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A PRELIMINARY DESIGN STUDY OF A LAMINAR FLOW CONTROL WING OF COMPOSITE MATERIALS FOR LONG RANGE TRANSPORT AIRCRAFT

G. R. Swinford

April 1976

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16. Abstract

The results of an 11 1/2 month preliminary design study are reported. The selected study airplane configuration is defined. The suction surface, ducting, and compressor systems are described. Techniques of manufacturing suction surfaces are identified and discussed. A wing box of graphite/epoxy composite is defined. Leading and trailing edge structures of composite construction are described. Control surfaces, engine installation, and landing gear are illustrated and discussed. The preliminary wing design is appraised from the standpoint of manufacturing, weight, operations, and durability. It is concluded that a practical laminar flow control (LFC) wing of composite material can be built, and that such a wing will be lighter than an equivalent metal wing. As a result of this study, a program of suction surface evaluation and other studies of configuration, aerodynamics, structural design and manufacturing, and suction systems are recommended. It is concluded that future development of composite primary structure should consider the requirements of LFC wings, and be coordinated with an LFC development program.

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FOREWORD

This is the final report for NASA Contract NAS1-13872, "A Preliminary Design Study of a Laminar Flow Control Wing of Composite Materials for Long-Range Transport Aircraft." The program was accomplished during the time period from April 1975 to March 1976 and was monitored by Mr. Richard D. Wagner of the Systems Technology Branch, Aeronautical Systems Division, NASA Langley Research Center.

The individuals who made technical contributions to the program and their areas of activity are as follows:

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B. L. Reynolds . . . . Deputy Program Manager
W. Pfenninger . . . . Aero and LFC Specialist
R. R. June . . . . Composite Specialist
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J. R. Curran . . . . Design
J. R. Bender . . . . Structural Analysis
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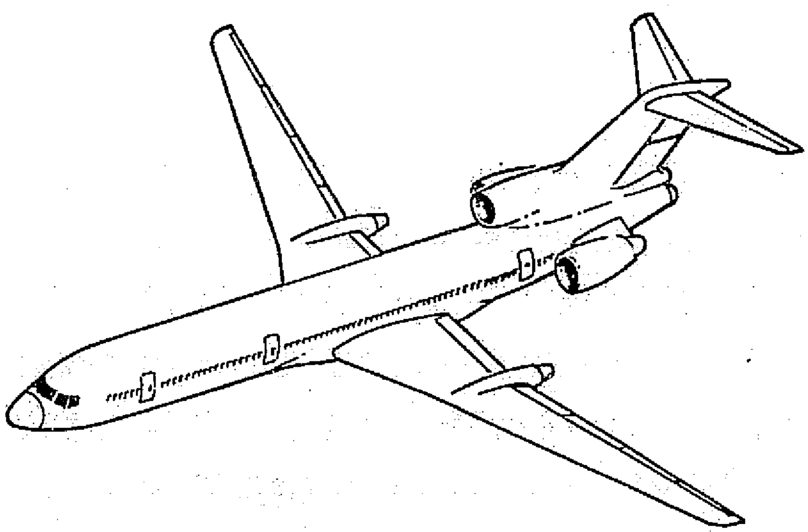
A Preliminary Design Study Of A Laminar Flow Control Wing Of Composite Materials For Long Range Transport Aircraft

G. R. Swinford

1.0 SUMMARY

A preliminary design study of a laminar flow control wing of composite materials was performed. The wing was designed for a long range (5,500 nautical mile) 200 passenger transport on which laminar flow was to be achieved by boundary layer suction.

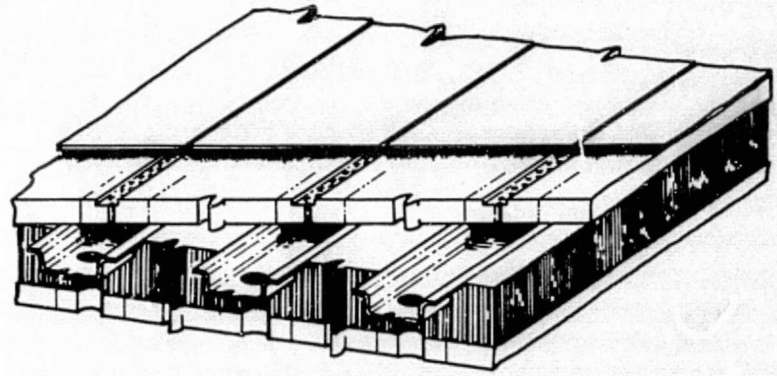
An airplane configuration was selected for the study, largely using the previous LFC transport studies of the LFC specialist, Dr. Werner Pfenninger, and Boeing's experience in large commercial transports.



LFC Airplane Configuration

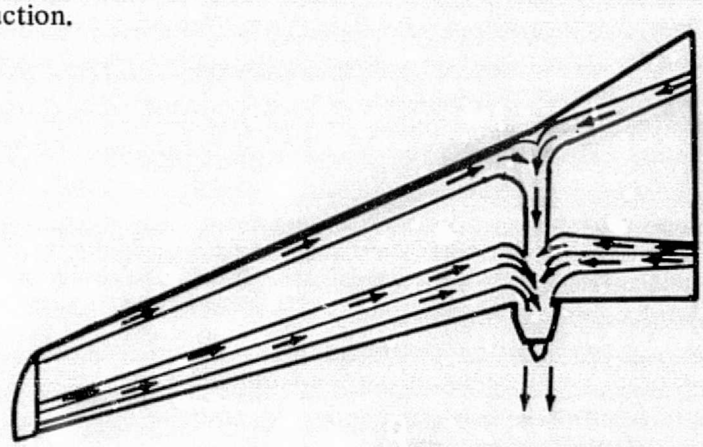
Features of the airplane include: three body mounted propulsion engines, two wing mounted suction engines, simple trailing edge high lift devices, integral suction surface structure, and laminarization of 95% of the exposed wing area.

A wing preliminary design was accomplished based on the results of design trade studies. Concepts were developed for leading and trailing edges, the wing box, suction surfaces, ducting, landing gear installation, laminarized movable surfaces, and major wing joints.



Integrated Structures and Spanwise Ducts

In the recommended design approach the wing suction surfaces were integrated into the advanced composite faced aluminum honeycomb sandwich skin panels. The outer surface of the entire wing was sheathed with thin titanium sheet for electrodynamic and environmental protection. Interior structure was also of advanced composite/honeycomb construction.



Main Duct System

Suction air passes through the outer skin into plenums, spanwise tributaries, chordwise collectors, and into the main spanwise ducts. These main spanwise ducts, which are integral with the leading edge and fixed trailing edge, carry the suction air to a single suction engine/compressor installation on each wing.

From this study the following conclusions were reached: an advanced composite LFC wing can be designed and manufactured; advanced composite structure facilitates the integration of suction ducts and plenums into the primary skin panels; the recommended honeycomb sandwich panels will maintain smooth aerodynamic contours to a high degree of accuracy.

As a result of this study, Boeing recommends a program of in-flight service evaluation of various suction surface configurations, to resolve questions of clogging, contamination, flow degradation, and other environmental effects. Other studies in the areas of configuration and aerodynamic development, structural design and manufacturing, and suction system development are also recommended. Future composite primary structure development programs should be coordinated with a continuing LFC development program.

2.0 INTRODUCTION

Laminar flow control (LFC), the maintenance of laminar flow through controlled suction on the airplane skin , offers the largest gains to be made in aircraft performance and reduced fuel consumption of any of the currently envisioned advances in aircraft technology. The LFC concept was proven in flight experiments on the X-21 airplane in the mid 1960's.

In the succeeding decade advances in many fields of manufacturing technology have brought the manufacture of suction surfaces within the realm of commercial application. In the same time period development of advanced structural composites has progressed to a point where these high strength, lightweight materials are being considered for commercial aircraft primary structure.

This study addressed the manufacturing and structural design problems of the suction surfaces and the integration of the suction surfaces and duct system with an LFC wing structural design, in which advanced composites were the principal structural materials. The study consisted of (1) an evaluation of various approaches to the design of the wing box and ducting system of a subsonic transport wing with laminar flow control, (2) a preliminary wing design, and (3) an evaluation of design feasibility.

3.0 SYMBOLS AND ABBREVIATIONS

A	Area, $m^2 (ft)^2$
b	Span, $m(ft^2)$
c	Chord
C_L	Wing Lift Coefficient
C. G.	Center of Gravity
c_l	Section Lift Coefficient
C_P	Wing Pressure Coefficient, $P_{local} - P_\infty/q_\infty$
C_q	Suction Coefficient, $\rho_{local}\ U_{local}/\rho_\infty U_\infty$
c_{SL}	Streamwise Chord Length
E	Modulus of Elasticity, GPa (lb/in^2)
F	Stress Pa (lb/in^2)
G	Modulus of Elasticity-Shear, GPa, (lb/in^2)
g	Acceleration Due to Gravity, 9.807 m/sec^2 (32.172 ft/sec^2)
I	Moment of Inertia, cm^4 ($in.^4$)
J	Torsional Constant, cm^4 ($in.^4$)
L	Lift, Newtons (lbs)
LFC	Laminar Flow Control
LRA	Load Reference Axis
l	Streamwise Airload/Unit Span, N/mm (lb/in.)
M	Mach Number
MSI	$Lb/in.^2 \times 10^6$
N	Panel End Load, kN/m (lb x 10^3/in.)
n	Load Factor
P	Pressure, Pa ($lb/in.^2$)
P_T	Total Pressure, Pa ($lb/in.^2$)
q	Dynamic Pressure, Pa ($lb/in.^2$)
q	Shear Flow, N/m (lb/in.)
T	Temperature, K^o (R^o)
t	Thickness, mm (in.)
u	Velocity, m/sec. (ft/sec)
w	Mass, kg (lb)
x	Distance Normal to Y Axis
y	Distance Normal to X Axis
Δ	Increment
δ	Angular Deflection
ϵ	Strain
η	Fraction of Semi-Span
Λ	Angle of Wing Sweep @ .25c
$\Delta P/q$	Duct Loss Coefficient
$W\sqrt{T}/P_T A$	Duct Flow Function

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SUBSCRIPTS

c	Chordwise, Compression
n	Factor
r	Root
t	Tension, Tip
w	Wing
ULT	Ultimate Load, 1.5 x Limit Load
LIMIT	Limit Load, 2.5 g
S	Shear
SU	Shear Ultimate

LAMINATE ORIENTATION CODE

The laminate orientation code used in this report is the "Basic Condensed Code" of Ref. 4. This code is used to describe the proportionate distribution of laminae among a number of specified orientation angles, without regard to layup sequence or the total number of laminae per se. Features of the code are briefly described as follows:

Each lamina is denoted by a number which represents the orientation of its filament direction relative to the X-axis. In this report the X-axis is chosen to represent the direction of principal axial loading.

Plus and minus signs are used when filament orientation angles are equal in magnitude but opposite in sign. Each sign represents a single lamina.

Laminae with different angular orientations are separated by a slash.

The number of laminae at each angle is denoted by a numerical subscript. When more than one subscript appears in a Basic Condensed Code Laiminate description, the common factors are removed (e.g., the combination $0_{12}/60_3/90_6$ would be listed as $0_4/60/90_2$ by dividing out the common factor 3).

The complete laminate designation is enclosed in brackets. In this report the subscript n may be used to indicate that multiples of the laminate combination are used to make up the required thickness.

Example: $[0_2/\pm 45/90]_n$ represents two layers of fibers at 0°, one layer at $+45^\circ$, one layer at -45°, and one layer at 90°, the entire combination repeated n times.

4.0 AIRCRAFT CONFIGURATION

4.1 MISSION DEFINITION

The LFC mission was defined by NASA/LRC in the statement of work. The LFC aircraft was required to carry approximately 200 passengers over a range of 10 186 km (5500 nautical miles) at a cruise mach number between 0.7 and 0.8.

4.2 CRITERIA

The following criteria were adopted by Boeing as guidelines to assure a commercially viable configuration:

- Adherence to FAR 25 requirements.
- Airplane layout to be conventional.
- Complexity of LFC features to be minimized.
- Airplane to be compatible with present airport facilities.
- Best existing power plant technology to be used.
- Fuselage width selected by drag considerations.
- Laminarized areas to be free of protuberances and gaps to the greatest practical extent.

4.3 CONFIGURATION SELECTION

The LFC aircraft configuration was based on past LFC transport studies by Dr. Werner Pfenninger, the program LFC specialist; the contractor's experience and expertise in the design of large commercial aircraft; and an extrapolation of current trends. Airplane configuration optimization studies were not within the scope of this contract.

The wing area and geometry, thrust requirements, and takeoff gross weight were selected on the basis of previous unpublished parametric LFC studies. The principal dimensions of the selected configuration are listed in table 1 below, and the configuration is illustrated in figures 1, 2 and 3.

A 10% first class/90% tourist ratio was assumed. This is consistent with the ratio currently in use by some international operators and reflects the expected future impact of SST service on long international routes.

Table 1. Airplane Characteristics

Dimensions		
Wing span	59.44m	(195 feet)
Tailplane Span	16.7m	(54.8 feet)
Length — overall	61.63m	(202.2 feet)
Height — overall	12.9m	(42.5 feet)
Wheelbase	24.8m	(81.5 feet)
Wheel track	6.4m	(20.9 feet)
Body diameter	4.06m	(13.3 feet - 160 inches)
Cabin length	44.6m	(146.4 feet)
Cabin width — maximum	3.8m	(12.5 feet - 150 inches)
M.A.C.	5.5m	(18.08 feet)
$c_t/c_r = .25$		
Sweep Angle (Λ) = 22° at .25 Chord		
Areas		
Wing (aero ref.)	293.4 m^2	(3158 $feet^2$)
Wing (gross)	313.2 m^2	(3371 $feet^2$)
Horizontal tailplane	49.1 m^2	(528.3 $feet^2$)
Vertical tailplane	52.0 m^2	(560 $feet^2$)
Propulsion		
Main engines	(3)	CFM 56 @ 9,980 kg (22,000 pounds) takeoff thrust
or	(3)	JT10D = 11,340 kg (25,000 pounds) takeoff thrust (center engine drives empennage suction compressor)
LFC engines	(2)	TF34 derivative driving - 3-stage suction compressor for upper and lower wing surface. 1020.6 kg (2250 pounds) thrust engine at takeoff
Passenger accommodations		
20 First class — 4 abreast @ 106.7 cm (42 inch) pitch		
183 Tourist Class — 6 abreast @ 86.4 cm (34 inch) pitch		
203 Total passengers		
Entry doors — type "A" 1.07 m x 1.93 m (42 x 76 inch) - 3/side		

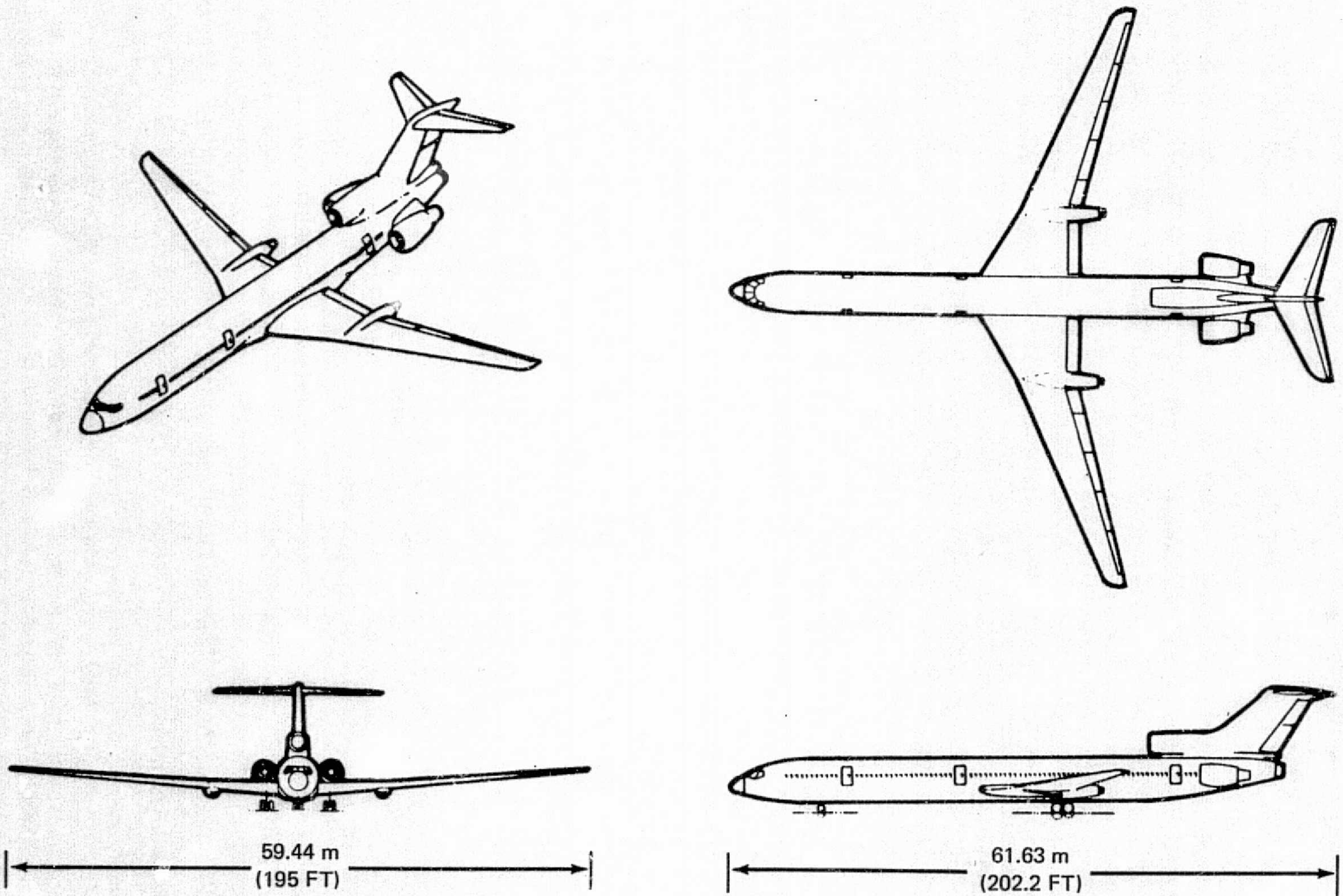

Figure 1. General Arrangement

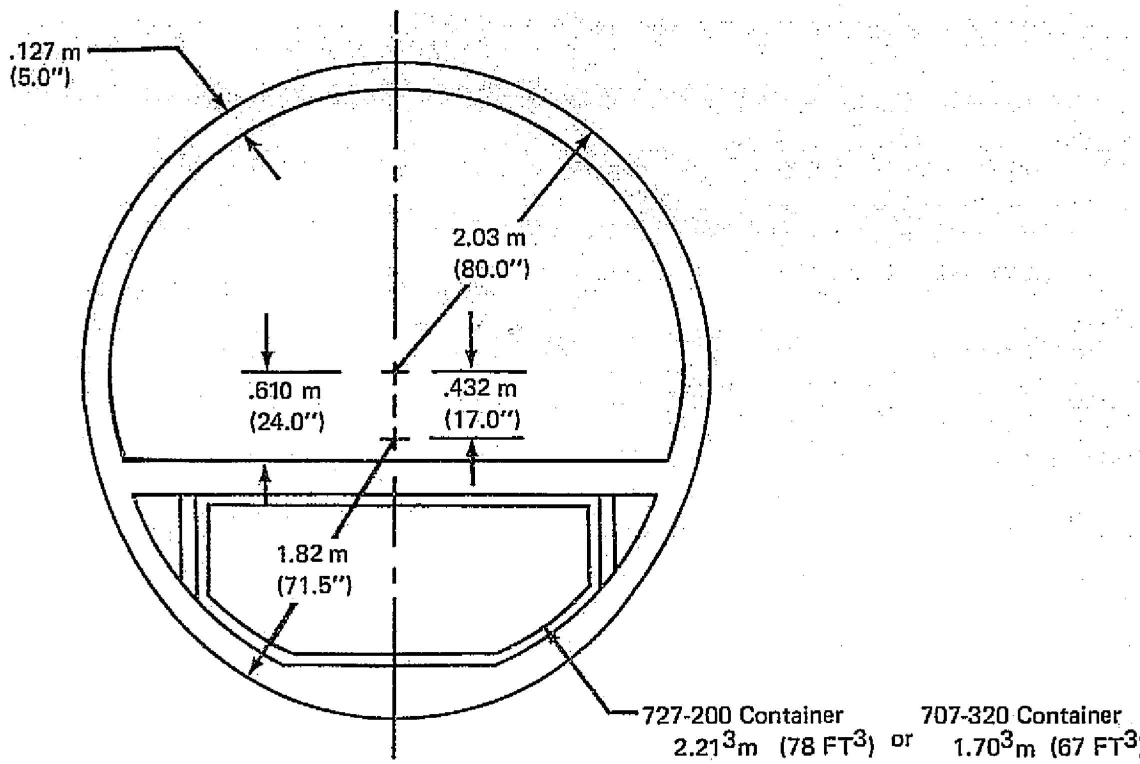

Figure 2. Body Cross Section

A single aisle cabin arrangement, with six abreast seating, was chosen to minimize fuselage wetted area. The cabin length was determined by a floor plan layout (figure 3) which included galley, lavatory, and closet space per the contractor's preliminary design criteria fo. long-range airplanes. Door and cross-aisle requirements of FAR 25 were incorporated. Type "A" entry doors, 1.07 m 1.93 m (42 in. x 76 in.), were selected to minimize the number of drag-producing entrances, with their requirements for cabin cross-aisle ai... All entry doors were located in the constant section portion of the fuselage, so identical assemblies could be used. The number of such doors (3 per side) was chosen to meet current FAR requirements (100 persons per Type "A" door, with half the doors not usable) while permitting some growth in seating capacity. The cargo-baggage capacity of the lower fuselage area is approximately 38% greater than the baggage volume required for 203 passengers, based on the use of existing Boeing 727-200 containers $2.21 m^3$ (78 ft^3) in volume, and on a requirement for $.14 m^3$ (5 ft^3) per passenger.

A conventional wing-body fairing was required to provide adequate cross section for landing gear stowage. The aft body closure angle is steeper than conventional practice dictates, as a result of the assumption that suction will be used to delay separation in this area.

Tailplane volume coefficients are comparable to those of the contractor's current production tri-jet, the 727-200. The tailplane weight and drag reductions which would result from the use of full-time stability augmentation were recognized. However, in the absence of pitch-up and deep stall recovery studies, no such reductions were assumed. The selected tailplane configuration places the laminarized horizontal tail well above the wing wake and engine inlet disturbances.

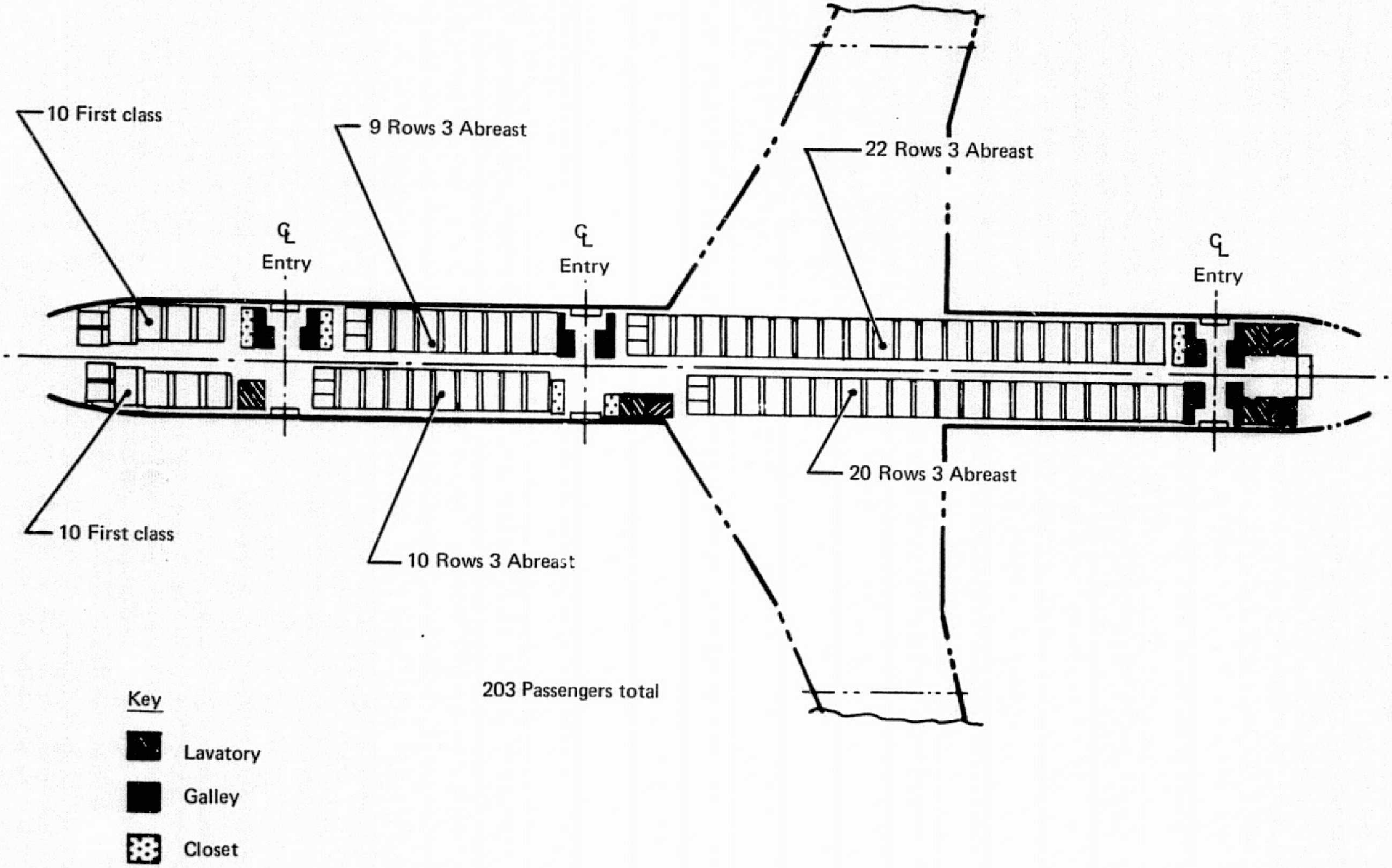

Figure 3. Interior Arrangement

The wing provides volume for approximately 46 135 kg (101 711 lbs) of fuel carried inboard of 70% semispan, between the spars. The fuel cell arrangement is illustrated in figure 35 (Sect. 5.0). Fuel requirements for the defined mission were estimated from previous parametric studies to be approximately 45 360 kg (100 000 lbs). It is assumed that 85% of the internal volume of the fuel bays is actually available for fuel.

