further seal travel. The seals will then stay in the same position, relative to the flap they are mounted in, for any nozzle area greater than the 16,780.6 cm^2 (2601 $in.^2$). The seal at the flap/pylon interface is fixed in the axial closeout of the flap (see Figure 82). This seal will be adjusted at assembly of the outer duct and nozzle to the pylon to have good contact with the sealing face on the pylon. This seal being fixed necessitates the pylon sealing face being perpendicular to the upper flap center line.

The flap was analyzed for the design conditions as shown in Table XXV. The critical forward thrust case was arrived at by comparing flap differential pressures at several points along the maximum flight envelope. This determined that the case of Mach number = 0.92 at 6.4 km (21,000 ft) would give the greatest flap loading. The plus or minus 20 g buffet factor was arrived at from experience in designing components for the C-5A Transport and the DC-10 aircraft. A similar factor was used in those designs. For the reverse thrust case, while the normal landing speed is 41.2 m/s (80 knots), it was decided that the 77.2 m/sec (150 knots) (M = 0.227) case would sufficiently cover an emergency landing or aborted takeoff condition.

The mechanical properties of the materials used are taken from several sources. Metallic components allowables are taken from MIL-HDBK-5, "Metallic Materials and Elements for Flight Vehicle Structures;" the graphite/epoxy allowables are from the "Advance Composites Design Guide" as prepared for the Advanced Development Division, Air Force Materials Laboratory; and the Kevlar 49 properties are from data issued by the E.I. duPont de Nemours and Co., Inc., Wilmington, Delaware, developer and supplier of Kevlar 49. Table XXVI shows critical calculated stresses/loads and margins of safety for the various flap components.

5.3.4 Core Cowl

The core cowl doors define the inner boundary of the fan air flowpath from the fan frame to the core nozzle. They are also used as sound attenuators. Maintenance access to the core engine is also provided. A schematic of the core cowl is shown in Figure 83.

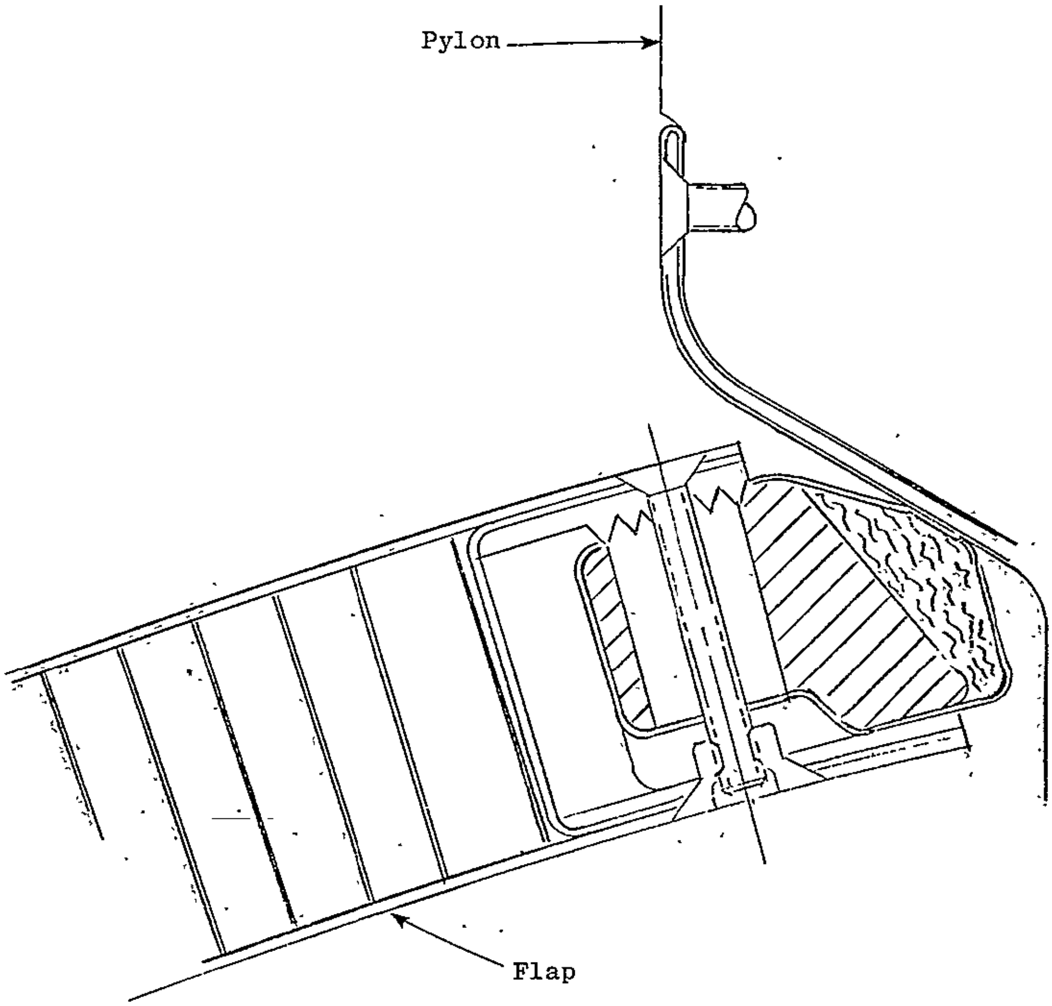


Figure 82. Flap/Pylon Interface, Seal.

Table XXV. Flare Nozzle Flap, Design Conditions.

- Forward Thrust
  - M = 0.92 @ 6.4 km (21K) MXCR
  - Maneuver Loads ∿ MIL-E-5007
    - 10 g down
    - ±2 g aft
    - 1.5 g side
  - Buffet Load
    - ± 20 g
    - C-5A & DC-10 experience
  - Single jammed actuator
- Reverse Thrust
  - M = 0.227 @ SL ∿ MX
  - Rejected T/O or emergency landing

Table XXVI. Flare Nozzle Flap, Load and Stress.

Component	Stress Mode	Stress Load	Allowable Stress Load	Margin of Safety	
Flap Skin	Bending	3,230 N/cm^2 (4,684 psi)	10,343 N/cm^2 (15,000 psi)	+2.2	(yield)
Hinge Lug	Shearout/Bearing	47,989 N (10,789 lbs)	48,052 N (10,803 lbs)	+ 0.001	(ultimate)
Hinge Pin	Bending	165,687 N/cm^2 (240,300 psi)	178,581 N/cm^2 (259,000 psi)	+ 0.077	(ultimate)
Hinge Lug	Shear	9,008 N/cm^2 (13,064 psi)	22,409 N/cm^2 (32,500 psi)	+ 0.2	(ultimate)
Hinge Lug	Bending	10,260 N/cm^2 (14,880 psi)	24,615 N/cm^2 (35,700 psi)	+ 1.4	(ultimate)
Clevis Lug	Shearout/Bearing	11,431 N (2,570 lbs)	35,895 N (8,070 lbs)	+ 2.14	(ultimate
Clevis Lug	Compression	19,051 N/cm^2 (27,630 psi)	24,615 N/cm^2 (35,700 psi)	+ 0.29	(yield)
Clevis Pin	Bending	92,531 N/cm^2 (134,200 psi)	178,581 N/cm^2 259,000 psi	+ 0.28	(ultimate)
Worst Case: Jammed actuator at M = 0.92 @ 6.4 km (21,000 ft)					

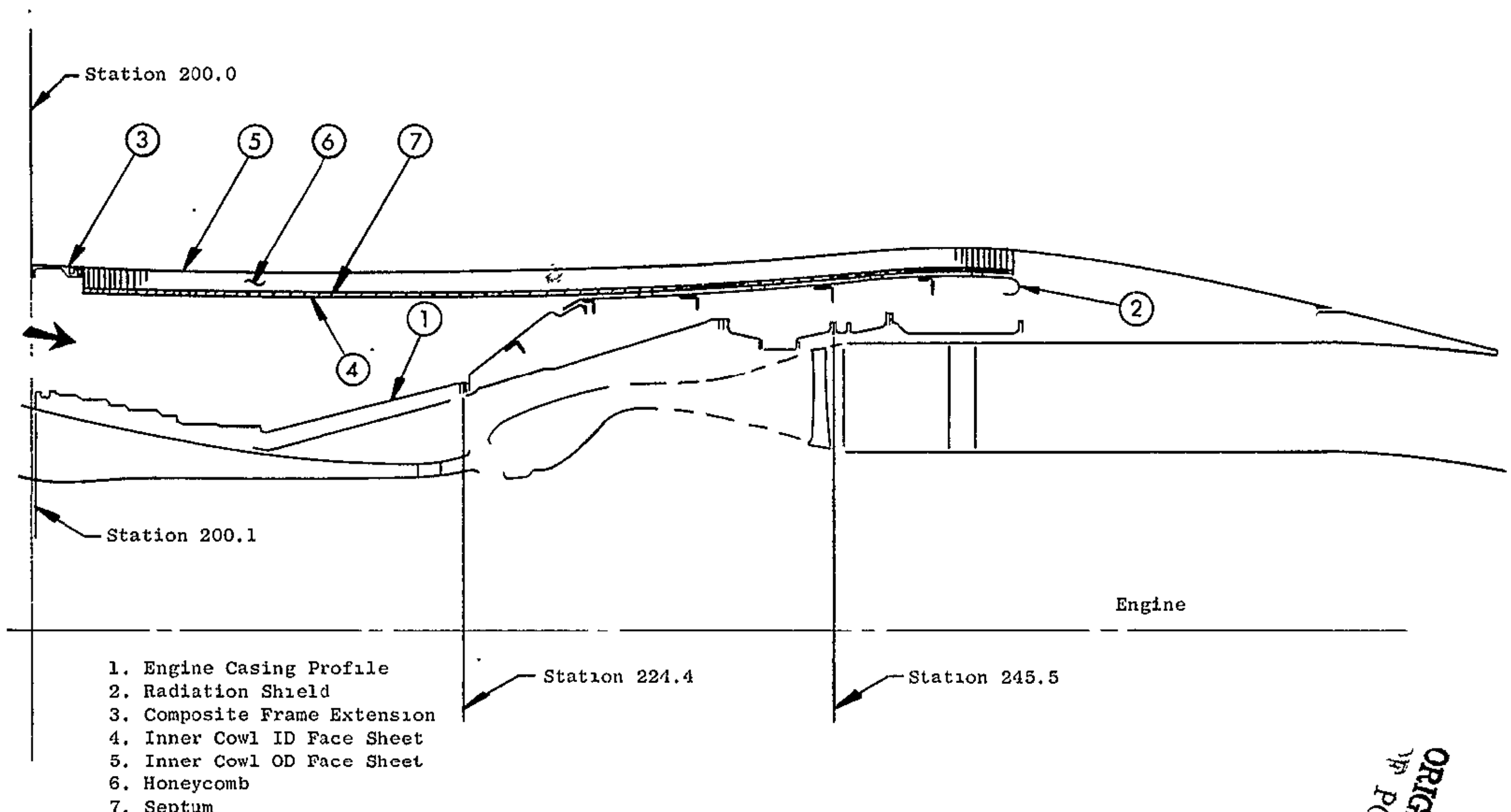


Figure 83. Core Cowl.

Temperature considerations were of primary importance in determining cowl door construction materials and configuration. The inner surfaces of the doors are exposed to radiant and convective heat from engine casings and are insulated from cooling effects of the fan air stream by the honeycomb sandwich walls. A heat transfer analysis showed (Figure 84) that a radiation shield, plus cooling air, is required to keep the maximum cowl skin temperatures within the capabilities of composite materials. Therefore, a stainless steel radiation shield is mounted off the cowl door aft at engine station 235 at a distance of 0.635 cm (0.25 in.) from the inner surface of the door. Cooling air is bled in between the core cowl wall and the radiation shield and exhausted through the slip joint. The total amount of cooling air required to keep the maximum temperature at the desired level is approximately 0.45 kg (1 lb) per second. With the temperatures thus provided, the cowl doors can be constructed of graphite/PI using Dupont's 3003 polyimide resin system or equivalent. High temperature honeycomb (HRH 327) and adhesives (FM34 or HT434) are used to provide the required elevated temperature strength and stability.

5.4 DIGITAL CONTROL

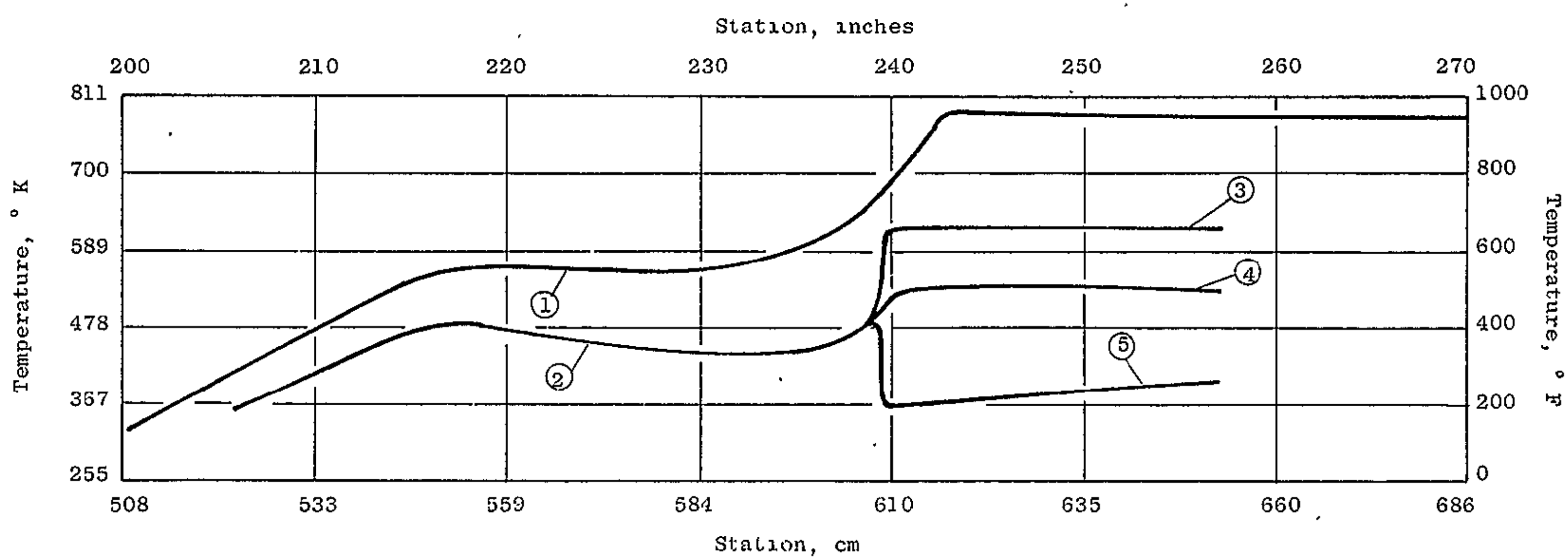
The control system for the flight version of the UTW propulsion system consists of a full authority digital control, and appropriate flow control valves, actuators and sensors. All system components are mounted on the propulsion system. A schematic of the system is shown in Figure 85.

The digital control manipulates three engine variables (fuel flow, fan pitch, and fan nozzle area) in a closed loop fashion to:

- Set percent of rated thrust
- Provide rapid thrust response
- Provide low idle thrust
- Reduce noise level
- Provide positive control of engine limits
- Reduce pilot workload

Studies during the design of the experimental engine have established the sets of controlled and manipulated variables. These sets of variables will be utilized in the flight system after verification on the experimental engine. The prime set of controlled and manipulated variables in the forward thrust regime are:

- Percent of rated thrust is controlled by engine fuel flow.
- Inlet Mach number is controlled by fan nozzle area.



1. Max. Estimated Engine Casing Temperature.
2. Estimated Unshielded Cowl ID Max. Temperature with 1 Chg/Min Air.
3. Estimated Unshielded Cowl ID Max. Temperature with 380 Chg/Min Air.
4. Estimated Cowl ID Max. Temperature with Radiation Shield & 292 Chg/Min Air.
5. Estimated Cowl ID Max. Temperature with Radiation Shield, Thermal Blanket & 180 Chg/Min Air.

Figure 84. Core Cowl, Operating Temperature.

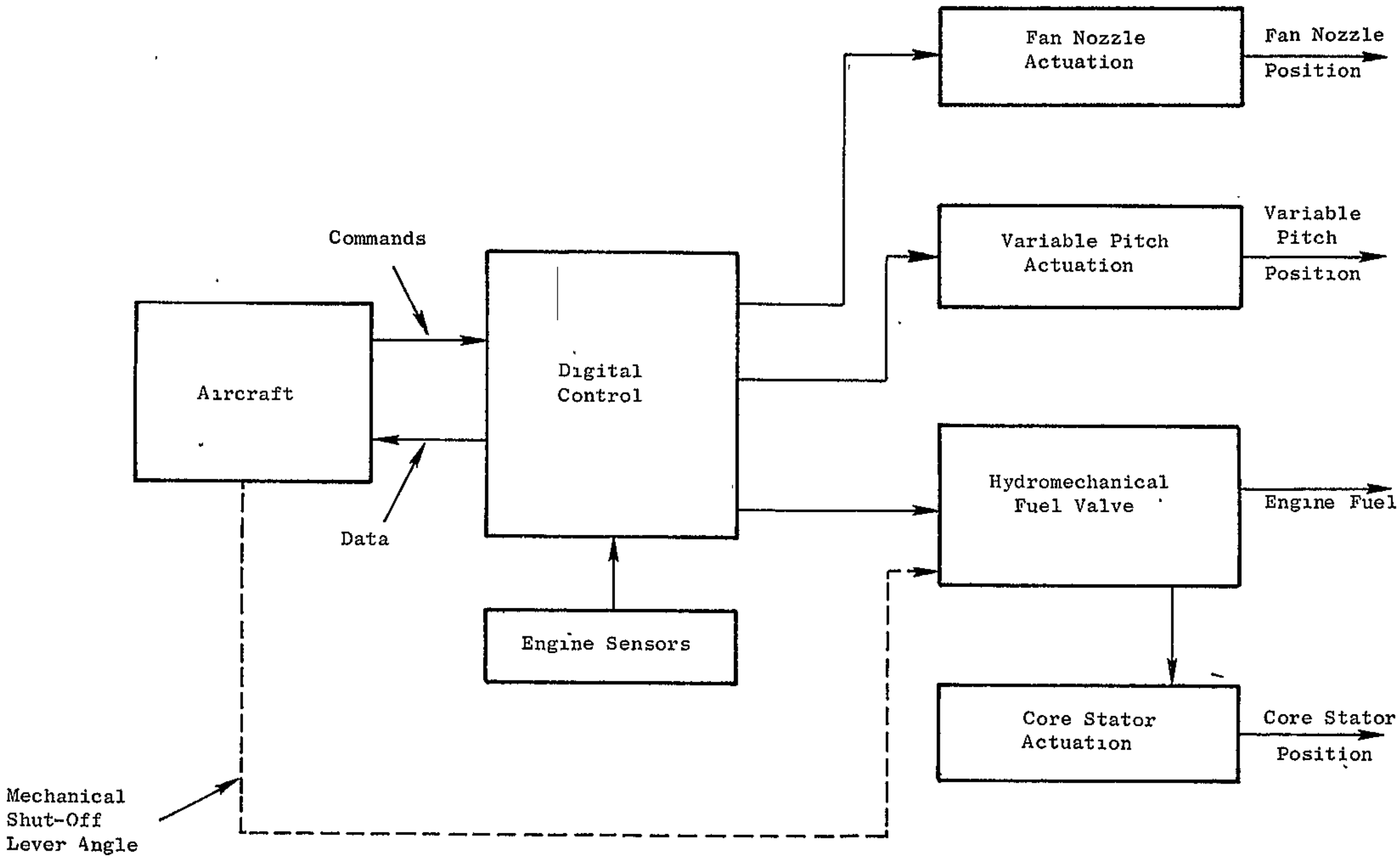


Figure 85. Control System Schematic.

- Corrected fan speed is controlled by fan pitch angle.

- Core stator angle is scheduled by corrected core speed.

In the reverse operating mode thrust is set by commanding a corrected fan speed which is controlled by varying fuel flow while fan pitch is scheduled to the reverse position and fan area is scheduled to the full open position.

5.4.1 Safety Features

The digital control system incorporates functions to automatically prevent the engine from exceeding design limits. The specific engine parameters which are protected by the digital control are low pressure turbine speed (LPT), core speed, and high pressure turbine inlet temperature. The LPT and core speed limits are mechanized by sensing the value of these parameters and comparing to a reference limit within the digital control. If the sensed value tries to exceed the limit, engine fuel flow is cut back. The turbine temperature limit is mechanized by computing the value of turbine inlet temperature from fuel flow and compressor discharge temperature and pressure measurements. Engine fuel flow is retarded if the calculated temperature exceeds a reference value in the digital control. If desired, this reference limit could be scheduled as a function of fan inlet temperature to extend the life of hot section parts. The control system is designed to allow stable, continuous operation on the limit. However, the engine is not expected to run on the limits during normal operation unless engine performance has deteriorated. In addition to the above operational limits, the digital control incorporates automatic fuel flow cutback features to prevent engine damage. These features are:

- Engine fuel flow is limited through a compressor discharge sensor and digital control logic to prevent compressor overpressure. In addition, fuel flow is limited as a function of compressor discharge pressure to reduce damage from turbine overtemperature as a result of compressor surge.

- Engine fuel flow is automatically cut back to idle in the event the actual fan pitch angle differs from the commanded angle by a prescribed amount (i.e., inadvertent reverse).

- Engine fuel flow is automatically cut off if the rate of change of low pressure turbine speed exceeds a prescribed value or the level of low pressure turbine speed exceeds a prescribed value. This event could occur if the low pressure turbine load was lost.

The digital control will incorporate an advanced failure indication corrective action concept to prevent loss of operational capability in the event a control system sensor fails. Essentially, this concept consists of calculating the values of the sensed parameters from the engine inputs by use of a simplified model of the engine which is contained in the digital control.

memory. The calculated values of the sensed parameters are compared with the actual values and the model is updated to zero the error as long as the variation between the calculated and sensed values do not exceed a prescribed limit.

If the prescribed limit is exceeded on a given parameter the sensed parameter is replaced with the calculated value and a signal is sent to the cockpit noting that a specific control sensor has failed. Incorporation of this concept will prevent engine damage and allow normal operation with a control system sensor failure. The concept permits use of the digital controls inherent computational capability and will be more cost effective than redundant or backup sensors.

To allow propulsion system operation in the event of a digital control failure, the flight system incorporates fail-fixed electrohydraulic servovalves, and a simplified hydromechanical fuel control valve. Since the system is a full authority digital control there is an interface device between the electrical commands and geometry actuators. This device is an electrohydraulic servovalve which has been modified to temporarily lock the output stage if the electrical input is zero or hardover in either direction. With this device incorporated, the following events will occur with a digital control failure:

- Fuel flow will be initially held at the level at time of failure and then over a period of several minutes drift upward until it intersects the level required to satisfy the core speed command set by the hydromechanical fuel shut-off lever. Core speed and engine power may be retarded or advanced by movement of the cockpit manual power lever.

- The fan nozzle area will be initially held at the area at time of failure and then over a period of several minutes drift open to approximately takeoff area where the load on the actuators is balanced.

- The fan pitch will be locked in the position it held at the time of failure by the variable-pitch actuator "no back" since the pressure drop across the hydraulic motor will be low.

- The core compressor stators will initially be locked in the position held at the time of failure and then will track with core speed through a function in the hydromechanical fuel valve.

With the above features, the propulsion system will provide, with a digital control failure, nearly normal power in the takeoff regime; altitude thrust, however, will be substantially reduced due to the open nozzle. Transient performance will be limited, pilot monitoring of fan speed and turbine temperature will be required, and reverse operation will not be possible with a digital control failure. Although propulsion system operation will be limited with a digital control failure, the design is based on the assumption that a satisfactory level of reliability will be attained prior to introduction of the digital control. With achievement of the required reliability, the all digital system will be cost effective since cost, failures, and maintenance associated with a complete hydromechanical back-up system will be eliminated.

5.4.2 Condition Monitoring

The digital control will serve as an accumulator and processor of engine condition information. The specific data list has not been finalized but will consist of rotor speed, fuel flow, engine vibration, geometry positions, core engine pressures and temperatures, and lube system temperature and pressure. This data will be transmitted to the aircraft via a multiplex link. In addition to the information processed by the digital control, the following parameters will be sent to the cockpit on separate hardwired cables to allow engine operation with digital control failure: low pressure turbine speed, low pressure turbine discharge temperature, lube oil pressure, and fan pitch. This data will be monitored by the pilot.

5.4.3 Aircraft Interface

The propulsion control will have mechanical and electrical interfaces with the aircraft. The two mechanical interfaces will be on the engine fuel valve and will serve as a fuel shut-off and core engine speed command input when the digital control is inoperative. The electrical interface will occur at the digital control. This interface will receive power command and operating mode from the aircraft and will transmit engine data to the aircraft. This interface problem has been studied by Douglas Aircraft Company. Results of the study are outlined below:

Introduction - The Douglas Aircraft Company has conducted a status projection of the technologies for the 1980 to 1990 time period to be expected in advanced propulsion control systems. In addition to QCSEE-type digital electronic controls, Douglas foresees the use of digital data buses for command and response (control-by-wire), advanced selectable displays (many in graphic form), and both short- and long-term onboard engine monitoring and diagnosis. The primary criterion for introducing any of these features will be that future detailed studies will have shown a high probability for a significant reduction in overall cost of ownership when employing these advanced concepts. In advance of such detailed studies, the ensuing discussion cannot be considered fully definitive, but it is indicative of what is believed will be the major influences on the airplane-to-engine digital control interface.

Architecture - The current Douglas concept relating to how control subsystems will be arranged on mid-to-late 1980 airplanes is based upon the use of multiplexing transmission techniques implemented as a set of (serial-digital) separate data buses. One data transmission bus will serve the propulsion control system, another the flight management system, another the electrical power management system, and so forth. When two subsystems are required to interact, the respective data transmission buses will be linked through a dedicated secondary data path. For example, when the propulsion and flight management subsystems must interact, the secondary link between the Flight Management Computer Complex (autopilot) and the Required Thrust (autothrottle) Computer is connected. A propulsion control system concept featuring this form of architecture for a four-engine (QCSEE type) commercial transport is shown in Figure 86. Redundancy considerations are not yet included in this concept.

Data Format - The data handling subsystem (data bus) used for propulsion control will be in a format compatible with proposed MIL-STD-1553, Reference 4, that is, it will feature:

1. One megaHertz bit rate

2. Manchester biphase level (per MIL-STD-442) encoding

3. Pulse Code Modulation

4. 20 bit words:

    3 synchronization-pattern bits
    16 data bits
    1 parity bit

Data transmission on the data bus will be bidirectional so that both commands to the engines and responses from the engines would be carried on the same medium (two conductor, twisted, shielded, jacketed cable) in the above format.

Description - The engine and aircraft elements of the propulsion control system are tied together by the digital data bus introduced above and shown in Figure 86. The control system for each engine is comprised of the engine-dedicated digital electronic controller and its associated sensors, actuators, and fuel metering servo. In normal operation, all thrust and mode commands from the aircraft are directed to each engine through its digital controller. The outputs of all engine sensors whether for control and monitoring purposes or for monitoring purposes only are routed through the engine digital controller to the data bus. Outputs of those sensors required by the crew to operate the engine when the digital controller has failed are carried on additional, dedicated, hard-wired channels to the cockpit. The required data includes fan speed to be used as the thrust-setting parameter, engine health information such as exhaust gas temperature and engine oil pressure. Under these circumstances thrust commands are conveyed directly to the engine fuel metering servo through a dedicated link with the normal thrust lever. The dedicated backup link may be combined with the same, high reliability one used for fuel shutoff. Reliability and simplicity of the action required of the pilot are the prime requirements for the backup system.