The wing profile selected was based on a Boeing developed transonic airfoil family. In the absence of configuration optimization studies, representative wing twist, twist distribution, and wing root profile were adopted. High lift devices consist of plain trailing edge flaps and drooped outboard ailerons, with suction employed to prevent separation on takeoff and approach. Hinges and actuators were placed within the wing airfoil section to eliminate turbulance problems from external mechanisms. No leading edge devices or spoilers were included, although future trade studies of weight, range, and field length requirements might indicate the complexity added by such devices to be worthwhile.

Landing gear truck and tire size, track, and wheelbase were determined by the usual ground stability, tire capacity, and pavement loading criteria. Pavement loading is comparable to that of existing international transports and would require no improvement to existing facilities. Static ground clearance and takeoff rotation angles are in accordance with Boeing's current practice. Thus, landing gear strut lengths were kept to a minimum and airplane door sill heights were made compatible with existing loading and passenger boarding facilities.

The decision to laminarize virtually the entire wing surface led to the aft fuselage engine location. This engine location separated engine-related ducting and equipment from the wing suction installation and minimized the effect of engine noise on laminarization. A twin engined layout using engines in the 209 066 N (47 000 lb) takeoff thrust class was considered, but since the defined long-range mission implies operation over water and inhospitable terrain and, since the single engine cruise altitude would not be great enough for efficient LFC performance, the twin configuration was not pursued.

There are two modern high bypass ratio engines of suitable size for the 3-engined LFC airplane, the CFM 56 and the JT10D, both having approximately 97 860 N (22 000 lb) takeoff thrust ratings. The two engines are nearly identical in their external (cowled) appearance.

The TF-34 was selected as the basis for the suction powerplant. Two additional compressor stages were added to replace the fan which was deleted. A coaxial three-stage suction compressor was connected to the fan turbine. The compressor power and RPM requirements differed from those of the deleted propulsion fan, so the fan turbine diameter was assumed to be reduced. Since it is not thermodynamically efficient to require the suction powerplant to ingest low energy suction surface air, an S-shaped turbine inlet duct was provided to draw engine air from the outside airstream.

The use of two suction powerplants was influenced by the desire to minimize the complexity and maintenance cost of the LFC system, and by the size of the current engine which is assumed to serve as the suction powerplant core. Boeing's experience with a particular large commercial transport powered by modern high bypass ratio engines indicates that well over half the operational maintenance and schedule interruption cost (excluding overhaul) is attributable to powerplant and APU systems, thus it is important to minimize the number of suction powerplants. Additionally, the assumed suction powerplant is somewhat oversize for an airplane of the size dictated by the mission requirements. The excess capacity could be absorbed in driving pressurization equipment or electrical generators, or the core engine could be derated for longer life and greater reliability. In the absence of systems trade studies, a modest residual thrust was assumed to be available.

The suction powerplant location was dictated by considerations of duct size, duct length, and available wing volume. The location chosen provided adequate wing box depth to accommodate the cantilevered engine mounting and duct system while avoiding wing-body intersection disturbances. It was assumed that the aft location of the suction engine nacelle would provide some beneficial area distribution affect.

The LFC suction surface includes virtually all the wing surface, extending from 0.8% chord rearward to 95% chord. The suction engine nacelle fairings are laminarized by suction as far aft as the wing trailing edge. The suction surface of the horizontal tailplane also extends aft to 95% of its chord, or to as great a percentage of chord as the available section depth permits. The vertical tailplane and center engine inlet are also laminarized by suction. Suction will be used to minimize separation at the turbulent wing-body intersection and the aft body closure area. Aft fuselage and tailplane suction was assumed to be provided by a suction compressor mechanically driven by the center engine, exhausted overboard above the center engine nozzle.

5.0 DESIGN STUDIES

5.1 SUCTION DUCTING

The suction duct system in the LFC wing must carry a large volume of air from the wing surface to the suction compressors with minimum energy expenditure. The large ducts required occupy a substantial portion of the wing cross section. The location and routing of these ducts becomes a critical consideration in the structural layout of the LFC wing. For this reason, a substantial portion of this design study effort was devoted to suction duct sizing and arrangement.

It was initially decided, for reasons of pumping efficiency and operational flexibility, to keep upper surface and lower surface suction air separated, passing each stream through a separate compressor stage. This decision dictated a separate system of chordwise ducts and major spanwise ducts for each surface.

Because suction system optimization was beyond the scope of the contract effort, conservative assumptions were made for external pressure distribution, maximum duct velocity and suction mass flow.

The duct sizes incorporated in the preliminary design were determined, by later analysis, to be larger than necessary.

All duct size calculations were made for M = .8 cruise at 11 582 m (38 000 ft) in standard atmosphere. The selected cruise altitude represents a flight level lower than ideal for the LFC transport, but one which might be required due to conflicting traffic enroute. Provision of adequate suction capacity under this set of conditions ensures more than adequate capacity at higher altitudes or lower cruise mach numbers.

A similar conservative approach was taken in defining the extent of laminarization. The suction surface was assumed to extend from 0.8% to 95% of the chord length.

The large suction requirement of the rearmost 30% of the wing surface accounts for nearly half the total duct cross section area required (figure 4).

Four airflow system computer programs, "LFC", "LFC1", "EJLFC", and "LFCDUC", were used to provide the following information:

- Upper and lower surface suction flow rates, relative to chordwise location.
- Wing surface pressure relative to chordwise location.
- Slot width, spacing, and number of slots for each surface, relative to chordwise location.
- Internal flow losses and duct velocities.
- Ejector performance for flow mixing.
- Chordwise duct size.

These four programs are general in nature and may be applied to future LFC studies. They are described by figures 5 through 8.

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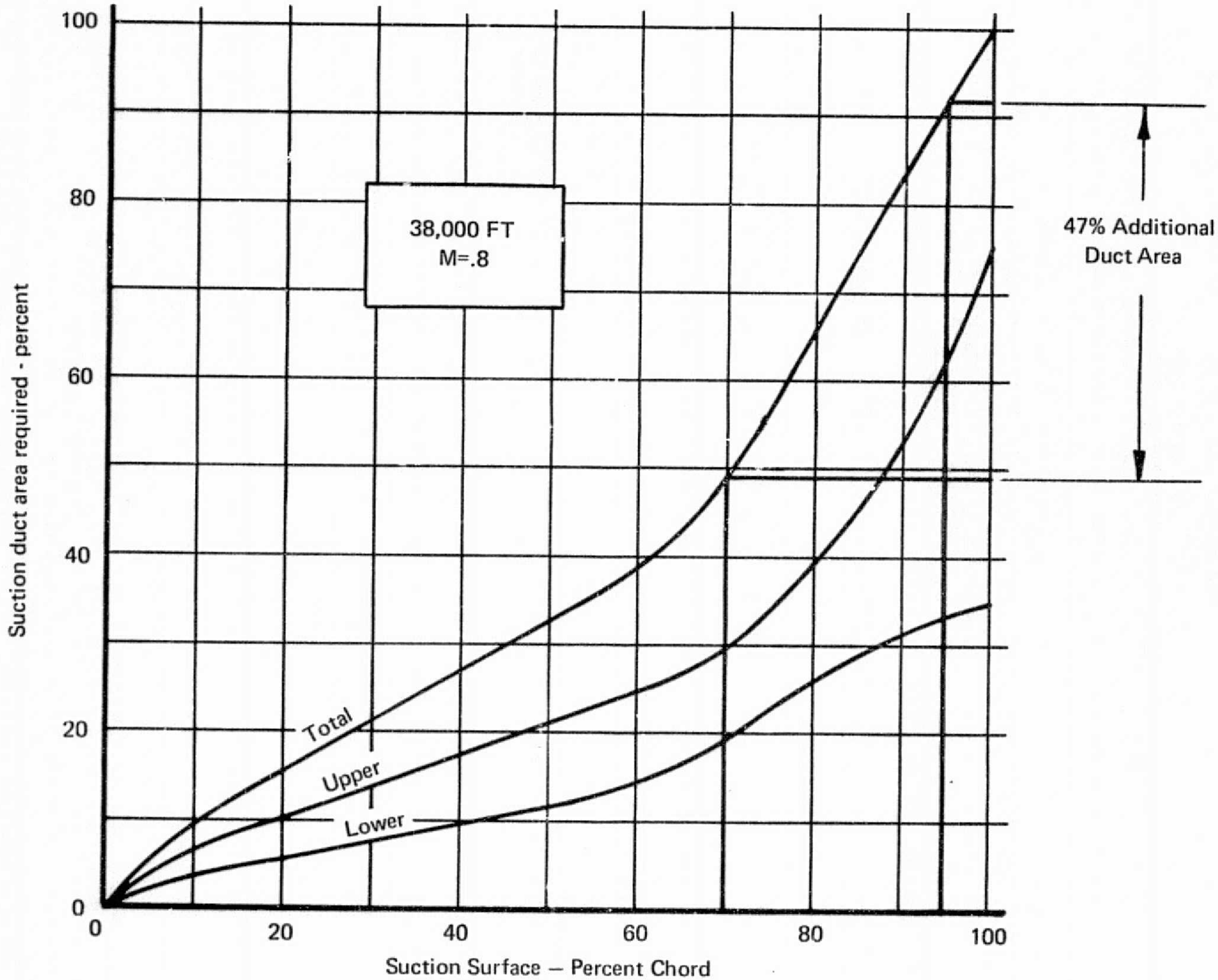


Figure 4. Suction Duct Area

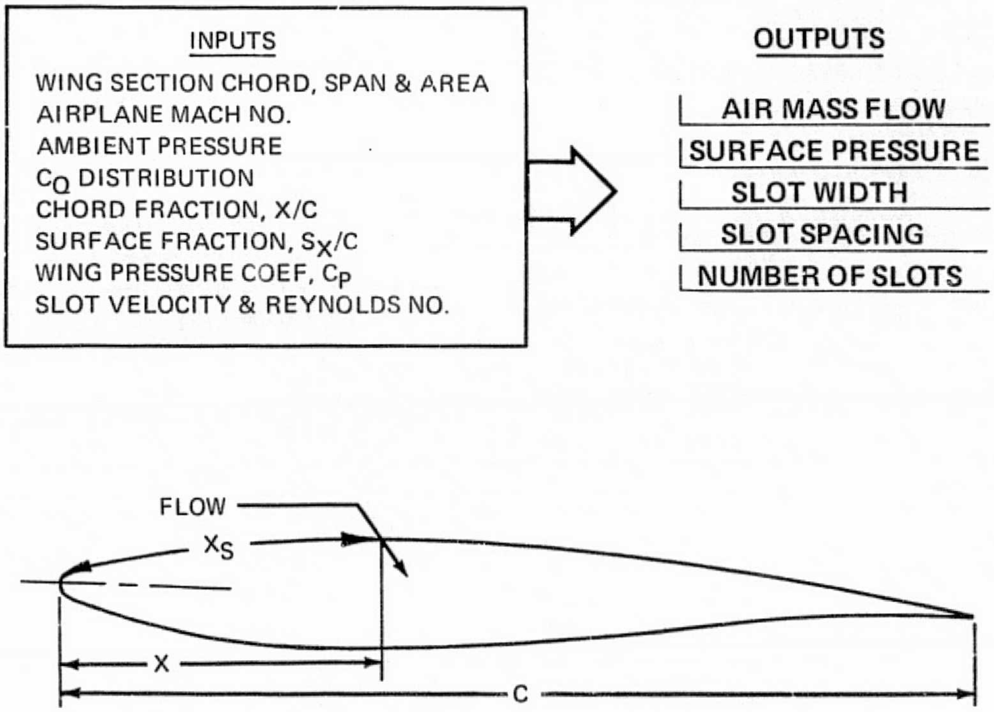


Figure 5. Computer Program – Code LFC

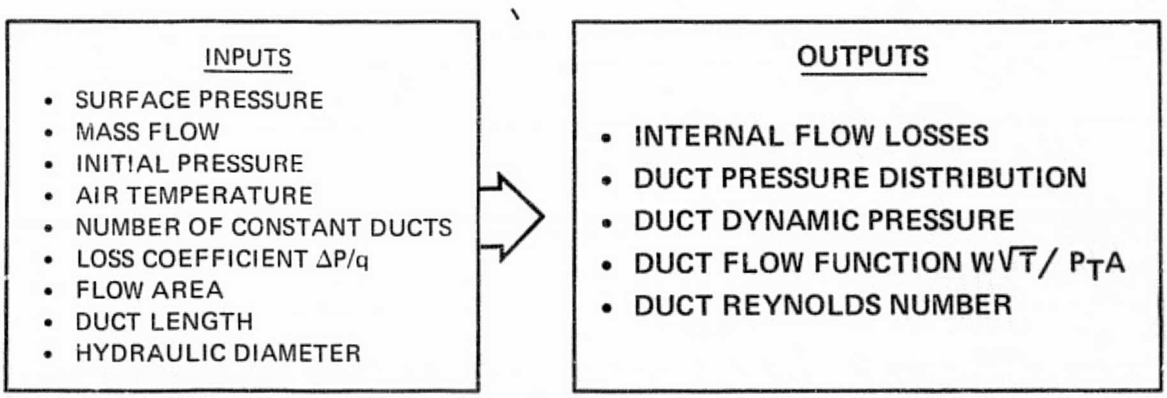


Figure 6. Computer Program – Code LFC1

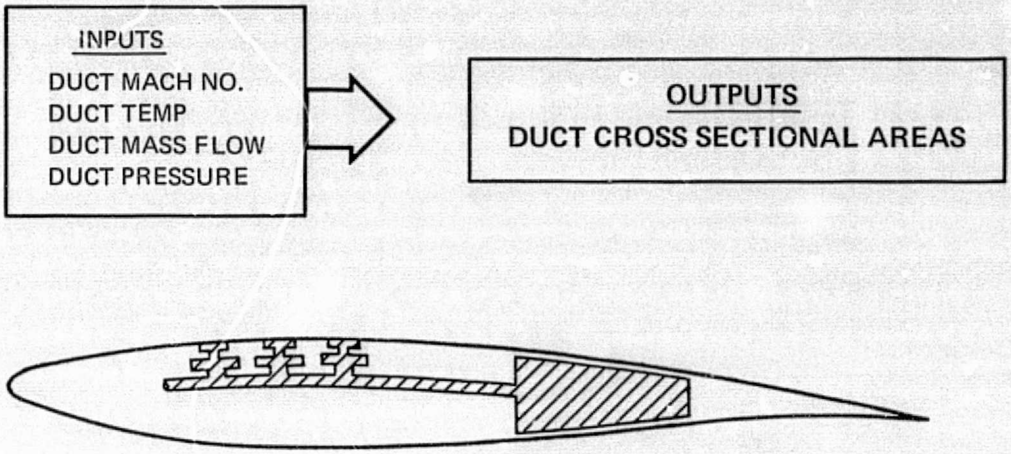


Figure 7. Computer Program - Code LFCDUC

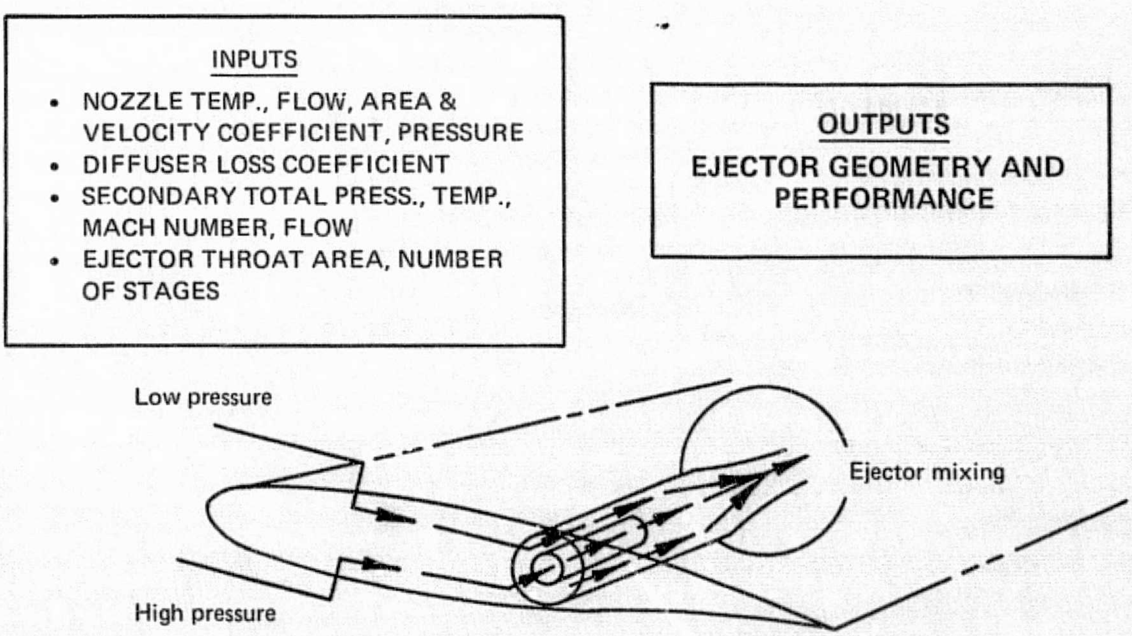


Figure 8. Computer Program — Code EJLFC

Pressure in each suction duct must be reduced to a level below the lowest pressure on the suction surface being served by that duct. For efficiency, surface areas with substantially different pressure levels are served by different ducts. The air in these ducts must at some point be brought to a common pressure level. It appeared advantageous to utilize the concept of ejector mixing, utilizing the energy in one duct air stream to pump air to from another duct to a common pressure level. The computer program EJLFC (Fig. 8) was written to evaluate this concept. A diffuser, with length proportional to the duct cross section, is required downstream from the ejector. In the main ducts the required diffuser length proved to be difficult to accommodate. Ejector mixing was also considered for use in the smaller ducts, but optimization of the ejectors and diffusers was beyond the contract scope. In addition, it was the opinion of the LFC specialist that separate individually modulated compressors or compressor stages would provide a greater degree of flexibility for off-design suction cases. Ejector mixing appears to be a valid concept which requires further investigation.

At the beginning of the suction duct study, it appeared desirable to divide the wing into three independent regions, served by separate main ducts (concept I) in order to more readily accommodate varying spanwise pressure distributions by suction adjustment (figure 9). Each region was to have one large chordwise duct into which the spanwise tributary ducts delivered their air. After preliminary duct sizing calculations and layout it was realized that the required tributary cross sections were excessive, and the separate main duct system unduly complex. In addition, the large chordwise ducts required large spar penetrations.

Another concept, in which each rib bay was served by a small chordwise duct, resulted in small tributary ducts and small spar penetrations. This concept was retained through all subsequent studies.

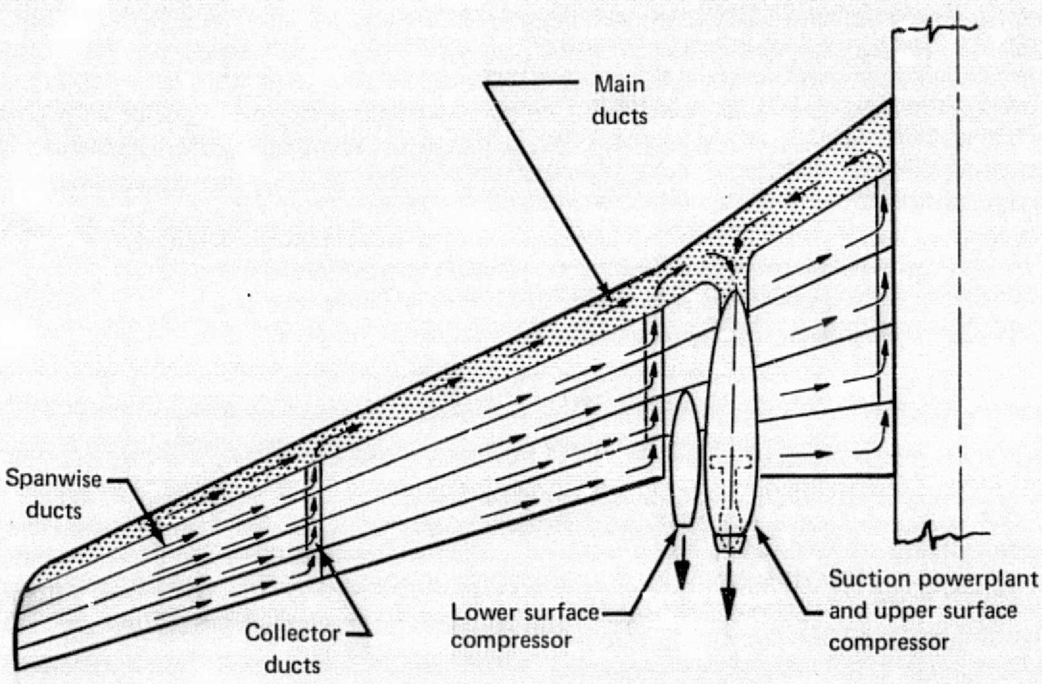


Figure 9. Wing Suction Concept I

A suction duct layout (concept II) with all main ducts located aft of the rear spar was studied. To accommodate the large ducts, the rear spar was moved forward from the initially assumed 70% chord point to 55%; the front spar being moved from 19% to 10% to keep the wing box cross section and volume approximately the same. Main ducts were sized for a maximum duct velocity M = .25. A separate lower surface compressor was provided, driven by a geared cross shaft from the larger upper surface compressor (figures 10 and 11). Chordwise ducts were provided by double-walled ribs in which the cavity between the webs was divided into upper and lower plenums (figure 12).

A consideration of the relationship between duct velocity and duct pressure drop (figure 13) led to reconfiguration of the system with duct velocities limited to M = 0.2 (concept III). With the resulting increase in main duct cross sectional area, the volume available aft of the rear spar was not sufficient to contain all the main ducts. Upper surface air from 55% chord forward was carried forward by the upper surface chordwise ducts to a main duct ahead of the front spar (figure 14). The forward upper surface duct then penetrated both spars to reach the suction compressor. The chordwise plenums in the double-walled ribs were changed to chordwise ducts of efficient shape located in the center of each rib bay (figure 15).

At this time a reconfiguration of the suction engine and compressor system was made. An axial, 3-stage suction compressor was adopted, eliminating the weight and complexity of a geared cross shaft drive. The suction compressor, coaxial with the suction engine, was directly connected to the low pressure turbine shaft (figure 16).

Suction system concept III was adopted for incorporation with the wing structure in all subsequent studies. The significant features of this concept include chordwise collector ducts in the center of each rib bay; the incorporation of a main spanwise duct ahead of the front spar, requiring passage of a large duct through the front and rear spar; and the co-axial arrangement of the 3-stage suction compressor and the suction power plant.

5.2 SUCTION SURFACE

For the purposes of this study, the suction surface was considered to be the external aerodynamic surface containing openings to admit boundary layer air into the suction system, and the plenums immediately underlying the external surface.

The following criteria were established for selection of the LFC suction surface:

- Aerodynamic suitability.
- Future commercial producibility.
- Adaptability to structural concepts.
- Environmental stability.
- Maintainability.
- Inspectability.
- Reasonable product cost.
- Minimum weight penalty.
- Durability.

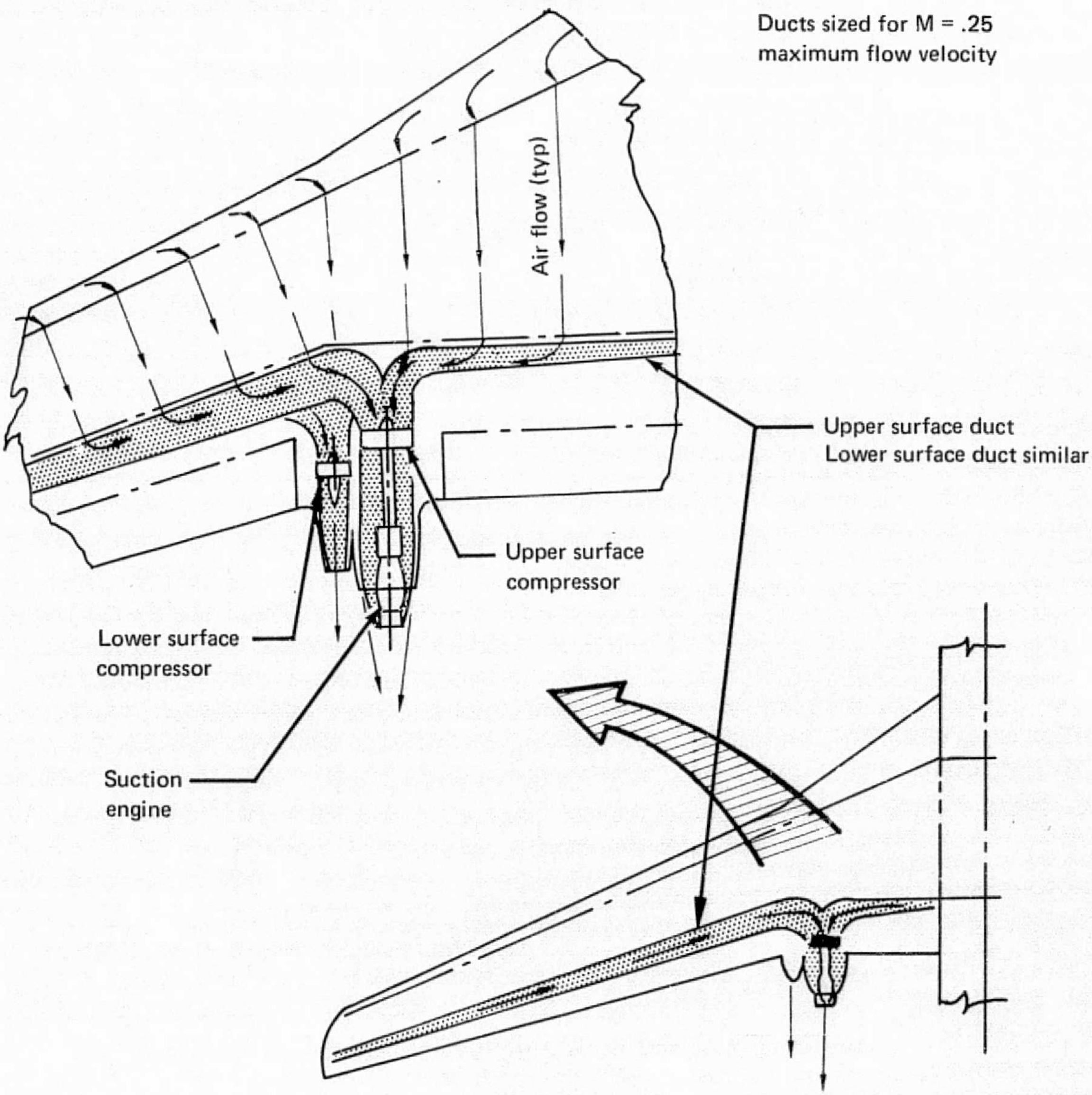
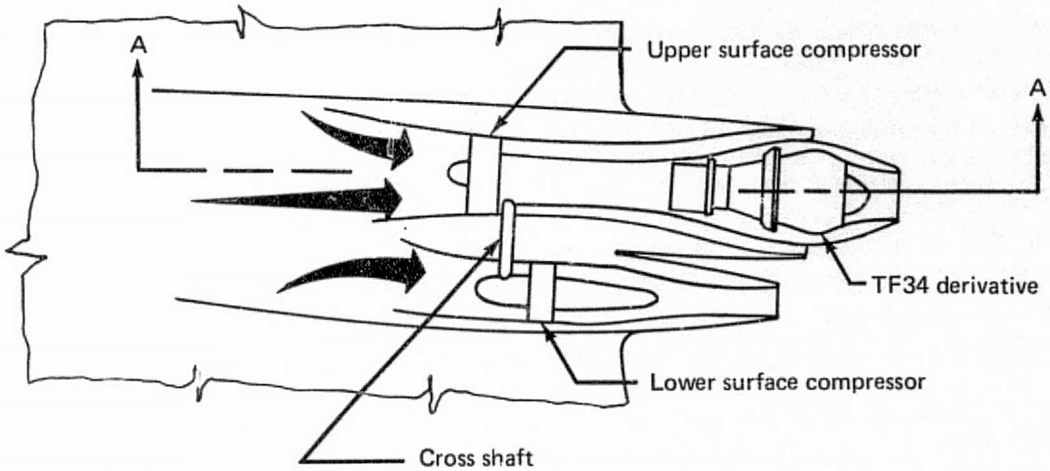
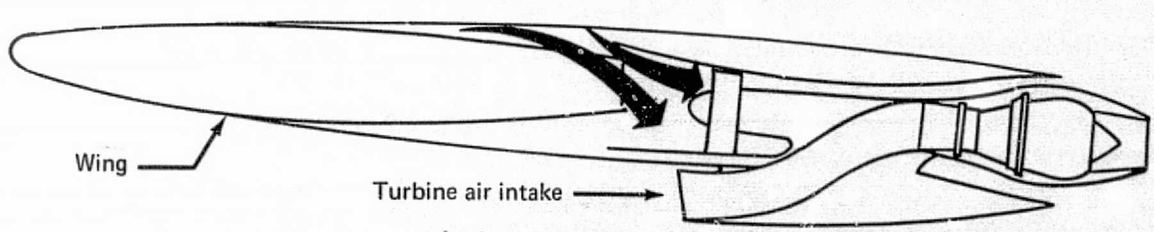


Figure 10. Suction System Concept II
Main Duct and Compressor Arrangement



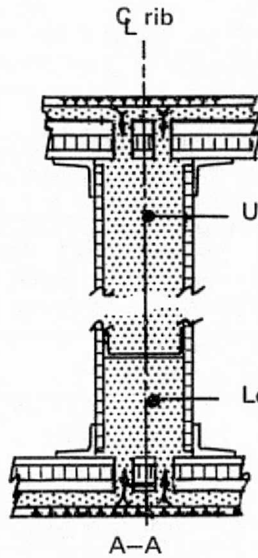
Plan view of installation

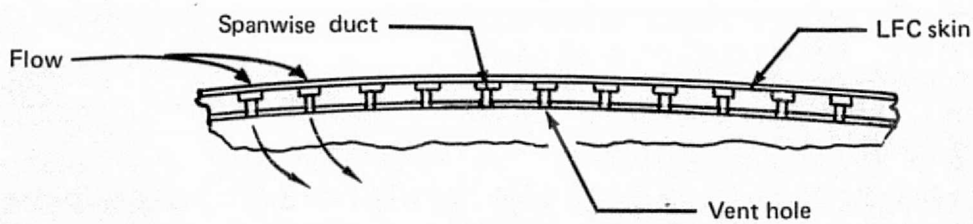


A–A

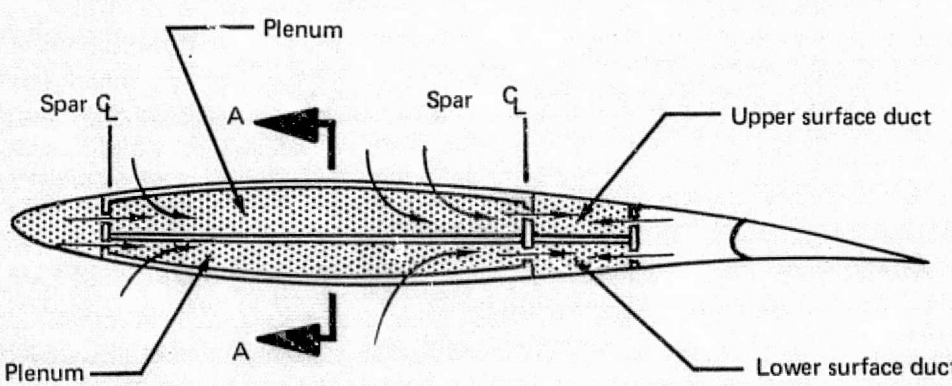
Cross section of upper surface compressor

Figure 11. Suction System Concept II Suction Engine and Compressors





Typical inspar panel
(at plenum [rib])



Section thru rib

Main ducts sized for
M = .25 duct velocity

Figure 12. Suction System Concept II
Rib and Duct Detail

(Aft upper surface outboard panel duct shown)

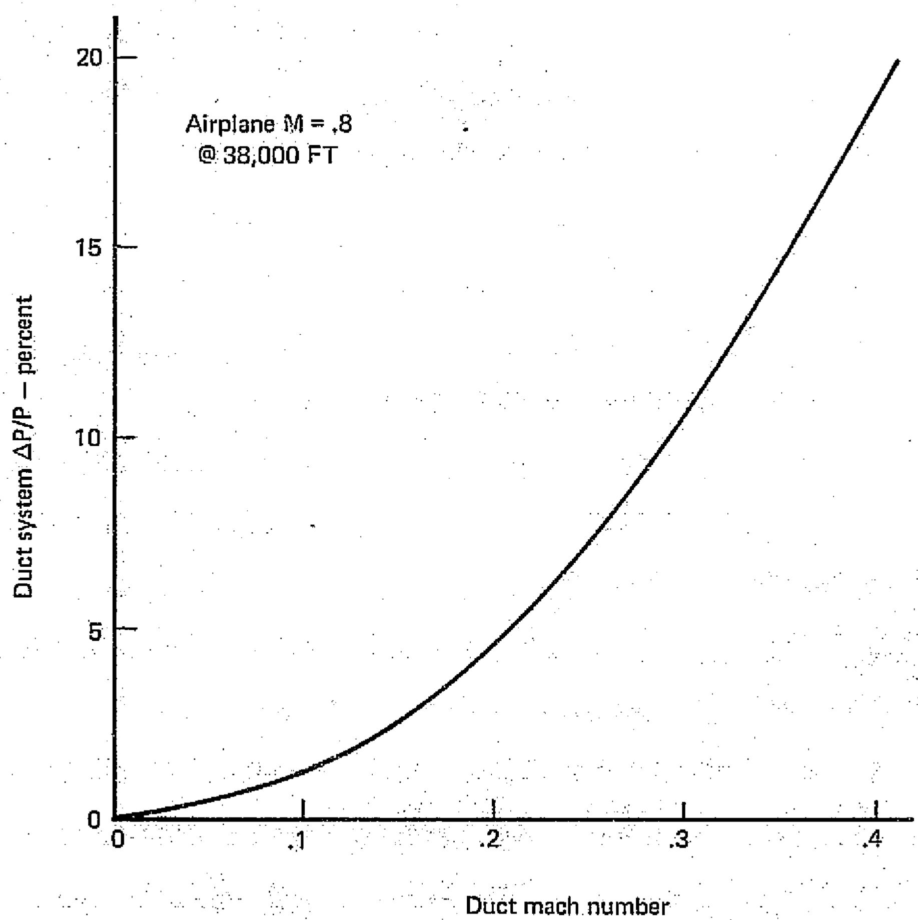
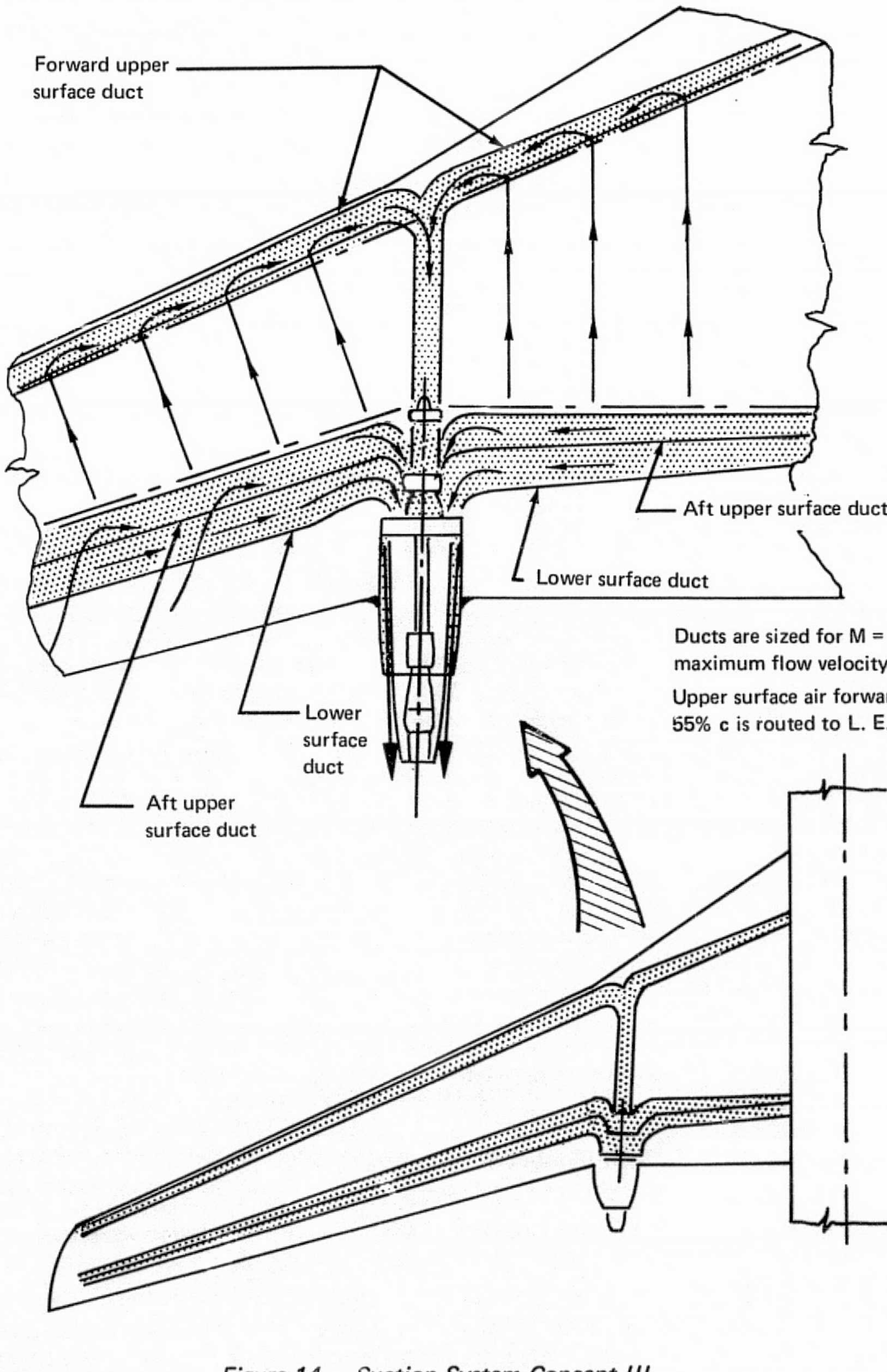


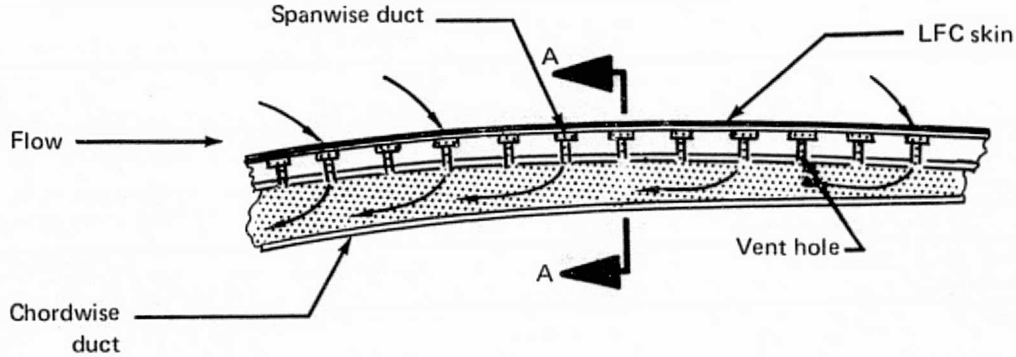
Figure 13. Duct Loss Characteristic



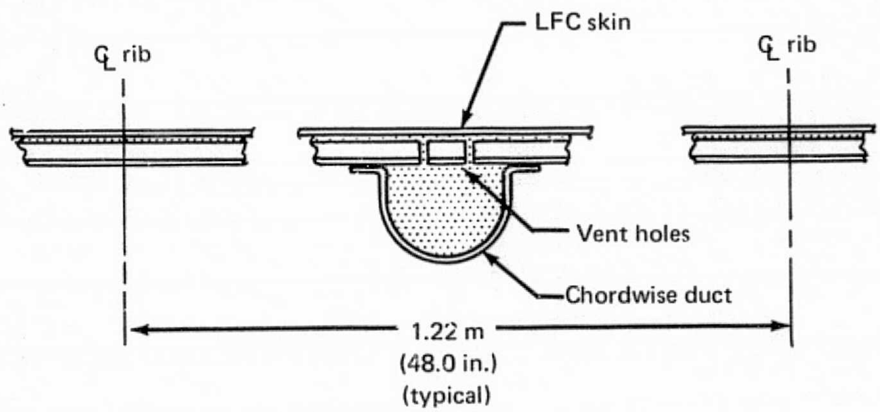
Ducts are sized for M = .20 maximum flow velocity

Upper surface air forward of 55% c is routed to L. E. duct

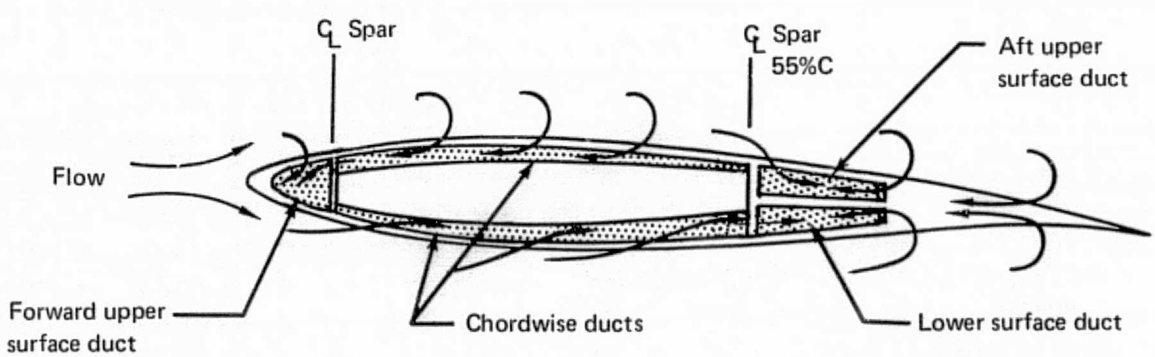
Figure 14. Suction System Concept III
Main Duct and Compressor Arrangement



Typical inspar panel at duct



A–A



Section thru wing

Figure 15. Suction System Concept III Chordwise Duct Details

(Engine arrangement)

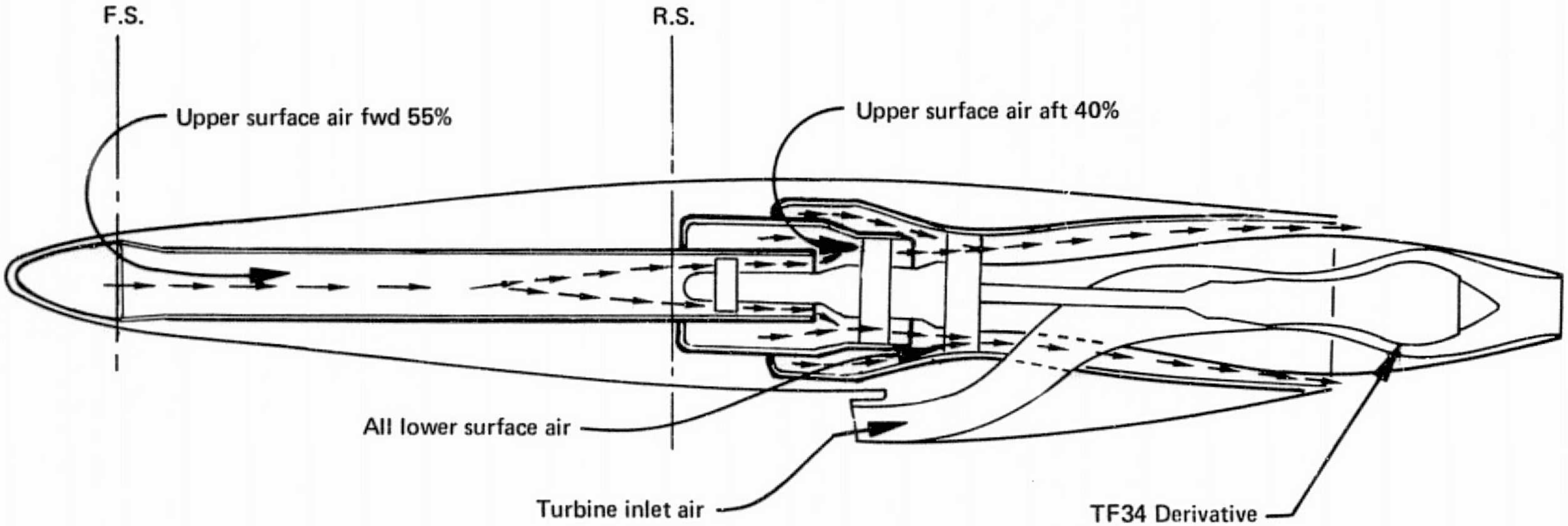


Figure 16. Suction System Concept III
Suction Engine and Compressors

Three classes of LFC suction surfaces; slotted, perforated, and porous, have been recognized in previous investigations, and were considered in this program.

5.2.1 SLOTTED SURFACE

Proven criteria for the design of slotted LFC surfaces, available since the completion of the X-21A program (Ref. 1) were updated to reflect subsequent study by the program LFC specialist, and were used in the computer program "LFC" to determine suction system parameters.

Several methods of generating suitable slots are available, utilizing existing technology. These methods, discussed in detail in section 5.4, include slitting or sawing, laser beam cutting, slitting with a high pressure water jet, and chem-milling. Insertion of pre-finished slot-plenum assemblies is a demonstrated possibility. Slots may be inspected visually with various optical aids, and such devices as leaf gauges.

The uninterrupted, closely-spaced slots render the slotted material ineffective in resisting wing torsion; however, it may be effectively used to resist bending.

The moderate slot flow velocity (not greater than 30.48 m/sec) indicates that slot erosion should not present a service problem. Impact erosion on leading edges must be considered. Relative to most airborne particles, slot width is large indicating that slot blockage due to such particles should not be an operational problem. X-21A flight tests revealed no slot erosion or contamination problems.

The slotted surface consists of a number of slots .10 mm (.004 in.) to .20 mm (.008 in.) in width, oriented approximately parallel to the wing spars, and with slot-to-slot spacing ranging from 12.7 mm (0.5 in.) to 203 mm (8.0 in.). These slots, each of which lies over a suction plenum, can be produced with present day manufacturing methods. Such a surface has been demonstrated, with positive results, under conditions representative of the mission defined for this program. Since such a surface could be immediately applied to an LFC wing, with minimum technical and operational risk (as compared to porous or perforated surfaces), it has been regarded as the baseline suction surface for this study (figure 18).

5.2.2 PERFORATED SURFACE

An ideal perforated surface would consist of an array of holes with diameters on the order of .13 mm (.005 in.) or smaller, with hole-to-hole spacing approximately 10 diameters. Problems of attaching such a surface to the underlying structure and of serving it with suitable plenums require some compromise with this ideal arrangement. A more practical arrangement consists of bands of perforations interrupted by unperforated zones to which the underlying structure is attached, preferably by adhesive bonding. Such a surface is illustrated in figures 17 and 19.

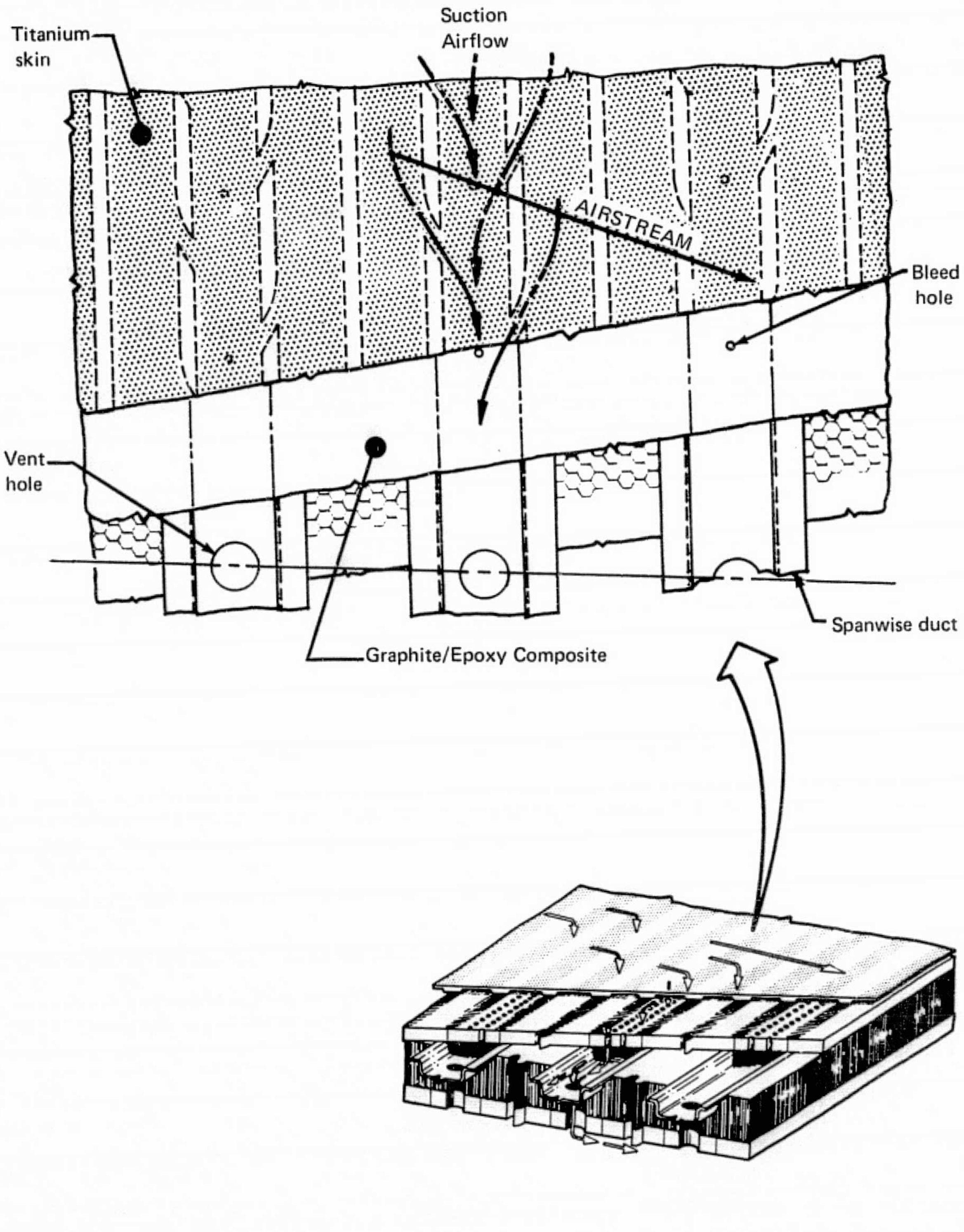


Figure 17. Perforated Skin Concept

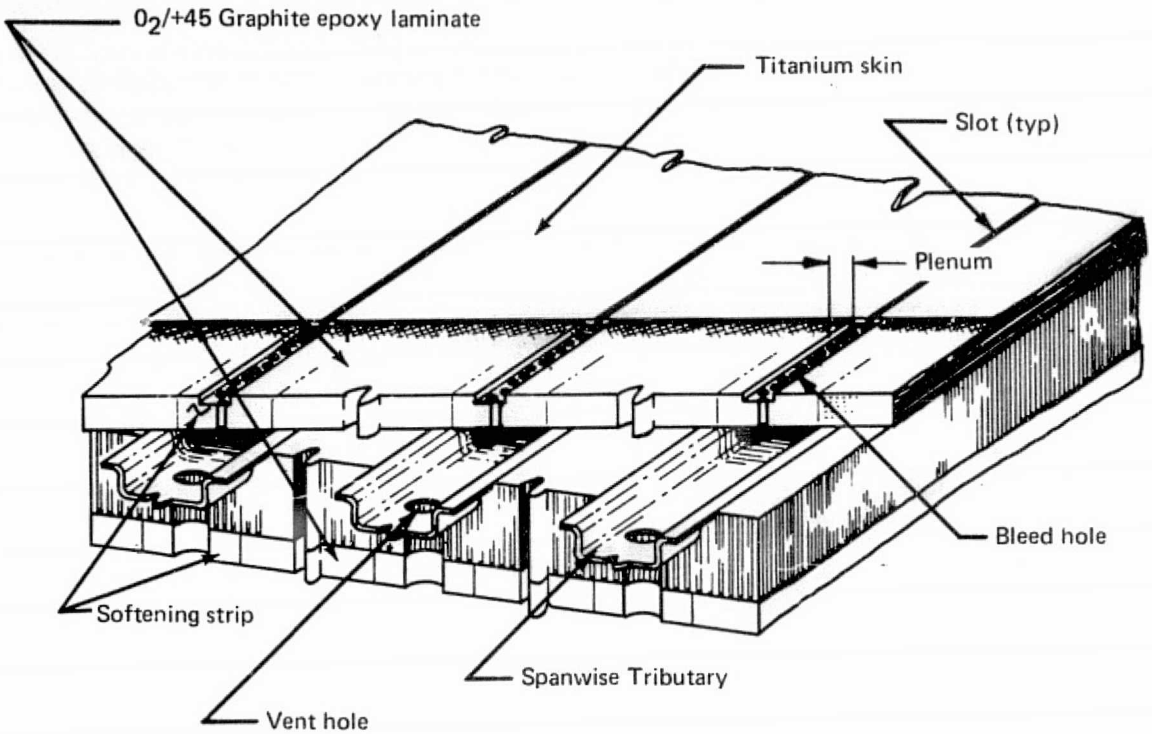


Figure 18. Skin Panel Concept

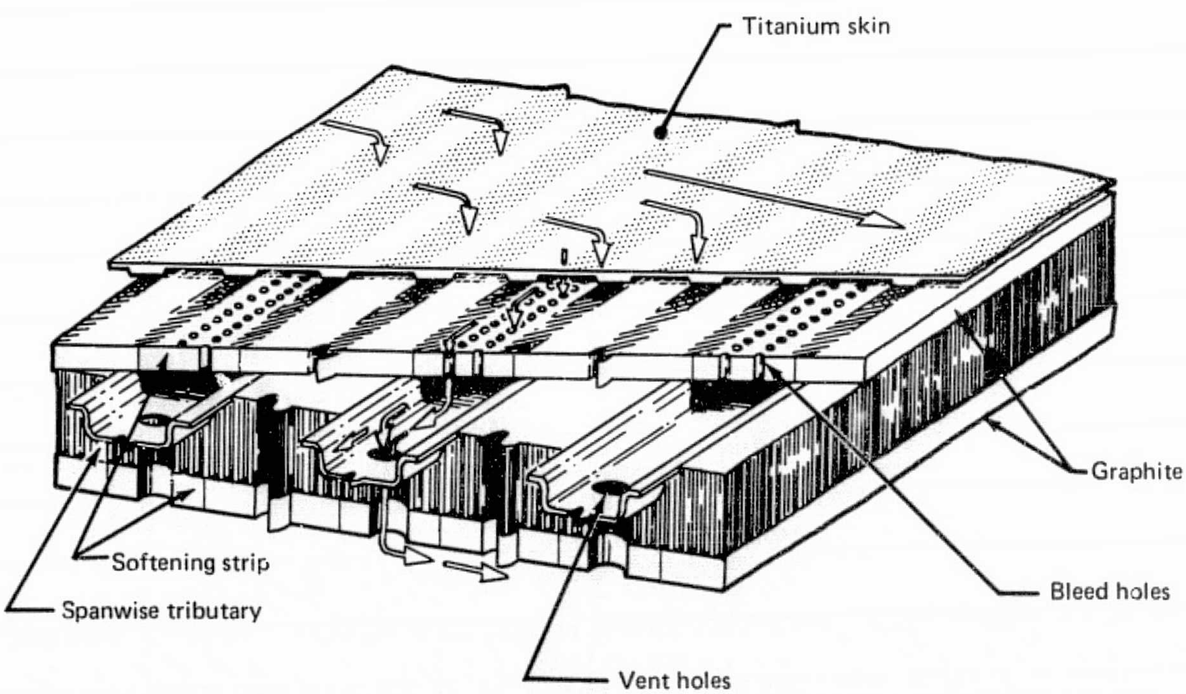


Figure 19. Perforated Skin Panel Concept

The aerodynamic suitability of a perforated surface has been demonstrated by wind tunnel test and theoretical studies (Ref. 2). The surface material can be made to resist both wing torsion and wing bending, but the presence of an array of open holes over the whole surface, acting as stress risers, implies a relatively low allowable stress level.