The aircraft elements of the propulsion control system include the Required Thrust Computer (currently called the Autothrottle and Speed Control Computer on the DC-10 airplane), the Data Bus Traffic Controller, the Central

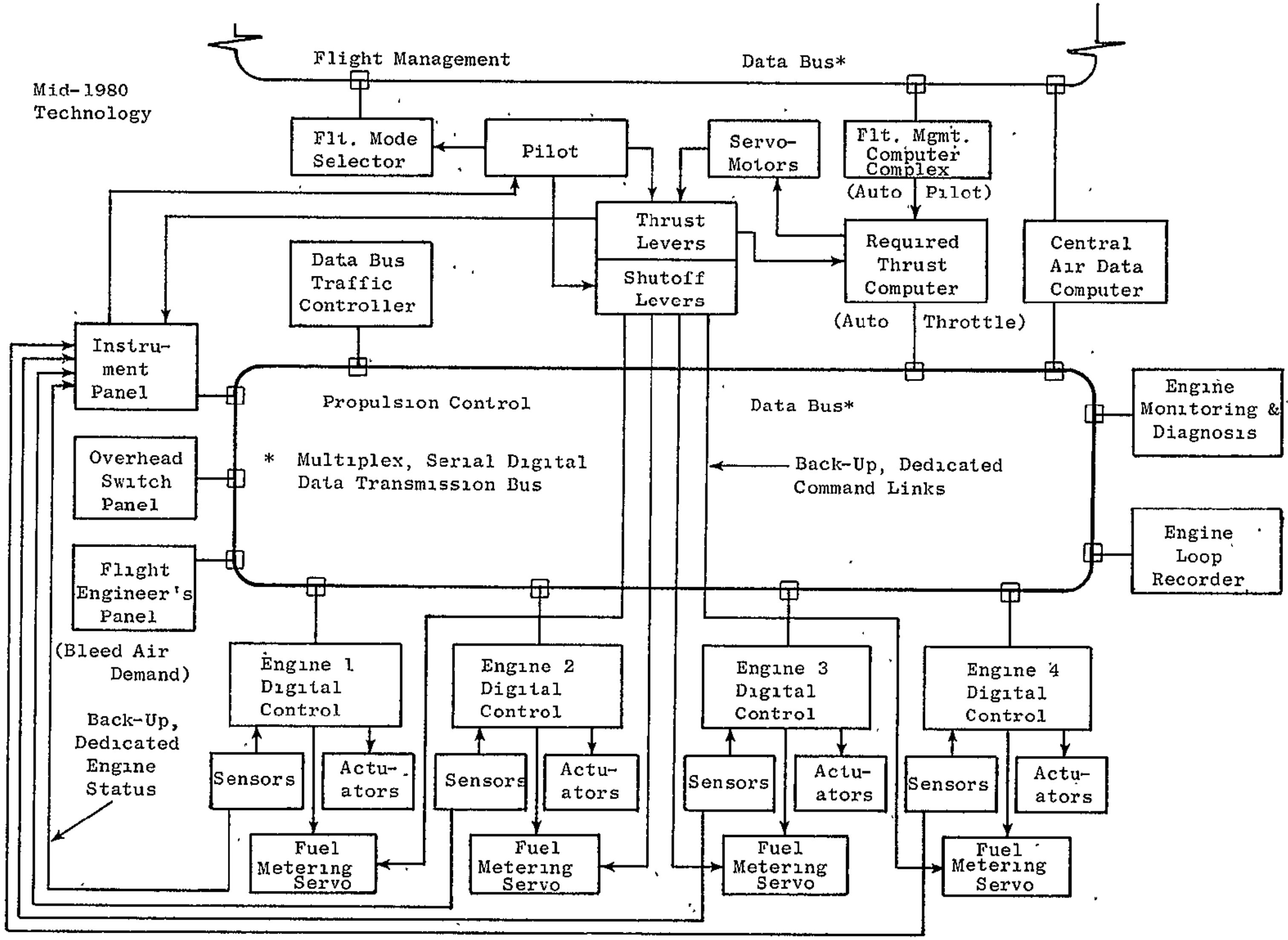


Figure 86. Digital Control System.

Air Data Computer, the Engine Short-Term Data Loop Recorder, the Engine Monitoring and Diagnosis Unit, and the cockpit displays and controls. Thrust commands are fed into the data transmission bus by the Autothrottle Computer. Manual commands are sent to the Autothrottle Computer by the pilot's thrust levers in the form of analog electric signals and are encoded into serial-digital format for transmission to each engine. Automatic commands enter the Autothrottle Computer from the Flight Management Computer via the secondary link in the form of serial-digital signals. These are distributed on the data bus to each engine's digital controller, and are also converted to analog signals to position each cockpit thrust lever via the thrust lever servomotors. Signals generated by the autopilot depend on the flight mode (i.e., takeoff, climb, cruise, descent, approach, land, and reversing) selected by the pilot. The Autothrottle Computer makes possible automatic engine-out compensation in flight and on the ground and automatic reversing during the landing roll. In flight, the actual thrust parameter output of each engine is compared within the engine controller to the thrust parameter command from the Autothrottle Computer. The output is maintained at the commanded value or limited to the available thrust as computed based on selected flight mode, actual flight conditions, stored maximum ratings, and selected percent of derating.

The Data Bus Traffic Controller determines the routine and contingency priority for command and status messages that are placed on the data transmission bus. This follows the general requirements now evolving in MIL-STD-1553. Its operation is highly critical in the propulsion control system. Consequently, suitable redundancy of this function will need to be provided.

The outputs of the Central Air Data Computers are shared with other airplane systems. These provide a variety of computed data based on measured flight conditions to the Autothrottle Computer and the Rated Thrust Computers. This latter function will be discussed further under OPERATION. The Autothrottle Computer will use data such as airspeed, vertical speed, Mach number, and altitude. The thrust rating function of the digital engine controllers will use data such as airplane Mach number, altitude, and ambient total air temperature. In addition, a bleed air demand signal will be required for the thrust rating computation. This is particularly important for the high bypass ratio QCSEE, where bleed demands impact dramatically on performance. This demand can be deduced from bleed valve positions and control switch settings on the system operator's (flight engineer's) panel or it may prove feasible to compute it as part of the onboard reference engine model envisioned for use in the Engine Monitoring and Diagnosis Unit.

Engine data enter the data bus through the digital engine controllers. Selected parameters are recorded at predetermined rates on the Engine Short-Term Data Loop Recorder which stores on a continuous tape loop the most recent 20 minutes of high-rate engine data for incident analysis purposes. Once or more during each flight, when stabilized conditions prevail, engine data are recorded by the Engine Condition and Diagnosis Unit over a brief interval to provide a "snapshot" of engine status. These data are used to update the diagnosis of trends in engine condition. It is planned that this function will be performed onboard the airplane. This onboard diagnosis feature is considered vital for long-range over-water operators. However, it may be more cost-effective for the short-to-medium range domestic carriers (who are most likely to use a QCSEE-powered airplane initially) to perform the diagnosis on the ground. In this case, only a long-term engine trend recorder is required onboard the aircraft. The engine tape would be removed periodically for analysis of engine health trends in a ground facility dedicated to that purpose.

Engine data are also directed to various displays in the cockpit via the digital data transmission bus. This includes engine failure indications and corrective action advisories generated in the Engine Monitoring and Diagnosis Unit and the individual engine digital controllers.

The concept of the redundant data transmission bus implies that special consideration must be given to determining the level of redundancy required. In addition, a selection technique must be developed to resolve which of the command values going to each engine on each redundant data bus to use, and how to use it to maintain balanced thrust across the wing.

Operation - The Autothrottle Computer processes and relays to the data bus both the manual thrust commands originated by the pilot and automatic thrust commands originated by the Flight Management System (autopilot). In automatic control modes, the thrust levers are back-driven to the correct positions by signals from the Autothrottle Computer via the thrust lever servomotors. This enables easy and smooth manual override by the crew.

From the operational viewpoint, it is desirable for thrust to be set and indicated as a percent of each maximum rating. This gives the crew a clear picture of the demands they or the Autothrottle Computer are placing on the engine in each flight regime. It also simplifies the process of setting thrust levels which are derated from the maximum by a selected percentage. Five separate ratings corresponding to the principal engine operating modes must be considered:

Maximum takeoff (static and in-flight)

Maximum continuous

Maximum climb

Maximum cruise

Maximum reverse

These maximum ratings will be precomputed and stored within a computer in the propulsion control system as functions of the flight conditions and aircraft bleed demands. For a selected engine operating mode, flight conditions and selected derating percentage, available thrust can be computed. The arrangement for providing this available or rated thrust computation function is part of each engine digital controller. This permits closed-loop control of the engine to the rated thrust parameter without response and command data having to be sent to the airframe and back. Engines of different ratings can be mixed on the same aircraft. However, aircraft air data and bleed demand data has to be sent to each engine.

If the engine is being operated to hold a certain percent of maximum takeoff thrust, say 97%, the engine digital controller will hold 97% of the corrected (installed) maximum takeoff rating available under given ambient conditions. If the engine is being operated to hold an airplane parameter such as indicated airspeed, Mach number, or for following a selected path such as programmed climb profile, the engine digital controller will maintain that airplane parameter or path unless the required thrust would exceed the available thrust as determined by the installed maximum climb ratings.

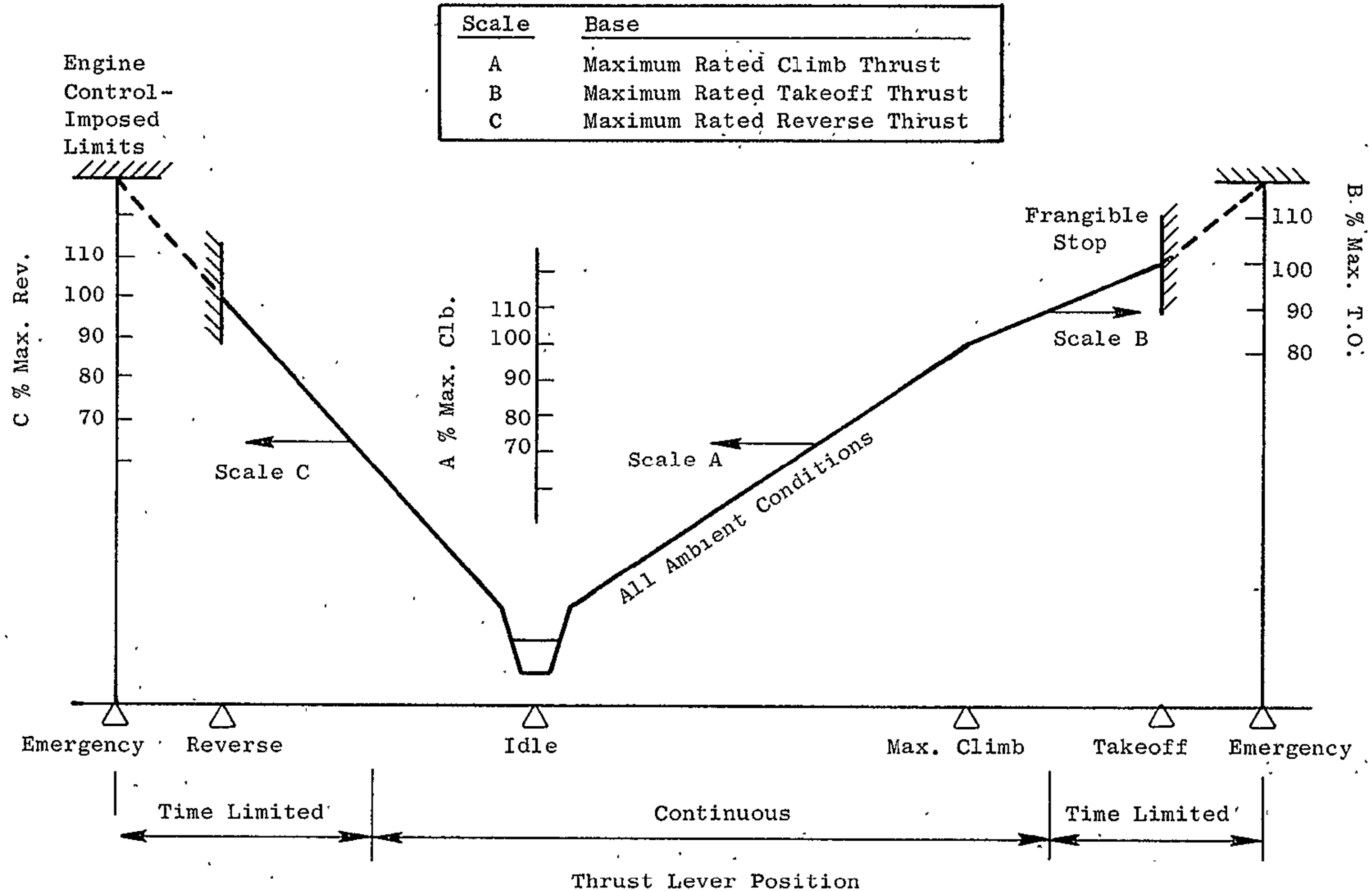
It is desirable to keep thrust mode and thrust level selection a simple procedure. The approach to manual operation would be to divide the thrust lever quadrant into "mode ranges". Moving the lever into a given mode range simultaneously generates a mode signal and a thrust level signal in terms of percent of mode rating. The higher-power end of the range would correspond to 100% of rating for that mode and would be marked by a detent. The takeoff and reverse mode ranges would be terminated by a frangible stop for normal operation. In emergencies, this stop can be broken to obtain all available power. A simplified illustration of this concept is shown in Figure 87. The maximum continuous and maximum cruise ratings are omitted for clarity. In this form, it corresponds to an approach suggested by Kamber, Reference 3.

Failure of electronic controller would require the pilot to close the thrust setting loop and perform overboost protection using rotor speeds and exhaust gas temperature to control thrust. Automatic operation of the engine would be selected by the crew through the Flight Mode Selector Panel which, through the Flight Management System, would provide thrust mode and thrust level signals to the Autothrottle Computer which would obtain and monitor the desired engine operation.

The percent of the appropriate base thrust selected for each engine is displayed on the instrument panel in digital form along with an indication of the actual percent thrust to permit visual checks of the performance of each digital engine control.

For reversing, the thrust levers are manually or automatically moved from the approach thrust position to reverse and can be modulated between reverse idle and maximum reverse. The thrust indicators will display the percent of maximum reverse at which the engines are operating. This is also shown in Figure 87 as the scale C range. Reversing ratings and procedures will have to be determined for a given airplane installation so as to be compatible with limitations created by the potential of ingesting foreign objects from the runway that have been disturbed by the reversing efflux.

Benefits - Implementation of the foregoing airplane-to-engine digital control interface concept on a QCSEE-powered airplane is expected to reduce



'igure 87. Control, Thrust Setting.

significantly the costs of ownership over current aircraft and at the same time be consistent with the highest safety standards. These specific payoffs are foreseen:

1. Reduced Engine Operating Costs - Precise thrust control will save fuel. Precise thrust rating adherence will extend engine life. Expanded monitoring and diagnosis will reduce unscheduled engine removals and secondary damage by showing in advance significant degradation trends and impending failure conditions. Continuous automatic supervision of critical engine parameters will prevent engine distress from over-limits operation.

2. Reduced Crew Workload - Automatic protection against operating extremes and automatic adherence to ratings makes more crew time available for contingencies and monitoring of the total airplane. Automatic engine-out compensation and automatic reversing on the ground helps keep the crew "on top" of contingencies. These features contribute to making it safe and practical for a two-man crew to fly a short-haul airplane powered by four QCSEE-type engines.

3. Improved Terminal Area Flight Control - Precise airspeed control provided by improved thrust control enables reduced traffic spacing when necessary. Precise altitude control provided by improved integration of thrust and aerodynamic controls enhances automatic landing performance capabilities. The close tie of precise thrust control with advanced flight management techniques and advanced scanning microwave landing systems will enable use of steep, curving approaches and departures without compromising passenger comfort while reducing the noise "footprint" on the ground and avoiding population concentrations in the vicinity of the airport.

5.5 FUEL/LUBE SYSTEM

QCSEE bearings and seals components, other than the main reduction gear, can operate throughout the flight map with oil supply temperature of 149° C (300° F) and oil scavenge temperatures as high as 177° C (350° F). Local bearing temperatures rise above 177° C (350° F) during some portions of a flight. These components generate 61,600 J/sec (3500 Btu/min) at takeoff power.

The AISI 9310 gears and bearing outer races in the main reduction gearbox have 149° C (300° F) metal temperature limits at all conditions. These components require a lube system with the ability to cool the oil below 93° C (200° F) during most flight conditions. This system must accommodate an additional 70,400 J/sec (4000 Btu/min) from the gearbox during take off.

Fuel system requirements are:

- 50° C (122° F) supply max at all conditions.

- Fuel heating to 0° C (32° F) at filter inlet with -40° C (-40° F) soak and -48.3° C (-55° F) flight.

The following system was selected to meet QCSEE requirements.

- Fuel recirculation to aircraft fuel tanks.
- Split bypass return fuel to A/C tanks - returning portion of lube heat to A/C tanks.
- Ambient cooling of A/C wing surfaces sufficient to avoid tank temperature rise.
- Priority arrangement of reduction gearbox and main lube heat exchangers.
- Combined function oil-to-fuel heating for filter ice protection.

Results of the UTW flight study are summarized here. The selected system exceeds all system requirements.

- Two CF6-size fuel-oil heat exchangers 12 kg/engine (2616) dry weight aluminum removable core - nonbrazed.
- Supplemental oil cooling (recirculation) at cruise and idle descent only.
- No fuel tank overheating.
- 127° C (260° F) AGMA scoring temperature (climb). 131° C (268° F) bearing race temperature (approach).
- Fuel heating capability (de-ice) -41.1° C (-42° F) soak
-53.8° C (-65° F) flight

The system shown in the UTW fuel/oil schematic (Figure 88) was selected to meet QCSEE requirements. Oil discharged from the lube supply pump follows two parallel paths. One path directs oil to all the normal lube system components; the other path routes oil through a supplemental cooler prior to entering the main reduction gear. Thus, only the oil required for the reduction gear is cooled to lower temperature levels. All scavenge oil is routed through a common heat exchanger before returning to the oil tank. Interfaces with the airplane accessories cooling system is discussed in Section 3.2.14.

The heat study conditions as shown in Table XXVII were selected from the November 1, 1974 QCSEE Technical Requirements and the July 18, 1974 Curtiss-Wright Gearing Data.

Heat generation from the reduction gearbox, engine lube, and engine hydraulic systems have been calculated for each condition. Table XXVIII summarizes these results.

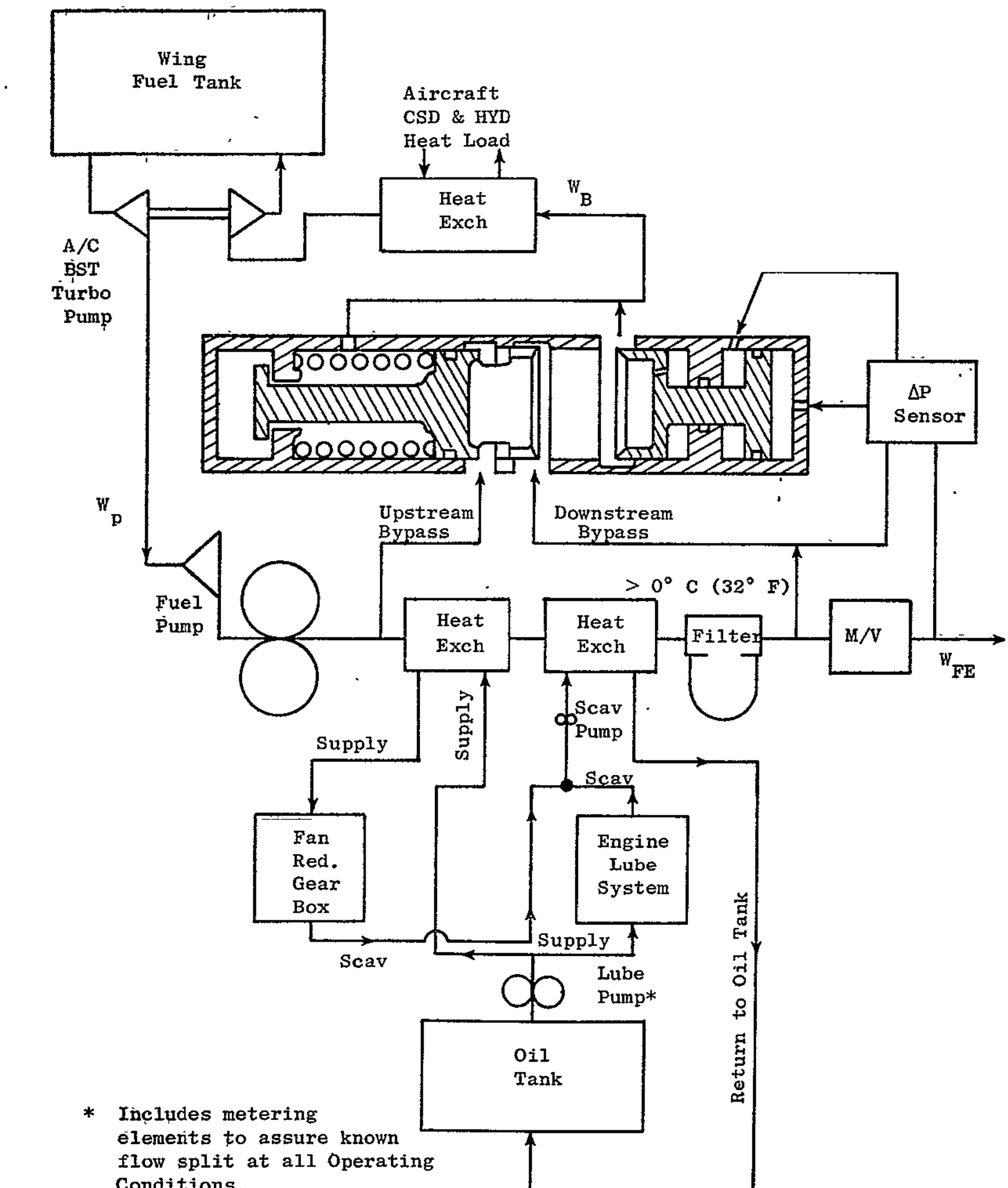


Figure 88. Fuel/Oil Schematic.

Table XXVII. Fuel/Oil Cooling, Heat Study.

Condition	Case No.	Altitude		Flight Mach No.	Reduction Gear Efficiency, %	LPT		Fan Speed, %	Core Speed, %
		m	ft.			MW	HP		
Ground Idle	112	SL	SL	0	98.96	0.08	101	27.8	50.3
Takeoff	1	SL	SL	0	99.30	9.80	13145	97.3	93.3
Climb	301	SL	SL	0.38	99.28	10.51	14097	92.0	92.8
Cruise	404	7620	25,000	0.7	99.24	6.34	8496	102.8	89.8
Descent	503	4572	15,000	0.6	98.99	1.75	2346	61.2	77.8
Approach	8	61	200	0.12	99.26	6.80	9125	95.2	87.4
Reverse	Table XXIX (Thru Stall)	SL	SL	0	99.19	5.47	7335	91.8	89.8
Max Fan Torque	928	SL	SL	0	99.30	11.80	15830	96.2	92.3

Table XXVIII. Fuel/Oil Cooling, Heat Generated.

Condition	Reduction Gearbox		Engine Lube		Engine Hydraulic		Total Q	
	J/sec	Btu/min	J/sec	Btu/min	J/sec	Btu/min	J/sec	Btu/min
Ground Idle	792	45	4,118	234	1,936	110	6,846	389
Takeoff	68,728	3,905	58,837	3,343	3,608	205	131,173	7,453
Climb	75,803	4,307	54,982	3,124	3,590	204	134,376	7,635
Cruise	48,224	2,740	56,126	3,189	3,485	198	107,835	6,127
Descent	17,688	1,005	23,707	1,347	3,010	171	44,405	2,523
Approach	50,442	2,866	47,432	2,695	3,379	192	101,253	5,753
Reverse	44,370	2,521	49,544	2,815	3,467	197	97,381	4,433
Max. Fan Torque	82,773	4,703	56,320	3,200	3,573	203	142,666	8,106

Fuel and oil flows (Table XXIX) are tabulated for the same operating conditions. While upstream bypass fuel is returned to the tanks during the entire mission, heated downstream bypass fuel (after the heat exchanger) is only returned during cruise and descent.

Using the previously tabulated heat rejection rates and fuel/oil flows a fuel tank temperature rise rate was calculated in Table XXX for each condition in the flight map. It was assumed that 14 m^2 (150 ft^2) of wing cooling surface was available for each engine. During all conditions except cruise and descent, the only heat returned to the tanks is from the fuel pump.

Calculated reduction gear bearing and gear temperatures are well within their 149° C (300° F) limits. As seen on Table XXXI, maximum calculated metal tempratures never exceed 132° C (269° F).

A final design objective was to assure that under cold fuel conditions, the fuel entering the filter would be at least 0° C (32° F). As can be seen in Table XXXII, the system meets the design objectives of -40° C (-40° F) ground soak and -48.3° C (-55° F) in-flight cool-down.

5.6 PROPULSION SYSTEM WEIGHT

The UTW "flight" propulsion system is projected to meet the following:

Uninstalled thrust	81,400 N (18,300 lb)	$F_N/Wt = 6.0$
Uninstalled weight	1384 Kg (3,051 lb)	
Installed thrust	77,395 N (17,400 lb)	$F_N/Wt = 4.3$
Installed weight	1825 Kg (4,024 lb)	

The above projection is based on experimental engine thrust and weight (to be measured during the QCSEE Program), with suitable adjustments in weight to account for material and design changes that have been identified as applicable to the flight system.

Uninstalled weight includes all turbomachinery components and related fuel and electrical system.

Installed weight (See Table XXXIII) includes, in addition, the following:

- High Mach suppressed inlet
- Fan aft duct
- Flare nozzle
- Core cowl
- Core nozzle

Table XXIX. Fuel/Oil Flow.

Condition	Engine Metered Fuel WFE		Engine Supply Fuel WP		Total Tank Return Fuel WB		Heat Exchanger Tank Return Fuel		Reduction Gear* Oil Flow		Engine Lube & Hydraulic Oil Flow*	
	Kg/hr	pph	Kg/hg	pph	Kg/hr	pph	Kg/hr	pph	Kg/min	lb/min	Kg/min	lb/min
Ground Idle	288	634	4586	10,110	4298	9,476	0	0	39.9	88	33.6	74
Takeoff	2656	5855	8507	18,755	5851	12,900	0	0	73.9	163	53.5	118
Climb	2776	6121	8461	18,653	5685	12,532	0	0	73.5	162	53.1	117
Cruise	1476	3254	8187	18,050	6711	14,796	852	1879	71.2	157	51.7	114
Descent	577	1273	7093	15,638	6516	14,365	852	1879	61.7	136	46.3	102
Approach	1932	4259	7968	17,567	6037	13,308	0	0	69.4	153	50.3	111
Reverse	1847	4072	8187	18,050	6340	13,978	0	0	71.2	157	51.7	114

*886.8 kg/m³ (7.4 lbs/gal.)

Table XXX. Fuel Tank Temperature Rise Rate.

Condition	Temperature Heat Exchanger Fuel Inlet		Temperature Heat Exchanger Fuel Discharge		Temperature Return Fuel to Tank		Total Return Q to Tank		Fuel Tank** Ambient Cooling		Fuel Tank** Temperature Rise Rate
	° C	° F	° C	° F	° C	° F	J/Sec	Btu/min.	J/Sec	Btu/min.	° C/sec (° F/sec)
Ground Idle	50.5	123	91.1	196	50.5	123	2,411	137*	1,408	80	0.3 (0.6)
Takeoff	51.1	124	136.1	277	51.1	124	7,480	425*	1,408	80	2.3 (4.2)
Climb	51.1	124	133.9	273	51.1	124	7,550	429**	28,582	1624	-7.7 (-13.8)
Cruise	51.1	124	130.6	267	61.1	142	45,690	2596	94,547	5372	-18.0 (-32.4)
Descent	50.5	123	103.9	219	57.2	135	32,138	1826	76,102	4324	-16.3 (-29.4)
Approach	51.1	124	141.1	286	51.1	124	6,688	380*	25,450	1446	-7.0 (-12.6)
Reverse	51.1	124	141.7	287	51.1	124	6,794	386*	1,408	80	2.0 (3.6)

*Fuel Pump 'Q' Only - No Lube Heat

**Per Engine - 4649 kg (10,250 lbs. Fuel) - 14.m^2 (150 ft^2) Lower Wing Surface Cooling

Table XXXI. Lube System Temperatures.

Condition	Scavenge Return Oil		Lube Supply Oil		Reduction Gear Supply Oil		Bearing Race*		AGMA* Scoring Factor	
	° C	° F	° C	° F	° C	° F	° C	° F	° C	° F
Ground Idle	74.4	166	69.4	157	68.9	156	74.4	166	70.6	159
Takeoff	138.3	281	98.9	220	77.8	172	118.3	245	128.9	264
Climb	138.9	282	107.2	225	77.2	171	115.0	239	131.7	269
Cruise	134.4	274	101.1	214	81.7	179	121.7	251	117.8	244
Descent	97.8	208	81.1	178	72.8	163	89.4	193	88.3	191
Approach	142.2	288	112.2	236	92.2	198	131.1	268	131.1	268
Reverse	133.9	253	93.3	200	75.6	168	112.8	235	108.9	228

*9310 Steel 149° C (300° F) Limit

Table XXXII. Fuel Heating Capability.

Condition	Engine Fuel Supply*		Fuel Filter Inlet	
	° C	° F	° C	° F
Ground Idle	-41.1	-42	0	32
Takeoff	-86.1	-123	0	32
Climb	-83.9	-119	0	32
Cruise	-80.6	-113	0	32
Descent	-53.9	-65	0	32
Approach	-91.1	-132	0	32
Reverse	-91.7	-133	0	32

*Design Objectives: -40° C (-40° F) Ground Soak
-48° C (-55° F) In-Flight Cooldown

Table XXXIII. Propulsion System Weight.