An electron beam piercing process, capable of generating suitable perforations in metallic surfaces, is currently being demonstrated on a small scale. If large vacuum chambers are available, there is no apparent technological barrier to large size application of this process. Susceptibility to clogging in an operational environment has not been tested and must be recognized as a potential problem.

5.2.3 POROUS SURFACE

The aerodynamic suitability of porous LFC surfaces was verified by early wind tunnel experiments (Ref. 3). Sintered metal surfaces were used for these experiments. A number of porous materials are currently being produced for use as filters, screens, acoustic liners, and for other applications. Susceptibility to clogging and methods of cleaning clogged porous surfaces in an LFC airplane environment have not been tested. Little information on the strength and fatigue properties of porous materials is available, and no porous material with a suitable combination of properties was identified.

5.2.4 SURFACE MATERIALS

Each of the three classes of LFC suction surface may be made of metal or non-metallic materials. However, electrodynamic studies of composite structures indicate that inclusion of a conductive metallic layer is necessary to provide a path for static discharge and lightning strike currents, and to prevent induced current surges in wiring contained within the structure. The metallic layer may be applied by flame-spraying or plating, or by including a metal sheet or screen in the laminate lay-up. To provide surface protection from ozone, ultraviolet radiation and other environmental components, it will be most efficient to apply the required metal layer to the outer surface.

5.2.5 SUCTION SURFACE STRUCTURAL ALTERNATIVES

There are two basic concepts for the incorporation of the LFC suction surface with the wing structure. In the glove concept, the LFC surface is attached to the outside of the wing structure, and is not part of the primary load carrying material. In the opposing concept the suction surface is an integral part of the load-carrying skin panel. Features of the two concepts are discussed below.

In the glove concept the LFC surface construction may be chosen for aerodynamic properties, minimum weight, minimum cost or other similar considerations. Fatigue failure in the glove requires repair or replacement of the surface, but does not affect the integrity of the primary structure, so fail safe design features are not required.

Since the LFC glove must be designed to be removable for inspection, repair, renovation, and replacement, removable fasteners must be incorporated in the skin surface. Each such fastener is a possible source of turbulence. When fasteners are reinstalled each fastener head must be refaired to the surface to prevent turbulence. The glove must maintain aerodynamic smoothness and contour under cruise conditions to preclude loss of laminar flow. The glove material must be compatible with the strain induced by deflections of the primary structure or, the glove must be fitted with some form of chordwise slip joint for strain relief. Each such joint becomes a source of boundary layer transition, causing a turbulent wedge to spread from the forward edge of the joint. In addition the joints represent discontinuities in the spanwise duct system.

The suction glove structure occupies the position farthest from the wing neutral axis, yet does not contribute to wing bending strength. In order to furnish the required moment of inertia, the primary skin/stiffener panel must have greater cross sectional area, hence greater weight, than if it were at the wing outer surface. The suction glove, since it is not contributing to wing strength, must be lightly constructed. It is difficult to reconcile light construction with the necessary rigidity, durability and resistance to damage.

If LFC is to be commercially accepted, a minimum maintenance suction surface must be developed. By virtue of its light construction the suction glove will require considerable maintenance, therefore the fact that it is removable for such maintenance is not seen as a particular virtue.

If the suction surface is integral with the primary load-carrying structure, strain compatibility will be assured by proper selection of material combinations. The substantial thickness of the primary skin panels will minimize distortion of the aerodynamic contours and resist impact damage. The primary structural material will be disposed at the greatest possible distance from the wing neutral axis, for structural efficiency. The suction surface material will be structural weight, rather than parasite weight, therefore, the requirements for weight efficiency, and surface quality and durability will not be in conflict.

5.2.6 SUCTION SURFACE RECOMMENDATIONS

The integrated suction surface concept was felt to offer the greater potential, when combined with a composite primary structure. Development of this concept will also reveal structural questions not exposed by development of the glove concept. Accordingly, the integral concept was recommended for the preliminary wing design phase of the contract.

No porous material having suitable smoothness, uniform controlled porosity, weight, and physical properties could be identified. Therefore, the slotted and perforated suction skins were recommended for the preliminary wing design.

Since inclusion of metal is required for electrical protection, and since either slotted or perforated metal suction surfaces can be produced with currently available methods, use of a metal outer surface was recommended. For thermal and chemical compatibility with graphite/epoxy composites and, for environmental protection, titanium was the metal recommended.

It should be noted that for the recommended structural skin concept, either slotted or perforated outer skin may be incorporated with only minor effect on the structural skin (figure 18 and figure 19). The structural concept is also compatible with use of a porous skin, and with substitution of non-metallic outer skins of any of the three classes, if suitable skin materials are identified.

5.3 WING BOX

In order to maximize the weight advantage of advanced structural composites and to minimize the weight penalty associated with incorporation of LFC features it is necessary to integrate the LFC features with the basic structural box. Therefore, the wing box study was conducted in parallel with the suction surface and ducting studies and modified as those studies progressed.

For study purposes, the wing box was broken down into skin panels, ribs, spars, wing structural splice, and wing/body attachment.

5.3.1 WING PANELS

Two types of skin panels were considered; a skin and stiffener panel analogous to the X-21A construction, and a thick honeycomb panel with integrated suction ducting.

The skin and stiffener panel concepts, of which figure 20 is typical, were found to have a number of disadvantages which, in the end, ruled the concepts out of further consideration. These disadvantages included the following:

- The outer skin panel is essentially non-structural, hence should be as light as possible, yet it must maintain proper contour when the wing deflects.
- Both the outer skin panel and the underlying structural panel would have critical contour requirements and would have to be laid-up and cured in separate large tools.
- Upon final panel assembly, some adjustment; shimming, etc., would still be necessary, and inspection of the assembled skin panel would require use of a side-looking borescope or similar device to assure the integrity of each duct/stiffener area.

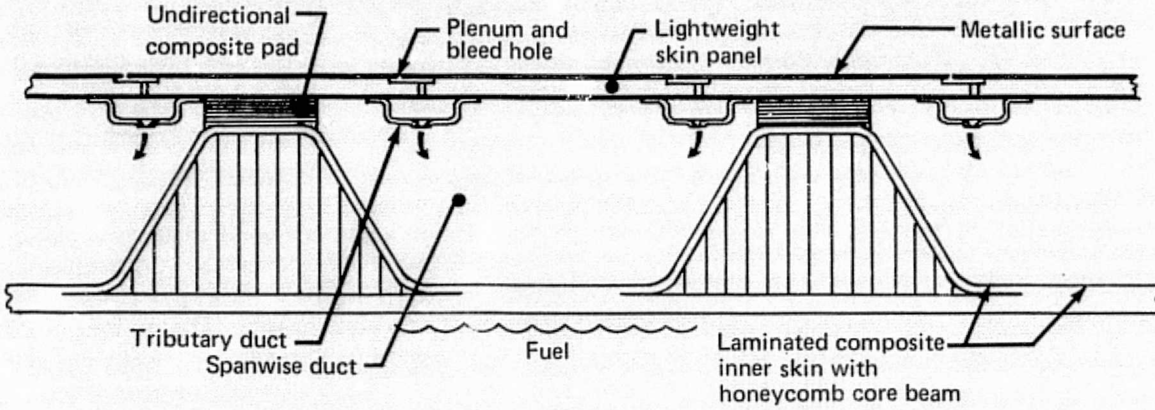


Figure 20. Skin and Stiffener Panel Concept

Layup of the stiffened structural panel would be complex, and not easily adapted to auto-mated tape laying machinery.

Several cycles of layup and curing would be required, with attendant requirements for balanced distribution of fibers in each sub-assembly, to prevent fabrication thermal stress and distortion.

Finally, an analysis of spanwise duct cross section requirements led to the conclusion that the duct depth required for long spanwise runs would place the inner torsion skin inefficiently near the neutral axis of the wing box, requiring an additional weight of torsional material and reducing the fuel volume available.

The thick honeycomb sandwich skin panel with integrated suction ducting, figures 18 and 21, may be compared with the skin stiffener panel as follows:

- Both faces of the honeycomb sandwich panel are used effectively to resist bending and torsion. Since the entire panel must strain as a unit, distortion of the aerodynamic contour due to wing deflection will be minimized.
- The outer face sheet of the sandwich will be laid up and cured in the same contoured tool used for the subsequent layup and cure of the entire sandwich panel assembly, eliminating contour coordination problems.
- Panel final assembly will have been accomplished when the core and inner face sheet are cured. Bonding pressure will eliminate gaps and voids in the bond line. Inspection will be by conventional methods now used on bonded honeycomb structure.
- Face sheet layup is suited to use of automated tape laying machinery. The core is continuous, with spanwise features which can readily be generated by numerically controlled machines.
- No more than two layup/cure cycles would be required to assemble the complete structural panel. Balanced fiber distribution can easily be obtained.
- The shallow tributary ducts are easily contained within the core depth required for panel stability.
- Rib shear and tension ties can be made without the interference of stiffeners.

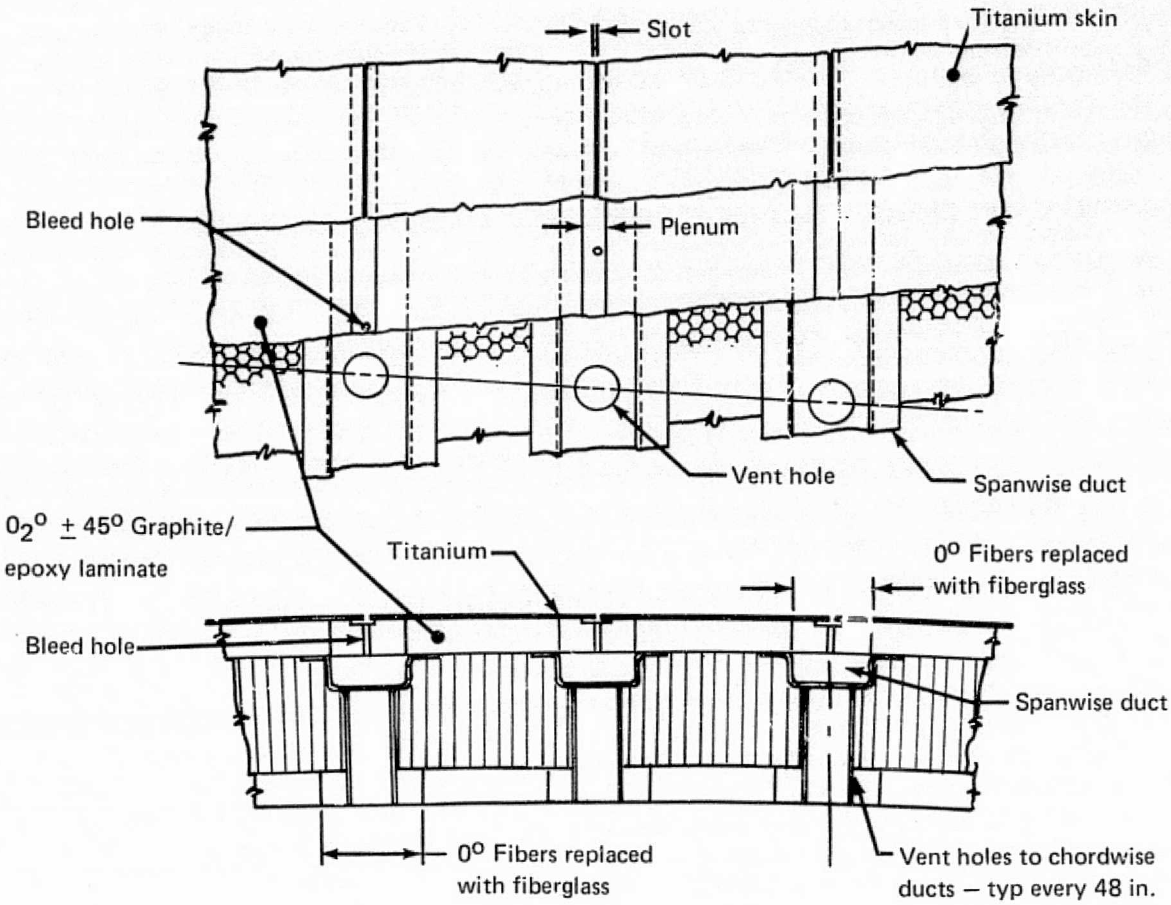


Figure 21. Honeycomb Skin Panel Concept

5.3.2 RIBS

In order to use composites to best advantage in the wing ribs, a simple honeycomb web with pultruded caps (Sect. 5.4.2) was adopted. Such a web can be fabricated on simple flat plate tooling, and edge closure and reinforcing insert design methods and fabrication techniques are well understood. The pultruded caps can be located and bonded to the skin panel assemblies during the fabrication of those assemblies, with rib chord to web assembly made by mechanical fasteners during wing box assembly, (figure 22).

5.3.3 SPAR WEBS

With the wing bending material distributed across the skin panels, the spars are reduced to shear webs which also serve to close the torsional box. For simplicity of fabrication, and to use the high strength composite to best advantage, a simple honeycomb spar web concept was chosen. Honeycomb depth is dictated by panel stability and fuel slosh requirements. Spar panel aspect ratios are controlled by rib spacing, which would be selected to optimize skin panel weight. The spar webs are mechanically fastened to skin panel flanges during wing final assembly, (figure 23).

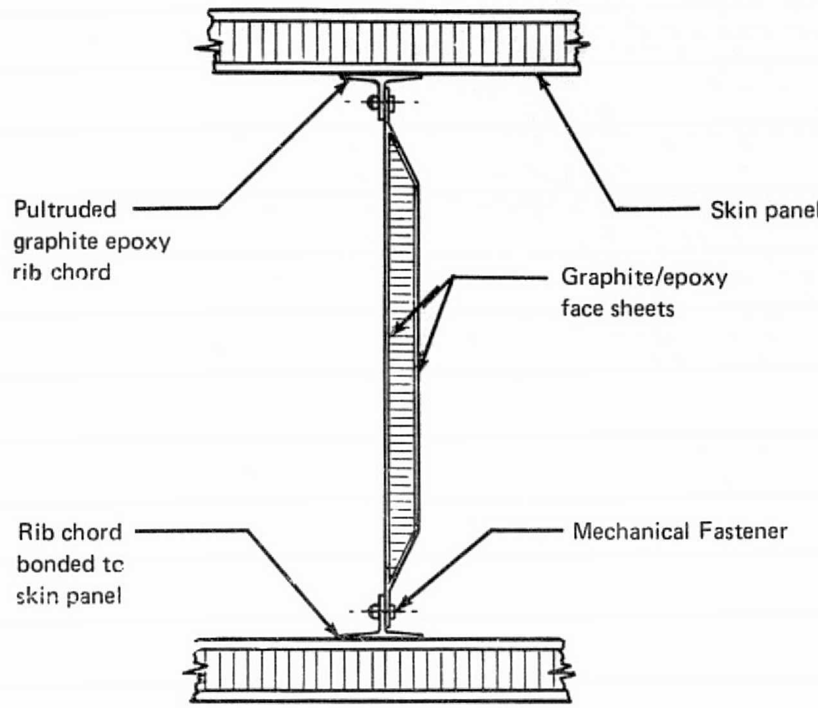


Figure 22. Rib to Skin Attachment

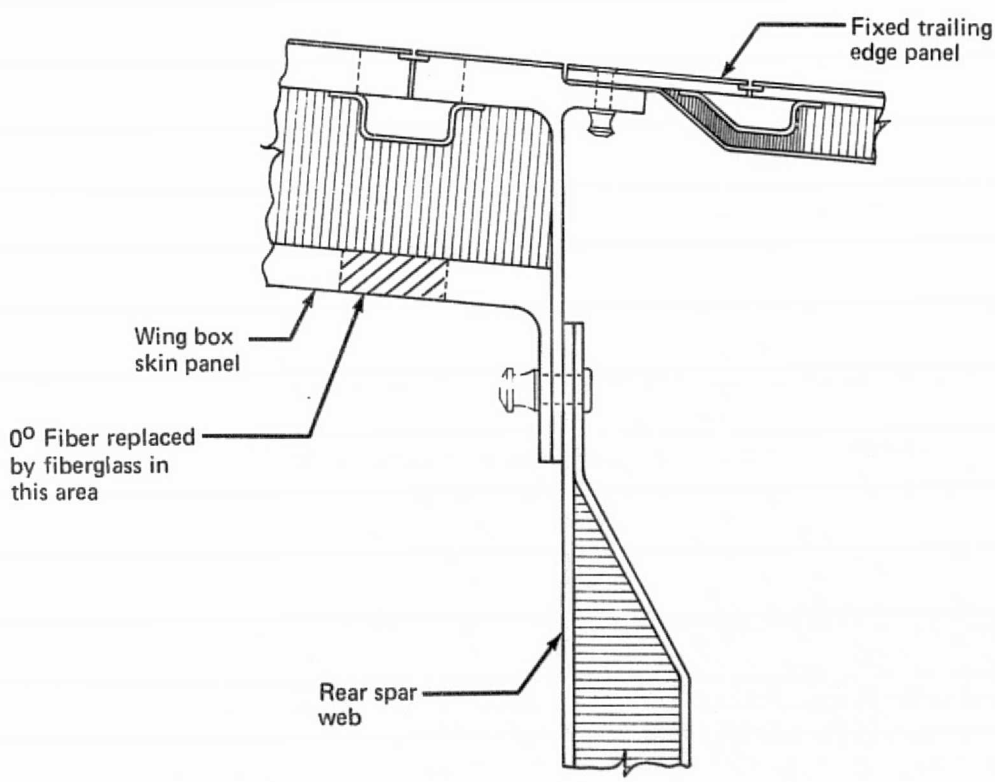


Figure 23. Spar Web to Skin Panel Attachment

5.3.4 WING SPLICE LOCATION

Boeing's conventional transport wings are spliced at the side of body (S.O.B.) rib, where spars and stringers change direction. Other manufacturers have located the wing production splice on the aircraft centerline. Both locations were considered for the composite LFC wing. Choice of a side of body joint results in two major structural splices and three major assemblies; the center wing box, and the left and right wing panels. A single centerline splice results in two major assemblies; the left and right wing panels. With either concept a robust side of body rib is required to react wing-body joint loads, and to serve as a tank end rib. Side of body joint concepts are shown by figures 24 and 25. The wing centerline splice concept is illustrated by figures 26 and 27.

One of the principal reasons for choosing a side of body joint on a swept wing of conventional construction is the necessity of splicing wing stiffeners at the side of body, where the stiffeners change direction. Since the selected composite skin panels have no separate stiffeners, and since skin panel contour changes may easily be incorporated in the major skin assembly fixtures, the practicality of a centerline splice appears to be enhanced. In order to evaluate possible reductions of complexity and non-optimum (joint) weight, both concepts were retained for further study.

5.4 MANUFACTURING TECHNOLOGY

The manufacturing technology effort on this study was expended in two principal areas. The first was the identification of at least one method which has commercial potential for manufacturing each class of suction surface (slotted, perforated, and porous) in the 1985 time period assumed for production of the LFC airplane. Some exploratory tests were performed to assess processes thought to be promising. The second area of study was identification of production techniques currently being developed for large composite structures which would be used for the LFC wing.