Nacelle Components	Equivalent Flight Weight	
	Kg	lb
Inlet	156	345
Fan Duct	74	163
Flare Nozzle	26	57
Core Cowl	44	98
Core Exhaust	64	142
Hydraulic System	10	21
Nozzle Actuation	20	44
Oil Cooler	18	40
Oil Tank	12	25
Fire Detection and Extinguishing	0	0
Instrumentation	11	25
Drains and Vents	6	13
Total Installation	441	973
Engine	1384	3051
Total Propulsion System	1825	4024

- Engine hydraulic system
- Nozzle actuation system (aircraft hydraulic supply)
- Oil cooler
- Oil tank
- Safety instrumentation
- Fluid drains and vents

Normally aircraft furnished components such as starter, aircraft accessory gearbox, bleed piping, and engine mounts are excluded.

6.0 PROPULSION SYSTEM PERFORMANCE

6.1 UTW FLIGHT ENGINE

Engine performance data were generated for the QCSEE Under-the-Wing Flight Engine for use in evaluation of the flight, economic, and acoustic characteristics of commercial, short-haul aircraft. Engine performance over a representative mission profile is shown in Table XXXIV. (For title definitions; see List of Symbols). The performance levels shown for the flight engine are based on component characteristics projected for the certification level engine.

The system performance in Table XXXIV includes the following installation factors:

- Ram recovery
- Pylon scrubbing drag (for that portion of the pylon washed by the engine exhaust flow)
- Core cowl scrubbing drag
- Core exhaust (centerbody) scrubbing drag
- Customer bleed (DAC estimated requirement)
- Customer shaft power extraction (DAC estimated requirement).

Ram recovery utilized in generating installed performance is shown in Figure 89. The inlet throat is sized to produce 0.79 Mach number at a corrected flow of 405.5 kg/sec (894 lb/sec). The loss characteristics are based on estimates provided by DAC.

The pylon drag is based on a scrubbed area of 0.755 m^2 (1170 $in.^2$) (total for both sides). This surface area was determined by projecting the fan exhaust plume outer streamline back to the pylon trailing edge. The outer edge of the plume was established by the constant ambient pressure streamline. Although the outer streamline shifts depend on nozzle position, a single value of surface area is used and the associated drag coefficient is adjusted depending on flight condition.

External nacelle form drag and friction drag are included in the aircraft drag characteristics.

6.2 BOTTOM-MOUNTED ACCESSORIES CONFIGURATION

An alternative engine configuration concept was investigated in which the accessory location was changed to the bottom of the engine. The effect on the engine cycle due to changes in the pylon fairing and the addition of a bottom

Table XXXIV. Performance Data Summary.

Altitude		Flight Mach No.	DTAMB		F_N		SFC		W_1		Bypass Ratio	T_{18}		P_{18}		Interstage Bleed* TB27		Interstage Bleed* PB27		CDP Bleed* TB3		CDP Bleed* PB3	
m	ft		°C	°F	N	lb	g/sN	lb/hr/lb	kg/sec	lb/sec		°C	°F	N/cm²	psia	°C	°F	N/cm²	psia	°C	°F	N/cm²	psia
0	0	0	-0.5	31	76,105	17,110	0.009	0.326	384	847	12.1	56.6	134	12.48	18.1	288.9	552	55.64	80.7	-	-	-	-
0	0	0	-17.7	0	76,105	17,110	0.0088	0.314	396	872	12.1	37.8	100	12.48	18.1	258.9	498	55.78	80.9	-	-	-	-
0	0	0.191	-0.5	31	55,311	12,435	0.0128	0.453	399	879	12.4	58.3	137	12.69	18.4	290.0	554	56.40	81.8	-	-	-	-
0	0	0.378	-17.7	0	45,036	10,125	0.0157	0.554	438	966	12.7	45.6	114	13.72	19.9	265.0	509	59.92	86.9	-	-	-	-
762	2,500	0.395	-17.7	0	42,145	9,475	0.0158	0.560	406	896	12.4	41.7	107	12.82	18.6	261.1	502	56.13	81.4	-	-	-	-
1524	5,000	0.413	-17.7	0	39,365	8,850	0.0160	0.566	377	831	12.3	38.3	101	11.93	17.3	256.7	494	55.40	76.0	-	-	-	-
2286	7,500	0.432	-17.7	0	36,696	8,250	0.0162	0.572	349	770	12.0	34.4	94	11.10	16.1	252.8	487	48.89	70.9	-	-	-	-
3048	10,000	0.452	-17.7	0	34,143	7,676	0.0164	0.579	323	713	11.8	31.1	88	10.27	14.9	248.3	479	45.51	66.0	-	-	-	-
		0.502	-17.7	0	32,466	7,299	0.0173	0.613	332	733	12.0	33.3	92	10.55	15.3	250.0	482	46.40	67.3	-	-	-	-
		0.541	-17.7	0	31,136	7,000	0.0181	0.639	340	750	12.2	35.0	95	10.76	15.6	251.1	484	47.02	68 2	-	-	-	-
4572	15,000	0.593	-17.7	0	26,688	6,000	0.0185	0.654	294	648	11.9	28.9	84	9.38	13.6	242.2	468	40.82	59.2	-	-	-	-
6096	20,000	0.651	-17.7	0	22,574	5,075	0.0190	0.670	254	561	11.7	22.8	73	8.14	11.8	232.8	451	35.10	50.9	-	-	-	-
7620	25,000	0.700	-17.7	0	19,015	4,275	0.0192	0.680	217	479	11.6	15.6	60	6.90	10 0	221.7	431	29.72	43.1	-	-	-	-
9144	30,000	0.700	-17.7	0	16,235	3,650	0.0188	0.664	178	392	11.3	5.0	41	5.65	8.2	207.2	405	23.99	34.8	-	-	-	-
9449	31,000	0.700	-17.7	0	15,675	3,524	0.0188	0.663	171	376	11.2	3.8	38	5.45	7 9	204.4	400	22.89	33.2	-	-	-	-
					11,898	2,675	0.0197	0.695	161	355	12.1	-2.2	28	5.10	7.4	-	-	-	-	305.6	582	47.37	68.7
					11,565	2,600	0.0198	0.699	160	353	12.2	-2.8	27	5.10	7 4	-	-	-	-	302.2	576	46.61	67.6
7925	26,000	0.700	-17.7	0	10,119	2,275	0.0218	0.772	189	416	13.5	3.9	39	6.00	8.7	-	-	-	-	293.3	560	48.95	71.0
7559	24,800	0.681	-17.7	0	2,744	617	0.0439	1.550	166	367	16.3	-3.9	25	5.52	8.0	-	-	-	-	226.7	440	31.85	46.2
6066	19,900	0.649	-17.7	0	3,189	717	0.0430	1.519	197	434	16.5	4.4	40	6.62	9.6	-	-	-	-	238.8	462	37.92	55.0
4511	14,800	0.591	-17.7	0	3,656	822	0.0418	1.477	226	498	16.8	10.6	51	7.72	11.2	-	-	-	-	246.7	476	43.30	62.8
3048	10,000	0.541	-17.7	0	4,670	1,050	0.0377	1.333	251	553	16.3	17.8	64	8.96	13.0	-	-	-	-	257.8	496	50.26	72.9
		0.452	-17.7	0	5,120	1,151	0.0310	1.098	222	490	15.6	12.8	55	8.48	12 3	-	-	-	-	243.3	470	45.58	66.1
1524	5,000	0.413	-17.7	0	5,907	1,328	0.0303	1.069	256	564	15.9	20.6	69	10.00	14.5	-	-	-	-	252.8	487	52.13	75.6
762	2,500	0.395	-17.7	0	6,321	1,421	0.0299	1.058	274	604	16.0	24.4	76	10.83	15.7	-	-	-	-	257.2	495	55.50	80.5
0	0	0.378	-17.7	0	6,752	1,518	0.0296	1.046	293	647	16.2	28.3	83	11.72	17.0	-	-	-	-	261.7	503	59.02	85.6
9144	30,000	0.610	-17.7	0	9,732	2,188	0.0194	0.685	148	327	12.7	-9.4	15	4.76	6.9	-	-	-	-	275.0	527	40.06	58.1

*Bleed flow 0.45 kg/sec (1 lb/sec); shaft power extraxtion, 37 kW (50 HP).

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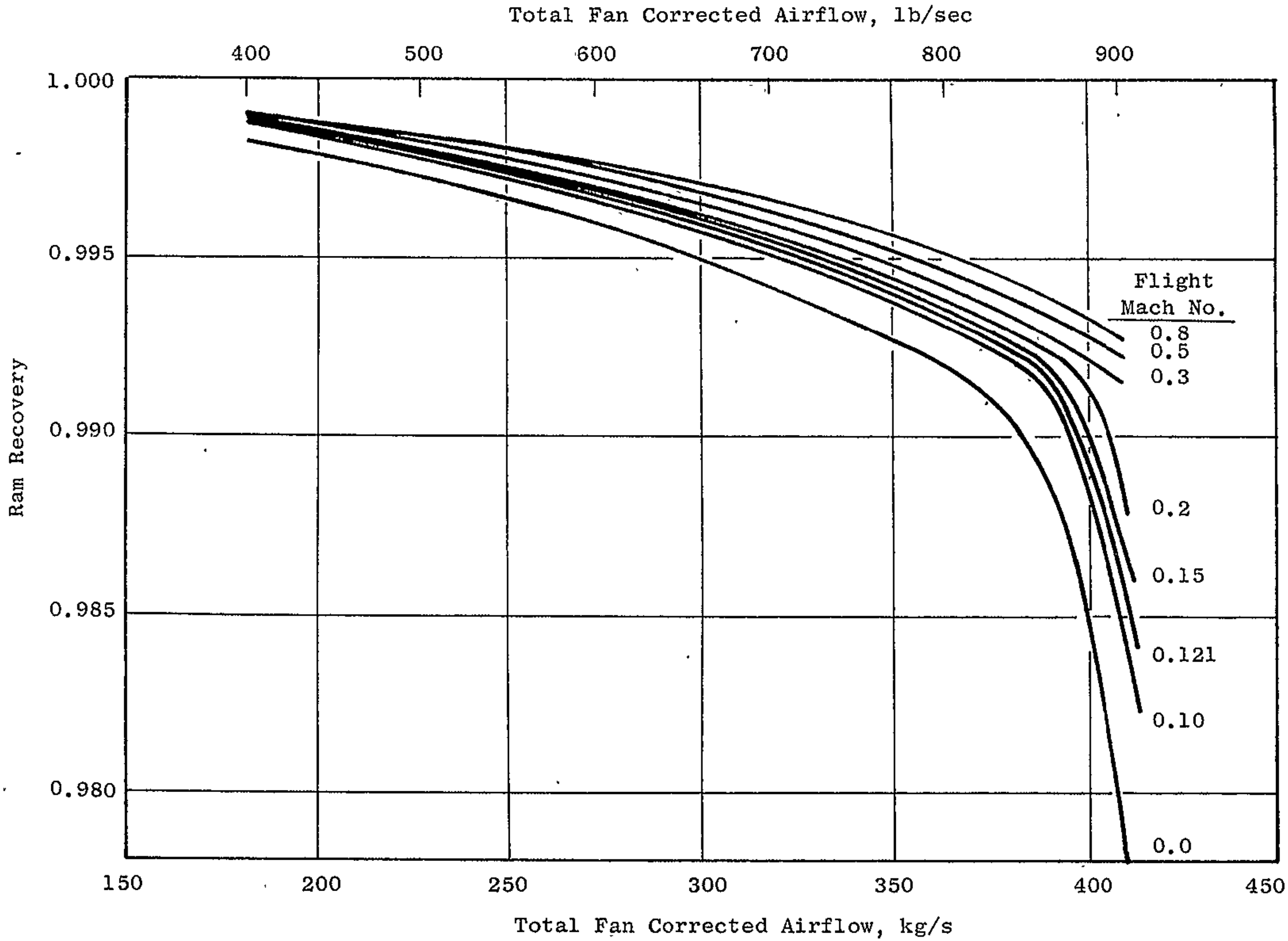

Figure 89. Ram Recovery.

pylon was to increase the fan duct pressure drop by approximately 3 percent. Resulting system performance at important flight conditions is shown in Table XXXV. The first five data points in the table are run to the corresponding engine temperature, the last three to the corresponding thrust level. Effects on engine installation, weight, and system economics are discussed in Section 8.1.

6.3 REDUCED SUPPRESSION CONFIGURATION

Another alternative engine system was investigated in which the acoustic treatment was reduced, allowing reduced inlet and duct lengths but resulting in increased noise levels. The effect on the engine cycle was estimated to be a 6.5 percent reduction in the fan duct pressure drop, and a 50 percent reduction in the turbine exhaust duct pressure drop. The resulting performance is shown in Table XXXVI. The first five data points shown are run to the corresponding engine temperature level the last three cases are run to the corresponding thrust level. Effect on engine installation, weight, and system economics is discussed in Section 8.2.

Table XXXV. Performance, Bottom-Mounted Accessories.

Altitude		Flight Mach No.	DTAMB		F_N		SFC		W_1		Bypass Pressure Ratio	T_{18}		P_{18}		Interstage Bleed						CDP Bleed					
																WB27		TB27		PB27		WB3		TB3		PB3	
m	ft		°C	°F	N	lb	g/sN	lb/hr/lb	kg/sec	lb/sec		°C	°F	N/cm²	psia	kg/sec	lb/sec	°C	°F	N/cm²	psia	kg/sec	lb/sec	°C	°F	N/cm²	psia
0	0	0.191	0.5	+31	58,496	13,151	0.012	0.439	398	878	12.2	59.4	139	12.9	18.7	0	0	298	568	60.2	87.3						
					55,284	12,429	0.013	0.452	399	879	12.4	58.3	137	12.7	18.4	0.5	1.0	290	554	56.5	81.9						
3048	10,000	0.502	-17.7	0	32,430	7,291	0.017	0.614	332	733	12.0	33.3	92	10.5	15.3	0.5	1.0	250	482	46.5	67.4						
6096	20,000	0.651	-17.7	0	22,556	5,071	0 019	0.671	254	561	11.7	22.8	73	8.1	11.8	0.5	1.0	233	451	35.2	51.1						
9449	31,000	0.700	-17.7	0	15,670	3,523	0 019	0.663	171	376	11.2	3.3	38	5.4	7.9	0.5	1.0	204	400	23.1	33.5						
					11,898	2,675	0.020	0.696	161	355	12.1	-2.2	28	5.1	7.4							0 5	1.0	312	593	47.8	69.3
					11,565	2,600	0.020	0.699	160	353	12.2	-2.8	27	5.1	7 4							0 5	1.0	309	588	47.0	68.1
9144	30,000	0.610	-17.7	0	9,732	2,188	0.019	0.686	148	327	12.7	-9.4	15	4.8	6.9							0.5	1.0	281	537	40.5	58.7

Special data for comparison studies.
First five cases are run to corresponding QCSEE turbine inlet temperature.
Last three cases are run to corresponding QCSEE net thrust.
First case has zero shaft power extraction, other cases have 37 kw (50 HP) extraction.

Table XXXVI. Performance, Reduced Suppression.

Altitude		Flight Mach No.	DTAMB		F_N		SFC		W_1		Bypass Pressure Ratio	T_{18}		P_{18}		Interstage Bleed						CDP Bleed					
																WB27		TB27		PB27		WB3		TB3		PB3	
m	ft		°C	°F	N	lb	g/sN	lb/hr/lb	kg/sec	lb/sec		°C	°F	N/cm²	psia	kg/sec	lb/sec	°C	°F	N/cm²	psia	kg/sec	lb/sec	°C	°F	N/cm²	psia
0	0	0.191	0.5	+31	58,696	13,196	0.012	0 437	399	879	12.2	59.4	139	13 0	18.8	0	0	298	568	60 1	87.2						
					55,453	12,467	0.013	0 451	399	880	12.5	58.3	137	12.7	18.4	0.5	1.0	290	554	56.4	81.8						
3048	10,000	0.502	-17.7	0	32,617	7,333	0.017	0.610	333	734	12.0	33.3	92	10.5	15.3	0.5	1.0	250	482	46.4	67.3						
6096	20,000	0.651	-17.7	0	22,689	5,101	0.019	0.667	254	561	11 7	22.8	73	8.1	11.8	0.5	1.0	233	451	35.2	51.1						
9449	31,000	0.700	-17.7	0	15,737	3,538	0.019	0.660	171	376	11.2	3.3	38	5.4	7 9	0.5	1.0	204	400	23.1	33.5						
					11.898	2.675	0 020	0.692	161	355	12.2	-2.2	28	5.1	7 4							0 5	1.0	311	592	47.6	69.0
					11,565	2,600	0,020	0 696	160	353	12.3	-2.8	27	5.1	7.4							0.5	1.0	308	586	46.8	67.9
9144	30,000	0.610	-17.7	0	9,732	2,188	0.019	0.682	148	327	12.7	-9.4	15	4.8	6.9							0.5	1.0	280	536	40.3	58.5

Special data for comparison studies.
First five cases are run to corresponding QCSEE turbine inlet temperature.
Last three cases are run to corresponding QCSEE net thrust.
First case has zero shaft power extraction, other cases have 37 kw (50 HP) extraction.

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7.0 FLIGHT SYSTEM ACOUSTICS

7.1 NOISE STUDY FOR DACo 914.4 m (3000 ft) RUNWAY AIRCRAFT

Noise predictions were made for the McDonnell/Douglas 914.4 m (3000 ft) runway aircraft employing four QCSEE UTW engines. The predictions were based on aircraft data (trajectory, flap angles, power settings) supplied by Douglas (see Appendix B) and corresponding General Electric engine cycle data. A 25° C (77° F) "acoustic standard day" was assumed. Noise objectives and measuring points are shown in Figure 90.

7.2 UTW NACELLE ACOUSTIC DESIGN

In order to meet the 95 EPNdB goal, the following engine noise suppression levels are necessary:

- Forward radiated fan noise - 5.0 PNdB suppression
- Aft radiated fan noise - 6.5 PNdB suppression
- Core exhaust noise - no suppression.

The required fan inlet suppression level is obtained with a high Mach inlet having a throat Mach number of 0.75 or greater. The inlet design selected for QCSEE has an inlet throat Mach number of 0.79 for suppression margin and good aerodynamic performance (see Section 3.3.1).

The required fan suppression is obtainable with acoustic treatment on the walls only. A fan exhaust acoustically treated splitter is not required. Acoustic features of the nacelle designed to meet the suppression requirements is shown on Figure 91.

7.3 SYSTEM NOISE LEVEL PREDICTIONS

At various points along the Douglas supplied flight path, predictions of the total system (engine + jet/flap) noise levels were made, employing the procedures outlined in Appendix I to the QCSEE Statement of Work. Tables XXXVII and XXXVIII summarize the predictions for the 152.4 m (500 ft) sideline, and are representative of the peak sideline noise levels to be measured on takeoff and approach for the Douglas aircraft. Suppression levels typical of the nacelle in Figure 90 were assumed. The goal of 95 EPNdB for peak sideline noise is met with this suppressed system, the peak level being 95 EPNdB on takeoff, and 90.6 EPNdB on approach. The single engine unsuppressed peak levels are 99.5 EPNdB on Takeoff, and 94.0 EPNdB on approach as shown in Tables XXXVII and XXXVIII.

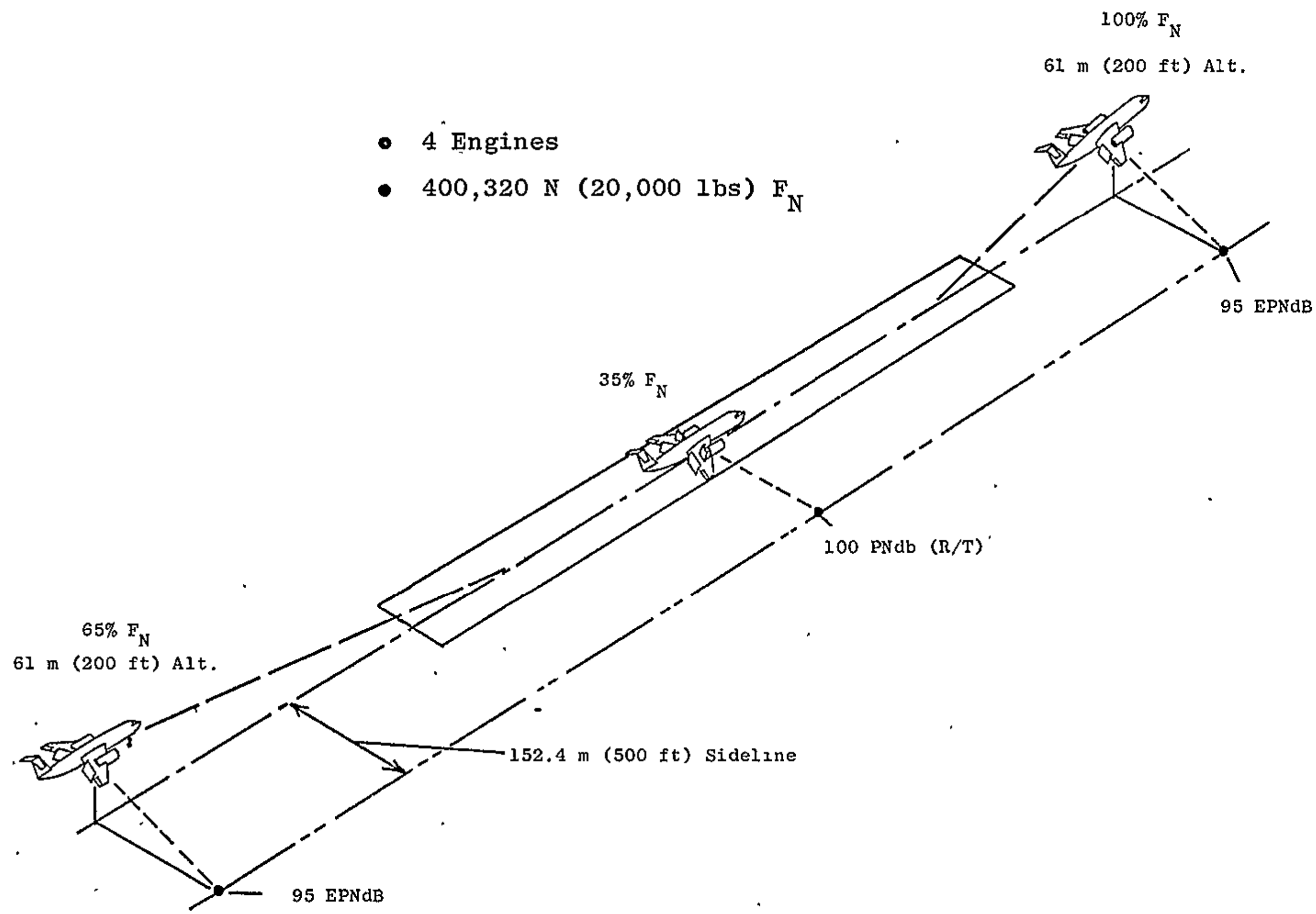

Figure 90. Acoustic Requirements.

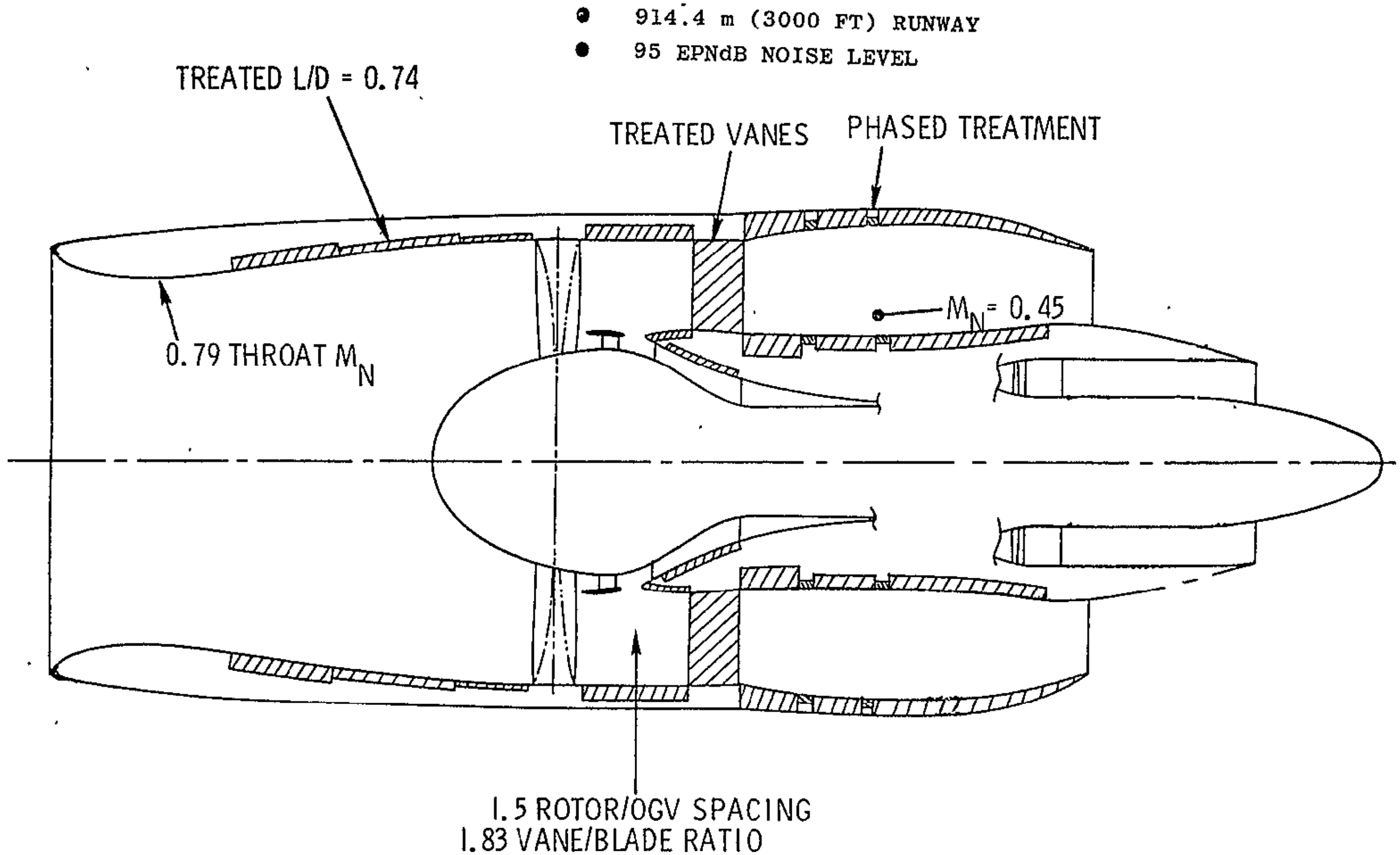

Figure 91. Acoustic Treatment.

Table XXXVII. Takeoff, Peak Noise Data Summary.

- Takeoff Power [F_N = 57,824 N (13,000 lb)]
- 152.4 m (500 ft) Sideline; 76.2 m (250 ft) Altitude
- Aircraft Velocity 66.75 m/sec (129.76 knots)
- Flap Angle = 30°
- V_{Core} = 248 m/sec (813.5 ft/sec)
  V_{Fan} = 209 m/sec (685.7 ft/sec)

	Max. Forward Angle (60°) PNdB				Max. Aft Angle (120°) PNdB			
	Fan	Turb	Comb	Jet/Flap	Fan	Turb	Comb	Jet/Flap
Single Engine Unsuppressed @ 61 m (200 ft) S/L (Includes Azimuth Angle Corrections to Jet/Flap Noise)	107.8	93.1	87.4	99.5	112.2	100.1	97.9	98.0
Corrections - Appendix I Procedure								
Steps 1 thru 5:								
152.4 m (500 ft) S/L, 76.2 m (250 ft) Alt. Single Engine	-9.1	-10.7	-7.9	-8.3	-12.5	-14.0	-11.3	-12.0
Step 6:								
4 Engines	+6.0	+6.0	+6.0	+6.0	+6.0	+6.0	+6.0	+6.0
Fuselage Shielding	-1.2	-1.2	-1.2	-1.2	-1.2	-1.2	-1.2	-1.2
Dirt/Grass Ground	-1.5	-1.5	-0.5	-0.5	-1.5	-1.5	-0.5	-0.5
Step 7:								
Inlet Cleanup	-3.0	--	--	--	--	--	--	--
Step 8:								
Jet/Flap Relative Velocity	--	--	--	-4.5	--	--	--	-4.5
Total Corrections	-8.8	-7.4	-3.6	-8.5	-9.2	-10.7	-7.0	-12.2
Corrected Levels (Unsuppressed)	99.0	85.7	93.8	91.0	103.0	89.4	90.9	85.8
Suppression	-5.0	--	--	--	-6.5	--	--	--
Suppressed System Levels	94.0	85.7	83.8	91.0	96.5	89.4	90.9	85.8

Step 10:	Max. Forward Angle (60°) PNdB	Max. Aft Angle (120°) PNdB
Sum Constituents		
- Unsuppressed	100.5	104.5
- Suppressed	97.1	99.1

Step 11:	
PNdB to EPNdB	EPNdB
- Unsuppressed	99.5
- Suppressed	95.0

Table XXXVIII. Approach, Peak Noise Data Summary.