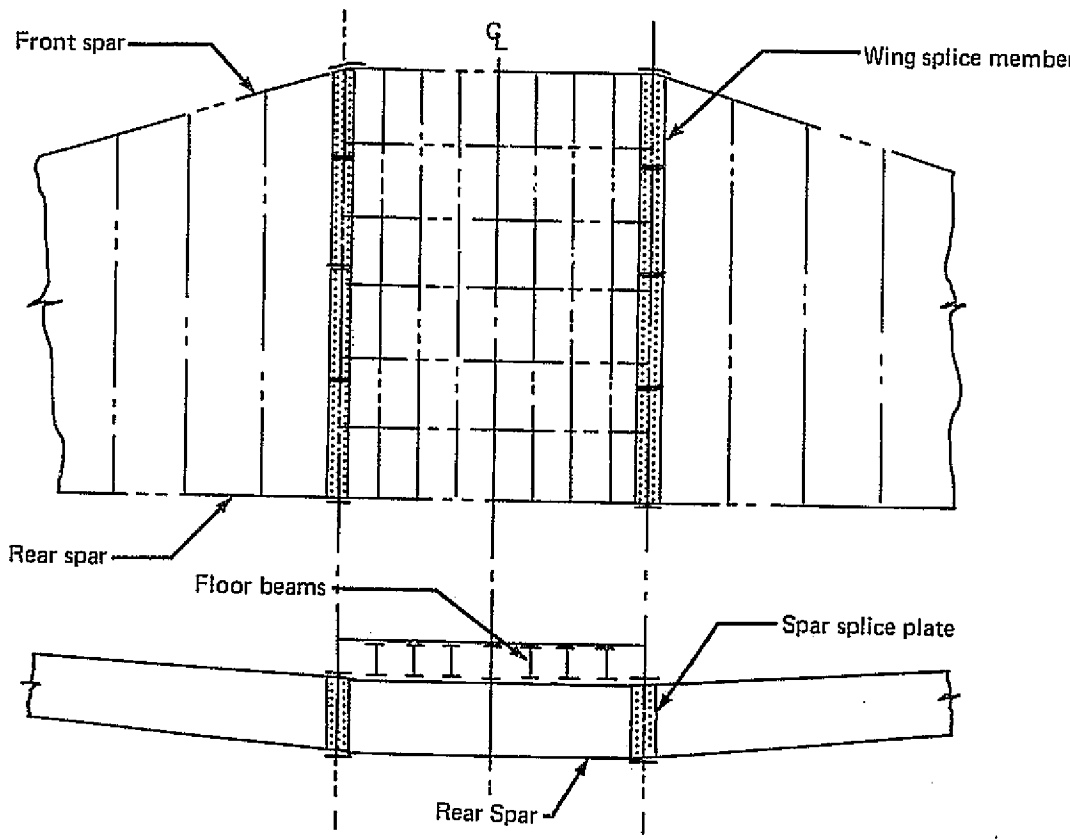


Figure 24. Side of Body Splice Concept

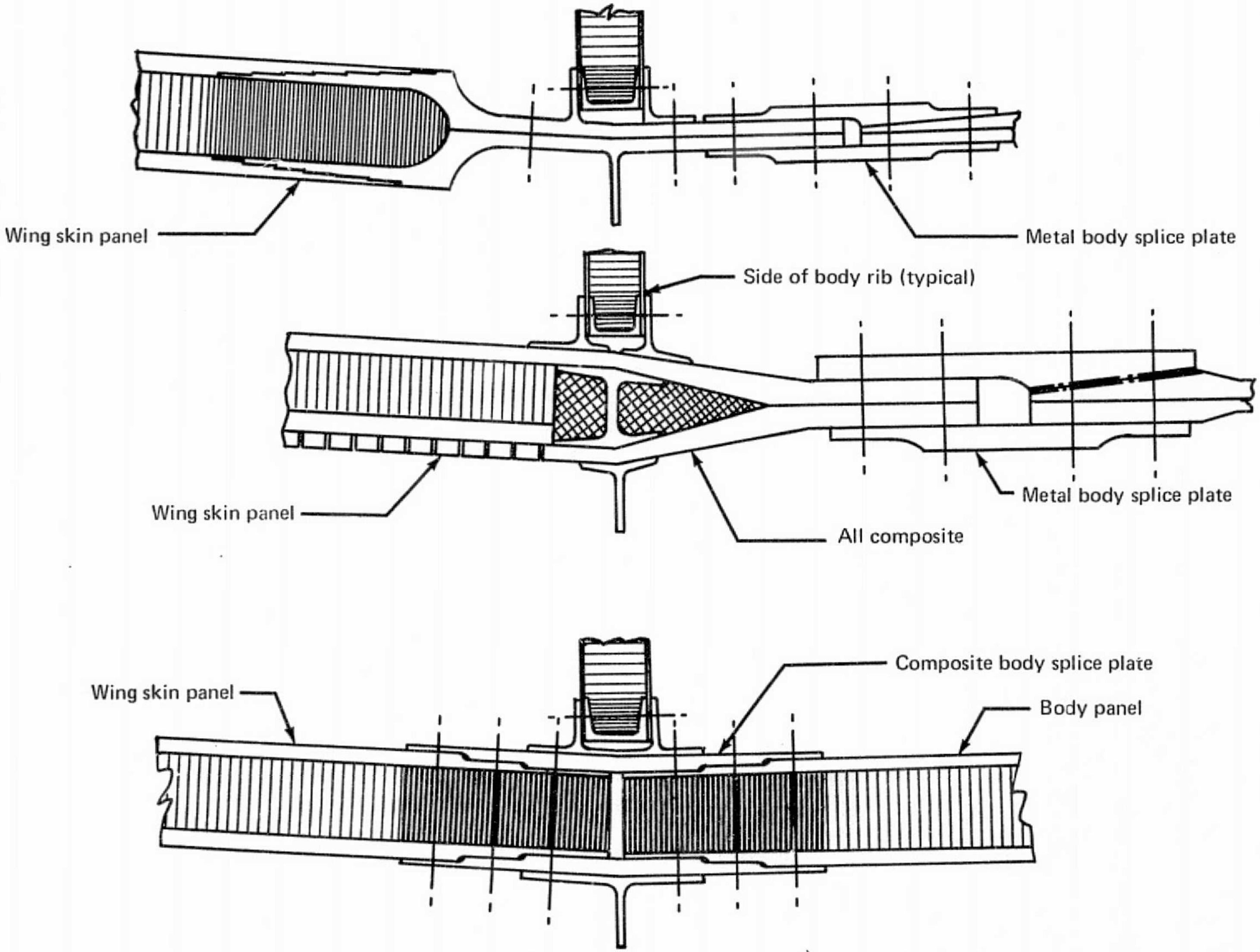


Figure 25. Wing Joint Concept – Side of Body

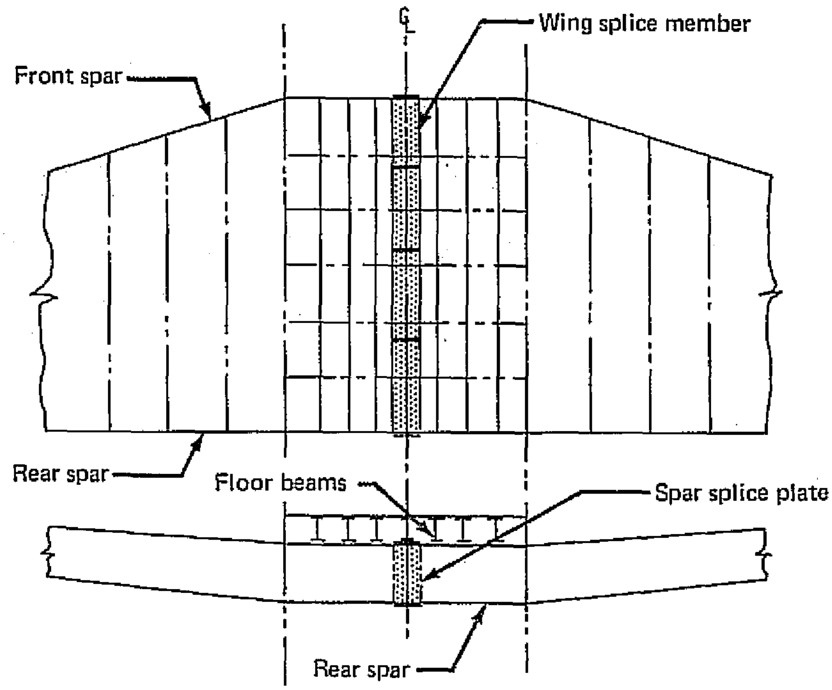


Figure 26. Center Line Splice Concept

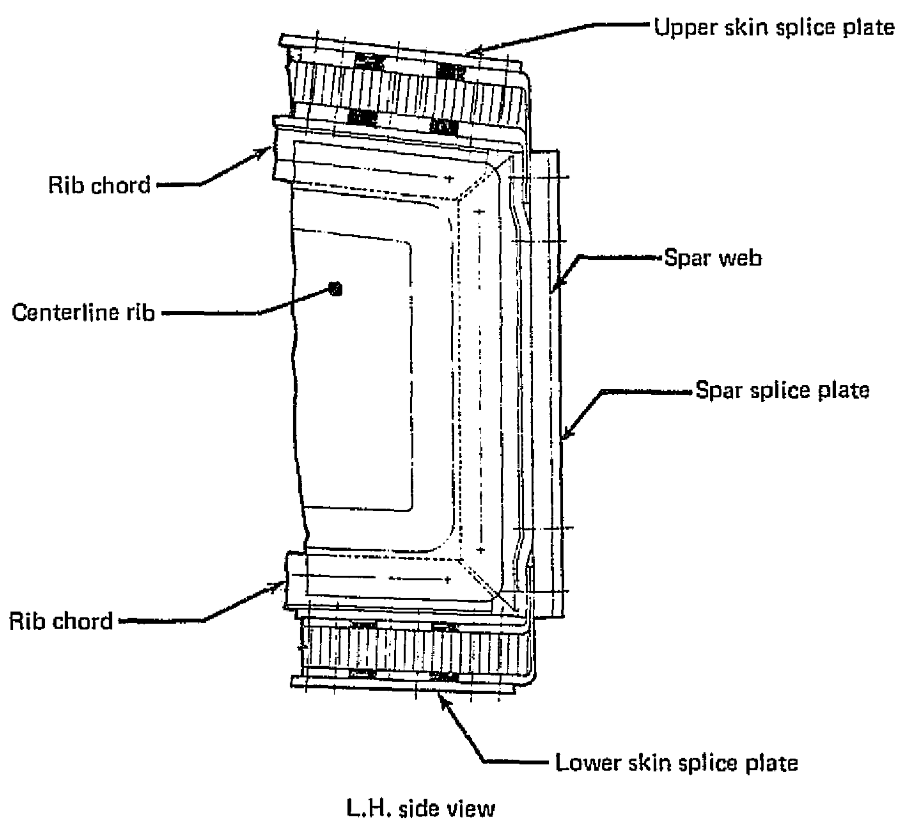


Figure 27. Center Line Splice Detail at Rear Spar

5.4.1 SUCTION SURFACE TECHNOLOGY

As the suction surface study (section 5.2) progressed, it became apparent that no suitable porous surface is presently available. Therefore the manufacturing study effort was focused on slotted and perforated surfaces. Table 2 lists processes considered to be candidates, and the more promising of these candidates which were investigated in some detail. In view of the apparent necessity to incorporate a conductive layer in the laminate, and the environmental protection obtainable by having such a layer on the outer surface, major emphasis was placed on processes suitable for slotting or perforating such a metallic layer. The metal was assumed to be a titanium alloy, since titanium is most nearly thermally and chemically compatible with a graphite/epoxy composite.

Table 2. Manufacturing Technology — Suction Surface Studies

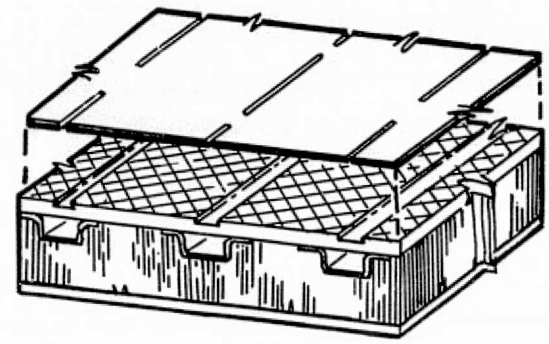
Candidate Processes	Selected for investigation	Verified Production Potential
Slots		
Sawing	√	√
Slitting	√	√
Electron beam cutting		
Laser cutting	√	√
Electro-chemical machining		
Chem-milling	√	√
Electrical discharge machining		
Water jet cutting	√	√
Slot-plenum insert	√	√
Perforations		
Drilling		
Electron beam piercing	√	√
Chem-milling	√	
Electro-chemical machining		
Electrical discharge machining		
Removable cores		
Water jet piercing		
Mechanical piercing		
Porous Material		
Porous Polyimide matrix		
Powder Metallurgy		
Metallic mesh		

Four methods of assembling the suction surface to the underlying primary structure are defined by figure 28. Compatibility of each assembly method with specific processes for generation of suction surfaces is considered in the discussion which follows.

Suction slots may be generated prior to skin panel assembly (method 1), following skin panel assembly as on the X-21A (method 2), or the slot material may be prepared separately in tape form and installed in a recess in the structural skin (method 3). The fourth method, similar to method 3, calls for the use of a prefabricated slot and plenum which is bonded into a properly shaped recess in the structural skin. Prefabricated slots and plenums of proper dimensions and tolerance appear to be producible by the "collimated hole structure" process; a vendor's proprietary technique in which a material cross section is reduced while retaining geometric similarity to the original shape. The existence of this process was discovered late in the study and only small representative samples were obtained.

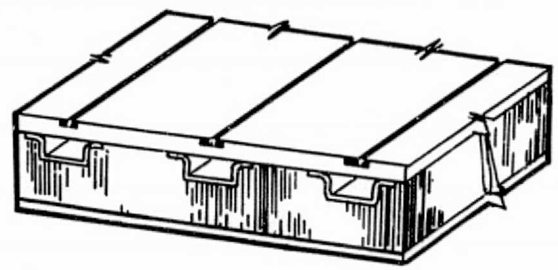
Production of surface slots by sawing was demonstrated on the X-21A program and no notable advance in the state of that art was identified. Small circular cutters, commonly known as jeweler's saws, are guided along the intended slot centerline. Slot width is a function of cutter width and runout. Production rate in titanium alloy is estimated to be 2.54 cm (1.0 in.) per minute. X-21A aluminum skins were slotted at six to seven times this rate. Machining chips in the plenums and hanging burrs on the slot edges were problems encountered on X-21A and should be anticipated in future use of the slot sawing technique. This technique is judged to be suitable only in conjunction with method 1, figure 28, which permits deburring prior to assembly.

Mechanical slitting is particularly suitable to the slot tape concept, method 3. With this technique, the slot material, prebonded to a backing strip, is drawn past a single point tool which cuts the slot. Cutting rate in titanium alloy may be as great as 30.5 m (100 ft) per minute if carbide tools are used. Chips and burrs may be easily removed from the finished cut. Slot width would be controlled by tool point dimensions.



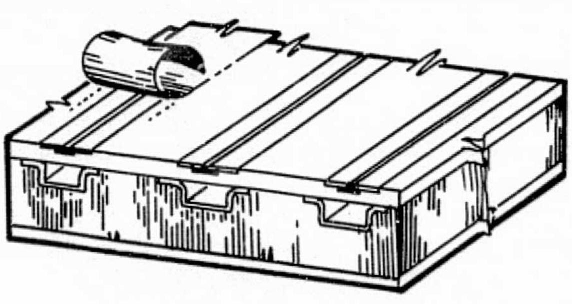
Method 1

Intermittent cuts permit handling - slots completed after bonding.



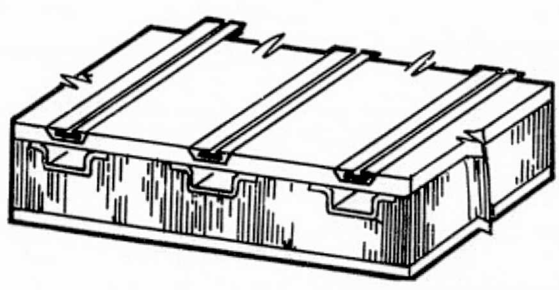
Method 2

Slots cut after bonding.



Method 3

Backing strip holds slot tape - removed after bonding.

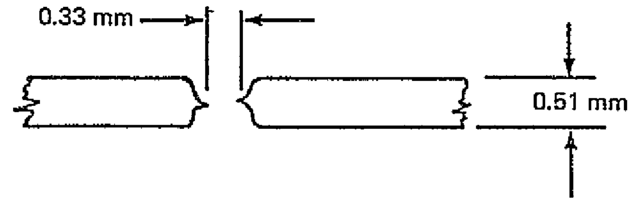


Method 4

Slot - plenum insert

Figure 28. Suction Slot Methods

Chem-milling, the removal of metal by etching exposed surfaces in a chemical bath, is particularly applicable to method 1, figure 28. Exploratory tests were performed on chemically pure (CP) titanium sheet 0.51 mm (0.020 in.) thick to assess the possibility of milling slots 0.20 mm (0.008 in.) in width. The minimum slot width attained by use of current chem-milling processes and techniques was 0.33 mm (0.013 in.). An attempt to produce a smaller slot by reducing the photo template line pattern (used to selectively remove photosensitive maskant) to 0.025 mm (0.001 in.) was not successful due to (1) traces of maskant remaining in the area to be etched following pattern development; (2) surface tension preventing free access of solution to the titanium surface to be etched; and (3) solid particle blockage in the slot area. The slots which were produced show the typical hourglass cross section associated with chemical milling of an orifice from both sides of a sheet, as shown in the sketch below. The hourglass shaped slot would not be suitable for the LFC surface, without some modification.

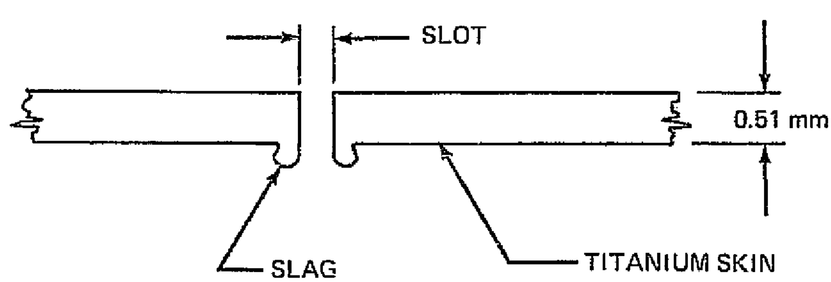


Typical Chem-milled Slot Shape

Several samples of CP titanium were slotted with a CO_2 laser, using the following operating parameters:

- Power – 1,100 watts.
- Effective focal length – 6.35 cm (2.5 in.)
- Speed – 635 cm/min (250 in./min.)
- Gas assist – argon, both sides.

Slot widths were satisfactory, however, slag or dross remained on the back side of the slot, as shown in the following sketch. Clean, slag-free slots were produced in an aluminum alloy test piece. In the opinion of the LFC specialist, a small, tightly adhering slag buildup, which does not reduce the slot width, might prove to be acceptable. If laser cutting is used with method 1, the slag is accessible to deburring operations and does not present a critical problem.



Laser Cut Slot

If laser cutting is used in association with method 2, some means of protecting the underlying surface from the laser beam becomes essential. The study did not reveal any suitable technique for defocusing the beam. One possible method of protecting the underlying surface is the use of a shielding strip which can be withdrawn from the plenum before the panel end is closed out. Figure 29 illustrates this technique.

High pressure water jets have been used in cutting and trimming composite laminates. An experimental set-up was available, so water jet slot cutting was attempted. A test specimen consisting of a graphite/epoxy skin laminate with a suction plenum and a bonded 0.51 mm (0.020) titanium skin was fabricated. A slot 0.254 mm (0.010 in.) wide was produced by a water jet operating at 4.137×10^8 N/m^2 (60 000 psi) pressure. The fluid was pure water, passing through an orifice 0.076 mm (0.003 in.) in diameter. The nozzle was positioned 7.62 mm (0.30 in.) above the surface of the titanium. The cut was made in two passes, at a speed of 2.54 cm (1.00 in.) per minute per pass. A titanium shielding strip was inserted in the plenum to protect the laminate. The shielding strip was grooved but not penetrated by the water jet. In this exploratory test, slot edges were not acceptably smooth, however, the process shows potential for successful development.

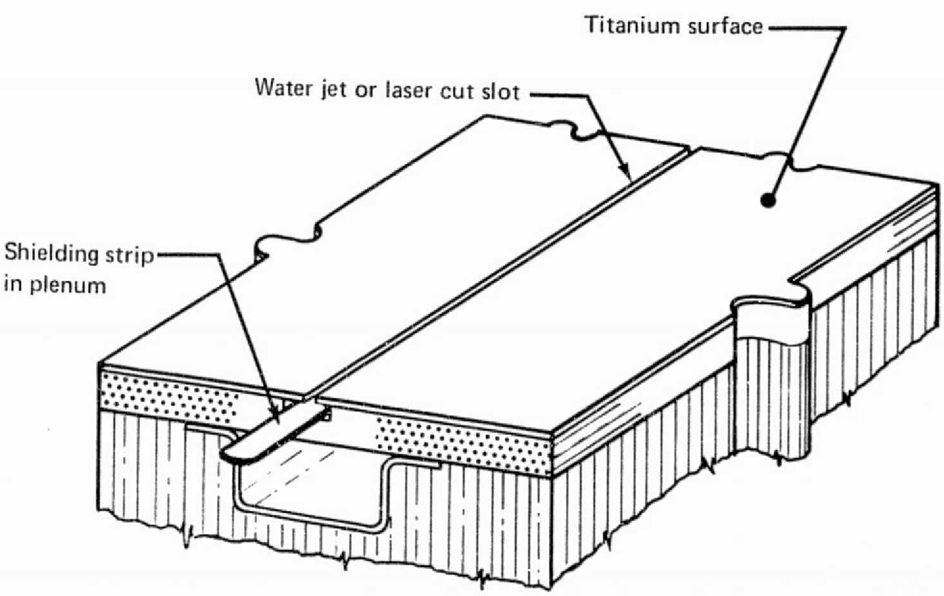


Figure 29. Shielding Strip in Plenum

A perforated surface might be generated after skin panel assembly (method 2). However, the one process for which production potential was identified requires the workpiece to be perforated in a vacuum, hence is most suitable for perforating suction surface material prior to assembly. This technique, the Steigerwald process, is a proprietary means of electron beam piercing and is currently being demonstrated by a U.S. licensee of its European developer. This licensee evaluated the typical perforated array shown in figure 30 and gave a price quotation for a test specimen 15.2 by 30.5 cm (6 by 12 in.) in 0.51 mm (0.020) thick commercially pure titanium. The sample was not procured; however, the price quotation gives assurance that material, thickness, and hole diameter requirements are compatible with the present state of development of this process. It should be noted that the electron beam process is capable of producing holes of such a size and spacing that the surface may be considered to be porous rather than perforated.

The "collimated hole structure" process previously referred to has the capability of producing excellent porous or perforated specimens for LFC test panels. The process consists of reducing a precisely shaped cross section, consisting of a base alloy and a filler, by drawing through reducing dies. The filler material is then etched away or otherwise removed. Cross sections sliced from the resulting drawn section may then be used as screens, nozzles, filters, etc., depending on the configuration selected. Unfortunately, for full scale LFC applications, the final cross section is limited to approximately a 10.16 cm (4.00 in.) diameter enclosing circle, and no feasible method of building up large skin panels was recognized.

5.4.2 COMPOSITE STRUCTURE TECHNOLOGY

Fabrication of high strength composite structures will be accomplished by extension of manufacturing techniques which have been developed for conventional fiberglass composite structures, and by new techniques which will evolve from continuing composite manufacturing effort. Necessary development of such items as large automated tape laying equipment, large bond assembly tooling, and low temperature curing resin systems is being accomplished in connection with other programs, and will not be discussed as part of this study.

One such process will be discussed, however, because it was assumed to be used for the Manufacturing Appraisal phase of the study, and because it offers substantial cost reduction potential for such structural members as rib chords, stiffeners, and attach members. This process is pultrusion.

Pultrusion is a low cost production process developed by Boeing (patents pending), which consists of pulling precisely cut composite preimpregnated tapes through a high energy (microwave) chamber containing a compaction/shaping die. Composite members may be pultruded in continuous lengths at speeds of 20.3 to 25.4 cm/min (8 to 10 in./min.). Fiber orientation of the tapes may be varied to suit the intended application, and non-metallic sandwich core material may be included. Cost projections based on current technology show pultruded graphite/epoxy structural shapes to have one third the fabrication cost of autoclave processed shapes. Process flow time is only 10% of autoclave process time. Pultruded sandwich fabrication costs in large scale production will be less than half that of autoclave processed sandwich. Methods for pultruding tapering sections are currently being refined. Curved members such as rib chords will be combined, shaped, compacted and partially cured by the pultrusion process, then curing will be completed in contoured dies. Figure 31 is a schematic representation of the pultrusion process developed by Boeing.

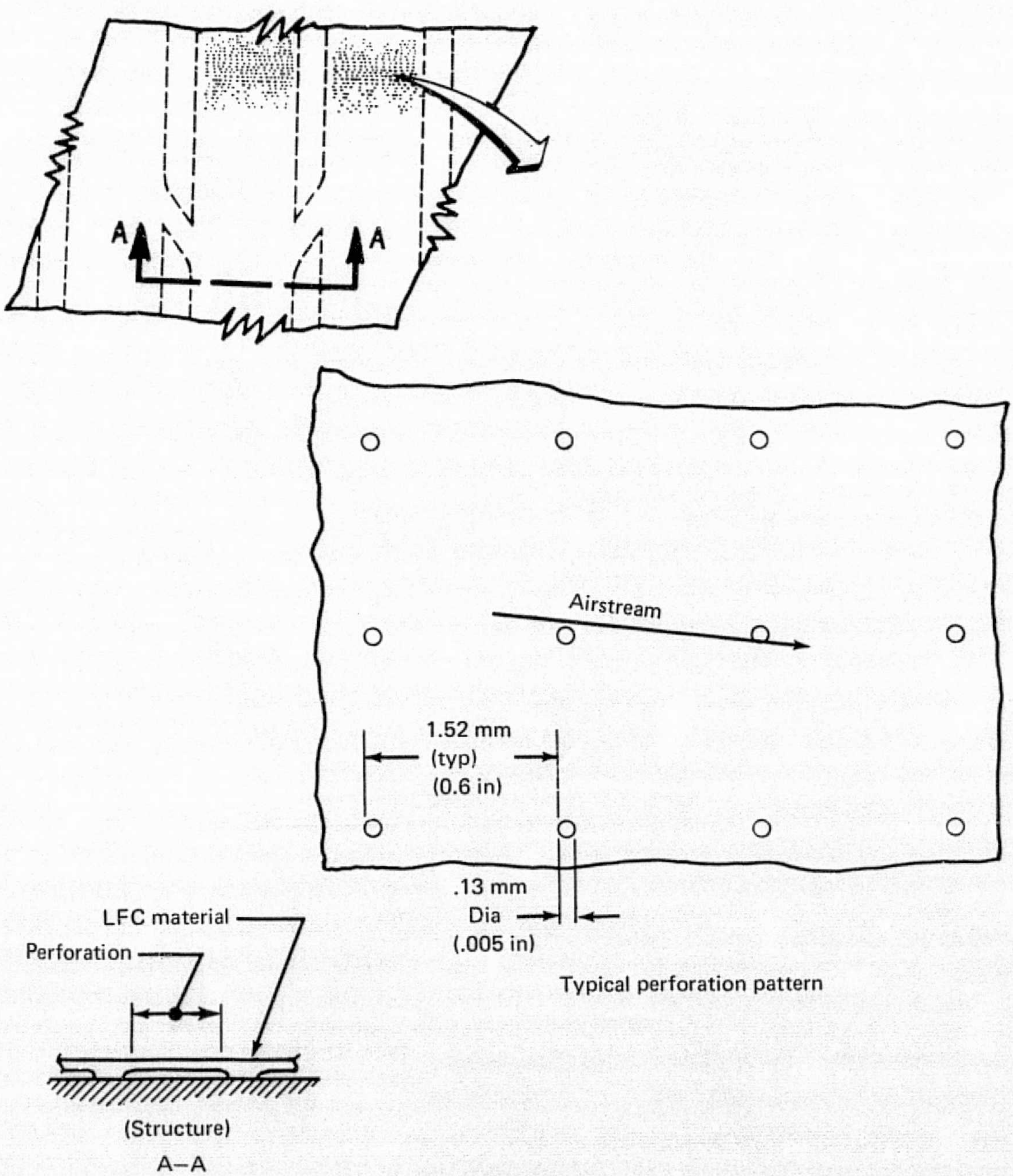


Figure 30. Suction Skin Perforations

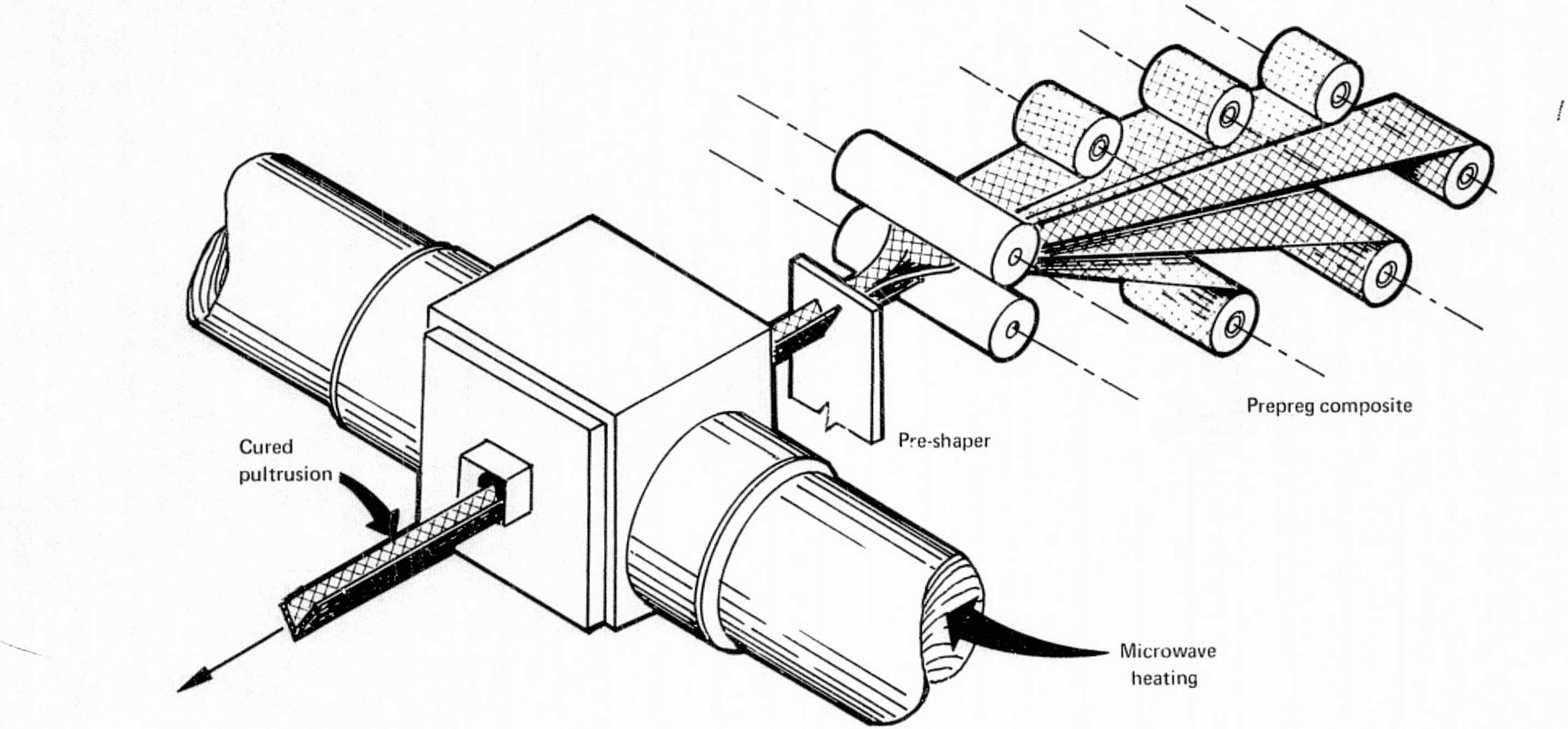


Figure 31. Pultrusion Process

Figure 32 illustrates how one of the configurations developed in this study would be manufactured. Pultruded splice plates would be held in contact with the wing skin panels during the curing process. Intimate fit up, without shims, is thus assured. When cured, the splice plates are removed for inspection, trimming, drilling, and other secondary operations.