- Approach Power [F_N = 34,939 N (7,855 lb)]
- Takeoff Fan Speed
- 152.4 m (500 ft) Sideline; 36.6 m (120 ft) Altitude
- Aircraft Velocity 50.45 m/sec (98.1 knots)
- Flap Angle = 48°
- V_{Core} = 186 m/sec (609.8 ft/sec);
  V_{Fan} = 160 m/sec (525.4 ft/sec)

	Max. Forward Angle (60°) PNdB				Max. Aft Angle (120°) PNdB			
	Fan	Turb	Comb	Jet/Flap	Fan	Turb	Comb	Jet/Flap
Single Engine Suppressed @ 61 m (200 ft) S/L (Includes Azimuth Angle Corrections to Jet/Flap Noise)	100.5	87.2	83.9	96.6	105.7	97.0	94.3	92.0
Corrections - Appendix I Procedures								
Steps 1 thru 5:								
152.4 m (500 ft) S/L, 36.6 m (120 ft) Single Engine	-10.2	-11.6	-8.7	-9.2	-12.7	-14.1	-10.9	-12.0
Step 6:								
4 Engines	+6.0	+6.0	+6.0	+6.0	+6.0	+6.0	+6.0	+6.0
Fuselage Shielding	-1.2	-1.2	-1.2	-1.2	-1.2	-1.2	-1.2	-1.2
Dirt/Grass Ground	-1.5	-1.5	-0.5	-0.5	-1.5	-1.5	-0.5	-0.5
Step 7:								
Inlet Cleanup	-3.0	--	--	--	--	--	--	--
Step 8:								
Jet/Flap Relative Velocity	--	--	--	-2.6	--	--	--	-2.6
Total Corrections	-9.9	-8.3	-4.4	-7.5	-9.4	-10.8	-6.6	-10.3
Corrected Levels (Unsuppressed)	90.6	78.9	79.5	89.1	96.3	86.2	87.7	81.7
Suppression	-5.0	--	--	--	-6.5	--	--	--
Suppressed System Levels	85.6	78.9	79.5	89.1	89.8	86.2	87.7	81.7

Step 10:	Max. Forward Angle (60°) PNdB	Max. Aft Angle (120°) PNdB
Sum Constituents		
- Unsuppressed	93.9	97.8
- Suppressed	91.9	93.9

Step 11:	
PNdB to EPNdB	EPNdB
• Unsuppressed	94.0
• Suppressed	90.6

The predicted system noise levels along the sideline, along with those for a direct flyover, were used to generate the estimated noise contours for 90, 95, and 100 EPNdB. These contours are presented in Figure 92 both for the suppressed and unsuppressed engines.

- 914.4 m (3000 ft) Runway Aircraft
- Douglas Aircraft Flight Path
- 25° C (77° F) Standard Day

EPNdB Contour Areas in km² (Acres)

	Unsuppressed			Suppressed		
Contour	Takeoff	Approach	Total	Takeoff	Approach	Total
90	29.2 (721)	7.1 (176)	36.3 (896)	15.1 (374)	5.1 (126)	20.2 (500)
95	12.7 (314)	3.6 (90)	16.3 (404)	6.3 (155)	2.4 (59)	8.7 (214)
100	5.4 (133)	1.5 (37)	6.9 (170)	2.4 (60)	0.9 (22)	3.3 (82)

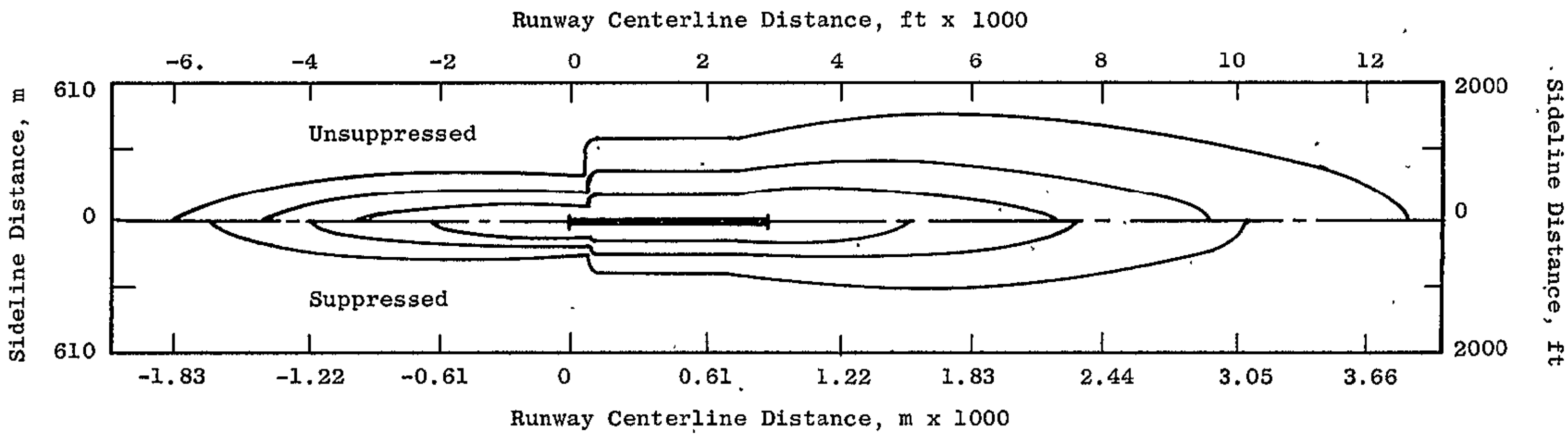

Figure 92. Flight System Noise Contours.

8.0 ECONOMICS

8.1 PAYLOAD-RANGE AND DIRECT OPERATING COST

Payload-range performance for the airplane is shown in Figure 93 for three operational condtions:

- Design takeoff gross weight: M = 0.7 cruise
- Maximum takeoff gross weight: M = 0.7 cruise
- Maximum takeoff gross weight: long-range cruise speed

Maximum payload consists of 162 passengers and bags at 91 kg (200 lb) each plus 1588 kg (3500 lb) of cargo. All fuel is carried in the wing outside the fuselage. Maximum fuel capacity is 18,600 kg (41,000 lb). With this amount of fuel, ferry range exceeds 3700 km (2000 Nm).

Figure 94 shows how direct operating cost (DOC) varies with stage length. The airplane reaches the maximum gross weight limit at 1389 km (750 Nm). For longer stage lengths, payload must be reduced as fuel is increased so that this gross weight is not exceeded. The payload causes DOC to increase.

A breakdown of the direct operating cost for the design point is shown in Table XXXIX. Fuel cost is the largest single contributor to DOC, making up almost a third of the total. A fuel cost of 7.93 cents/liter (30 cents per gallon) was used in the calculations. The following method, based on the operating experience of Douglas customers, was used for calculating DOC. QCSEE engine maintenance requirements are assumed to be the same as those of current turbofan engines. All costs are in 1974 dollars.

¢/ASNM = ($/flight) x 100/NS/R

where:

Ns = number of seats = 162
R = stage length Km (N)

Trip cost in $/flight is made up of the following components:

Crew:

$$\$/\text{flight} = 1.504\ [\text{TOGW}(5 \times 10^{-5}) + 100]\ t_b$$

where

TOGW = maximum takeoff gross weight kg (lbs)
t_b = block time (hr)

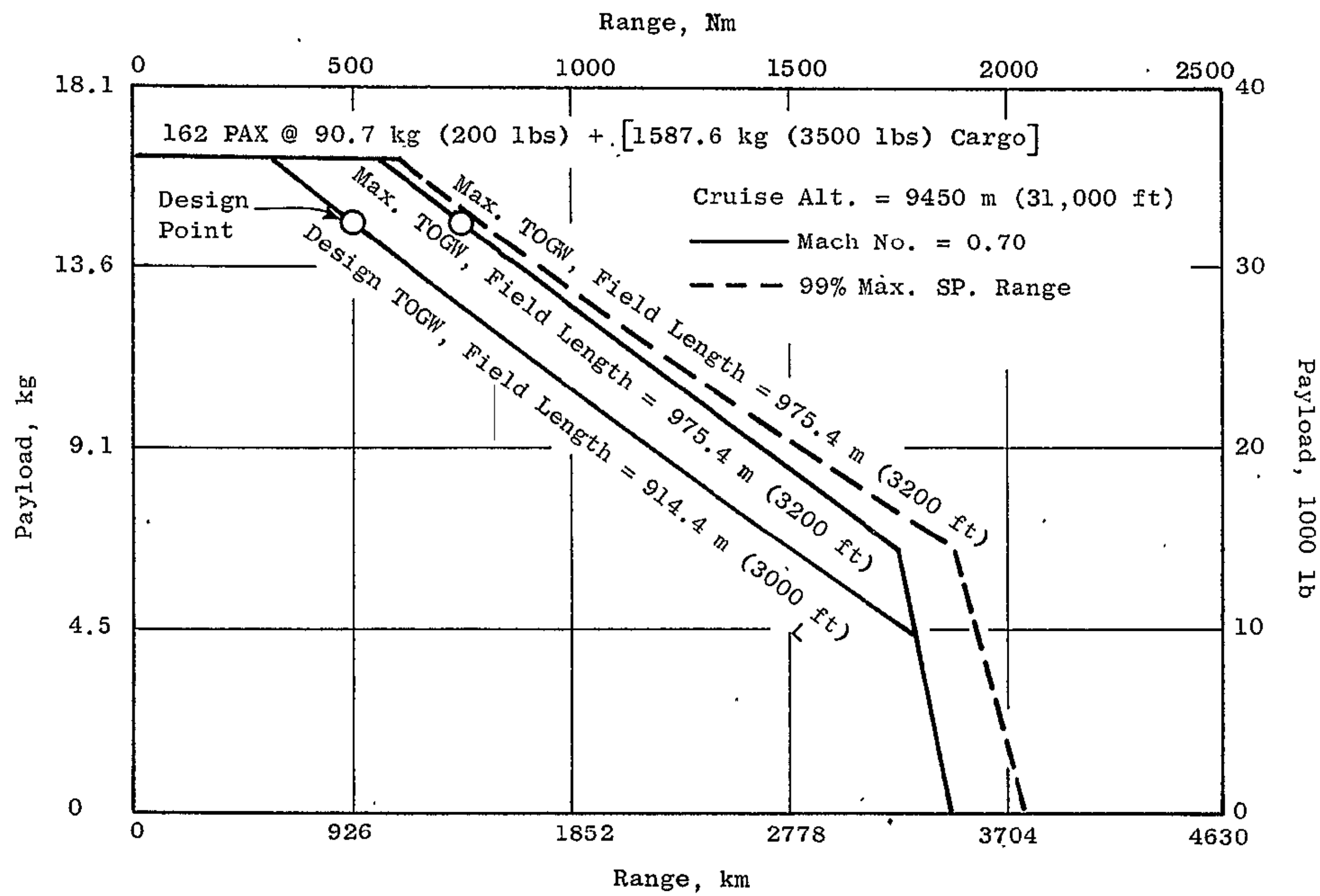

Figure 93. Payload, Range.

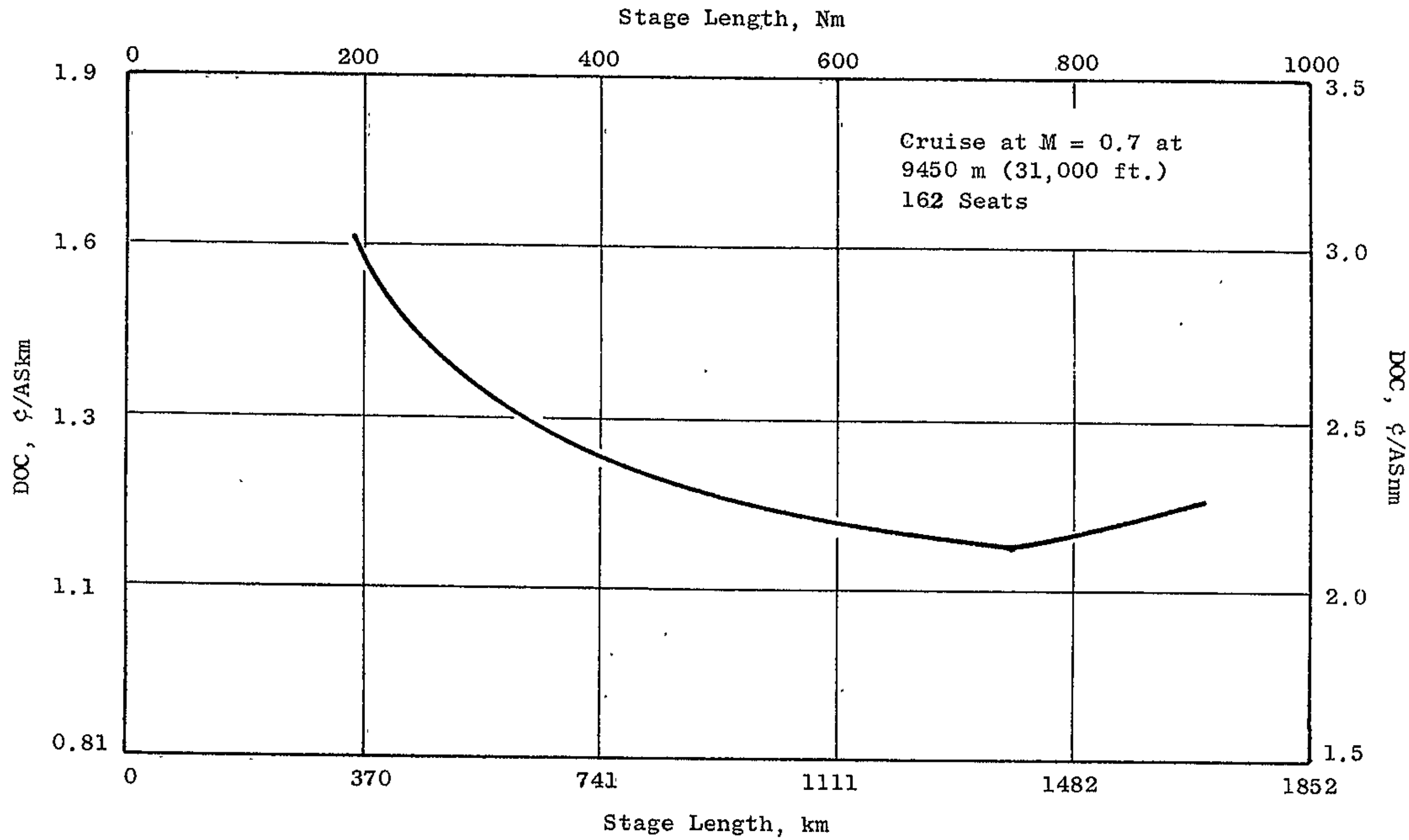

Figure 94. Direct Operating Cost vs. Stage Length.

Table XXXIX. Direct Operating Cost Breakdown.

Component	\$/Flt Cycle	Percent of Total
Crew	230	12
Insurance	73	4
Depreciation	499	27
Fuel	546	30
Airframe Maintenance	223	12
Engine Maintenance	267	15
Total	1843	100

(¢/ASKM = 1.23)

(¢/ASNM = 2.27)

Insurance:

$\$/\text{flight} = IC_t\ t_b/U$

where

I = annual insurance rate = 0.01
C_t = total aircraft first price (\$) = 13.2×10^6 for design point
U = annual utilization (hr/yr) = 2555

Depreciation:

$\$/\text{flight}\ (1 - R)\ (C_a\ K_a + N_e\ C_e\ K_e)\ t_b/D/U$

where

R = residual value ratio = 0.1
C_a = airframe price (\$) = $C_t - N_eC_e$
K_a = airframe spares factor = 1.06
N_e = number of engines = 4
C_e = bare engine price (\$) = 1.063×10^6 for design point
K_e = engine spares factor = 1.3
D = depreciation period (yr) = 15

Maintenance - Airframe

Labor: (includes burden)

$$(\$/\text{flight} = L_r\ W_r\ (1.942 \times 10^{-4} - W_a \times 6.9 \times 10^{-10}) \times (t_f + W_a \times 1.54 \times 10^{-5} - 0.606)$$

where

L_r = labor rate (\$/hr) = 7.20
W_a = aircraft empty weight less engines (lb) = 89,460 for design point
t_f = flight time (hr) = $t_b - 0.13333$

Materials:

$$\$/\text{flight} = 1.75 \times 10^{-6}\ C_a\ (t_f + 0.95 + 9.06 \times 10^{-6}\ W_a)$$

Maintenance - Engines

Labor: (includes burden)

$$\$/\text{flight} = 1.68\ N_e\ L_r\ (t_f\ (1 + T/60{,}000) + 0.5)$$

where

T = rated thrust per engine
= 81,400 N (18,300 lbs) at design point

Materials:

$$\$/\text{flight} = 23.6\ N_e\ C_e\ (t_f + 0.33)\ 10^{-6}$$

Fuel

$$\$/\text{flight} = C_f\ F_b/655$$

where

C_f = fuel price = 79.26 ¢/cnm (30¢/gal)
F_b = block fuel (lb)

A 463 km (250 NMi) flight cycle representative of an average operational mission is shown in Figures 95 and Table XL. Note that this specific mission is slightly less severe, in turns of engine duty cycle than the general requirements mission in Section 4.1. Mach number, pressure altitude, and engine net thrust are plotted as a function of time. Cruise Mach number is 0.70 at 7620 m (25,000 ft). A noise-reduction-procedure thrust-cutback is included in the climb portion of the mission.

Figure 96 compares the fuel burned per seat-mile for the QCSEE aircraft with two aircraft currently in service, the DC-9-30 and the DC-10-10. In terms of fuel consumption, the QCSEE aircraft is as good as or better than the DC-9 and DC-10 for stage lengths below 1389 km (750 NMi). Cruise Mach numbers are: QCSEE, M = 0.70; DC-9, M = 0.78; DC-10, M = 0.82. Fuel consumption for all of the aircraft would decrease slightly if they were flown at their long range cruise speeds.

8.2 PROPULSION-RELATED TRADE FACTORS

Trade factors or sensitivity factors in terms of aircraft gross weight, fuel burned, and direct operating cost at 926 km (500 NMi) were calculated to assess changes in engine weight, cost, and SFC.

A 10 percent increase in bare engine weight will increase

TOGW by 1.4%
Block fuel by 1.0%
DOC by 0.5%

A 10 percent increase in bare engine cost will increase

DOC by 2.0%

A 10 percent increase in climb and cruise SFC will increase

TOGW by 1.9%
Block fuel by 9.8%
DOC by 3.4 %

The aircraft if most sensitive to SFC due to the high fuel price 79.26¢/cu m (30¢/gal).

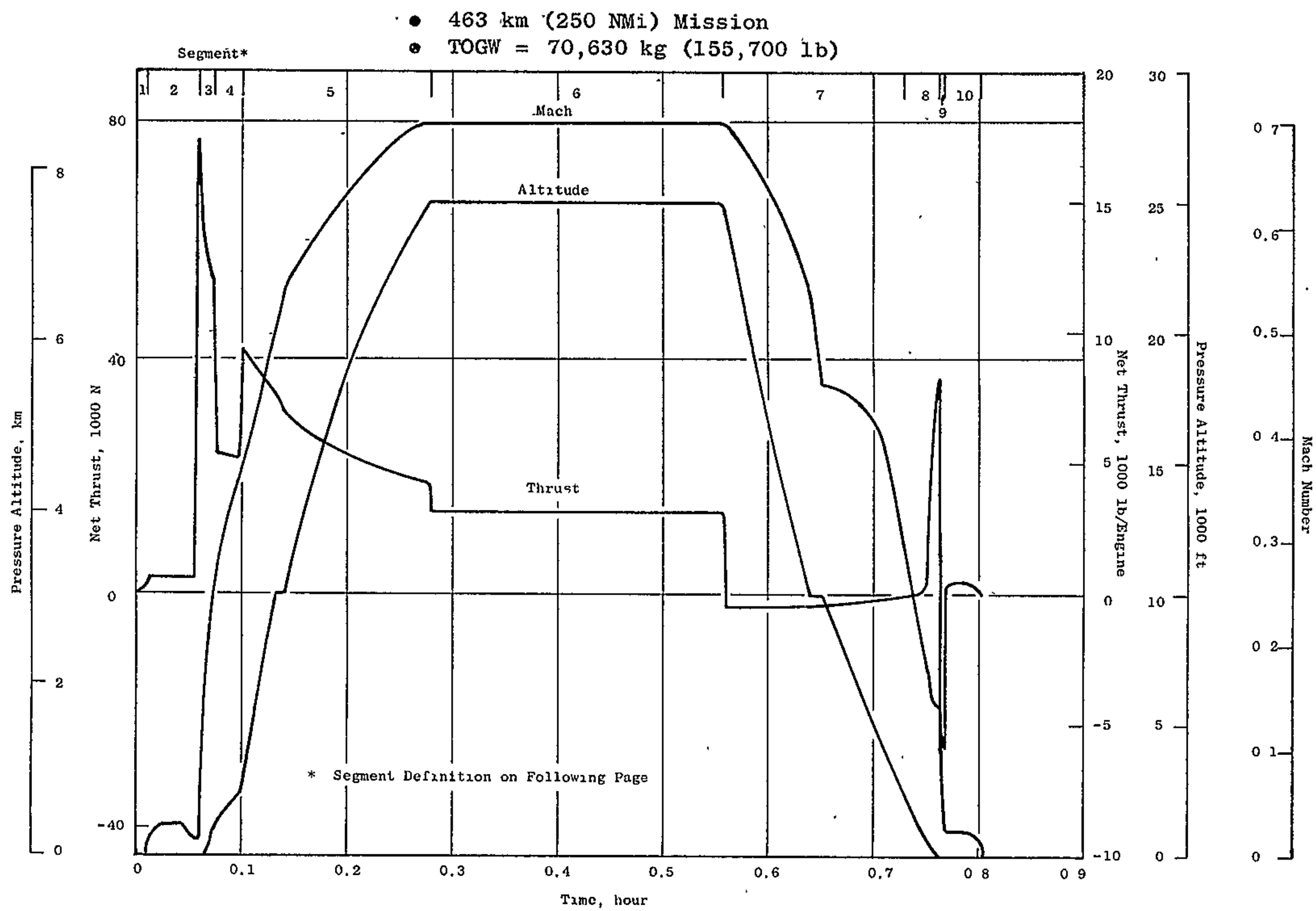

Figure 95. Typical Flight Cycle 463 km (250 nmi).

Table XL. Typical Flight Cycle 463 km (250 NMi) Mission.

Segment	Altitude		Mach No.	Thrust	Time, min.	Time, %
	m	ft				
Start	0	0	0		0.50	1.0
Taxi	0	0	0 - 0.03	Ground Idle	3.00	6.2
Takeoff	0 - 305	0 - 1000	0.02 - 0.25	Takeoff	0.85	1.8
Noise Reduction Thrust Cutback	305 - 762	1000 - 2500	0.25 - 0.37	-65% Takeoff	1.65	3.4
Climb	762 - 7620	2500 - 25,000	0.37 - 0.70	Max. Continuous	10.74	22.3
Cruise	7620	25,000	0.70	-80% Max. Cruise	16.68	34.6
Descent	7620 - 762	25,000 - 2500	0.70 - 0.30	Flight Idle	10.38	21.6
Approach	762 - 0	2500 - 0	0.30 - 0.15	Flight Idle - -65% Takeoff	2.02	4.2
Landing	0	0	0.15 - 0.03	Full Reverse	0.24	0.5
Taxi	0	0	0.30 - 0.0	Ground Idle	2.11	4.4
Total					48.17	100%

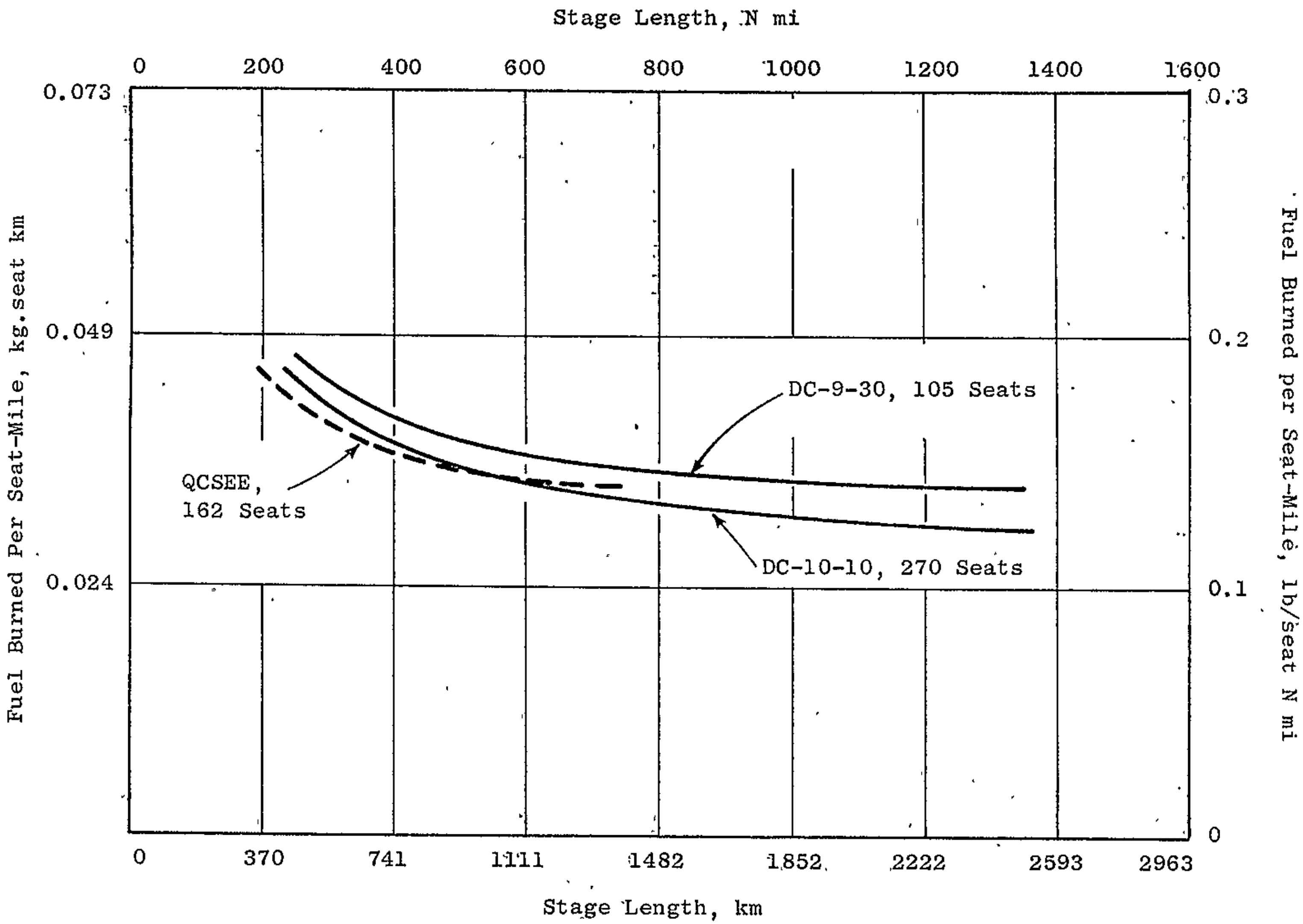

Figure 96. Fuel Consumption.

9.0 ALTERNATE NACELLE CONCEPTS

9.1 BOTTOM-MOUNTED ACCESSORIES CONCEPT

A study was made to identify the impact on the baseline airplane of moving the accessories to the bottom of the engine. The concept, depicted in Figure 97, envisions the internal passages of the pod and engine to be like the baseline QCSEE, except that a streamlined strut is provided at the 6 o'clock position in the fan outlet to enclose the umbilical of services that join the engine accessories to the core.

The exterior of the pod is identical to the baseline QCSEE except that a fairing is added to enclose the engine and airplane accessories, grouped at the bottom of the fan frame.

To minimize the size of this "accessory bump", and to reduce congestion in the accessories zone and also to provide a more favorable operating environment, the engine lube tank and digital control are located atop the fan frame, as in the baseline QCSEE concept.

The accessories fairing is shown extending beyond the fan nozzle to obtain an acceptable boattail angle on the afterbody without resorting to increasing the length of the fan duct, which would have caused a cascade of changes occasioned by moving the engine forward to maintain the desired gap between the fan nozzle and the wing leading edge. The fairing extension introduces the need to make provisions for a flap edge seal as is done for the nozzle flap/pylon intersection. In addition, the presence of the fixed-geometry fairing at the bottom of the nozzle reduces the area change attained when opening the nozzle. Thus the nozzle flap system would be modified to increase the flap deflection to attain the area needed for takeoff and for reversing.