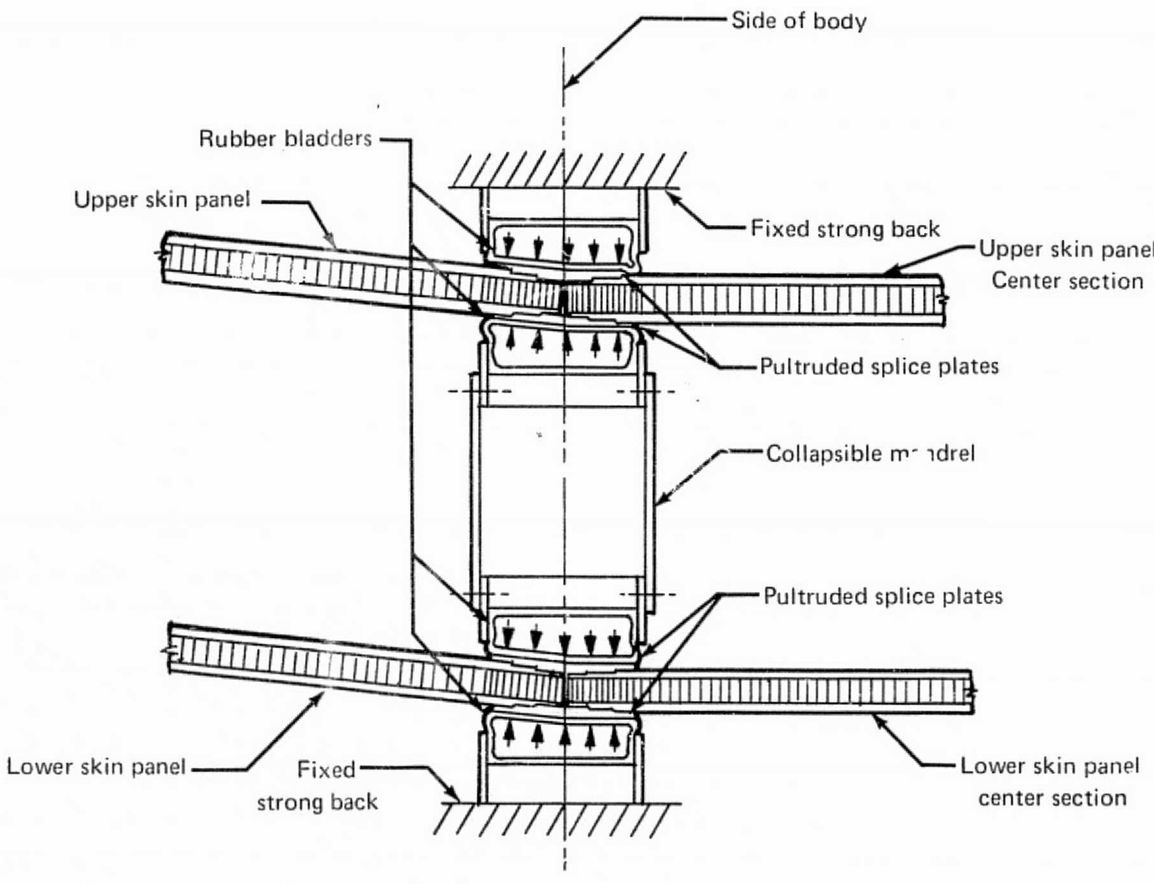


Figure 32. Side of Body Splice Plates

6.0 RECOMMENDED PRELIMINARY DESIGN APPROACH

A recommended preliminary design approach, incorporating conclusions drawn from suction surface, suction system, and wing box structural studies, was submitted to NASA, per contract provisions. The recommended wing geometry was the same geometry submitted as part of the configuration study (Figure 33). The recommended structural arrangement and fuel tank provisions are shown in figures 34 and 35. This design approach was approved by NASA and used as the basis for subsequent wing preliminary design.

The recommended wing box structure is made up of thick graphite/epoxy honeycomb skin panels carrying a slotted suction surface and containing suction plenums and tributary ducts, graphite/epoxy spar webs and graphite/epoxy honeycomb rib webs mechanically attached to pultruded caps and end members. Ribs are spaced at 1.22 m (48.0 in.) outboard of $\eta = .3$, with closer spacing inboard. The wing box inboard of $\eta = .7$ is used for fuel tanks. A dry bay at approximately $\eta = .3$ houses the upper forward suction air duct.

Each inspar (wing box) skin panel consists of an outer and inner graphite/epoxy face sheet bonded to an aluminum honeycomb core. The outer face sheet contains spanwise plenums which lie beneath each suction slot. Bleed holes pass through the face sheet to tributary ducts which are set into the honeycomb core. At intervals, typically 1.22 m (48 in.),vent holes pass from the tributary duct through the core and lower face sheet. The laminate buildup consists of 0° (spanwise) plies which resist wing bending loads, and plies laid at plus and minus 45° which serve to resist wing torsion. Along the line of the plenum the 0° plies are replaced by graphite or fiberglass 45° plies which serve as tear stopping softening strips. Bo. . face sheets incorporate such strips. Figure 21, section 5.3.1, shows the typical skin panel cross section, taken near $\eta = .3$. For preliminary design and analysis purposes the honeycomb core depth was assumed to be a constant 3.81 cm (1.50 in.).

The recommended suction duct layout is shown by figures 14 and 15, in section 5.1. Suction air is carried from the slots and plenums through bleed holes to the tributary ducts. From the tributary ducts air passes through the vent holes into chordwise ducts which pass through the spar webs to the major spanwise ducts. Air from the forward upper surface to the wing is carried forward; all other suction air is transferred aft to major ducts lying behind the rear spar web. The forward major duct is contained in the wing leading edge. Suction air from this duct is passed through the front and rear spar webs in a fore and aft duct to enter the first stage of the suction compressor. The major ducts behind the rear spar are led into the second and third suction compressor stages.

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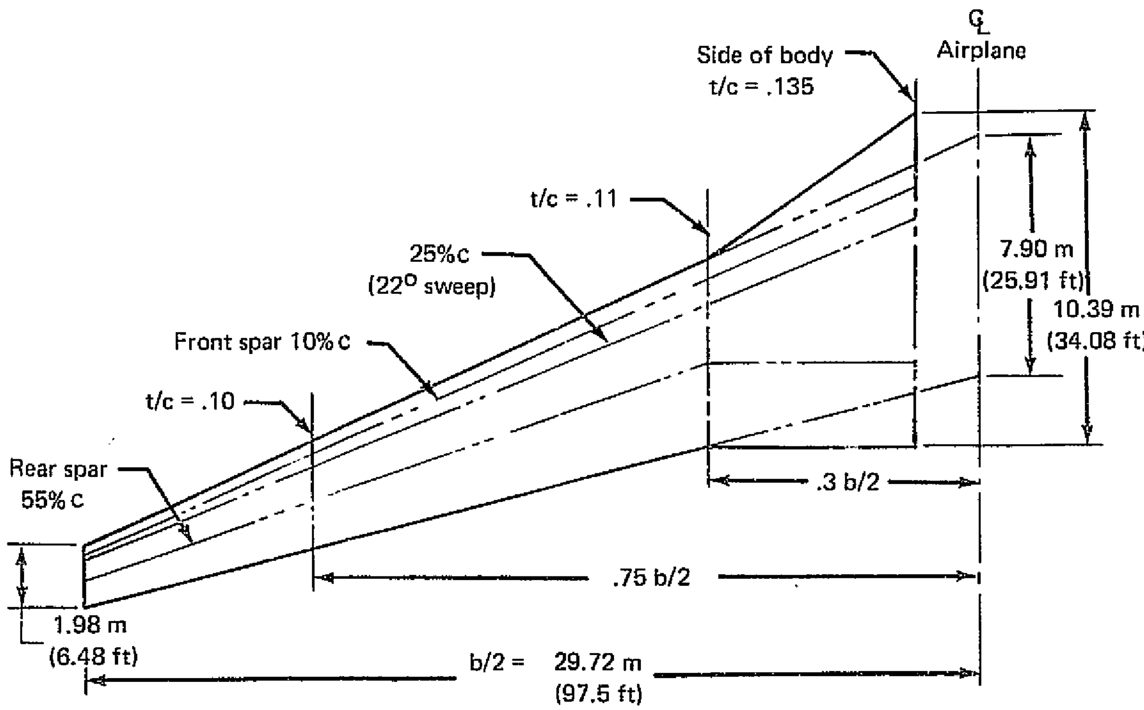


Figure 33. Wing Geometry

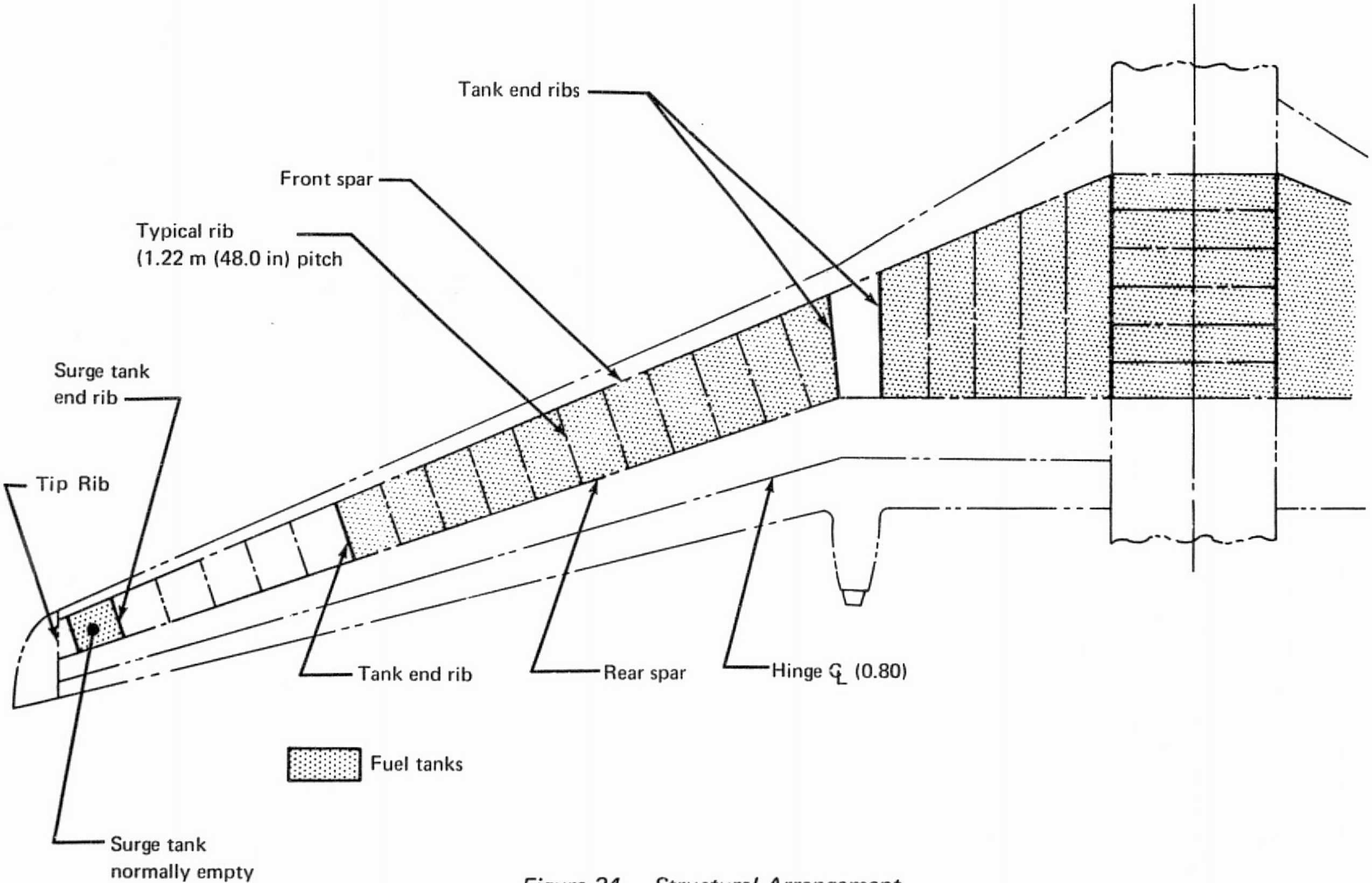
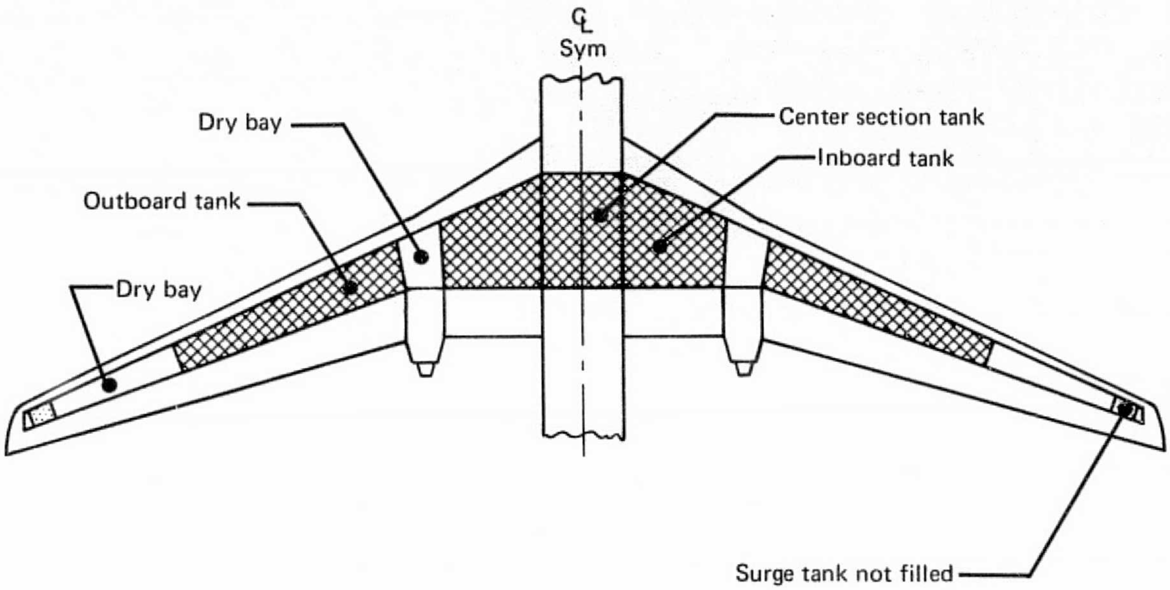


Figure 34. Structural Arrangement



Fuel quantity = 85% x fuel tank volume

TANK	QUANTITY			
	KG	POUNDS	METERS^3	U.S. GALLONS
CENTER	11 515	25 387	14.78	3906
INBOARD	21 690	47 818	27.85	7357
OUTBOARD	12 930	28 506	16.60	4386
TOTAL	46 135	101 711	59.23	15 649

Figure 35. Fuel Volume

The wing leading edge is a fixed structure provided with closely spaced suction slots aft of approximately 0.8% chord. It is attached to the wing box by fasteners which pass through angles attached to the spar web face. It houses the major spanwise duct serving the upper wing surface forward of the rear spar. For ease of assembly, and for inspection and maintenance, all of the leading edge structure is removable. The recommended leading edge, shown in figure 36, is a titanium covered, graphite reinforced, fiberglass honeycomb sandwich structure, without ribs. Upper surface tributary ducts are vented directly to the interior of the leading edge, which serves as the major spanwise duct. Suction air from lower surface tributary ducts is collected by chordwise ducts which pass through the front spar, collect lower wing box surface air and join the major spanwise ducts aft of the rear spar.

The recommended fixed trailing edge structure is shown in figure 37. The skin panels are graphite reinforced fiberglass sandwich structures with titanium skin, carrying suction slots and plenums. The interior of the fixed trailing edge is divided by diagonal webs which serve as duct walls. At each wing box rib station a rib chord member is attached to the wing box structure. The duct walls are locally stiffened in these planes, to serve as rib truss members. The upper and lower fixed skin panels are mechanically attached to the wing box skin panels and to the rib chord members. The lower skin panels may be removed for access. The honeycomb sandwich duct walls also contain access doors.

Access to the interior of the wing box structure for fabrication, maintenance and repair is gained through removable access panels on the wing lower surface. The panels are provided with suction slots and plenums feeding a collector duct, which is in turn connected to a lower surface chordwise duct. The access doors are non-structural elements. The lower wing skin sandwich is reduced to a single surface in the area of the access doors. Fiber orientation in this area is plus and minus 45° only. The reduced thickness area of the lower skin panel provides a fuel passage below rib chord and suction ducts. Figure 38 illustrates a typical access door, and the surrounding skin area.

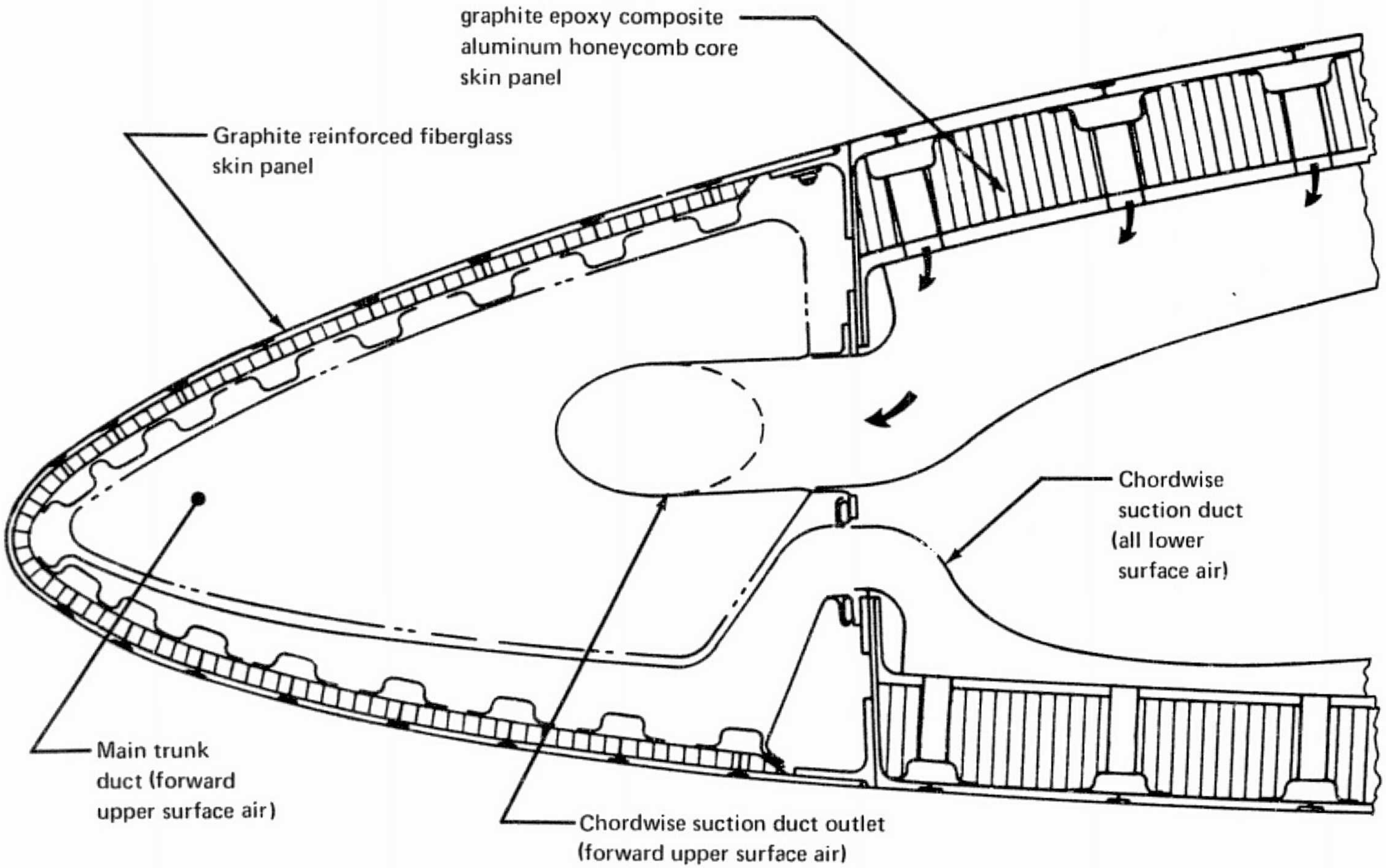


Figure 36. Fixed Leading Edge

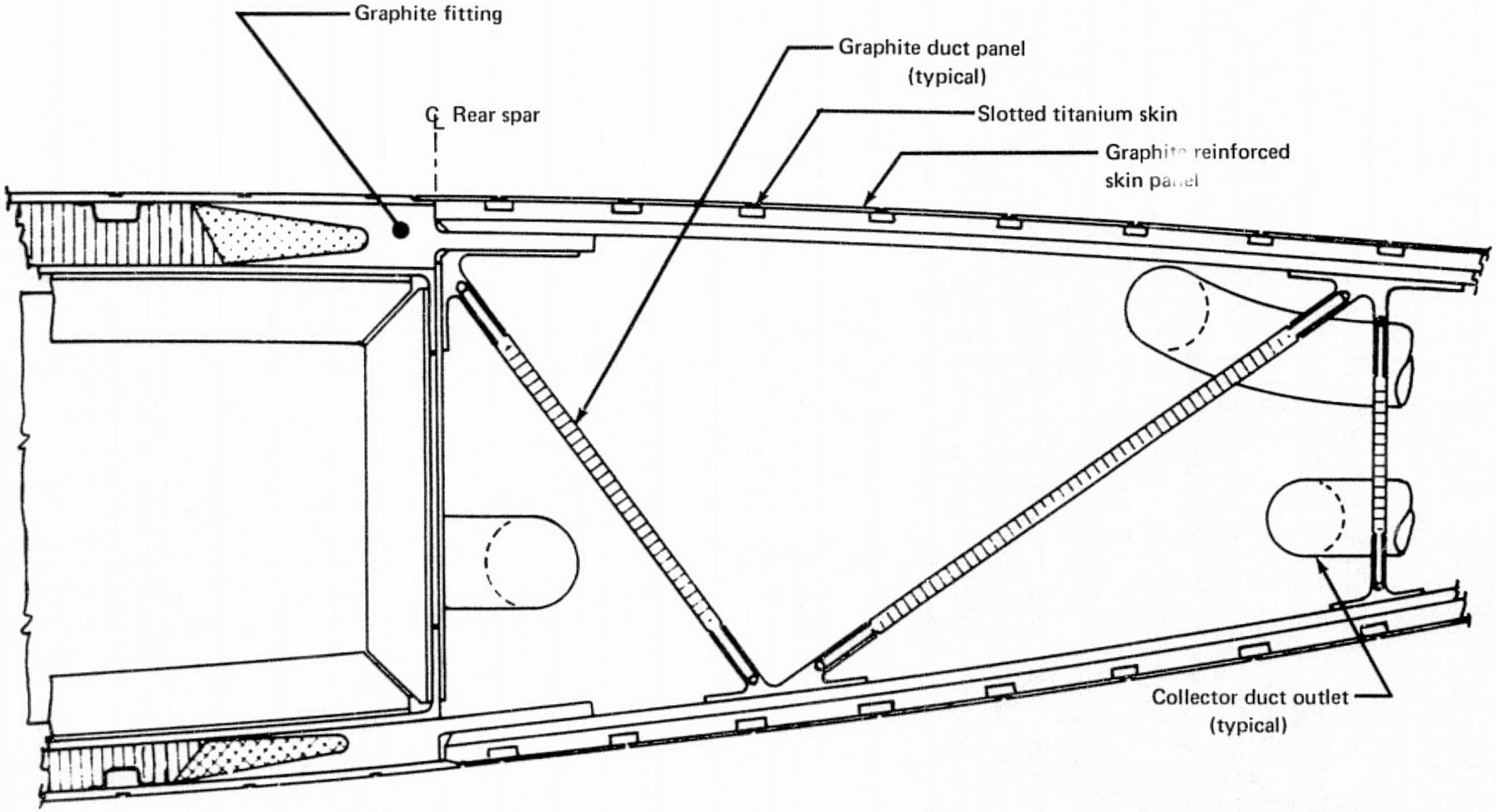


Figure 37. Fixed Trailing Edge

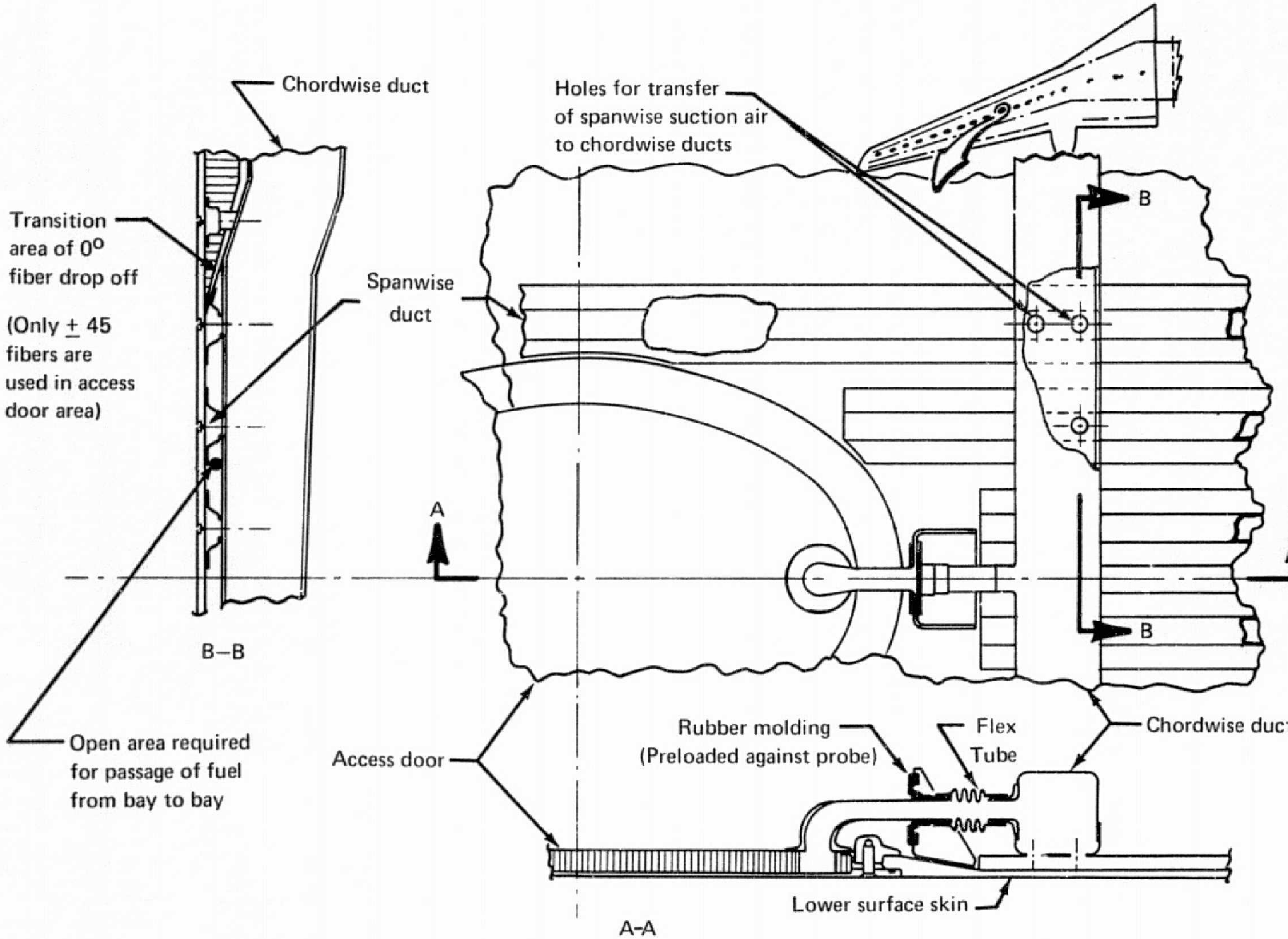


Figure 38. Access Door Area

7.0 WING PRELIMINARY DESIGN

7.1 STRUCTURAL ANALYSIS AND SIZING

7.1.1 WING GEOMETRY

Figure 39 shows the wing geometry and the twelve load panels that were used for this preliminary design analysis. The load reference axis (LRA) shown is a preliminary design estimate of the location of the wing elastic axis. The LRA is straight outboard of η = .4 and is located approximately midway between the front and rear spars. Inboard of η = .4 the LRA curves toward the rear spar. This is due to both the rear spar change in direction and the "root effect" which is typical of swept wings.

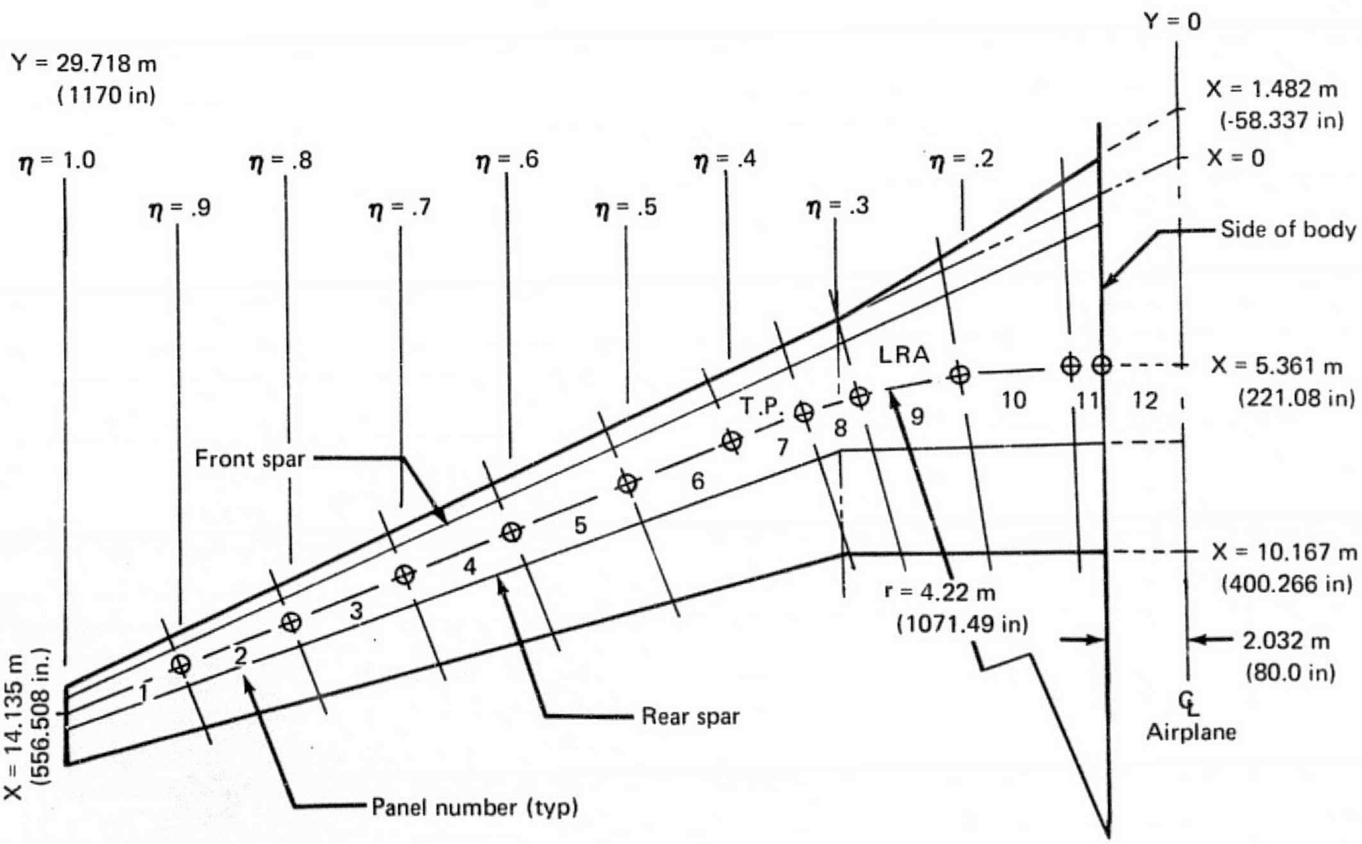


Figure 39. LFC Wing Load Panels and Analysis Stations

Table 3 shows the coordinates of the LRA at the analysis stations used in this structural sizing.

Figure 40 shows the height between upper and lower outside wing skin surface at the front spar, 35% line, and rear spar. These values were used for determining wing box section properties at the eleven analysis stations.

7.1.2 WING LOADS

In order to establish structural feasibility and member sizing, preliminary design loads were established. For preliminary inspar wing box sizing, a 2.5g positive maneuver condition was selected as the most critical design condition. This selection was based on the fact that the 2.5g positive maneuver condition is the most critical condition for most of the 727 airplane wing inspar box and the LFC airplane's similarity in overall configuration.

The estimated spanwise distribution of the wing lift coefficient used for this study was based on data for the 727 airplane wing and two other inhouse study wings with high aspect ratios and comparable sweep angles.Table 4 shows the spanwise distribution of c_l/C_L and l/L used for this study. Figure 41 illustrates the limit load spanwise lift distribution for the 2.5g manuever condition.

The airfoil used in this study is a supercritical shape with a cusped trailing edge. Figure 42 shows the idealized streamwise shape of the net pressure distribution used in this analysis. Using these spanwise and chordwise airload distributions, the air load panel loads were calculated. These panel loads and locations are shown in table 5.

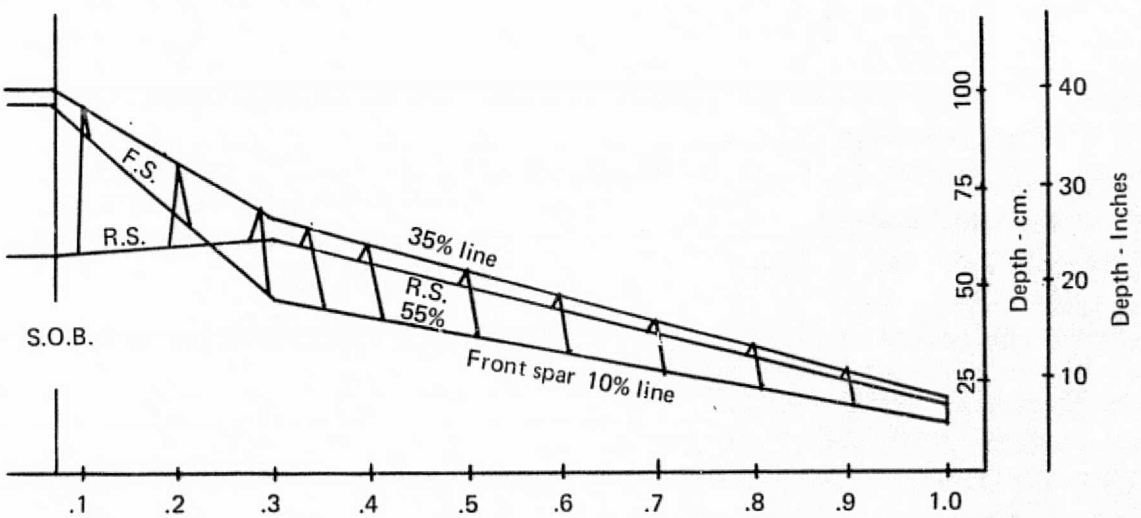
The 1g dead weight panel loads used in this analysis for the wing structure, LFC engine, and the fuel are given in tables 6 and 7. The structural dead weight panel loads are the result of several sizing iterations of the inspar box combined with a parametric estimate of the leading and trailing edge panel weights.

The air loads and dead weights were combined for the 2.5g positive limit maneuver conditions. The usual safety factor of 1.5 was used to determine the ultimate shear, moment, and torsion at the LRA for each analysis station. These values are shown in table 8 and plotted in figure 43.

Preliminary investigation of the wing trailing edge design loads conditions indicated that, due to the cusped trailing edge shape of the airfoil used in this study, the 2.5g limit maneuver condition is critical for design of the fixed trailing edge, flaps, and low speed outboard flaperon. The movable surfaces are in the faired position for this condition. The loads for flaps down approach with surfaces deflected are slightly lower. Figures 43 and 44 show the spanwise and chordwise load distributions that were used for all of the trailing edge structure, except the high speed inboard flaperon. The high speed inboard flaperon maximum design load condition was for full deflection, $\delta = 15^o$, at $V_e = 198.06$ m/sec (385 knots), which was $\Delta_p = 47.6 kN/m^2$ (6.9 psi) at the hinge line, with a linear decrease to zero at the trailing edge.

Table 3. Coordinates of Wing Load Reference Axis (LRA)

η_{LRA}	X Meters	X Inches	Y Meters	Y Inches
1.0	14.1353	556.508	29.718	1170
.9	12.9554	510.056	16.7461	1053
.8	11.82 374	465.604	23.7744	936
.7	10.6719	420.153	20.8026	819
.6	8.8316	374.701	17.8308	702
.5	8.3629	329.249	14.859	585
.4	7.2084	283.797	11.8872	468
.337	6.561	258.306	10.0229	394.6029
.283	6.1205	240.965	8.4139	331.2965
.2	5.644	222.205	5.9436	234
.1	5.377	211.72	2.9718	117
S.O.B.	5.3614	211.08	2.032	80
0	5.3614	211.08	0	0



$$n = \frac{2Y}{b}$$

Wing inspar box depth

Figure 40. Distance Between Upper and Lower Outside Wing Skin Surfaces

Table 4. Spanwise Lift Distribution

η	c_{SL}		c_l/C_L	l/L		$l_{2.5}$	
	Meters	Inches		10^{-6}/mm	10^{-6}/Inches	N/mm	Lbs/In
0	11.65	458.6	.838	29.9	759	112	638
.068376	10.39	409	.887	28.2	716	105	601
.10	9.806	386.1	.904	27.1	689	101	579
.20	7.963	313.5	.971	23.7	601	88.5	505
.30	6.121	241	1.019	19.1	485	71.3	407
.35	5.824	229.3	1.037	18.5	469	69.0	394
.40	5.529	217.7	1.054	17.8	453	66.6	380
.50	4.937	194.4	1.086	16.4	417	61.3	350
.60	4.344	171.0	1.115	14.8	376	55.4	316
.70	3.752	147.7	1.135	13.0	331	48.7	278
.80	3.160	124.4	1.127	10.9	277	40.8	233
.90	2.568	101.1	1.07	8.4	213	31.4	179
.95	2.272	89.5	.948	6.57	167	24.5	140
.975	2.124	83.6	.81	5.28	134	19.8	113
.99	2.035	80.1	.53	3.3	84	12.3	70
1.00	1.976	77.8	0	0	0	0	0

$$l/L = \frac{c_{SL}}{A_\omega}\left(\frac{c_l}{C_L}\right)$$

L_n = n (Gross wt.) + Balancing Tail Load

L_n = 2.5 (1334.4) + 400 = 3736 KN
or 2.5 (300 000) + 90 000
=(840 000 Lbs)

$$l_n = \frac{l}{L}(L_n)$$

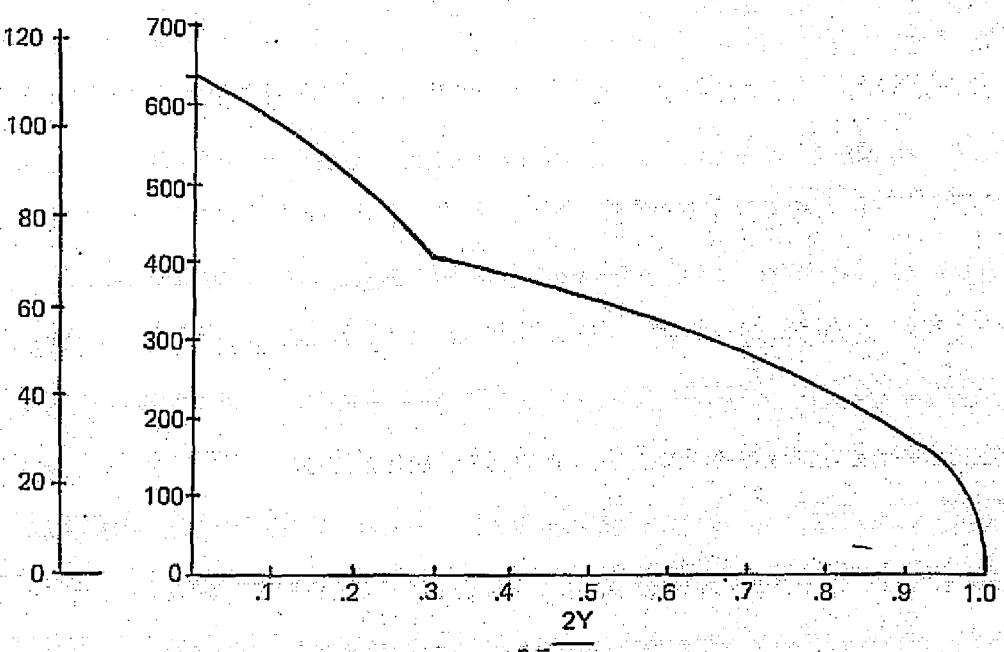


Figure 41. Spanwise Life Distribution - 2.5g Maneuver, Limit Air Load

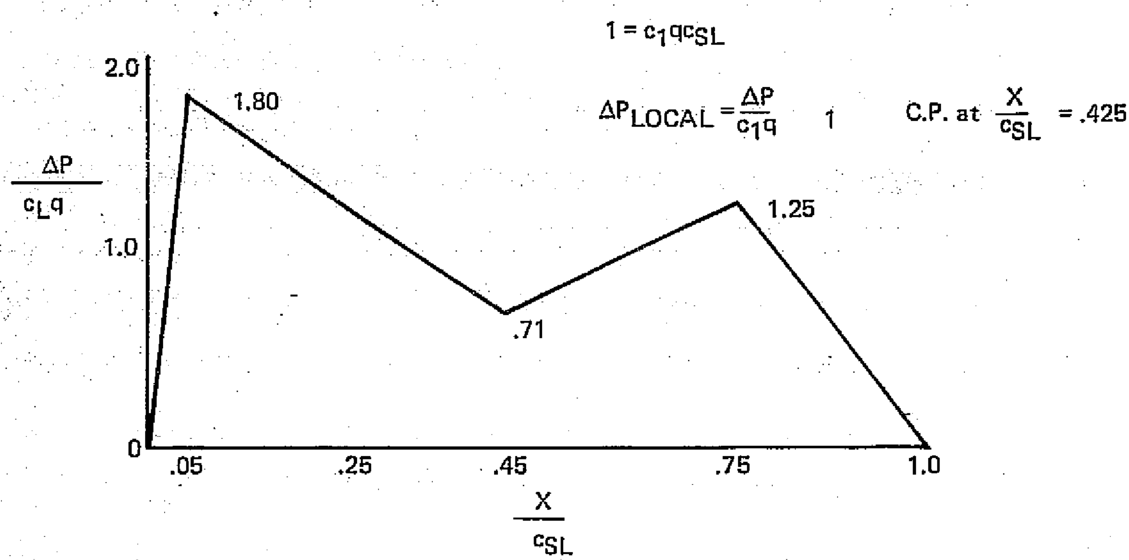


Figure 42. Streamwise Pressure Distribution - Flaps up, Cruise

Table 5. Airloads, n = 1 Panel Loads

Load panel no.	Panel loads		Load centroid location			
			X		Y	
	kN	Lbs	Meters	Inches	Meters	Inches
↑	3.65	820	14.25	561	29.13	1147
1	6.76	1520	13.92	548	28.48	1122
↓	16.97	3816	13.48	530.8	27.36	1077
2	43.43	9764	12.65	498.1	25.09	987.8
3	53.82	12100	11.57	455.7	22.11	870.6
4	62.75	14108	10.48	412.6	19.13	753.1
5	70.56	15864	9.36	369.3	16.14	635.6
6	77.48	17420	8.28	326	13.16	518
7	54.60	12276	7.366	290	10.62	418
8	45.23	10168	6.782	257	8.99	354
9	79.26	17820	6.096	240	7.11	280
10	112.5	25296	5.18	204	4.34	171
11	36.56	8220	4.57	180	2.54	100
12	83.66	18808	3.12	123	.99	39

L = 1 (1334.4) + 160 = 1494.4 kN

or

L = 1 (300 000) + 36 000 = 336 000 Lbs

Table 6. 1 g Dead Weight Loads, Wing Structure, LFC Suction Engine and Compressor

Panel number	1g Structural Dead weight loads		C. G. Location			
			X		Y	
	kn	Lbs.	Meters	Inches	Meters	Inches
1	2.5	560	13.61	536	28.12	1107
2	5.6	1260	12.52	493	25.15	990
3	8.1	1820	11.38	448	22.15	872
4	10.3	2320	10.26	404	19.18	755
5	12.9	2900	9.12	359	16.21	638
6	15.3	3440	8.00	315	13.21	520
7	10.4	2340	7.01	276	10.8	425
8	9.6	2160	6.375	251	9.14	360
9	17.6	3960	5.79	228	6.99	275
10	27.0	6080	5.18	204	2.44	160
11	10.1	2260	5.08	200	4.06	96

LFC engine 1 g dead weight load = 17.79 kn (4000 lbs)
C. G. location X = 10.21 m (402 in.)
y = 8.59 m (338 in.)

Table 7. 1 g Dead Weight Panel Loads, – Fuel

Panel number	1 g Fuel dead weight loads		C. G. location			
			X		Y	
	kn	Lbs	Meters	Inches	Meters	Inches
4	14.1	3180	10.06	396	19.25	758
5	16.2	3644	8.92	351	16.28	641
6	23.9	5384	7.77	306	13.31	524
7	17.1	3844	6.81	268	10.92	430
8	17.7	3976	6.375	251	9.14	360
9	43.9	9860	5.72	225	7.01	276
10	65.1	14644	4.95	195	3.21	170
11	26.2	5880	4.67	184	2.49	98

Table 8. Wing Ultimate Design Loads, 2.5g Limit Maneuver Condition, S.F. = 1.5

η_{LRA}	Shear (V)		Moment (M)		Torsion (T)	
	kn	KIPS	MN - m	10^6 In.- Lbs.	MN - m	10^6 In. - Lbs.
.9	93.36	20.99	.122	1.08	-.027	-.24
.8	253.9	53.03	.635	5.62	-.072	-.64
.7	406.7	91.43	1.65	14.6	-.141	-1.25
.6	550.2	123.7	3.17	28.1	-.238	-2.07
.5	705.9	158.7	5.17	45.8	-.35	-3.10
.4	849.1	190.9	7.66	67.8	-.479	-4.40
.337	950.5	213.7	9.35	82.8	-1.22	-10.8
.283	951.4	213.9	10.8	95.7	-1.62	-14.3
.2	1018.	228.9	13.0	115 0	-2.87	-25.4
.1	1094.	246.0	16.0	141.7	-4.51	-39.9
S.O.B.	1096.	246.3	16.8	148.5	-5.02	-44.4

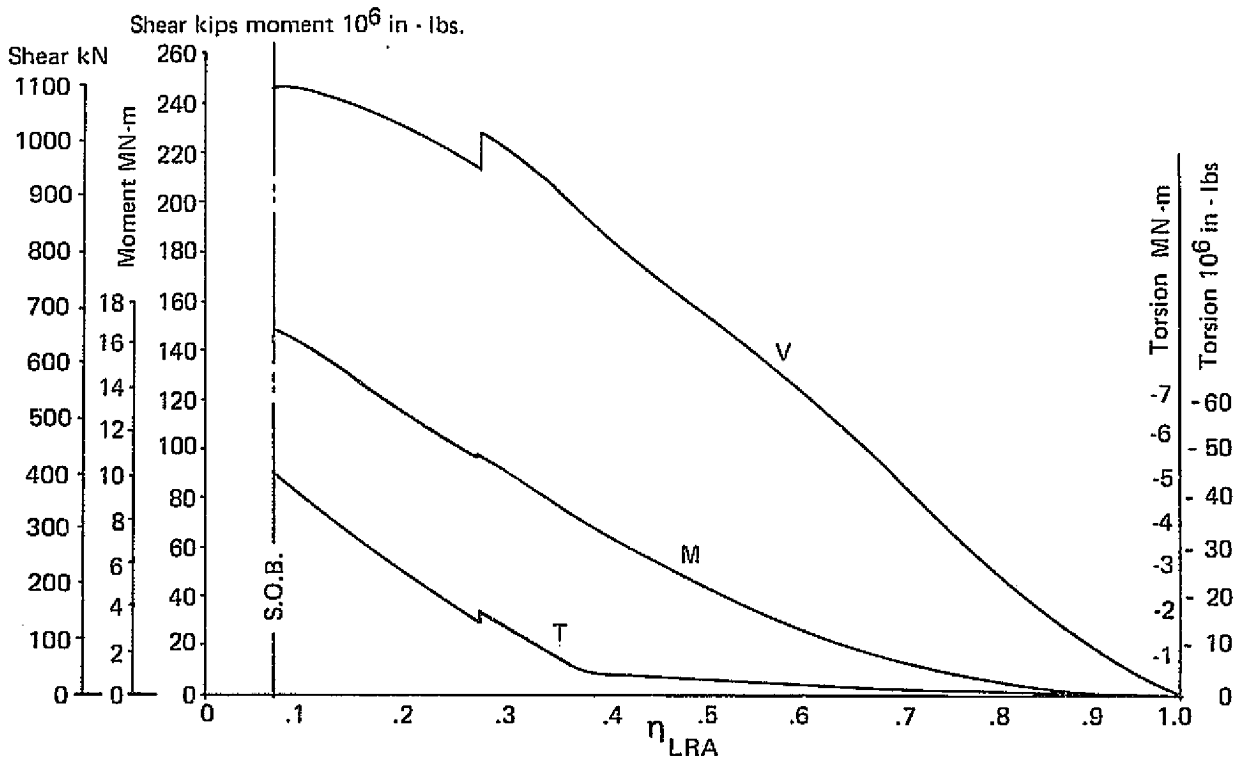


Figure 43. Ultimate Loads - 2.5g Limit Maneuver Conditions

These preliminary design loads for the fixed trailing edge and movable surface structure were used to approximate the hinge, actuator and member sizes in order to determine feasibility of installations and space requirements.

7.1.3 INTERNAL LOADS

For this study, it was assumed that the inspar box carries all of the wing shear moment and torsion. The leading edge and trailing edges were conservatively assumed to be ineffective due to removable replaceable maintenance panels required for a commercial airplane application.

A single cell, box beam solution was used to evaluate the panel end loads, (n) and skin panel and spar shear flows (q). An appropriate wing root correction was made for the increased loading of the skin panels aft of the LRA, from η = .4 in to the side of body rib.

Table 9 contains the resultant wing box internal loads used for member sizing.

7.1.4 INSPAR BOX SKIN PANEL CONCEPT

The skin panel structural concept trade studies showed that a graphite faced, aluminum honeycomb core panel concept provided the best functional and structural concept for the LFC wing design. Figure 21, Section 5.3, shows a typical cross section of this concept.