For this study, the accessories arrangement was based on using the angle drive, transfer shaft, and accessories gearbox from the CFM56 engine. It was assumed the services (wire harnesses, hydraulic plumbing, fuel lines, starter turbine air supply duct, electric power feeders, etc.) needed to link the airplane and accessories section could be accommodated in cavities at the rear of the fan frame. This concept is shown in Figure 98. The concept envisions that access covers on the exterior surface, would be provided to close the cavities in the fan frame that house these services. The fuel tube shroud drains and the accessories shafts seal drains would be connected to a drain mast protruding from the bottom of the accessory fairing as is typical of current practice with airplanes such as the DC-10.

The accessory fairing is split and latched on the bottom centerline. The two fairing halves are hinged to the fan frame so the accessories can be exposed for servicing as shown in Figure 99. The section of fairing extending beyond the fan frame is carried on the fan duct cowl panels. Since no equipment components are located in the section, these are merely dummy

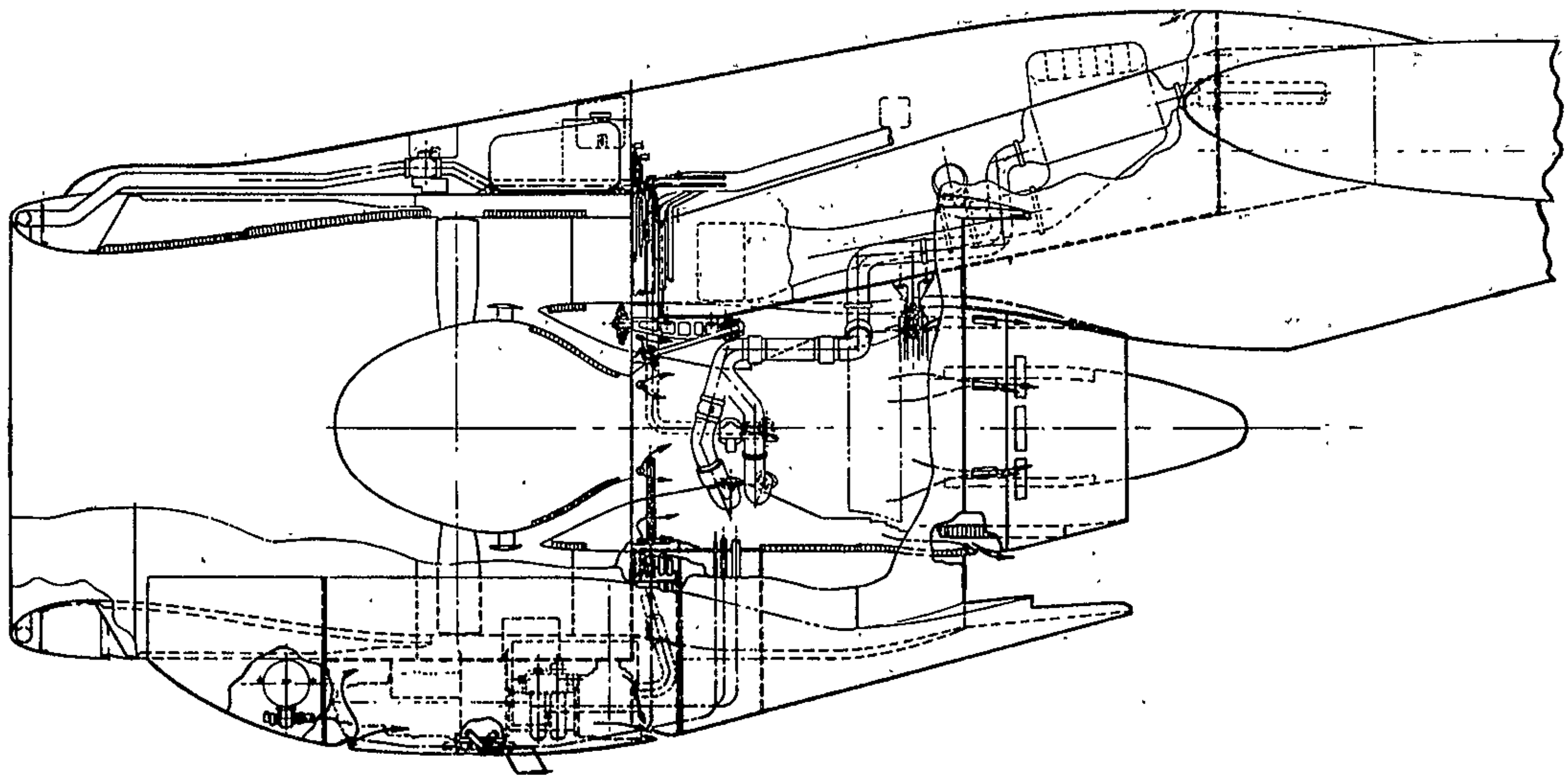

Figure 97. Propulsion System, Bottom Mounted Accessories.

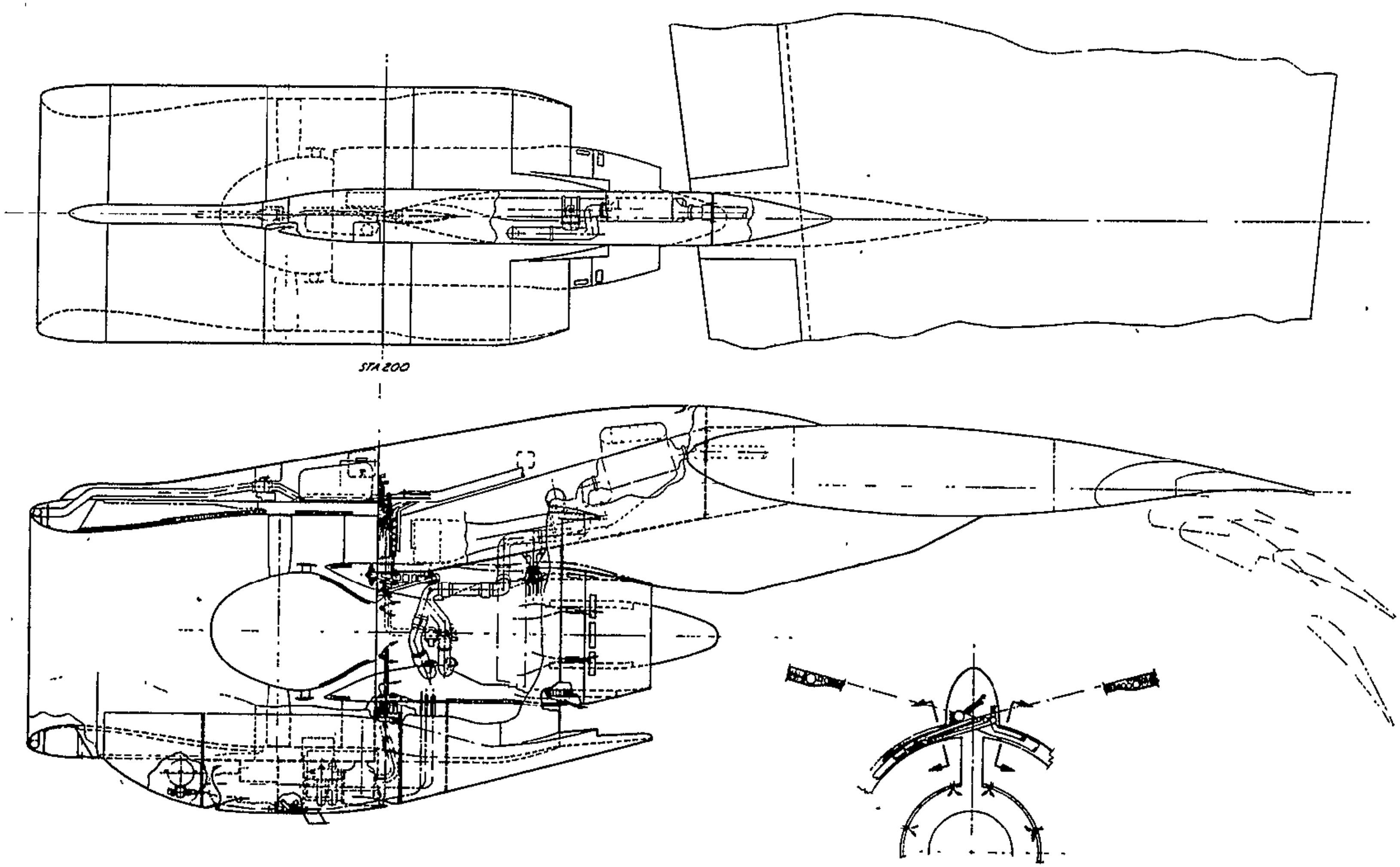

Figure 98. Bottom Accessory, Installation Features.

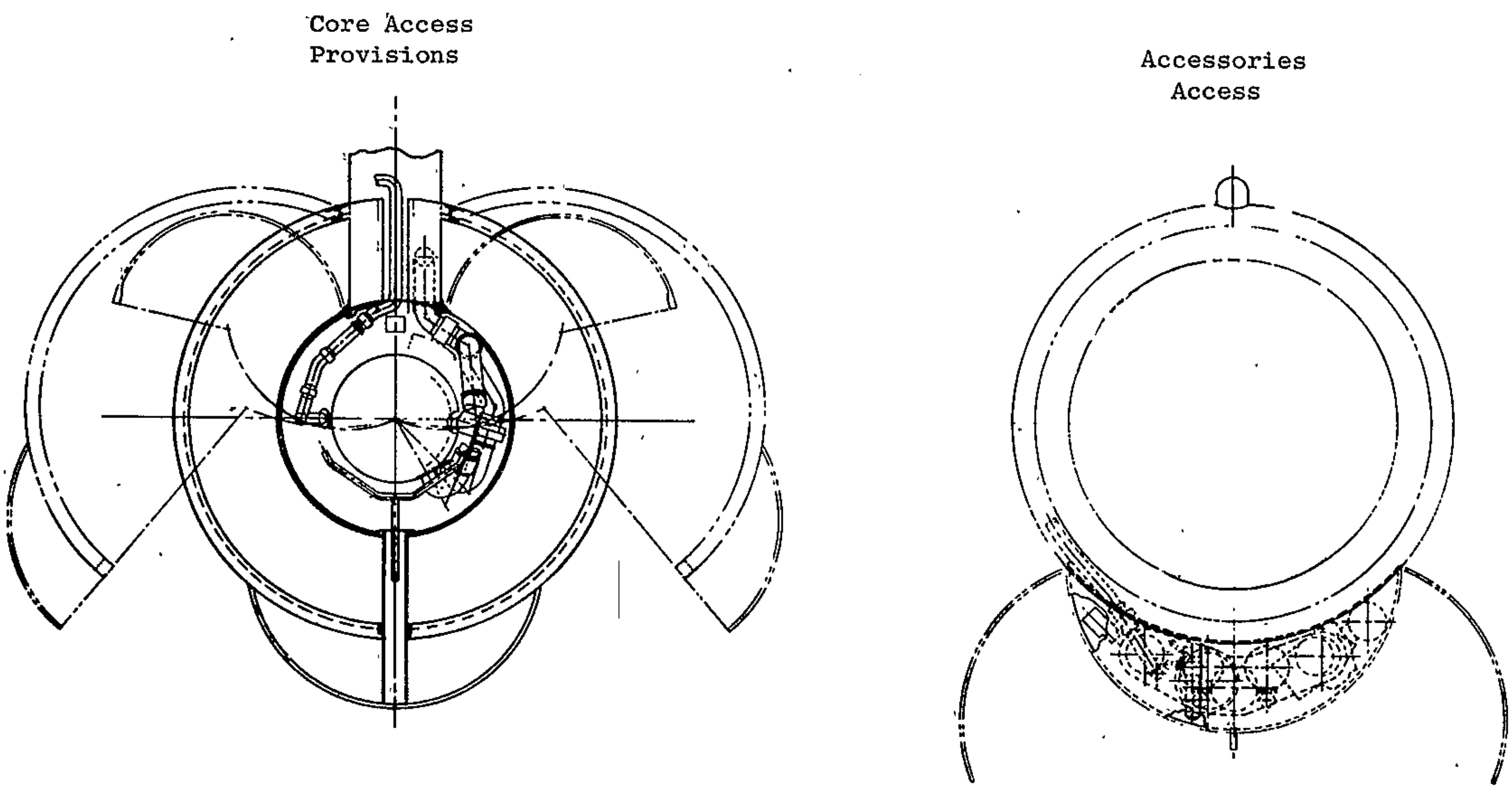

Figure 99. Bottom Accessory, Accessibility.

fairings appended to the fan panels. The space in the leading edge section of the fairing, which underlies the inlet duct has been utilized to house the fire extinguisher agent containers. This serves to simplify the fire extinguisher distributor system by locating the agent containers immediately adjacent to the accessory compartment (in a space isolated by a burn-through resistance bulkhead) and by eliminating the need to make the distributor system traverse the engine/airplane interface, as it would need to do if the agent containers were retained in the wing leading edge as in the top-mounted accessories concept.

The width of the pylon upper fairing was reduced from the 50.8 cm (20 in.) needed to accommodate the QCSEE accessories in the baseline concept to the 40.6 cm (16 in.) needed to fair the 40.6 cm (16 in.) wide pylon beam. The profile of the fairings was lowered as permitted by the removal of the engine and airplane accessories. An assessment of the propulsion system drag indicated that the changes made to the pylon upper fairing width and profile reduced that component of the drag enough to offset the increase attributable to the accessory fairing added to the bottom of the pod.

The weight chargeable to the propulsion system increased 103.4 kg (228 lb). The airplane was then resized to maintain the capability to operate on the 914.4 m (3000 ft) runways. This process showed these changes occur when the accessories are moved to the bottom of the fan frame.

Wing area increases 0.6%
Operator's empty weight increases 0.4% (190.5 kg) (420 lb)
Design TOGW increases 0.3% (195.0 kg) (430 lb)
Mission fuel burn increases 0.1%
Direct operating cost (DOC) increases 0.3%

Maintainability of the Bottom-Mounted Accessories Variant - The study indicates the maintenance burden increases as a consequence of the additional man-hour effort needed to prepare a new engine, or one coming from overhaul, for installation on the wing. The effect is also felt when it is necessary to strip an engine that has been removed for overhaul. The presence of the airplane accessories on the quick engine change unit (QECU) creates congestion, which can be expected to reduce the efficiency of inspection and maintenance actions and thus further increase the maintenance burden.

To help put this discussion in proper perspective, the components added to the QECU when the accessories are located at the bottom of the fan frame are listed below.

Hydraulic Pump

Suction, pressure, and case drain lines
EDV wire harness
Filter module and pressure sensing wire harness
Related attachment hardware

Air Turbine Starter

- Control valve
- Wire harness
- Related attachment hardware
- Air supply duct
- Related clamps and attachment hardware

60 KVA IDG

- Power feeders
- Generator control wire harness
- Decoupler wire harness
- Oil cooler and related tubes and fittings
- Condition sensor wire harness
- Related clamps and attachment hardware

Fuel

- Fuel supply tubes
- Fuel bypass tubes
- Related fittings, clamps, and attachment hardware

Controls

- Throttle push-pull control
- Fuel shutoff push-pull control
- Related brackets, cranks, and attachment hardware

Fire Extinguishers

- Fire extingisher agent containers
- Actuation wire harness
- Distributor tubes
- Related clamps and attachment hardware

Fire Detector

- Heat sensor
- Related clamps, brackets, and attachment hardware

Drains

- Drains mast
- Tubes, brackets, fittings, and attachment hardware

The bottom-mounted accessories concept also has significant impact in the area of the engine/airframe interface by adding the numerous fluid carrying tubes and ducts and wire harnesses related to the hydraulic pump, pneumatic starter, power generator and its constant speed drive, and the fire detection and suppressuion systems. These additional interface connections represent an increase in the work load related to changing an engine, and a decrease in subsystem reliability.

Hydraulics

Pump suction line
Pump pressure line
Case drain line
Filter pressure sensing wire harness
EDV control wire harness

Starter

Air supply duct
Control valve wire harness

Electric Power

Power feeders
Generator control and condition indicating wire harness
Drive decoupler control and condition indicating wire harness

Fire Protection

Heat detector wire harness
Fire extingishing container control wire harness

Workstands - The impact of the bottom-mounted accessories on maintenance workstand concepts was studied. The results indicate that heavy, on-wing maintenance can be accomplished with the aid of a single level platform of the type shown in Figures 100 and 101. This arrangement places the workstand floor in approximately the same relationship to the engine as is the ground to the engines on the current low-wing wide-body transports. The reduced complexity of this stand as compared with the two-level workstand envisioned for the baseline QCSEE is apparent. The concept retains the hinged platform sections and the absence of bracing at the rear, to accomodate spotting an engine transport dolly below the pylon and raising and lowering the engine as in the baseline concept.

Figures 102 and 103 indicate that the portable workstands in current use by airlines will conveniently accommodate the needs related to performing ramp servicing and light maintenance.

9.2 REDUCED ACOUSTIC TREATMENT

A study was made to identify the impact of a reduced acoustic treatment concept on the baseline airplane. It was determined that if the 95 EPNdB sideline noise requirement were relaxed to 100 EPNdB, the following changes could be made to the baseline design:

- The acoustic treatment in the inlet and exhaust passages was eliminated.

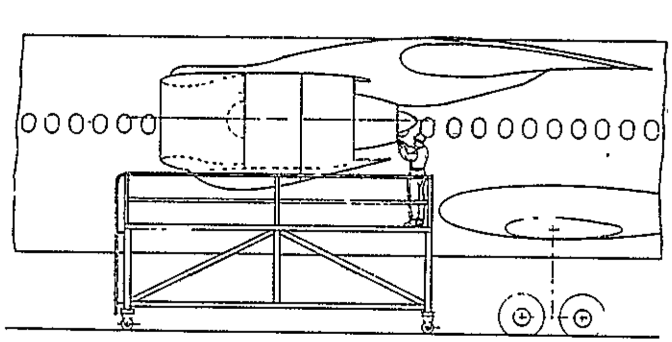

Figure 100. Bottom Accessory, Maintenance Stand, Heavy, Side.

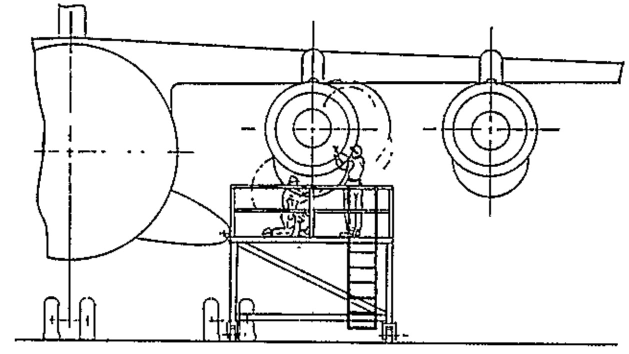

Figure 101. Bottom Accessory, Maintenance Stand, Heavy, Front.

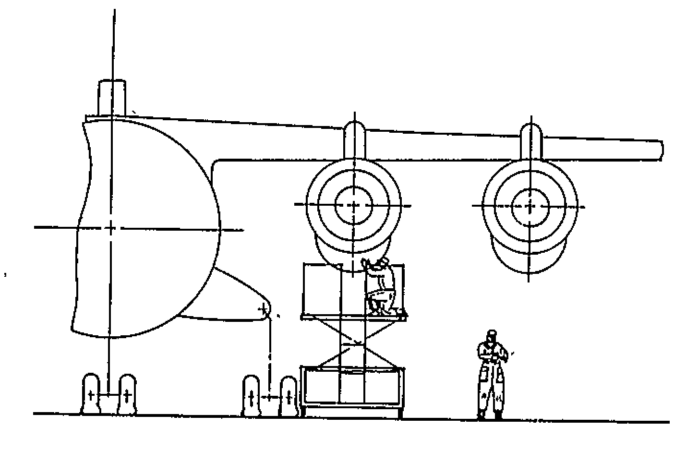

Figure 102. Bottom Accessory, Maintenance Stand, Light.

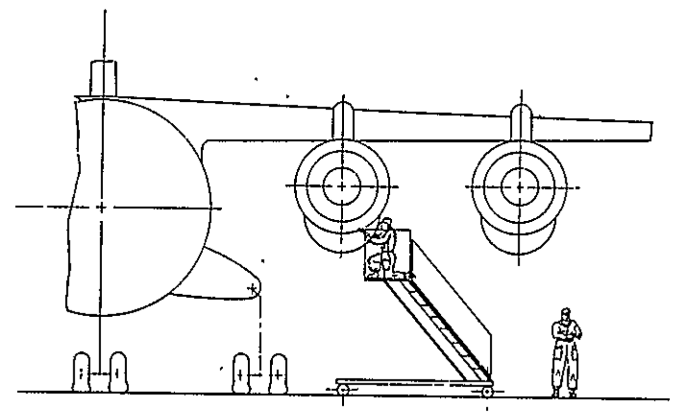

Figure 103. Bottom Accessory, Maintenance Stand, Light (Alternate).

- The inlet duct was shortened 30 cm (11.8 in.) and the throat area increased to reduce the Mach number to 0.62 from 0.79. The inlet cowl maximum diameter was increased to 211 cm (83 in.) to maintain the same 21% thick lips on the inlet as on the baseline QCSEE inlet.

- The fan exhaust duct was shortened 33 cm (13 in.). The turbine exhaust duct was shortened 17.8 cm (7 in.) and the nozzle plug replaced by a turbine disc exit fairing. The concept is shown by Figure 104. These changes reduced the weight of the pod by 88 kg (195 lb).

- The changes to the engine pod allowed the pylon to be shortened 33 cm (13 in.) to maintain the desired relationship of the exhaust stream with the EBF lifting surfaces. This reduced the weight of the pylon (box beam and fairings) by 20 kg (45 lb).

The net effect was:

Airplane drag was reduced 372 cm^2 (0.4 ft^2)
Propulsion system weight was reduced 441 kg (972 lb)
Propulsion system cost was reduced $140,000

The airplane was then resized to maintain the capability to takeoff and land on 914.4 m (3000 ft) long runways. This process indicated these to be the effects of reducing the acoustic treatment.

The wing area decreases 2.3%
The operators empty weight decreases 1.6% (771 kg) (1700 lb)
The design TOGW decreases 1.3% (907 kg) (200 lb)
The mission fuel burned decreases 1.7%
The direct operating cost decreases 1.2%

The airplane performance for the reduced acoustic treatment was calculated using engine performance adjusted only for differences in installation losses resulting from the reduced treatment. It is possible that the lower inlet Mach number would permit higher engine thrust levels during climb and cruise. This would result in further improvements in the aircraft, with potential noise reduction at the takeoff measuring point.

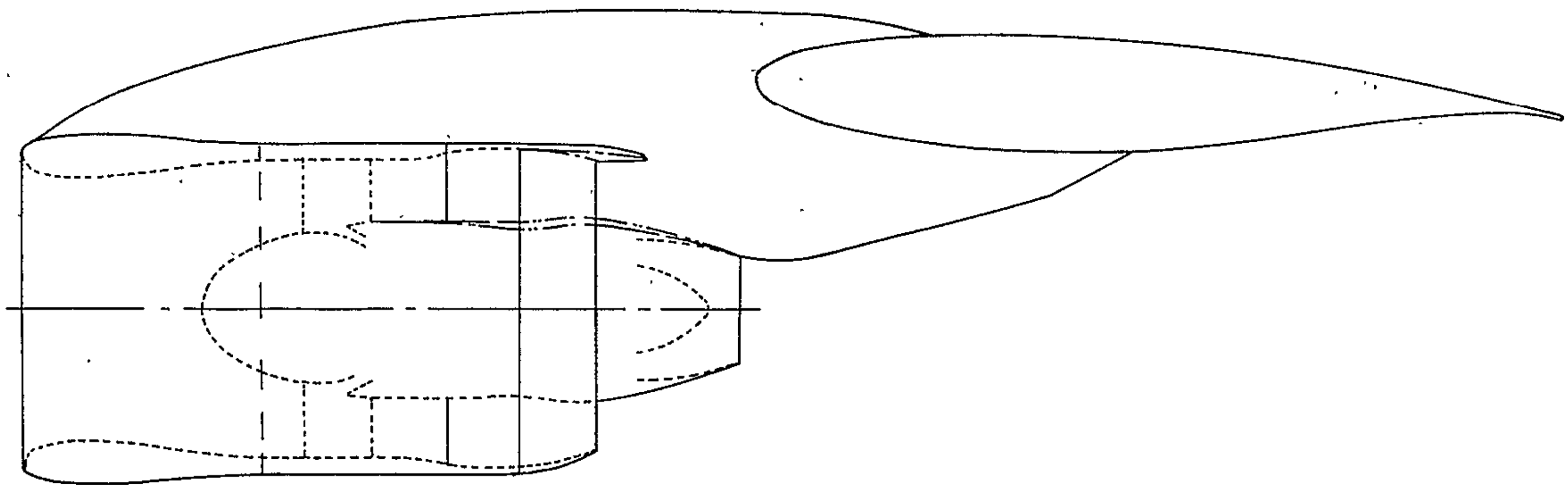

Figure 104. Propulsion System, Reduced Suppression.

10.0 CONCLUSIONS

The system integration studies have strengthened the conclusions reached in earlier conceptual studies that a production derivative of the UTW QCSEE promises to be an economical propulsion system for short-haul, STOL aircraft using powered lift. The increased concern about energy conservation, compounded by escalating fuel prices, has decreased the emphasis on increasing cruise speeds. Studies are being addressed to advanced technology CTOL aircraft that achieve significant reductions in fuel usage at current and reduced cruise speeds.

Specific conclusions applicable to a short-haul airliner fitted with engines based on utilization of the UTW/QCSEE technology, that can be drawn from this study are these:

- Top-mounted accessories are technically sound and look attractive for high-wing aircraft. This arrangement simplifies and streamlines the quick engine change unit and the engine/airplane interface. There is no significant propulsion system aerodynamic drag difference between top-mounted and bottom-mounted accessories configurations.

- Electronic controls offer the means by which the complex variable geometry features can be accommodated without increased pilot work load.

- The thrust available from QCSEE for takeoff, climb, and cruise is well matched to the requirements of the baseline short-haul powered-lift airliner and consistent with the projected commercial aircraft requirements of the mid 1980's.

- The inlet geometry required for satisfactory inlet operation dictates an internal lip radius of $R_{HL}/R_i = 1.21$ and a lip shape in form of 2:1 ellipse. External cowl with a DAC-1 shape having $R_{HL}/R_{max} = 0.90$ and cowl length ratio of $X/D_{max} = 0.22$ is required.

- This external cowl shape will provide good internal performance at high angle-of-attack conditions and also exhibit low external drag at cruise and at engine-out takeoff-climb conditions. This can be attained with a cowl having a maximum diameter no larger than that of the QCSEE integral fan frame.

- For efficient powered-lift and cruise performance, the nacelle position relative to the wing should be with the engine thrust axis parallel to the wing reference plane, the fan nozzle 20 percent of the wing chord ahead of the wing leading edge, and the fan exhaust clearing the drooped leading edge and entering the flap slots in the high-lift configuration.

- Ground handling qualities of baseline short-haul airplane would be acceptable with idle thrust level of 3469 N (780 lb)/engine or approximately 4.5% of takeoff thrust.

- Stopping the baseline airplane on a 914.4 m (3000 ft) long runway without brakes requires a thrust reversing effectiveness $\geq$ 0.35.

- The proposed QCSEE air bleed capacity meets the baseline airplane demand, however special valves and operating procedures are required based on predicted QCSEE Experimental Engine capability. A flight system may require additional capability to assure economical commerical operation at all conditions.

- Structural acoustic panels and the use of composite materials resulted in significantly reduced nacelle component weights - QCSEE-type inlet 25 to 20% lighter than equivalent metal "state-of-the-art" inlet.

- The thin-wall nacelle and relatively small maximum nacelle diameter for a 180.3 cm (71 in) diameter fan result from the high Mach inlet and the integration of nacelle and engine components.

- A radiation shield, plus under cowl fan cooling air, is required in the after area of the core cowl to keep the cowl door temperatures within the operating limits of polymeric composities.

- The variable fan nozzle provides an efficient and unique method of controlling forward thrust and providing sufficient airflow for the reverse thrust mode.

- Frequent exposure of nacelle components to the possibility of incidental damage resulted in the selection of Kevlar 49 because of its greater toughness. Glass and/or graphite is used in areas requiring higher strength or stiffness.

- The proposed compressor discharge bleed limit results in a deficiency of sufficiently hot bleed air for ice protection at some altitude holding conditions. Unless the limit can be relaxed during these conditions, it will be necessary to adopt special engine operating procedures and to develop a more complex air bleed control system.

- Interstage bleed pressure during some cruise conditions is lower than is typically supplied to the air conditioning systems of contemporary aircraft. This will require the use of compressor discharge bleed during these conditions, with attendant SFC penalty, unless some other way can be found to resolve the problem.

- Separate compressor bleed ports for interstage and compressor discharge air (as opposed to coaxial ports) are needed for compatibility with practical bleed system concepts.

- A separate compressor discharge bleed port, dedicated to engine inlet ice protection only, is needed.

Special areas of interest requiring further development and/or demonstration:

- Identify characteristic of thrust transient during change from approach to reverse thrust.

- Wind tunnel model testing to identify airplane lift and drag characteristics in the reverse thrust mode and in presence of a ground plane.

- QCSEE propulsion system with EBF wing section in the proximity typical of an EBF powered-lift airplane to obtain measurements of the acutal heating effects of the turbine exhaust stream.

- Further study of the standby generator concept to determine whether a standby IDG should be running at no-load condition or be at rest, decoupled from the accessory gear box with, for instance, a fill and drain fluid coupling.

- Further study to develop concepts that will assure compatibility between the environmental control and ice protection system air bleed demands, and the proposed engine bleed capacity.

- Configuration changes to provide independent interstage and compressor discharge bleed ports (for environmental control and airframe ice protection) and a separate compressor discharge bleed port for engine inlet ice protection.

- Further study of geared engine heat rejection rates and fuel tank heat dissipation capacity to ensure adequate oil cooling at all flight conditions.

APPENDIX A

AMERICAN AIRLINES

OPERATIONAL SCENARIO AND GENERAL REQUIREMENTS FOR MULTI-ENGINE STOL PASSENGER TRANSPORT AIRPLANE FOR INTRODUCTION IN 1980-1982

This document was used as a guideline in the baseline aircraft sizing study.

February 13, 1974

(Revised 4/10/75)

1.0 GENERAL

The following is intended as a guide to airplane/engine manufacturers submitting design studies for a large multi-engine, short-medium range high reliability transport. These requirements will be used as a basis for evaluating the qualities of such designs and each submittal should address itself to them. In the event a particular design does not meet one or more of these requirements, cogent reasons and justification for non-compliance should be included in the submittal.

The basic configuration of the airplane shall be tailored for the carriage of passengers. The economy and utility of the airplane should not significantly be compromised by considerations for carriage of cargo.

With due regard to the restraints imposed by the detailed recommendations made below, the airplane is expected to have a capacity of 150 to 170 passengers in a 100% Coach Class at 91.4 cm (36 in.) pitch. The airplane will be used to expand commercial airline service into smaller airports where such service is not currently available and it must therefore be as quiet as is technically reasonable.

2.0 MISSION REQUIREMENTS

The airplane should be capable of operating regularly non-stop design missions, similar to the following, with a full load of passengers, plus their baggage and 1587.6 kg (3500 lbs) of cargo in belly containers.

(a) New York - Detroit

(b) New York - Chicago

The requirements given below give quantitative technical definitions consistent with the above policy.

2.1 The airplane should be capable of carrying a payload of (N x 200) + 1587.6 kg (3500 lbs) - where N is the number of passenger seats in a full coach class interior - over the ranges and under the conditions tabulated on the following page.

2.2 Fuel Reserve (Standard)

The total fuel remaining over intended destination shall be assumed to be the sum of the following fuel quantities:

1. Fifteen minutes holding at 1.52 km (5000 ft), at best holding speed, calculated at the predicted destination landing weight.

2. Commencing at predicted destination landing weight, climb from Sea Level at the preferred climbing speed and cruise at not less than 4.57 km (15,000 ft) at preferred Mach number for a total

	A NYC-DTW	B NYC-CHI
Equivalent Still Air Distance (NM)	1250 km (675 NM)	1250 kg (675 NM)
Assumed T/O Runway Length Available	914.4 m (3000 ft)	Greater than 914.4 m (3000 ft)
Takeoff Elevation	0	0
Takeoff Ambient Temperature	305.6° K (90° F)	305.6° K (90° F)
Enroute Ambient Temperature	ISA + 10° C	ISA + 10° C
Cruise Mach Number	0.65 Min	0.65 Min
Cruise Altitude	Not Less Than 7.62 km (25,000 ft)	Not Greater Than 9.45 km (31,000 ft)
Fuel Reserves	See Para 2.2	See Para 2.2

*Note: Cruise Mach number should be optimized on the basis of maximum seat miles per 378.5 m^3 (100,000) gallons of fuel consumed.

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diversion distance (climb and cruise) of 277.8 km (150 NMi). No descent allowance.

3. Forty-five minutes holding at 1.52 km (5000 ft), at best holding speed, commencing at the weight corresponding to the end of diversion cruise.

2.3 Fuel Capacity

Sufficient fuel tankage at 779 kg/m^3 (6.5 ppg) shall be provided in the wing structure to permit loading the airplane to maximum takeoff weight at a zero fuel weight equivalent to a weight no greater than the EOW plus 50% of the space limited payload.

2.4 Fuel Tank Location

No fuel tankage shall be permitted outside the wing box in the under-floor area of low wing designs or center wing areas of the aircraft on high wing designs.

3.0 EXTERNAL NOISE CHARACTERISTICS

3.1 Noise

The airplane shall be capable of complying with the appropriate noise regulations as called for by Federal statutes at the time of application for an original type certificate or when revenue service is initiated, whichever are the more stringent.

3.2 The takeoff climb flight paths used to show compliance with the above requirements shall not require a pitch angle, that is, the angle between the passenger cabin floor and the horizontal, greater than that necessary to achieve aircraft optimum climb performance (goal 20 degrees maximum).

3.3 The overall sound pressure levels and speed interference levels in the control cabin and passenger cabin, at altitudes and speeds specified below, and with representative engine cruise thrust on both engines typical airline interior configuration, and with standard materials, should not exceed the following values:

Pressure Altitude, km, (ft)	9.4 (31,000)		6.1 (20,000)	
Speed	0.74	Mach	0.65	
db	OASPL	SIL	OASPL	SIL
Pilot's Seat, Head Level	85	67	90	74
First Class Window Seats, Head Level	82	62	81	69
Tourist Class Window Seats, Head Level	90	65	89	69

4.0 APPROACH AND LANDING PERFORMANCE

4.1 The following requirements shall be met in the landing configuration with maximum landing flaps and at maximum landing weight with and without propulsive lift:

1. Stall speed ($V_{S_{1g}}$) shall not exceed (TBD) Kts. EAS.

5.0 POWER PLANTS

Engines of the high bypass fan type in the 88,960 N (20,000 lb) thrust size are recommended. Maintenance features directed at minimizing the engine servicing time are necessary. Special attention shall be given to the starting system, reverser system and instrumentation to maximize reliability. The cost for a delay shall be considered to be 300 dollars for up to 10 minutes, 700 dollars for 20 minutes and 2,000 dollars for 30 minutes or over. The requirements set forth in NASA CR-12134, as amended for STOL peculiar requirements, shall be used as a general guide in system design and program planning. Final engine design and selection must give due consideration to the following:

a. long term specific fuel consumption performance retention capability
b. noise characteristics
c. pollution emission (visible and invisible) as specified by EPA and/or FAA requirements

d. durability, reliability
e. low maintenance costs
f. applicability to other aircraft as a total propulsion system
g. long term warranties

5.1 The engines should be able to provide rated takeoff thrust automatically at full throttle, at any ambient temperature and altitude, approved for takeoffs. The aircraft shall also be certificated for reduced thrust takeoffs of not less than 10% of rated thrust with a goal of 20% rated thrust according to the available performance margins. Certification and appropriate power setting procedures for reduced power takeoffs shall cover ambient temperatures up to at least ISA + 5° C (+41° F), and thrust levels from maximum takeoff thrust down to maximum climb thrust for all takeoff altitudes. It shall be possible to obtain the reduced takeoff thrust with the throttle at the takeoff thrust within two seconds from the appropriate ground and flight idle positions is desired.

5.2 It shall be a design goal to actuate the thrust reversing system from the forward full thrust position to the reverse full thrust position, and vice versa, within one second at all speeds on the ground, up to the maximum touch down speed for a no-flap landing, and to obtain maximum available thrust relative to the thrust lever position within a total elapsed time of five seconds. In addition, actuation out of reverse in flight shall be possible up to at least 102.9 m/r (200 knots) indicated airspeed at all altitudes. Due consideration must be given to the impact of reversible pitch fans and the consequences of inadvertent reversal in flight and suitable protection measures must be provided. Consideration should also be given to the use of reverse thrust in flight for achieving aircraft deceleration and high rates of descent, particularly in the approach mode.

5.3 It is particularly important that the engines be protected against ingestion of foreign objects during all phases of flight operations. Resorting to operating procedures to provide this protection, particularly during thrust reverser operation, shall be minimized.

6.0 OVERALL DIMENSIONS

The overall dimensions of the airplane should be in the order of:

- a wing span of 32.9 m (108 ft)
- an overall length of 48.8 m (160 ft)
- an overall height of 17.5 m (57-1/2 ft)

7.0 GROSS WEIGHT/AIRPORT COMPATIBILITY

At a weight 5% in excess of the ramp weight required to perform mission 'B', with the center of gravity in the most adverse location, and with all other relevant items in the most adverse configuration (e.g., tire pressure, etc.), the airplane shall be capable of operating on all taxiways, runways, (including pier structures), aprons, etc. at LGA airport, without restrictions in maximum allowable taxi or takeoff weight due solely to pavement loading limitations.

7.1 All external servicing connections shall be physically interchangeable and compatibly located on the airplane with respect to fixed and portable airport facilities which will service B727 and DC-9 aircraft.

7.2 All passenger, cargo, and galley service doors shall be positioned horizontally and vertically such that they are compatible with fixed B727 and DC-9 terminal facilities.

8.0 INTERIOR ARRANGEMENT

8.1 Provisions shall be made to vary the percentage mix between First Class and Coach passengers and to vary the seat pitch. First Class/Coach ratios between 0 and 30% should be considered, with seat pitches varying between 81.3 cm (32 in.) and 101.6 cm (40 in.).

8.2 A single deck passenger cabin is preferred.

8.3 Passenger compartment aisle(s) should not be less than 50.8 cm (20 in.) wide.

8.4 Seat units may incorporate more than 2 seats; i.e., the basic interior may include triple seats or adjacent double and/or single seats. If triple seats are used, minimum overall seat width should be not less than 165.1 cm (65 in.) [i.e., 747 triple seat]. Consideration should also be given to increased seat pitch, and/or improved seat design, to provide equivalent passenger comfort and adequate in/out access for window seat passengers. Adjacent seat units, if installed, shall not be closer than (TBD) inches, between armrests, to provide for tables and stowage.

8.5 Emergency evacuation provisions should meet all requirements of the latest FAR.

8.6 An Adequate number of lavatories should be provided. The formula N-20/40 + 1, where N is the total number of passenger seats in the basic mixed class configuration provides guidance on the number required.

8.7 Galley capacity, where required, should be sufficient to provide meal service to the same standards as provided on B727 and DC-9 on a per passenger basis. The design should consider handling all food servie from carts.

8.8 Coat hanging space equivalent to 1.8N cm (0.7N inches) should be provided, where N is the total number of passenger seats in the basic mixed class condiguration.

8.9 Consideration shall be given to the incorporation of under seat overhead and special interior baggage storage racks for all baggage on the short-haul service. These provisions should be arranged to permit rapid and convenient access by enplaning and deplaning passengers using either one pair or two pairs of opposing passenger entry doors.

8.10 An airstair system shall be provided to rapidly load and unload passengers.

9.0 BAGGAGE AND CARGO HANDLING

All passenger baggage shall be considered to be of the carry-on type. A preloaded cargo handling system shall be provided to accommodate 14.15 m^3 (500 ft^3) of cargo. Consideration should be given to having the system capable of accepting LD-3 type containers. If the containers specified for the airplane are not actually LD-3 type containers, they should be interchangeable with the LD-3 type, and must be capable of being carried by DC-10, B747 and L-1011 aircraft.

9.1 The cargo containers shall be fuselage structure constrained without the requirement for locking devices, except for a doorway roll-out-stop.

9.2 The cargo handling/container system shall be designed so that containers can be loaded and unloaded by one man at each compartment opening, external to the opening.

9.3 Cargo doors shall be outward opening, canopy type, on the right-hand side of the airplane and incorporate positive latching to prevent opening in flight.

10.0 OPERATING FEATURES

10.1 The aircraft may normally operate from a parallel parked position with ground level passenger loading, Consideration should also be given to making the aircraft compatible with existing nose-in parking, second level loading for B727 and DC-9 aircraft.

10.2 The airplane shall be capable of a 20 minute through flight, or a 30 minute turn as a maximum.

10.3 Consideration shall be given to a self-contained means for backing the aircraft from its parked position.

10.4 Ground Maneuvering

1. The aircraft shall be capable of easily executing a 180° turn within a maximum pavement width of (TBD) feet, at its maximum taxi weight, with its center of gravity in the most adverse location. The radius of the circle described by the extremity of the aircraft's planform under these conditions shall not exceed (TBD) feet. Best efforts shall be exercised to achieve significantly lower turning radii than these specified maxima. Additionally, it shall be a design goal to achieve these characteristics with a nose landing gear position located as nearly as possible below the pilots' seats to aid in visually taxiing the aircraft.

2. The aircraft shall be capable of executing the turn of paragraph 10.4.1 from a standing start on a level dry pavement, with cold tires, at thrust levels that will not cause jet blast velocities in excess of 35.1 m/s (115 fps) along a line perpendicular to the aircraft centerline passing through the rearmost point on the aircraft, or 25.9 m/s (85 fps) along a line perpendicular to the aircraft centerline, and 15.2 m (50 ft) aft of the rearmost point on the aircraft. It shall be a design goal to achieve levels no greater than 27.4 m/s (90 fps) and 18.3 m/s (60 fps) at these locations, respectively.

10.5 Special consideration shall be given to providing redundancy and fault isolation capability in the aircraft systems to an extent sufficient to ensure that mechanical delays are minimized, and to provide operational characteristics, performance capability, and airworthiness characteristics with all engines operating comparable, insofar as practical, to a four engine aircraft with all engines operating, and in the case of an engine out, to a four engine airplane with one engine out.

10.6 Any major component (i.e. engine, LRU, etc.) should be replaceable within a four hour period.

10.7 The airplane shall be designed for a V_{mo}/M_{mo} of not less than (TBD).

10.8 The airplane shall be designed to be fully operational by a two man cockpit crew, but with separate and sufficient controls and systems monitoring devices for adequate functioning of a third crew member. At least one observer seat should also be provided and should be suitably positioned for check-training supervision.

10.9 The airplane shall be equipped with automatic spoiler extension for landing and aborted takeoffs, and with automatic safety retraction for go-around. The system should be similar to that of the B747-123.

10.10 Cockpit design shall pay particular attention to the requirements of SAE ARP 268D, Location and Actuation of Flight Deck Controls for Commercial Transport Type Aircraft; and SAE AS 580A, Pilot Visibility from the Flight Deck - Design Objectives for Commercial Transport Aircraft.

The airplane shall be designed to display the highest state-of-the-art in flying qualities and be certified for Category II operation prior to initial delivery. In addition, the system should be designed to anticipate Category III operations by being configured to facilitate post delivery modification, or incorporation of the following features:

- Best state-of-the-art autoland autopilot system (preferably fail operative)

- Control (force) wheel steering

- Automatic directional and lateral control guidance throughout takeoff and landing roll

- Automatic pilot go-around, pilot initiated

- Provisions for dual, windshield glass type flight director heads-up display

- Automatic wheel braking system, pilot adjustable for desired deceleration

- Electronic attitude director indicator incorporating visibility enhancement presentation. Color CRT preferred for symbology.

10.11 The main landing gear shall incorporate features designed to make the airplane land soft. It should be possible to land the airplane at substantially higher sink rates than are average on current jet aircraft without incurring greater than average normal vertical accelerations (G's). A minimum normal acceleration of 0.4G at a minimum sink rate of five feet per second is suggested as a criterion.

11.0 OTHER REQUIREMENTS

11.1 The maximum Design Zero Fuel Weight shall not be less than the sum of 110% the Operating Weight Empty and the space limited payload. The space limited payload is defined as: 200M + 10 [total containerized + bulk (if any) baggage and cargo volume] where N is defined as the total number of passenger seats in the basic mixed class configuration and baggage/cargo volume is in cubic feet.

The maximum Design Landing Weight shall not be less than the sum of 110% of EOW plus space limited payload plus the fuel reserves of paragraph 2.2. Consideration should also be given to the possible requirement to through fuel at one or more stations for operational or economic reasons.

11.2 Cabin differential pressure shall not be less than (TBD) p.s.i.

11.3 For performance calculation purposes, the total cabin air bleed in cubic feet per minute will be assumed to be 20 times the number of passenger seats in the maximum coach configuration. Power extraction corresponding to an electrical load of 100 kW should be assumed.

11.4 Space and wiring provisions should be included for a pictorial map type display centrally located and visible to both pilots, and which could be operated by either of two area navigation type computers.

11.5 Space, wiring, and structural provisions should be included for a performance recorder and maintenance monitoring system.

11.6 High intensity coded anti-collision condenser discharge type external lighting shall be provided.

11.7 Consideration should be given to assuring the quietness of operation of all systems and mechanically functioning components of the airplane as perceived within the airplane.

11.8 The aircraft should be equipped with inflight speed brakes which will not induce longitudinal or lateral trim changes, or otherwise adversely affect flight characteristics, over the entire design flight envelope. Such brakes shall be usable at all airspeeds and will not result in noticeable or objectional buffeting.

11.9 Consideration should be given to incorporating an inerting system to afford fire and explosion protection to the under-floor cargo compartments, wheel wells, fuel tanks, vents, and engines. Consideration should also be given to the use of nitrogen for maintaining oxygen in a liquid state, should cryogenic systems be incorprated.

11.10 All elements of the engine control system and aircraft flight control system which either pass through or are adjacent to the cabin floor or ceiling shall be protected against the catastrophic consequences that may arise from failure or significant deformation of the floor or ceiling.

12.0 LOADABILITY

Using the passenger seating and cargo/baggage loading assumptions defined below and with operating items in normal location, having weights consistent with Appendix B, and subject to limitations on total load set by design weights, the aircraft center of gravity shall remain within certified limits for takeoff, flight and landing under the following conditions:

a. Any number of passengers from zero to maximum in the First Class compartment, plus

b. Any number of passengers from zero to maximum in Coach compartment, plus

c. 13.6 kg (30 lb) of carry-on baggage per passenger for any passenger load, plus

d. Any quantity of cargo up to 1587.6 kg (3500 lb) to be distributed about the centroids of the forward and/or aft containers as required to maintain the center of gravity within limits, plus

e. Any quantity of fuel from zero to the maximum tank capacity, except that for takeoff, a fuel quantity of less than 4536 kg (10,000 lb) need not be assumed.

The following passenger seating arrangements shall be covered, with window seats occupied first, aisle seats next, and reamining seats last.

1. Passengers in each compartment, loaded from front to rear.

2. Passengers in each compartment, loaded from rear to front.

FAR regulatory allowances shall be made for adverse passenger and crew movement in flight and for gear and flap retraction and extension, as appropriate. Passengers are assumed to weight 77.1 kg (170 lb) each. Baggage and cargo stowage density is assumed throughout to be 160.3 kg/m^2 (10 lb/ft^3). The forward and aft containerized baggage compartments are assumed to house a full complement of baggage containers.

12.1 Aircraft tip-up characteristics, both rolling and static, will be such that no special precautions in loading, unloading, or in operating procedures, or in the use of special ramp equipment will be required during normal airline operation.

12.2 An on-board weight and balance system shall be incorporated that will provide an instantaneous read-out of gross-weight and center of gravity position. Full scale accuracies of approximately 0.08% are suggested as a design goal. The system shall also provide instantaneous visual indication of landing impact loads, including a sustained (resetable) maximum impact load. Consideration should be given to utilizing the flight/maintenance recorder for this purpose.

APPENDIX B

LANDING AND TAKEOFF PROFILES

The tabulation enclosed herein was generated as a basis for the baseline airplane noise estimates. This version was produced on 14 February 1975.

TAKE OFF CONDITION

TERMINAL AREA PERFORMANCE PROGRAM

FEBRUARY 14, 1975

WEIGHT	= 155700.	THRUST/ENG (CRUISE)	= 18300.	THRUST/ENG (LIFT)	= 0.
WING AREA	= 1541.00	NUMBER OF ENGINES	= 4.0000	NUMBER OF ENGINES	= 0.0
WING LOADING	= 101.038	SCALE FACTOR	= 1.0000	SCALE FACTOR	= 0.0
THRUST-TO-WEIGHT	= 0.4701	INCLINATION	= 0.0	INCLINATION	= 0.0
FLAP ANGLE	= 20.00	TAU	= 1.0000	TAU	= 0.0
AIRPORT ALTITUDE	= 0.	AIRPORT DTEMP (F)	= 18.00		

WEIGHT FUEL	TIME DIST HEIGHT	V(KTAS) V(KEAS) MACH	V/VS R/C Q	THETA GAMMA ALPHA	FLAP GAMDOT N	CMU CL/CLM A	CL CD CLMAX	DCL DCD DCLM	FG FR WF	NE TAU	TEMP PRESS CS	BANK DIRECT SIDE	
155700.	21.70	119.30	0.0	20.000	20.000	1.0490	2.5786	0.0	75201.	4.0000	536.54	0.0	0.0
0.	2534.0	117.22	25.10	7.160	1.7858	0.7742	-0.6076	0.0	22100.	1.0000	2113.54	0.0	0.0
	35.0	0.1773	46.52	12.840	1.1873	0.4231	3.3306	0.0	0.		672.78	0.0	0.0
155700.	22.70	119.84	0.0	17.500	20.000	1.0403	2.1310	0.0	75181.	4.0000	536.45	0.0	0.0
0.	2733.9	117.70	28.12	7.993	-0.0118	0.6412	-0.6959	0.0	22180.	1.0000	2111.48	0.0	0.0
	62.0	0.1781	46.90	9.507	0.9891	1.3548	3.3233	0.0	0.		672.72	0.0	0.0
155700.	23.70	120.89	0.0	15.000	20.000	1.0245	1.8504	0.0	75198.	4.0000	536.35	0.0	0.0
0.	2935.2	118.69	26.15	7.363	-1.0780	0.5590	-0.7273	0.0	22347.	1.0000	2109.39	0.0	0.0
	89.4	0.1797	47.63	7.637	0.8723	2.2403	3.3099	0.0	0.		672.66	0.0	0.0
155700.	23.70	120.89	0.0	15.000	20.000	1.0233	1.8626	0.0	75205.	4.0000	536.35	0.0	0.0
0.	2935.2	118.69	26.15	7.363	-1.0177	0.5629	-0.7245	0.0	22361.	1.0000	2109.39	0.0	0.0
	89.4	0.1797	47.69	7.637	0.8791	2.2580	3.3090	0.0	0.		672.66	0.0	0.0
155700.	25.70	123.61	0.0	15.000	20.000	0.9814	1.9867	0.0	75324.	4.0000	536.18	0.0	0.0
0.	3345.0	121.28	22.91	6.305	-0.1279	0.6070	-0.6687	0.0	22844.	1.0000	2105.71	0.0	0.0
	137.6	0.1838	49.81	8.695	0.9793	2.3108	3.2727	0.0	0.		672.55	0.0	0.0
155700.	27.70	126.26	0.0	15.000	20.000	0.9431	1.9637	0.0	75446.	4.0000	536.01	0.0	0.0
0.	3764.2	123.79	23.96	6.455	0.1325	0.6066	-0.6382	0.0	23317.	1.0000	2102.16	0.0	0.0
	184.3	0.1878	51.91	8.545	1.0090	2.1105	3.2372	0.0	0.		672.45	0.0	0.0
155700.	28.53	127.28	0.0	15.000	20.000	0.9295	1.9457	0.0	75486.	4.0000	535.94	0.0	0.0
0.	3940.6	124.75	24.64	6.586	0.1841	0.6034	-0.6288	0.0	23490.	1.0000	2100.63	0.0	0.0
	204.4	0.1893	52.70	8.414	1.0149	2.0211	3.2245	0.0	0.		672.40	0.0	0.0
155700.	28.53	127.28	0.0	15.000	20.000	0.9297	1.9429	0.0	75484.	4.0000	535.94	0.0	0.0
0.	3940.6	124.75	24.64	6.586	0.1689	0.6025	-0.6565	-0.0270	23486.	1.0000	2100.63	0.0	0.0
	204.4	0.1893	52.69	8.414	1.0131	2.4712	3.2248	0.0	0.		672.40	0.0	0.0
155700.	30.31	129.76	0.0	15.000	20.000	0.8967	1.6815	0.0	75601.	4.0000	535.78	0.0	0.0
0.	4323.5	127.10	26.55	6.963	0.2201	0.5891	-0.6364	-0.0270	23927.	1.0000	2097.17	0.0	0.0
	249.9	0.1930	54.71	8.037	1.0188	2.2397	3.1938	0.0	0.		672.30	0.0	0.0
155700.	30.31	129.76	0.0	15.000	20.000	0.8971	1.8822	0.0	75599.	4.0000	535.78	0.0	0.0
0.	4323.5	127.10	26.55	6.963	0.2201	0.5893	-0.6366	-0.0270	23922.	1.0000	2097.17	0.0	0.0
	249.9	0.1930	54.69	8.037	1.0188	2.2421	3.1941	0.0	0.		672.30	0.0	0.0
155700.	31.31	131.25	0.0	15.000	17.500	0.8789	1.7357	0.0	75667.	4.0000	535.68	0.0	0.0
0.	4542.1	128.51	26.59	6.894	-0.2739	0.5655	-0.6568	-0.0270	24176.	1.0000	2095.15	0.0	0.0
	276.6	0.1952	55.87	8.106	0.9598	2.8050	3.0694	0.0	0.		672.24	0.0	0.0
155700.	32.31	133.08	0.0	15.000	15.000	0.8567	1.6470	0.0	75774.	4.0000	535.59	0.0	0.0
0.	4763.7	130.25	25.46	6.508	-0.4753	0.5581	-0.6620	-0.0270	24505.	1.0000	2093.18	0.0	0.0
	302.7	0.1980	57.40	8.492	0.9356	3.3780	2.9512	0.0	0.		672.18	0.0	0.0
155700.	33.31	135.25	0.0	15.000	12.500	0.8316	1.5808	0.0	75918.	4.0000	535.50	0.0	0.0
0.	4988.7	132.32	23.83	5.991	-0.5468	0.5564	-0.6583	-0.0270	24897.	1.0000	2091.30	0.0	0.0
	327.3	0.2012	59.24	9.009	0.9268	3.9125	2.8409	0.0	0.		672.13	0.0	0.0

TERMINAL AREA PERFORMANCE PROGRAM FEBRUARY 14, 1975

WEIGHT FUEL	TIME DIST HEIGHT	V(KTAS) V(KEAS) MACH	V/VS R/C Q	THETA GAMMA ALPHA	FLAP GAMDOT N	CMU CL/CLM A	CL CD CLMAX	DCL DCD DCLM	FG FR WF	NE TAU	TEMP PRESS CS	BANK DIRECT SIDE	
155700.	34.31	137.71	0.0	15.000	10.000	0.8046	1.5243	0.0	76099.	4.0000	535.42	0.0	0.0
0.	5217.9	134.68	22.03	5.439	-0.5515	0.5564	-0.6484	-0.0270	25348.	1.0000	2089.57	0.0	0.0
	350.2	0.2049	61.38	9.561	0.9260	4.3849	2.7396	0.0	0.		672.07	0.0	0.0
155700.	35.31	140.42	0.0	15.000	7.500	0.7763	1.4710	0.0	76314.	4.0000	535.35	0.0	0.0
0.	5451.6	137.29	20.24	4.900	-0.5264	0.5555	-0.6336	-0.0270	25850.	1.0000	2087.97	0.0	0.0
	371.4	0.2090	63.79	10.100	0.9287	4.7817	2.6481	0.0	0.		672.03	0.0	0.0
155700.	36.31	143.35	0.0	15.000	5.000	0.7477	1.4178	0.0	76560.	4.0000	535.28	0.0	0.0
0.	5690.3	140.12	18.52	4.389	-0.4928	0.5522	-0.6150	-0.0270	26395.	1.0000	2086.51	0.0	0.0
	390.8	0.2133	66.45	10.611	0.9324	5.0996	2.5674	0.0	0.		671.98	0.0	0.0
155700.	37.31	146.45	0.0	15.000	2.500	0.7193	1.3654	0.0	76833.	4.0000	535.21	0.0	0.0
0.	5934.2	143.11	16.88	3.916	-0.4547	0.5464	-0.5936	-0.0270	26973.	1.0000	2085.17	0.0	0.0
	408.4	0.2179	69.32	11.084	0.9367	5.3334	2.4987	0.0	0.		671.94	0.0	0.0
155700.	38.31	149.65	0.0	15.000	0.0	0.6917	1.3138	0.0	77130.	4.0000	535.16	0.0	0.0
0.	6183.5	146.20	15.33	3.479	-0.4183	0.5360	-0.5698	-0.0270	27575.	1.0000	2083.95	0.0	0.0
	424.5	0.2227	72.36	11.521	0.9409	5.4797	2.4418	0.0	0.		671.91	0.0	0.0
155700.	38.31	149.65	0.0	15.000	0.0	0.6916	1.3133	0.0	77131.	4.0000	535.16	0.0	0.0
0.	6183.5	146.20	15.33	3.479	-0.4200	0.5378	-0.5697	-0.0270	27577.	1.0000	2083.95	0.0	0.0
	424.5	0.2227	72.37	11.521	0.9406	5.4782	2.4418	0.0	0.		671.91	0.0	0.0
155700.	40.31	154.08	0.0	20.000	0.0	0.6565	1.3800	0.0	77482.	4.0000	535.01	0.0	0.0
0.	6695.1	150.44	28.11	6.205	1.4413	0.6331	-0.5006	-0.0270	28407.	1.0000	2080.83	0.0	0.0
	465.5	0.2294	76.59	13.795	1.1977	2.9455	2.4191	0.0	0.		671.82	0.0	0.0
155700.	42.31	157.23	0.0	20.000	0.0	0.6330	1.3579	0.0	77650.	4.0000	534.77	0.0	0.0
0.	7216.4	153.36	37.31	8.082	0.5532	0.5649	-0.5078	-0.0270	28975.	1.0000	2075.82	0.0	0.0
	532.3	0.2341	79.61	11.918	1.0695	2.3960	2.4038	0.0	0.		671.67	0.0	0.0
155700.	44.31	159.84	0.0	20.000	0.0	0.6142	1.2779	0.0	77738.	4.0000	534.49	0.0	0.0
0.	7745.8	155.73	41.30	8.805	0.2903	0.5302	-0.5002	-0.0270	29434.	1.0000	2069.89	0.0	0.0
	611.2	0.2380	82.13	11.195	1.0307	2.0725	2.3914	0.0	0.		671.49	0.0	0.0
155700.	46.31	162.15	0.0	20.000	0.0	0.5987	1.2163	0.0	77775.	4.0000	534.19	0.0	0.0
0.	8282.5	157.77	44.01	9.253	0.1882	0.5108	-0.4909	-0.0270	29816.	1.0000	2063.48	0.0	0.0
	696.6	0.2415	84.31	10.747	1.0149	1.8376	2.3811	0.0	0.		671.30	0.0	0.0
155700.	48.31	164.21	0.0	20.000	0.0	0.5853	1.1801	0.0	77775.	4.0000	533.86	0.0	0.0
0.	8826.0	159.57	46.09	9.573	0.1403	0.4975	-0.4818	-0.0270	30144.	1.0000	2056.74	0.0	0.0
	786.8	0.2447	86.23	10.427	1.0071	1.6446	2.3722	0.0	0.		671.10	0.0	0.0
155700.	50.31	166.06	0.0	20.000	0.0	0.5736	1.1519	0.0	77744.	4.0000	533.53	0.0	0.0
0.	9375.5	161.14	47.92	9.824	0.1127	0.4872	-0.4735	-0.0270	30426.	1.0000	2049.73	0.0	0.0
	880.7	0.2475	87.93	10.176	1.0025	1.4775	2.3645	0.0	0.		670.89	0.0	0.0
155700.	52.31	167.72	0.0	20.000	0.0	0.5637	1.1296	0.0	77685.	4.0000	533.13	0.0	0.0
0.	9930.4	162.51	49.30	10.029	0.0935	0.4787	-0.4660	-0.0270	30667.	1.0000	2042.60	0.0	0.0
	977.9	0.2501	89.44	9.971	0.9991	1.3303	2.3577	0.0	0.		670.67	0.0	0.0

ORIGINAL PAGE IS OF POOR QUALITY

WEIGHT	TIME	V(KTAS)	V/VS	THETA	FLAP	CMU	CL	DCL	FG	NE	TEMP	BANK	
FUEL	DIST	V(KEAS)	R/C	GAMMA	GAMDOT	CL/CLM	CD	DCD	FR	TAU	PRESS	DIRECT	
	HEIGHT	MACH	Q	ALPHA	N	A	CLMAX	DCLM	WF		CS	SIDE	
155700.	52.75	168.07	0.0	20.000	0.0	0.5617	1.1243	0.0	77667.	4.0000	533.10	0.0	0.0
0.	10054.9	162.80	49.60	10.070	0.0905	0.4771	-0.4646	-0.0270	30713.	1.0000	2040.86	0.0	0.0
	999.9	0.2506	89.73	9.930	0.9985	1.3025	2.3564	0.0	0.		670.62	0.0	0.0
155700.	52.75	168.07	0.0	20.000	0.0	0.5374	1.1227	0.0	74309.	4.0000	533.10	0.0	0.0
0.	10054.9	162.80	49.60	10.070	0.0806	0.4798	-0.4412	-0.0270	30712.	1.0000	2040.86	0.0	0.0
	999.9	0.2506	89.73	9.930	0.9970	0.6348	2.3400	0.0	0.		670.62	0.0	0.0
155700.	53.75	168.12	0.0	18.000	0.0	0.4562	0.9444	0.0	63073.	4.0000	532.93	0.0	0.0
0.	10334.4	162.73	47.11	9.557	-0.9561	0.4133	-0.3798	-0.0270	28295.	0.8500	2037.22	0.0	0.0
	1048.6	0.2507	89.72	8.443	0.8346	-0.3897	2.2848	0.0	0.		670.51	0.0	0.0
155700.	54.75	167.64	0.0	16.000	0.0	0.3776	0.8637	0.0	51837.	4.0000	532.77	0.0	0.0
0.	10614.4	162.16	40.93	8.318	-1.4836	0.3874	-0.3090	-0.0270	25561.	0.7000	2033.93	0.0	0.0
	1092.8	0.2501	89.09	7.682	0.7615	-1.2018	2.2297	0.0	0.		670.41	0.0	0.0
155700.	55.09	167.38	0.0	15.333	0.0	0.3517	0.8508	0.0	48097.	4.0000	532.72	0.0	0.0
0.	10707.7	161.86	38.37	7.806	-1.5886	0.3848	-0.2842	-0.0270	24575.	0.6500	2032.98	0.0	0.0
	1106.0	0.2497	88.74	7.528	0.7472	-1.4259	2.2109	0.0	0.		670.38	0.0	0.0
155700.	55.09	167.36	0.0	15.333	0.0	0.3518	0.8526	0.0	48096.	4.0000	532.73	0.0	0.0
0.	10707.7	161.89	38.37	7.806	-1.5797	0.3856	-0.2842	-0.0270	24571.	0.6500	2032.99	0.0	0.0
	1106.0	0.2497	88.72	7.528	0.7486	-1.4184	2.2110	0.0	0.		670.38	0.0	0.0
155700.	60.09	166.93	0.0	14.093	0.0	0.3534	1.1191	0.0	47883.	4.0000	532.30	0.0	0.0
0.	12110.5	161.15	19.27	3.923	-0.1540	0.5059	-0.2644	-0.0270	24410.	0.6500	2024.07	0.0	0.0
	1226.7	0.2491	87.93	10.171	0.9739	0.0764	2.2121	0.0	0.		670.11	0.0	0.0
155700.	65.09	167.09	0.0	14.284	0.0	0.3527	1.1630	0.0	47743.	4.0000	531.95	0.0	0.0
0.	13516.7	161.08	19.22	3.907	0.0872	0.5258	-0.2593	-0.0270	24363.	0.6500	2016.99	0.0	0.0
	1322.8	0.2494	87.84	10.377	1.0111	0.0695	2.2117	0.0	0.		669.89	0.0	0.0
155700.	70.09	167.28	0.0	14.136	0.0	0.3520	1.1380	0.0	47606.	4.0000	531.61	0.0	0.0
0.	14924.4	161.03	16.95	3.850	-0.0595	0.5147	-0.2611	-0.0270	24318.	0.6500	2009.99	0.0	0.0
	1418.2	0.2498	87.77	10.289	0.9565	0.0655	2.2111	0.0	0.		669.68	0.0	0.0
155700.	75.09	167.48	0.0	14.327	0.0	0.3510	1.1658	0.0	47473.	4.0000	531.27	0.0	0.0
0.	16333.8	160.99	18.68	3.789	0.0969	0.5274	-0.2574	-0.0270	24283.	0.6500	2003.05	0.0	0.0
	1512.9	0.2502	87.76	10.538	1.0126	0.0173	2.2104	0.0	0.		669.47	0.0	0.0
155700.	80.09	167.67	0.0	13.961	0.0	0.3502	1.1160	0.0	47339.	4.0000	530.94	0.0	0.0
0.	17744.7	160.96	18.73	3.795	-0.1879	0.5050	-0.2616	-0.0270	24241.	0.6500	1996.19	0.0	0.0
	1606.8	0.2505	87.71	10.166	0.9688	0.0743	2.2099	0.0	0.		669.26	0.0	0.0
155700.	35.09	167.86	0.0	14.155	0.0	0.3494	1.1593	0.0	47206.	4.0000	530.61	0.0	0.0
0.	19157.4	160.91	18.17	3.677	0.0530	0.5248	-0.2565	-0.0270	24202.	0.6500	1989.38	0.0	0.0
	1700.4	0.2509	87.68	10.477	1.0060	0.0631	2.2092	0.0	0.		669.05	0.0	0.0
155700.	90.09	169.07	0.0	14.086	0.0	0.3486	1.1535	0.0	47074.	4.0000	530.28	0.0	0.0
0.	20571.9	160.89	18.51	3.742	0.0170	0.5223	-0.2563	-0.0270	24162.	0.6500	1992.67	0.0	0.0
	1792.7	0.2513	87.63	10.344	1.0005	0.0657	2.2087	0.0	0.		668.84	0.0	0.0

WEIGHT FUEL	TIME DIST HEIGHT	V(KTAS) V(KEAS) MACH	V/VS R/C Q	THETA GAMMA ALPHA	FLAP GAMDOT N	CMU CL/CLM A	CL CD CLMAX	DCL DCD DCLM	FG FR WF	NE TAU	TEMP PRESS CS	BANK DIRECT SIDE	
155700.	95.09	168.27	0.0	13.956	0.0	0.3477	1.1411	0.0	46944.	4.0000	529.95	0.0	0.0
0.	21988.0	160.86	18.37	3.708	-0.0550	0.5168	-0.2567	-0.0270	24124.	0.6500	1975.99	0.0	0.0
	1884.9	0.2517	87.61	10.289	0.9894	0.0697	2.2080	0.0	0.		668.63	0.0	0.0
155700.	100.09	168.48	0.0	14.095	0.0	0.3469	1.1687	0.0	46814.	4.0000	529.62	0.0	0.0
0.	23406.0	160.84	18.25	3.679	0.0973	0.5295	-0.2530	-0.0270	24087.	0.6500	1969.38	0.0	0.0
	1976.4	0.2521	87.58	10.415	1.0131	0.0695	2.2074	0.0	0.		668.42	0.0	0.0
155700.	105.09	168.71	0.0	13.811	0.0	0.3459	1.1177	0.0	46688.	4.0000	529.30	0.0	0.0
0.	24825.9	160.85	17.94	3.613	-0.1866	0.5065	-0.2571	-0.0270	24055.	0.6500	1962.86	0.0	0.0
	2066.9	0.2525	87.60	10.198	0.9690	0.0741	2.2067	0.0	0.		668.22	0.0	0.0
155700.	110.09	168.90	0.0	14.074	0.0	0.3451	1.1777	0.0	46560.	4.0000	528.98	0.0	0.0
0.	26247.5	160.80	17.93	3.605	0.1442	0.5338	-0.2504	-0.0270	24014.	0.6500	1956.44	0.0	0.0
	2156.2	0.2528	87.55	10.469	1.0205	0.0694	2.2061	0.0	0.		668.02	0.0	0.0
155700.	115.09	169.14	0.0	13.818	0.0	0.3441	1.1304	0.0	46435.	4.0000	528.66	0.0	0.0
0.	27671.0	160.82	17.84	3.584	-0.1181	0.5126	-0.2542	-0.0270	23984.	0.6500	1950.01	0.0	0.0
	2245.9	0.2533	87.56	10.234	0.9797	0.0698	2.2054	0.0	0.		667.82	0.0	0.0
155700.	120.09	169.36	0.0	13.978	0.0	0.3433	1.1837	0.0	46310.	4.0000	528.34	0.0	0.0
0.	29096.6	160.82	17.53	3.515	0.1764	0.5369	-0.2479	-0.0270	23948.	0.6500	1943.68	0.0	0.0
	2334.5	0.2537	87.55	10.463	1.0257	0.1024	2.2047	0.0	0.		667.62	0.0	0.0
155700.	125.09	169.55	0.0	13.739	0.0	0.3424	1.1303	0.0	46186.	4.0000	528.03	0.0	0.0
0.	30524.0	160.79	17.41	3.489	-0.1218	0.5128	-0.2525	-0.0270	23911.	0.6500	1937.47	0.0	0.0
	2421.6	0.2540	87.52	10.251	0.9791	0.0799	2.2041	0.0	0.		667.42	0.0	0.0
155700.	130.09	169.80	0.0	13.877	0.0	0.3413	1.1629	0.0	46066.	4.0000	527.72	0.0	0.0
0.	31953.2	160.82	17.53	3.507	0.0624	0.5278	-0.2482	-0.0270	23886.	0.6500	1931.25	0.0	0.0
	2509.0	0.2545	87.58	10.370	1.0079	0.0695	2.2033	0.0	0.		667.23	0.0	0.0
155700.	135.09	170.02	0.0	13.611	0.0	0.3405	1.1134	0.0	45942.	4.0000	527.41	0.0	0.0
0.	33384.4	160.81	17.19	3.434	-0.2137	0.5055	-0.2523	-0.0270	23848.	0.6500	1925.11	0.0	0.0
	2595.7	0.2549	87.55	10.177	0.9647	0.0701	2.2027	0.0	0.		667.03	0.0	0.0
155700.	140.09	170.21	0.0	13.844	0.0	0.3397	1.1682	0.0	45822.	4.0000	527.11	0.0	0.0
0.	34817.5	160.79	17.05	3.402	0.0876	0.5305	-0.2461	-0.0270	23812.	0.6500	1919.09	0.0	0.0
	2680.8	0.2553	87.53	10.442	1.0120	0.0695	2.2021	0.0	0.		666.84	0.0	0.0
155700.	145.09	170.42	0.0	13.552	0.0	0.3389	1.1146	0.0	45702.	4.0000	526.81	0.0	0.0
0.	36252.3	160.78	16.90	3.368	-0.2088	0.5063	-0.2505	-0.0270	23778.	0.6500	1913.09	0.0	0.0
	2765.8	0.2556	87.52	10.184	0.9655	0.0733	2.2015	0.0	0.		666.65	0.0	0.0
155700.	150.09	170.61	0.0	13.762	0.0	0.3381	1.1628	0.0	45582.	4.0000	526.51	0.0	0.0
0.	37688.8	160.76	16.83	3.351	0.0548	0.5283	-0.2450	-0.0270	23741.	0.6500	1907.19	0.0	0.0
	2849.6	0.2560	87.49	10.411	1.0069	0.0696	2.2009	0.0	0.		666.46	0.0	0.0
155700.	155.09	170.82	0.0	13.547	0.0	0.3372	1.1276	0.0	45463.	4.0000	526.21	0.0	0.0
0.	39127.1	160.75	16.64	3.308	-0.1403	0.5125	-0.2477	-0.0270	23707.	0.6500	1901.31	0.0	0.0
	2933.4	0.2564	87.48	10.239	0.9763	0.0793	2.2003	0.0	0.		666.27	0.0	0.0

TERMINAL AREA PERFORMANCE PROGRAM

FEBRUARY 14, 1975

WEIGHT FUEL	TIME DIST HEIGHT	V(KTAS) V(KEAS) MACH	V/VS R/C Q	THETA GAMMA ALPHA	FLAP GAMDOT N	CMU CL/CLM A	CL CD CLMAX	DCL DCD DCLM	FG FR WF	NE TAU	TEMP PRESS CS	BANK DIRECT SIDE	
155700.	159.09	170.96	0.0	13.620	0.0	0.3367	1.1432	0.0	45368.	4.0000	525.97	0.0	0.0
0.	40278.9	160.72	16.44	3.267	-0.0572	0.5197	-0.2456	-0.0270	23674.	0.6500	1896.67	0.0	0.0
	2999.5	0.2567	87.44	10.353	0.9893	0.0698	2.1999	0.0	0.		666.12	0.0	0.0

DECELERATING APPROACH CONDITION

TERMINAL AREA PERFORMANCE PROGRAM FEBRUARY 14, 1975

WEIGHT	=	155700.	THRUST/ENG (CRUISE)	=	18300.	THRUST/ENG (LIFT)	=	0.
WING AREA	=	1541.00	NUMBER OF ENGINES	=	4.0000	NUMBER OF ENGINES	=	0.0
WING LOADING	=	101.038	SCALE FACTOR	=	1.0000	SCALE FACTOR	=	0.0
THRUST-TO-WEIGHT	=	0.4701	INCLINATION	=	0.0	INCLINATION	=	0.0
FLAP ANGLE	=	25.00	TAU	=	0.1500	TAU	=	0.0
AIRPORT ALTITUDE	=	0.	AIRPORT DTEMP (F)	=	18.00			

TERMINAL AREA PERFORMANCE PROGRAM FEBRUARY 14, 1975

WEIGHT FUEL	TIME DIST HEIGHT	V(KTAS) V(KEAS) MACH	V/VS R/C D	THETA GAMMA ALPHA	FLAP GAMDOT N	CMU CL/CLM A	CL CD CLMAX	DCL DCD DCLM	FG FR WF	NE TAU	TEMP PRESS CS	BANK DIRECT SIDE	
155700.	0.0	175.00	0.0	-4.000	25.000	0.0747	1.0195	0.0	10706.	4.0000	527.75	0.0	0.0
0.	0.0	165.76	-26.92	-5.230	-0.3573	0.4105	0.0735	0.0	11863.	0.1500	1931.89	0.0	0.0
	2500.0	0.2623	93.02	1.230	0.9386	-1.6964	2.4836	0.0	0.		667.25	0.0	0.0
155700.	4.00	171.01	0.0	-2.948	25.000	0.0779	1.1345	0.0	10689.	4.0000	528.14	0.0	1.228
0.	1163.0	162.24	-26.19	-5.207	0.0279	0.4560	0.0765	0.0	11604.	0.1500	1939.56	0.0	-5.169
	2392.2	0.2562	89.08	2.258	1.0002	-1.6349	2.4880	0.0	0.		667.49	0.0	-2.860
155700.	8.00	167.32	0.0	-2.592	25.000	0.0810	1.1768	0.0	10676.	4.0000	528.51	0.0	1.159
0.	2300.2	158.99	-25.85	-5.251	0.0055	0.4722	0.0763	0.0	11370.	0.1500	1946.99	0.0	-5.172
	2288.1	0.2506	85.57	2.659	0.9966	-1.4802	2.4923	0.0	0.		667.72	0.0	-2.350
155700.	12.00	163.93	0.0	-2.148	25.000	0.0840	1.2217	0.0	10666.	4.0000	528.87	0.0	1.368
0.	3413.5	156.09	-25.21	-5.229	0.0022	0.4894	0.0767	0.0	11154.	0.1500	1954.30	0.0	-5.162
	2186.0	0.2454	82.39	3.081	0.9962	-1.3844	2.4965	0.0	0.		667.95	0.0	-1.942
155700.	16.00	160.76	0.0	-1.740	25.000	0.0870	1.2660	0.0	10660.	4.0000	529.23	0.0	1.333
0.	4504.9	153.22	-24.72	-5.228	-0.0000	0.5063	0.0773	0.0	10955.	0.1500	1961.47	0.0	-5.163
	2086.2	0.2406	79.48	3.488	0.9958	-1.2876	2.5007	0.0	0.		668.18	0.0	-1.535
155700.	20.00	157.81	0.0	-1.331	25.000	0.0900	1.3100	0.0	10657.	4.0000	529.58	0.0	1.299
0.	5575.7	150.63	-24.27	-5.227	0.0001	0.5230	0.0782	0.0	10771.	0.1500	1968.53	0.0	-5.165
	1988.2	0.2361	76.81	3.897	0.9959	-1.2077	2.5049	0.0	0.		668.40	0.0	-1.127
155700.	24.00	155.03	0.0	-0.925	25.000	0.0930	1.3535	0.0	10655.	4.0000	529.92	0.0	1.256
0.	6627.2	148.19	-23.84	-5.227	-0.0001	0.5395	0.0795	0.0	10598.	0.1500	1975.48	0.0	-5.167
	1892.0	0.2319	74.34	4.302	0.9958	-1.1401	2.5090	0.0	0.		668.61	0.0	-0.723
155700.	28.00	152.40	0.0	-0.522	25.000	0.0960	1.3955	0.0	10655.	4.0000	530.26	0.0	1.222
0.	7660.6	143.88	-23.44	-5.228	-0.0009	0.5557	0.0810	0.0	10435.	0.1500	1982.32	0.0	-5.169
	1797.5	0.2279	72.04	4.706	0.9957	-1.0833	2.5131	0.0	0.		668.83	0.0	-0.323
155700.	32.00	149.89	0.0	-0.120	25.000	0.0990	1.4402	0.0	10657.	4.0000	530.59	0.0	1.222
0.	8676.6	143.67	-23.06	-5.230	0.0021	0.5722	0.0828	0.0	10281.	0.1500	1989.08	0.0	-5.169
	1704.4	0.2240	69.68	5.111	0.9961	-1.0335	2.5172	0.0	0.		669.04	0.0	0.083
155700.	36.00	147.49	0.0	0.267	25.000	0.1019	1.4829	0.0	10657.	4.0000	530.92	0.0	1.198
0.	9676.3	141.57	-22.68	-5.227	-0.0002	0.5881	0.0844	0.0	10138.	0.1500	1995.74	0.0	-5.170
	1613.0	0.2204	67.85	5.494	0.9958	-0.9867	2.5212	0.0	0.		669.24	0.0	0.458
155700.	40.00	145.21	0.0	0.651	25.000	0.1049	1.5259	0.0	10659.	4.0000	531.24	0.0	1.159
0.	10660.1	139.56	-22.33	-5.228	-0.0000	0.6042	0.0862	0.0	10002.	0.1500	2002.32	0.0	-5.172
	1523.0	0.2169	65.94	5.879	0.9958	-0.9446	2.5253	0.0	0.		669.44	0.0	0.844
155700.	44.00	143.01	0.0	1.036	25.000	0.1079	1.5689	0.0	10663.	4.0000	531.55	0.0	1.126
0.	11626.9	137.64	-21.99	-5.228	-0.0001	0.6203	0.0882	0.0	9871.	0.1500	2008.80	0.0	-5.174
	1434.3	0.2136	64.13	6.264	0.9958	-0.9091	2.5294	0.0	0.		669.64	0.0	1.228
155700.	48.00	140.89	0.0	1.422	25.000	0.1109	1.6122	0.0	10667.	4.0000	531.87	0.0	1.097
0.	12583.3	135.77	-21.67	-5.228	-0.0001	0.6363	0.0904	0.0	9745.	0.1500	2015.22	0.0	-5.175
	1347.0	0.2103	62.41	6.650	0.9958	-0.8791	2.5336	0.0	0.		669.84	0.0	1.615

WEIGHT	TIME	V(KTAS)	V/VS	THETA	FLAP	CMU	CL	DCL	FG	NE	TEMP	BANK	
FUEL	DIST	V(KEAS)	R/C	GAMMA	GAMDOT	CL/CLM	CD	DCD	FR	TAU	PRESS	DIRECT	
	HEIGHT	MACH	Q	ALPHA	N	A	CLMAX	DCLM	WF		CS	SIDE	
155700.	52.00	136.84	0.0	1.811	25.000	0.1140	1.6560	0.0	10672.	4.0000	532.17	0.0	1.073
0.	13523.6	133.97	-21.35	-5.229	-0.0001	0.6525	0.0929	0.0	9623.	0.1500	2021.55	0.0	-5.176
	1260.9	0.2072	60.76	7.039	0.9956	-0.8542	2.5378	0.0	0.		670.03	0.0	2.006
155700.	56.00	136.84	0.0	2.205	25.000	0.1171	1.7005	0.0	10677.	4.0000	532.48	0.0	1.055
0.	14450.3	132.20	-21.05	-5.229	-0.0001	0.6689	0.0957	0.0	9504.	0.1500	2027.81	0.0	-5.177
	1176.1	0.2042	59.17	7.434	0.9558	-0.8344	2.5420	0.0	0.		670.22	0.0	2.404
155700.	60.00	134.88	0.0	2.612	25.000	0.1203	1.7444	0.0	10683.	4.0000	532.77	0.0	1.021
0.	15363.6	130.47	-20.71	-5.221	-0.0076	0.6850	0.0985	0.0	9386.	0.1500	2033.99	0.0	-5.179
	1092.6	0.2012	57.63	7.833	0.9549	-0.8255	2.5464	0.0	0.		670.41	0.0	2.774
155700.	64.00	132.93	0.0	3.028	25.000	0.1236	1.7923	0.0	10689.	4.0000	533.07	0.0	0.842
0.	16263.9	128.75	-20.41	-5.220	-0.0038	0.7026	0.1021	0.0	9270.	0.1500	2040.08	0.0	-5.189
	1010.4	0.1982	56.12	8.249	0.9954	-0.812	2.5509	0.0	0.		670.60	0.0	3.223
155700.	64.51	132.69	0.0	3.135	25.000	0.1240	1.9043	0.0	10690.	4.0000	533.10	0.0	0.894
0.	16377.2	128.53	-20.36	-5.215	0.0236	0.7071	0.1034	0.0	9255.	0.1500	2040.85	0.0	-5.186
	1000.0	0.1979	55.73	8.350	0.9987	-0.8249	2.5515	0.0	0.		670.62	0.0	3.347
155700.	64.51	132.69	0.0	3.135	25.000	0.1240	1.8035	0.0	10690.	4.0000	533.10	0.0	0.884
0.	16377.2	128.53	-20.36	-5.215	0.0197	0.7069	0.1023	0.0	9255.	0.1500	2040.85	0.0	-5.196
	1000.0	0.1979	55.93	8.350	0.9983	-0.8268	2.5515	0.0	0.		670.62	0.0	3.347
155700.	68.51	130.24	0.0	1.239	31.000	0.1284	1.8505	0.0	10690.	4.0000	533.38	0.0	-1.061
0.	17261.6	126.30	-19.41	-5.065	-0.0565	0.6676	0.1194	0.0	9101.	0.1500	2046.69	0.0	-5.283
	921.4	0.1942	54.01	6.304	0.9894	-1.1054	2.7724	0.0	0.		670.79	0.0	0.060
155700.	72.51	127.46	0.0	-0.683	37.000	0.1337	1.9322	0.0	10686.	4.0000	533.66	0.0	-2.597
0.	18128.2	123.75	-19.45	-5.187	-0.0360	0.6413	0.1410	0.0	8923.	0.1500	2052.50	0.0	-5.369
	843.5	0.1900	51.86	4.504	0.9917	-1.2695	3.0129	0.0	0.		670.97	0.0	-1.572
155700.	76.51	124.07	0.0	-2.024	43.000	0.1406	2.0411	0.0	10676.	4.0000	533.94	0.0	-2.854
0.	18974.1	120.59	-19.08	-5.228	-0.0066	0.6393	0.1752	0.0	8700.	0.1500	2058.27	0.0	-5.373
	766.3	0.1849	49.26	3.205	0.9951	-1.6194	3.1926	0.0	0.		671.14	0.0	-2.637
155700.	79.84	120.55	0.0	-3.401	48.000	0.1486	2.1023	0.0	10661.	4.0000	534.16	0.0	-2.927
0.	19659.6	117.29	-18.78	-5.296	-0.2436	0.6331	0.2046	0.0	8461.	0.1500	2062.97	0.0	-5.276
	703.4	0.1796	46.57	1.894	0.9684	-1.8377	3.3204	0.0	0.		671.28	0.0	-4.511
155700.	79.84	120.55	0.0	-3.401	48.000	0.1485	1.5762	0.0780	10661.	4.0000	534.16	0.0	-2.927
0.	19659.6	117.29	-18.78	-5.296	-2.4389	0.4734	0.2204	-0.0045	8462.	0.1500	2062.97	0.0	-5.376
	703.4	0.1796	46.57	1.894	0.7265	-2.0476	3.3296	0.0	0.		671.28	0.0	-4.511
155700.	81.84	117.66	0.0	1.210	48.000	0.2592	2.3762	0.0760	17743.	4.0000	534.31	0.0	2.234
0.	20059.7	114.55	-20.09	-5.806	0.4626	0.6623	0.2600	-0.0045	10669.	0.2500	2066.11	0.0	-5.118
	661.6	0.1753	44.42	7.016	1.0447	-2.6234	3.5880	0.0	0.		671.34	0.0	1.210
155700.	82.84	[illegible]	0.0	1.210	48.000	0.3170	2.4433	0.0760	21274.	4.0000	534.38	0.0	3.422
0.	20256.0	113.37	-18.21	-5.331	0.4792	0.6588	0.2328	-0.0045	11536.	0.3000	2067.54	0.0	-5.059
	647.4	0.1729	43.25	6.541	1.0466	-2.6035	3.7089	0.0	0.		671.42	0.0	1.210

TERMINAL AREA PERFORMANCE PROGRAM

FEBRUARY 14, 1975

WEIGHT FUEL	TIME DIST HEIGHT	V(KTAS) V(KEAS) MACH	V/VS R/C Q	THETA GAMMA ALPHA	FLAP GAMDOT N	CMU CL/CLM A	CL CD CLMAX	DCL DCD DCLM	FG FR WF	NE TAU	TEMP PRESS CS	BANK DIRECT SIDE	
155700.	82.84	116.11	0.0	1.210	48.000	0.3190	2.4432	0.0780	21274.	4.0000	534.38	0.0	3.422
0.	20256.0	113.07	-18.21	-5.331	0.4794	0.6588	0.2328	-0.0045	11536.	0.3000	2067.55	0.0	-5.059
	642.4	0.1729	43.28	6.541	1.0466	-2.6038	3.7088	0.0	0.		671.42	0.0	1.210
155700.	86.84	110.48	0.0	1.210	48.000	0.3962	2.5647	0.0780	23954.	4.0000	534.62	0.0	2.094
0.	21017.1	107.69	-16.94	-5.211	0.0015	0.6669	0.2077	-0.0045	11671.	0.3388	2072.68	0.0	-5.125
	573.9	0.1645	39.24	6.421	0.9960	-2.0671	3.8456	0.0	0.		671.57	0.0	1.210
155700.	90.84	106.21	0.0	1.210	48.000	0.5533	2.7723	0.0780	30980.	4.0000	534.86	0.0	1.788
0.	21744.9	103.63	-16.24	-5.196	0.0106	0.6800	0.1568	-0.0045	12785.	0.4388	2077.67	0.0	-5.141
	507.7	0.1581	36.33	6.406	0.9969	-1.5373	4.0767	0.0	0.		671.72	0.0	1.210
155700.	94.84	103.11	0.0	1.210	48.000	0.6828	2.9304	0.0780	36104.	4.0000	535.09	0.0	1.383
0.	22448.0	100.71	-15.76	-5.197	-0.0072	0.6882	0.1121	-0.0045	13425.	0.5116	2082.49	0.0	-5.161
	443.8	0.1535	34.31	6.407	0.9952	-1.0867	4.2577	0.0	0.		671.86	0.0	1.210
155700.	98.84	100.99	0.0	1.210	48.000	0.7913	3.0505	0.0780	40219.	4.0000	535.31	0.0	1.118
0.	23133.7	98.73	-15.47	-5.208	-0.0006	0.6936	0.0738	-0.0045	13901.	0.5697	2087.22	0.0	-5.174
	381.3	0.1503	32.98	6.418	0.9958	-0.7294	4.3980	0.0	0.		672.00	0.0	1.210
155700.	102.84	99.56	0.0	1.210	48.000	0.8679	3.1338	0.0780	42959.	4.0000	535.53	0.0	0.868
0.	23807.6	97.42	-15.25	-5.206	0.0039	0.6969	0.0466	-0.0045	14185.	0.6080	2091.87	0.0	-5.187
	319.9	0.1481	32.12	6.416	0.9962	-0.4868	4.4967	0.0	0.		672.14	0.0	1.210
155700.	106.84	98.61	0.0	1.210	48.000	0.9189	3.1870	0.0780	44700.	4.0000	535.75	0.0	0.612
0.	24473.6	96.57	-15.12	-5.211	-0.0017	0.6988	0.0292	-0.0045	14352.	0.6318	2096.47	0.0	-5.199
	259.2	0.1467	31.57	6.421	0.9957	-0.3269	4.5608	0.0	0.		672.28	0.0	1.210
155700.	110.84	97.96	0.0	1.210	48.000	0.9515	3.2240	0.0780	45761.	4.0000	535.96	0.0	0.473
0.	25134.3	96.03	-15.08	-5.234	0.0009	0.7007	0.0176	-0.0045	14445.	0.6459	2101.05	0.0	-5.206
	198.8	0.1457	31.21	6.444	0.9959	-0.2182	4.6009	0.0	0.		672.41	0.0	1.210
155700.	114.84	97.54	0.0	1.210	48.000	0.9762	3.2459	0.0780	46632.	4.0000	536.18	0.0	0.464
0.	25791.4	95.69	-14.99	-5.224	-0.0004	0.7009	0.0077	-0.0045	14537.	0.6571	2105.63	0.0	-5.207
	138.7	0.1450	31.00	6.434	0.9958	-0.1507	4.6311	0.0	0.		672.55	0.0	1.210
155700.	118.84	97.23	0.0	1.210	48.000	0.9910	3.2603	0.0780	47129.	4.0000	536.39	0.0	0.405
0.	26446.0	95.48	-14.94	-5.224	-0.0001	0.7013	0.0023	-0.0045	14587.	0.6630	2110.20	0.0	-5.210
	78.9	0.1445	30.86	6.434	0.9958	-0.1073	4.6489	0.0	0.		672.68	0.0	1.210
155700.	121.78	97.06	0.0	1.210	48.000	0.9985	3.2679	0.0780	47384.	4.0000	536.54	0.0	0.350
0.	26925.6	95.37	-14.92	-5.224	0.0016	0.7015	-0.0003	-0.0045	14614.	0.6657	2113.54	0.0	-5.212
	35.0	0.1443	30.79	6.434	0.9960	-0.0877	4.6582	0.0	0.		672.78	0.0	1.210

APPENDIX C

BASELINE AIRCRAFT FUEL SYSTEM CONCEPT

This appendix shows a schematic of a fuel system (Figure 105) which would be appropriate for the QCSEE baseline aircraft. The schematic is included here for the sake of completeness in documentation.

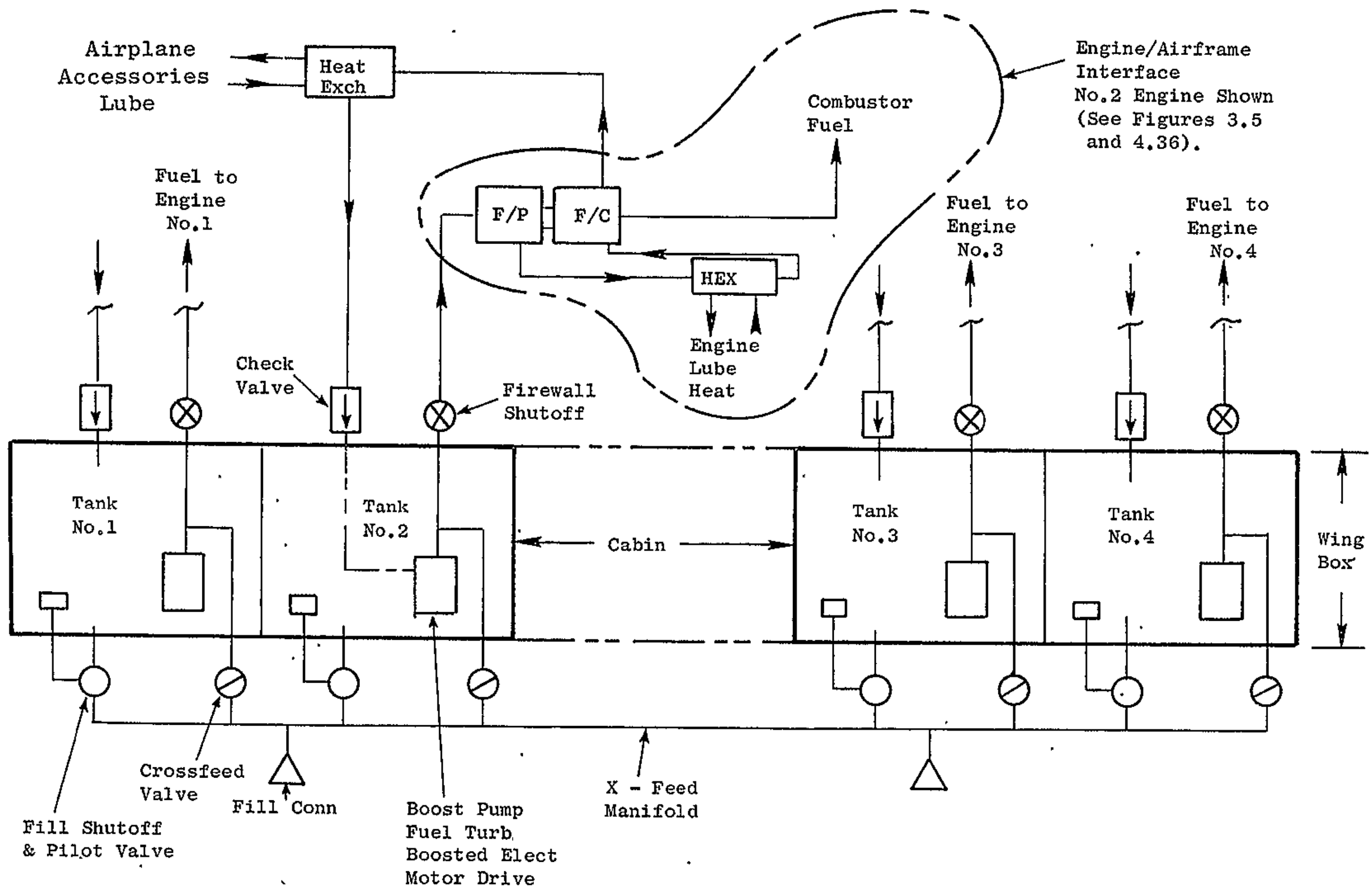

Figure 105. Baseline Airplane Fuel System Concept.

APPENDIX D

LOW MACH NUMBER INLET

This appendix discusses the low Mach number inlet for 50.8 cm (20 in.) Model Test that is consistent with QCSEE aerodynamic and acoustic requirements. The lower throat Mach number requires a larger inlet throat area for the same total engine airflow. The highlight to throat geometry is essentially unchanged based on the angle-of-attack and crosswind requirements of power-lift aircraft. The highlight to nacelle maximum diameter curvature based on aircraft cruise speed is also essentially unchanged, therefore, maximum diameter must increase. For this reason the low Mach inlet was not the number one choice for QCSEE.

QCSEE LOW MACH INLET DESIGN

AERODYNAMIC

The inlet geometry for an inlet design with a throat Mach number of 0.6 was recommended for a possible configuration for QCSEE 50.8 cm (20-in.) model tests. For the 0.6 M_i inlet, the internal lip thickness ratio of R_{HL}/R_i = 1.21 was used for low distortion at the high angle of attack and crosswind conditions at any power setting. The smaller lip thickness tested, with R_{HL}/R_i = 1.17, would have resulted in significantly higher distortion.

For external cowling, the best value of R_{max} was determined by calculating the cowl drag coefficient for a DAC-1 cowl shape at cruise conditions as a function of R_{max} or R_{HL}/R_{max} (Figure 106). It was found that as R_{max} was increased or R_{HL}/R_{max} decreased, the drag coefficient decreased rapidly, because of rapidly decreasing shock losses on the cowl, and then became nearly constant. The recommended value of R_{max} = 109.2 cm (43.0 in.) (R_{HL}/R_{max} = 0.88) was picked at the knee of the drag curve. (Note: R_{max} for the high Mach QCSEE inlet is 100.1 cm (39.4 in.).

For the calculations described above, the external-cowl nose shape was kept consistent with that of 30.5 cm (12 in.) models, as described in Reference 6, by choosing the cowl length ratio as a function of the cowl radius ratio from Figure 44.

Compared to the 0.79 M_i design, a smaller value of R_{HL}/R_{max} was required for the 0.6 M_i design due to the larger throat area of the inlet that resulted in a lower cruise mass-flow ratio. The inlet and cowl lines for the 0.6 M_i design are compared with the 0.79 M_i design in Figure 107.

Acoustic Design

The UTW nacelle acoustic design necessary to meet the QCSEE propulsion system goal of 95 EPNdB, with a low Mach inlet, must have the same suppression levels as stated in Section 6.0 for the proposed QCSEE Flight System.

- Forward radiated noise - 5.0 PNdB suppression
- Aft radiated noise - 6.5 PNdB suppression
- Core radiated noise - No suppression

The required low Mach number inlet suppression is obtained with acoustic wall treatment having a treated length-to-fan-diameter ratio of 0.50. The fan exhaust suppression is the same as shown in Section 6 with no acoustic splitter. A nacelle of this design is shown in Figure 108.

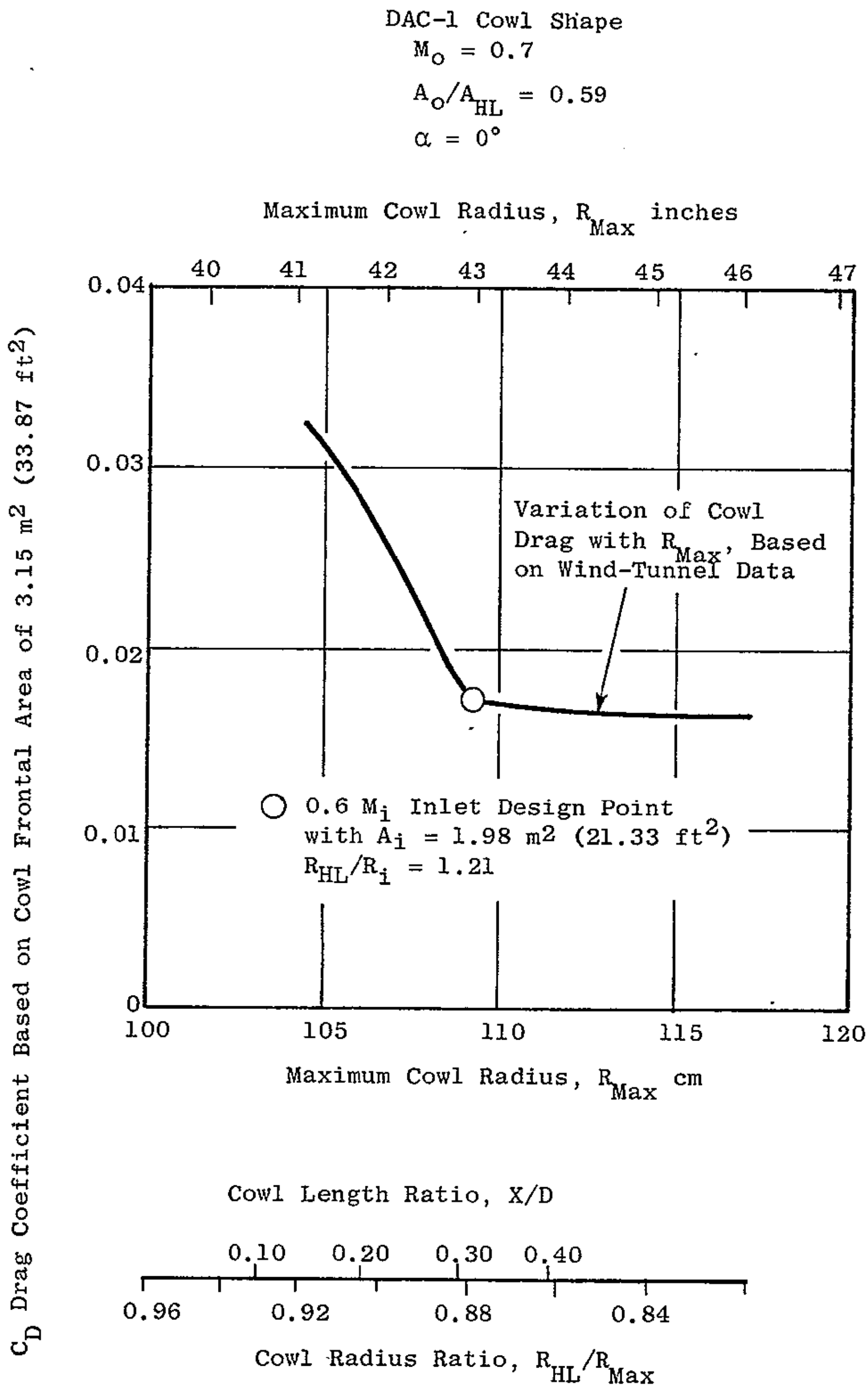

Figure 106. Inlet Cowl Drag for Throat Mach No. = 0.6.

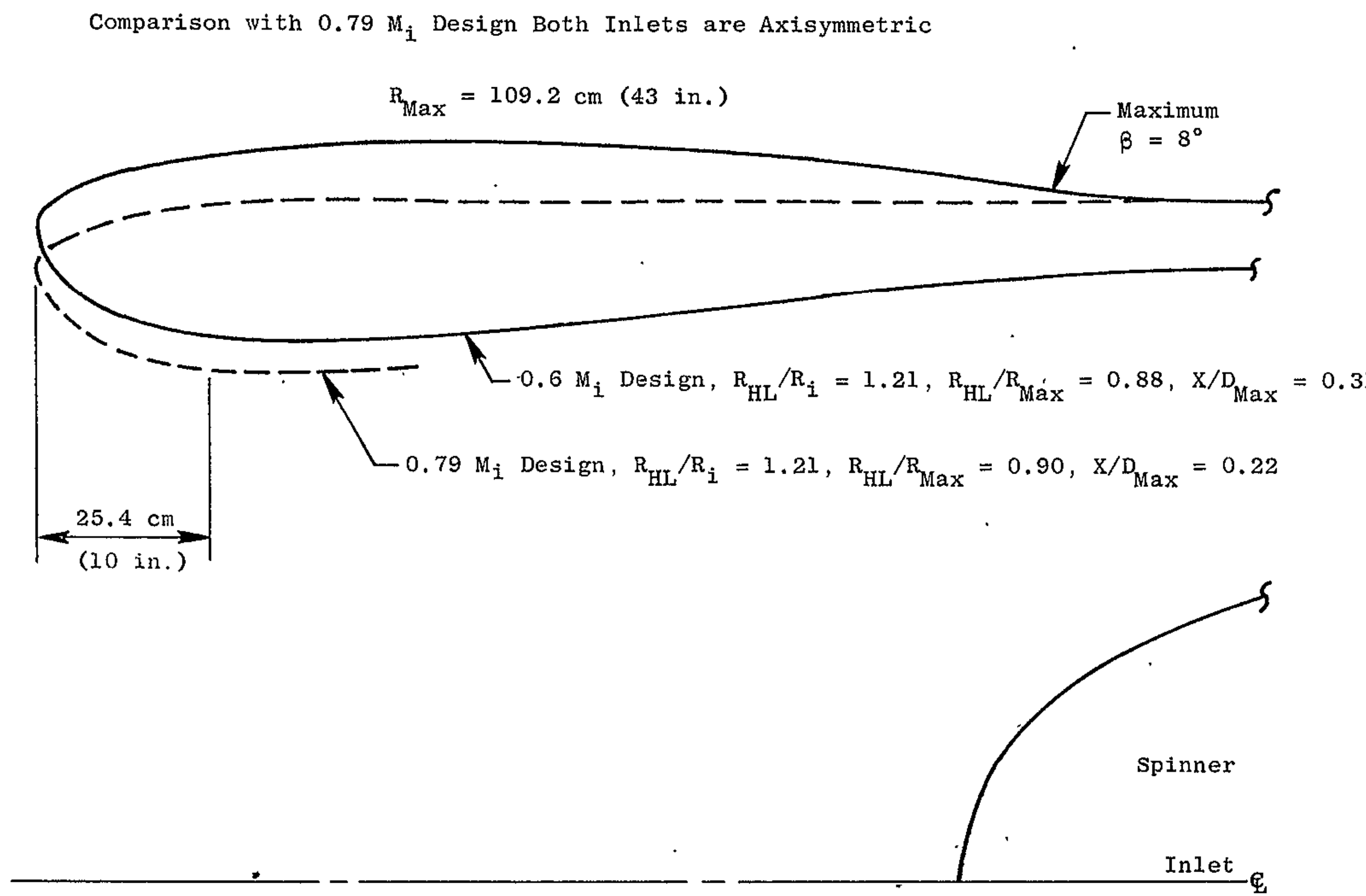

Figure 107. Inlet Design for Throat Mach No. = 0.6 vs. High Mach Inlet.

- 914.4 m (3000 ft) Runway
- 95 EPNdB Noise Level

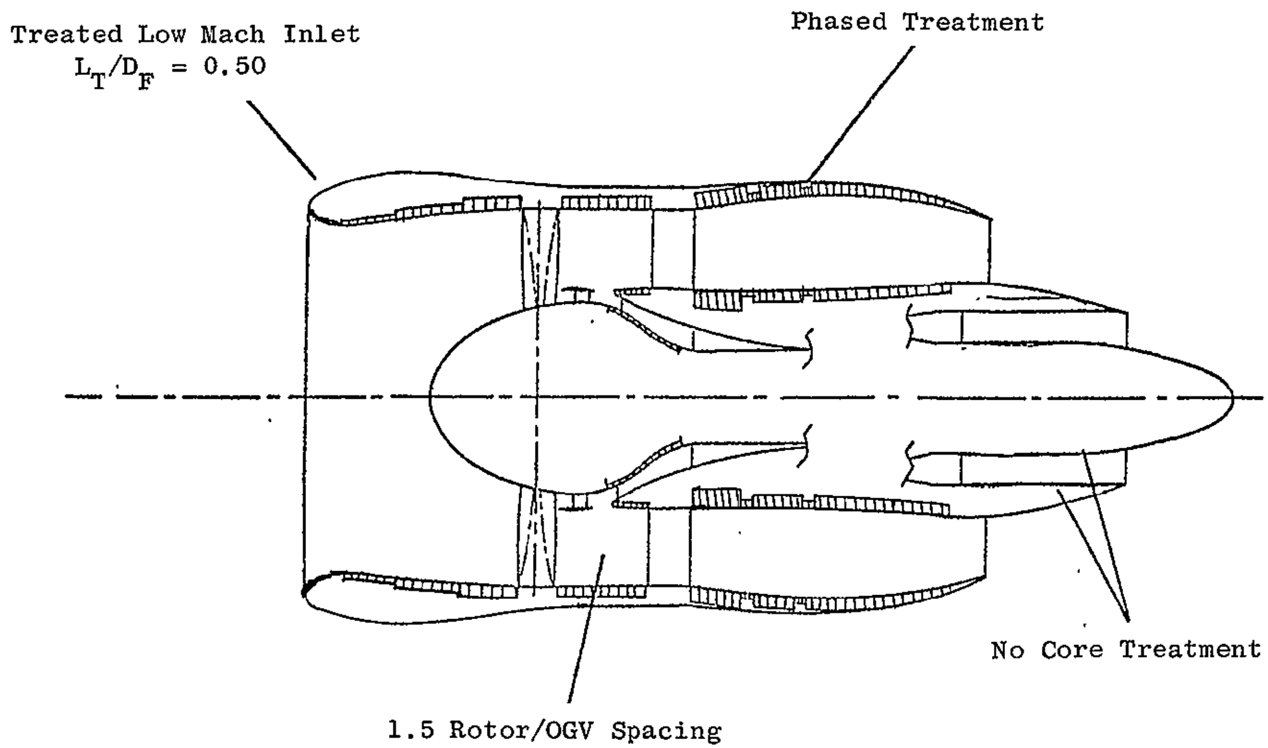

Figure 108. QCSEE UTW Engine Acoustic Treatment with Low Mach Inlet.

LIST OF SYMBOLS

ALT	Geopotential Pressure Altitude, km (feet)
APU	Auxiliary Power Unit
ASNM	Available Seat Nautical Mile
ASKM	Available Seat Kilometer
ASSM	Available Seat Statute Mile
BPR	Bypass Ratio (Bypass Stream Flow/Core Stream Flow)
C	Centigrade
CADC	Central Air Data Computer
C_D	Drag Coefficient
CIPR	Cubic Inch Per Revolution
c_f	Skin Friction Coefficient
CDP	Compressor Discharge Pressure
C_L	Lift Coefficient
$C_{L_{MAX}}$	Maximum Lift Coefficient
cm	Centimeter
C_O	Wing Chord
CPR	Compressor Pressure Ratio
C_μ	Gross Thrust Coefficient - gross thrust/(q S_W)
$C_{\mu O}$	Ram Drag Coefficient = ram drag/(q S_W)
D	Diameter
D_F	Fan Tip Diameter 180.3 cm (71 inch) For QCSEE OTW and UTW)
DAC	Douglas Aircraft Company
deg	Degrees
DLC	Direct Lift Control
DOC	Direct Operating Cost
DTAMB	Ambient Temperature Minus Standard Atmosphere Temp °C (°F)
EBF	Externally Blown Flap
ECS	Environmental Control System
EDV	Electric Depressurizing Valve
EPNdB	Effective Perceived Noise Level in Decibels

LIST OF SYMBOLS (Continued)

EVD	Elementary Vortex Distribution
F	Farenheit
FAR	Federal Air Regulations
F_G,Fg	Gross Thrust
F_N	Net Thrust
fpm	Feet Per Minute
fps	Feet Per Second
FRP	Fuselage Reference Plane
ft	Feet
g	Acceleration of Gravity
GE	General Electric
HP	High Pressure
IDG	Integral Drive Generator
I-P	Ice Protection
ISO	Isolation Valve
K	Nacelle Drag Form Factor
KCAS	Knots Calibrated Air Speed
kg	Kilogram
km	Kilometer
KVA	Kilovolt-Ampere
L	Length
Lo	Inlet Diffuser Axial Length From Throat
lb	Pound
Lt	Acoustic Treatment Axial Length
LP	Low Pressure
m	Meter
M	Mach Number
MTOGW	Maximum Takeoff Gross Weight
N	Newton
NASA	National Aeronautics and Space Administration
n	Load Factor
NMi	Nautical miles

LIST OF SYMBOLS (Continued)

OEW	Operators Empty Weight
OTW	Over the Wing
P	Rate of Roll
P18	Bypass Stream Exhaust Total Pressure, N/cm^2 (psia)
PB3	Core Compressor Discharge Bleed Pressure, N/cm^2 (psia)
PB27	Core Compressor Interstage Bleed Pressure, N/cm^2 (psia)
Pax	Passengers
PT	Total Pressure
PTO	Power Takeoff
q	Dynamic Pressure
QAD	Quick Attach-Detach
QCSEE	Quiet Clean Short-Haul Experimental Engine
QECU	Quick Engine Change Unit
R_{HL}	Inlet "Highlight" Radius-Center is Engine Axis
Ri	Inlet Throat Radius
R_{MAX}	Nacelle Maximum Radius
rpm	Revolutions Per Minute
sec	Second
SFC	Specific Fuel Consumption, kg/sec/N (lbs/hr/lb)
SL	Sea Level
SLS	Sea Level Static
S/O	Shut Off
SR	Specific Range
STOL	Short Takeoff and Landing
S_W	Wetted Area
T18	Bypass Stream Exhaust Total Temperature, °C (°F)
TB3	Core Compressor Discharge Bleed Temperature, °C (°F)
TB27	Core Compressor Interstage Bleed Temperature, °C (°F)
TO	Takeoff
TOGW	Takeoff Gross Weight

LIST OF SYMBOLS (Continued)

T/W	Thrust-to-Weight Ratio
UTW	Under the Wing
V	Velocity
V_1	Decision Speed
$V_{approach}$	Approach Speed
VP	Variable Pitch
V_{stall}	Stall Speed
W1	Engine Inlet Total Airflow, kg/sec (lbs/sec)
WB3	Core Compressor Discharge Bleed Flow, kg/sec (lb/sec)
WB27	Core Compressor Interstage Bleed Flow, kg/sec (lb/sec)
WRP	Wing Reference Plane
WTM	Wind Tunnel Model
W/S	Wing Loading - TOGW/S_W
X	Axial Distance From Inlet Highlight to Point on External Nacelle Surface/Flowpath
x	Axial Distance From Throat to Point on Inner Nacelle Flowpath
XM	Flight Mach Number
θ	Aircraft Pitch Attitude
μ	Coefficient of Friction
μ_{roll}	Rolling Coefficient of Friction
A_i	Inlet Throat Area
α	Inlet Angle of Attack
α_F	Airplane Angle of Attack Measured with Respect to Fuselage Reference Plane
β	Inlet yaw angle
c_f	Average skin-friction coefficient
Γ_2	Referred engine airflow, $W \sqrt{T_2/\delta_{T2}}$
δ_T	Relative absolute total pressure ratio
Δ	Finite difference

LIST OF SYMBOLS (Concluded)

θ_{eq}	Equivalent conical half angle of subsonic diffuser
θ_{max}	Diffuser wall maximum half angel
θ_T	Relative absolute total temperature ratio
K	Form factor used for nacelle and pylon drag calculations
L	Diffuser length from inlet throat to engine face
M_∞	Freestream Mach number
M_o	Inlet approach Mach number that differs from the freestream Mach number due to the wing flow field
$\frac{(P_{T_{max}}-P_{T_{min}})}{P_{T_{avg}}}$	Inlet steady-state total-pressure distortion number
q_F	Fan-jet dynamic pressure after isentropic expansion to freestream static pressure
q_∞	Freestream dynamic pressure
R_i	Inlet throat radius
R_{HL}	Radius of inlet high-light area or leading-edge area
R_{max}	Maximum external radius of inlet cowl
S_W	Wetted area
x	Axial distance along inlet centerline measured from inlet throat

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