For this preliminary sizing of the wing inspar panels, the basic face skin layup was selected as a $[\pm45/0^{o}_{2}]_{n}$ laminate. It was assumed that the rib chords and mid bay chordwise ducts will be primarily 0^{o} orientation pultrusions and will supply sufficient chordwise stiffness, requiring very few, if any, 90^{o} plies in the face skins. A panel test program will be required to verify this design assumption. The use of 50% of the laminate as $\pm45^{o}$ plies was chosen to provide high torsional stiffness for this long narrow wing box. The design and analysis refinement of this wing configuration and structural concept may show that some $\pm45^{o}$ plies are not needed and should be replaced with some 90^{o} plies, but the overall panel gages as sized in this study are not expected to change appreciably.

The zero degree plies of the panels occupy 5 cm (2.0 in.) of width in every 7.6 cm (3.0 in.) across the panel width. The remaining width is replaced by $\pm45^{o}$ plies. This gives strips of 2.6 cm (1.0 in.) wide full depth $\pm45^{o}$ layups which are the locations of the laminar control slots and holes in the panel skins. These full depth $\pm45^{o}$ strips have a lower spanwise modulus of elasticity than the basic laminate. Therefore, there is a lower stress in the area of the holes which reduces the potential for fatigue crack initiation. Because these 2.6 cm (1.0 in.) wide "softening strips" located on 7.6 cm (3.0 in) centers will also act as tear stoppers, for failsafe considerations, the skin panels are one piece with no spanwise or chordwise splices from the airplane centerline to the tip.

Table 9. Internal Ultimate Design Loads, — Inspar Wing Box

Analysis location η LRA	Skin panel end load N_T or $-N_C$ kN/m		Skin panel end load LB x 10^3		Spar shear flow: Rear spar kN/m	Rear spar LB x 10^3	Front spar kN/m	Front spar LB x 10^3
.9	641		3.66		333	1.90	193	1.10
.8	2007		11.46		608	3.47	380	2.17
.7	3592		20.51		851	4.86	543	3.10
.6	5003		28.57		975	5.57	606	3.46
.5	6185		35.32		1082	6.18	664	3.79
.4	6908		39.45		1114	6.36	665	3.80
▷1	N@R.S.	N@F.S.	N@R.S.	N@F.S.				
.337	7618	6882	43.5	39.3	1422	8.12	450	2.57
.283	7372	6024	42.1	34.4	1485	8.48	413	2.36
.2	5761	4255	32.9	24.3	1320	7.54	221	1.26
.1	4588	3065	26.2	17.5	1229	7.02	154	.88
S.O.B.	4571	2469	26.1	14.1	1205	6.88	154	.88

▷1 Load redistribution toward rear spar due to swept wing root effect and rear spar direction change.

The basic core is 9.13 kg/m^3 (5.7 lb/ft^3) aluminum honeycomb. Since the honeycomb core is a very small percentage of the total panel weight, a constant depth core was selected. The 3.8 cm (1.5 in.) depth selected for this design is a compromise which is not the optimum for panel column stability for either the low or the high end load areas of the panels. A constant core depth simplifies the fabrication of the splices in the basic core blanket and at dense core inserts at the rib locations. It also simplifies the machining of the slots for the spanwise ducts shown in figure 21. For this study, it was concluded that the additional costs generated by these manufacturing complexities were not worth the insignificant weight saving that might be possible by tapering the core depth.

7.1.5 SELECTION OF GRAPHITE-EPOXY COMPOSITE "PRELIMINARY DESIGN ALLOWABLES"–MATERIAL MATRIX

Several sources of graphite-epoxy composite laminate strength data were reviewed during this graphite LFC wing preliminary design study (Ref. 2, 3, and 4). The material matrix, table 10, shows the ultimate strains and elastic modulii for three different graphite-epoxy composite systems with three different fiber orientations. The graphite-epoxy system selected for use in final design will depend not only on the ultimate design strengths but also on evaluation and comparison of other data such as fatigue life, fracture resistance, impact resistance and cost.

Therefore, because the preliminary wing sizing in this study was based only on ultimate design strength, no particular material system was selected. Modulii of elasticity and allowable strain values were selected which are representative of many different combinations of graphite fibers and epoxy resins.

It was determined that, for cross plied laminates, a criterion which limits the maximum strain at limit load is necessary. The limit strains were selected to insure that no failure of the resin matrix occurs between adjacent cross plied fibers when strained by limit loads. Table 11 lists the ultimate strains used in this study based on this criteria.

The skin panel and shear web allowables shown on table 11 are based on the strain limits noted and the analytical evaluation of the modulus of elasticity based on fiber orientation and direction and type of loading. These allowables are "Preliminary Design Allowables" which are adequate for use in a preliminary design study.

"Firm" design allowables must be established for the selected graphite composite system prior to final design and release of detail drawings.

7.1.6 INSPAR SKIN PANEL ALLOWABLES AND SIZING

The main load carrying 0° graphite plies are located in the 5 cm (2.0 in.) widths at 7.6 cm (3.0 in.) spacing across the skin panels. Because of the difference in spanwise and chordwise modulus of elasticity of the panels, the honeycomb panel compression allowables were determined using honeycomb column equations instead of plate buckling equations. The panels were assumed to be simply supported at the ribs; have a column length equal to the rib spacing; and the panel E^c values oriented perpendicular to the analysis station.

Table 10. Material Matrix

	Materials considered									Design values used for this study. Strain limitations due to cross plied GR/EP laminates.		
	AS/3002 (Tape)[a]			T300/5208(Tape)[b]			HMF-330C/34 (Cloth)[c]					
Laminate orientation	Design ultimate strain	Elastic modulus		Avg. ult strain (B-basis values)	Typical elastic modulus		Typ. ult strain (B-basis values)	Typical elastic modulus		Ult design strain	Design elastic modulus	
		GPa	MSI		GPa	MSI		GPa	MSI		GPa	MSI
	ϵ_L^t = .0068	E_L^t = 135	19.6	.010 (.0085)	147	21.3	.010	69	10	.005	142	20.6
	ϵ_T^t = .0068	E_T^t = 6	.9	.0039 (.0032)	10.9	1.58	.0093	64	9.3	.005	7	1.0
0^o For tape	ϵ_L^c = -.0095	E_L^c = 110	16	-.0107 (-.0086)	131	19	-.0094	66.6	9.66	-.006	142	20.6
(0/90 for cloth)	ϵ_T^c = -.007	E_T^c = 21	3.1	-.011 (-.0078)	13	1.89	-.0088	61.9	8.98	-.006	14	2.0
	ϵ_s = .007	G = 5.5	.8	.014 (.0135)	6.4	.93	.008	7	1.0	.0075	6	.9
	ϵ_L^t = .0089	E_L^t = 20	2.9	.0054 (.0046)	26	3.78	.009	26	3.8	.005	25.5	3.7
±45	ϵ_L^t = .0089	E_L^c = 20	2.9	-.0073 (-.0054)	24.9	3.61	-.009	26	3.8	-.006	25.5	3.7
	ϵ_s = .00675	G = 28.6	4.15	.0124 (.0074)	31.5	4.57	N.A.	31	4.5	.0075	31	4.5
$[0/\pm45/90]_n$	ϵ^t = .0098	E^t = 43	6.3	.0074 (.0058)	56	8.1	.0092 (.007)		7.8	.005	52	7.5
Quasi-isotropic	ϵ^c = -.010	E^c = 43	6.3	-.0079 (-.0059)	54	7.8	-.0078 (-.0075)		7.0	-.006	52	7.5
	ϵ_s = .010	G = 18	2.6	.017 (.012)	17.4	2.52	N.A.	N.A.	N.A.	.0075	17	2.5

a = ref. 4 b = ref. 5 c = ref. 6

Table 11. Graphite Laminate "Preliminary" Design Allowables
Wing Inspar Skins & shear webs

Inspar skins							
Loading		Modulus of elasticity		Limit strain	Ultimate strain	Allowable stress ultimate	
Direction	Condition	GPa	MSI	ϵ limit	ϵ_{ult}	MPa	LB/In ^{2}
spanwise	tension	59.6	8.65	.0033	.005	298	43 250
spanwise	compression	59.6	8.65	-.004	-.006	-358	-51 900
chordwise	tension	27.6	4.0	.0033	.005	138	20 000
chordwise	compression	27.6	4.0	-.004	-.006	-167	-24 000
	shear	17.2	2.5	.005	.0075	129	18 750

Shear webs, $\pm 45^{\circ}$ Laminate $e_{s\ ult} = .0075$ G = 31GPa (4.5 MSI) F_{SU} = 233 MPa (33 750 LB/in^2)

The tension allowables inboard of η = .4 were also modified to account for variation of E^t caused by zero ply orientation and primary load direction.

The panel end loads and allowables were used to determine the skin thicknesses. Figures 44 and 45 show the face skin thicknesses of the equal faced honeycomb inspar skin panels and the front and rear spar web gages.

7.1.7 BENDING AND TORSIONAL STIFFNESS

The calculated EI and GJ values for this inspar wing box design are plotted in figure 46. This stiffness data was reviewed by the Boeing Structural Dynamics staff. Their comparison with parametric data indicated that these stiffnesses provided a slightly positive flutter margin for the study design. This evaluation of the flutter margin was of the same degree of refinement as the preliminary loads, allowables and sizing used in this study.

The refinement of this design would require an in-depth screening for all critical loading conditions, the establishment of firm design allowables, and a rigorous dynamic analysis of the wing structure.

7.2 STRUCTURAL DETAILS

As the recommended design approach was developed into a preliminary wing design, a number of structural details were defined. During the development some desirable changes to the recommended approach were recognized. The structural details and the changes to the recommended approach are discussed in the following paragraphs. Figure 47 is a structural diagram of the preliminary wing design.

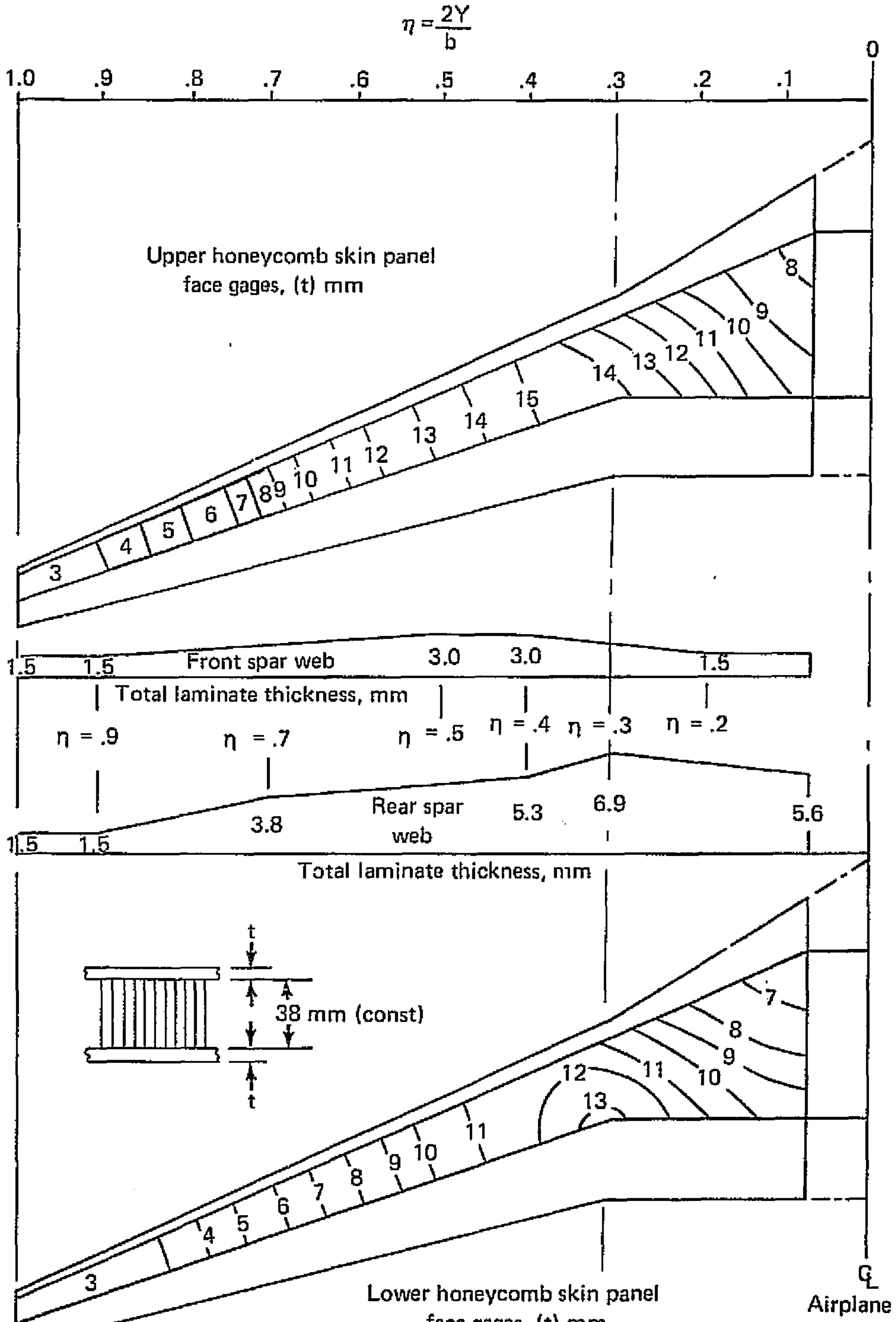


Figure 44. Inspar Box Member Sizes (SI Units)

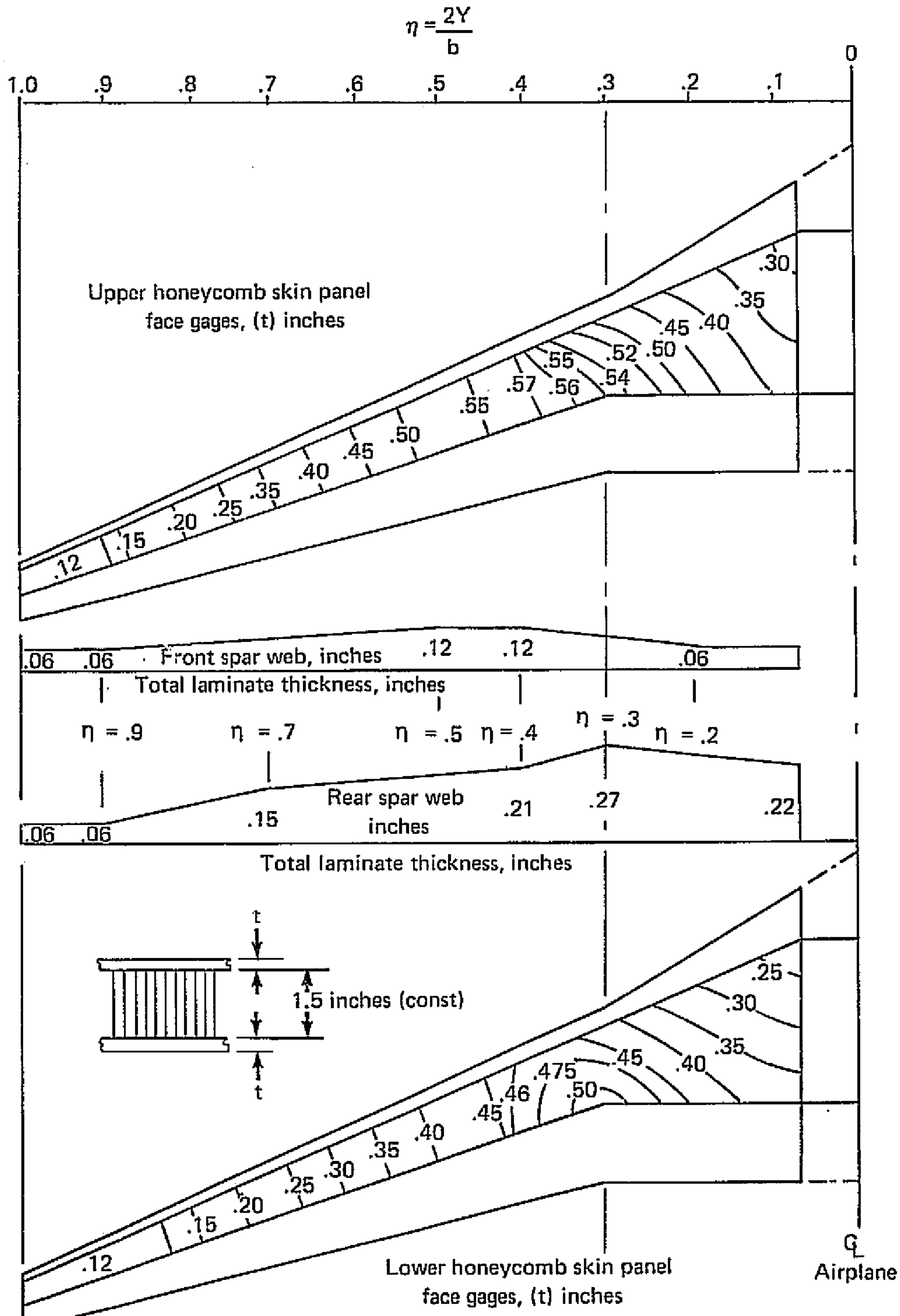


Figure 45. Inspar Box Member Sizes (U.S. Customary Units)

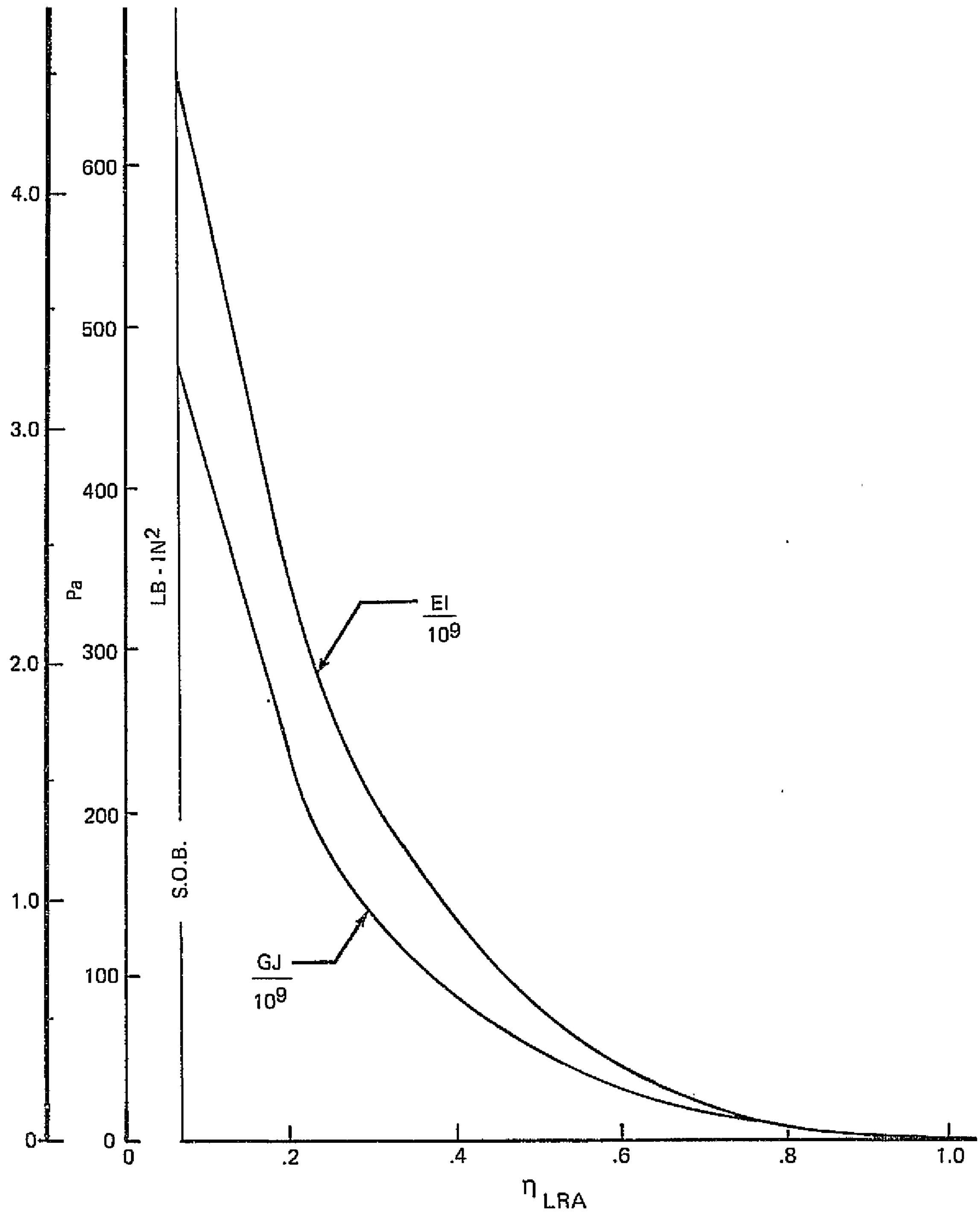


Figure 46. Bending and Torsional Stiffness

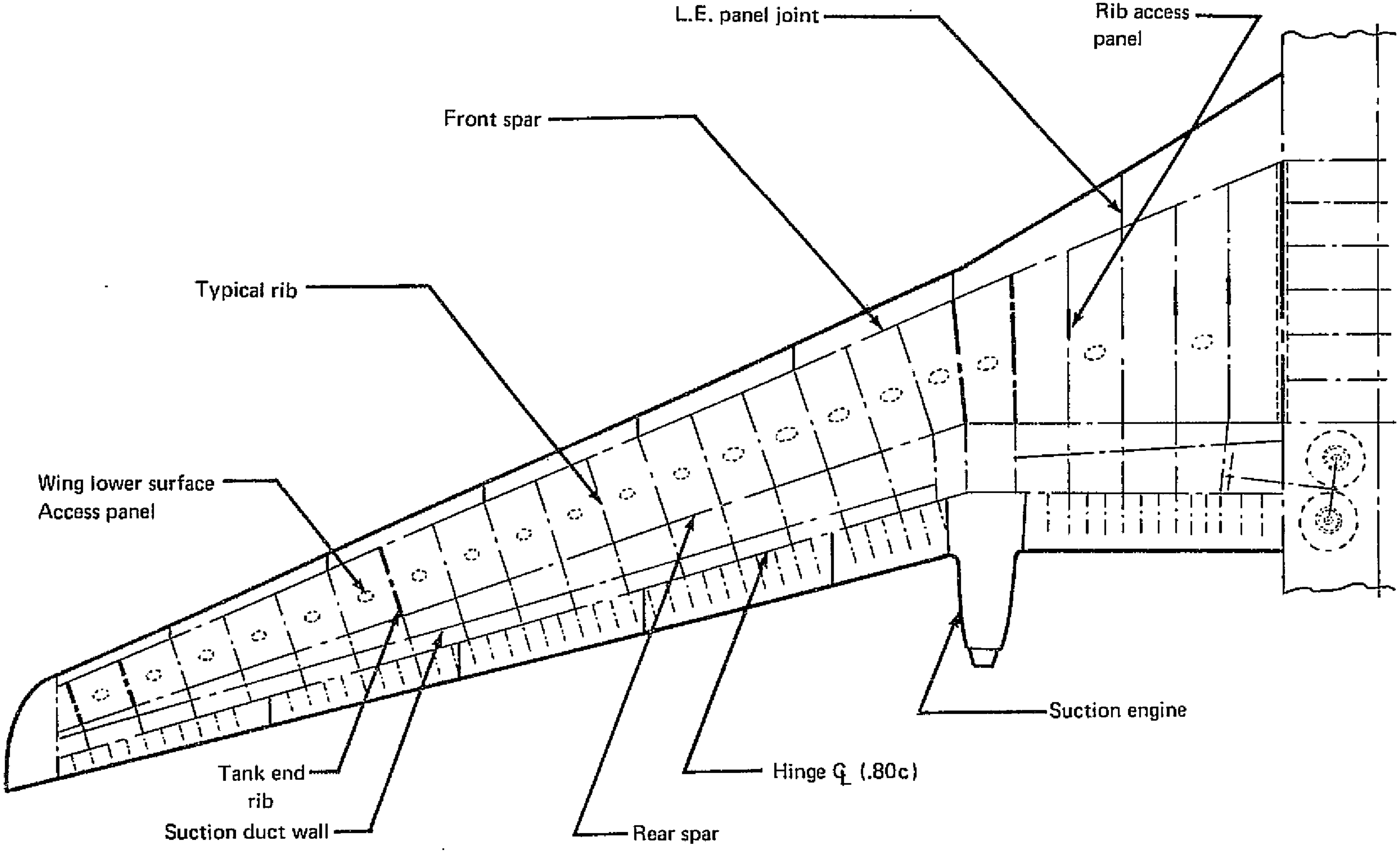


Figure 47. Structural Arrangement

7.2.1 WING SPLICE

In the recommended approach, in order to further explore the possibility of weight saving in this area, the wing splice location was not specified. In the final preliminary design the wing splice is at the usual side of body location for the following reasons:

- Skin panel contours must change at the side of body since the wing root taper cannot be carried into the airplane centerline without affecting the cabin floor height and the wing body fairing depth.

- Although the skin panel has no discrete stiffeners the softening strips beneath each slot serve the same purpose and must have continuity across the side of body area. In the exposed, sucked portion of the wing the slots and softening strips must be essentially parallel to the wing sweep angle, while across the center wing box they should be normal to the airplane centerline. The side of body splice members can provide the required continuity and change in load path direction.

- An increase in skin panel thickness (pad up) is required at the side of body to transfer cabin pressurization loads into the side of body rib. The weight added for pad up in this area is approximately equal to wing splice pad up weight, and serves a dual purpose if a side of body wing joint is employed. A centerline splice requires pad up at the wing centerline and the side of body.

- The side of body rib is required to react loads induced by the change in contour of the wing skin panels regardless of the choice of structural splice location. In either case it serves as a tank end rib. No appreciable rib weight saving is realized by incorporating a centerline splice.

Details of the final splice configuration are shown in figures 48 and 49. Division of the wing splice members into chordwise segments provides fail safe integrity across the joint.

7.2.2 WING CENTER SECTION

The recommended approach to the wing center section included chordwise ribs. When the upper skin of the wing center section supports a pressurized cabin floor, or is employed as a pressure bulkhead, the center section ribs transmit pressure loads to the side of body attachments of the fuselage frames. Transverse ribs, or beams, provide a more direct load path, resulting in a lower weight design. The transverse beams serve as slosh baffles. The pitch of body frames in the body center section will be half the pitch of the center section beams, allowing alternate frames to be attached directly to center section beams. With the elimination of a centerline splice rib, the transverse center section beams are continuous across the center wing box. The final arrangement of the wing center section is shown in figure 50.

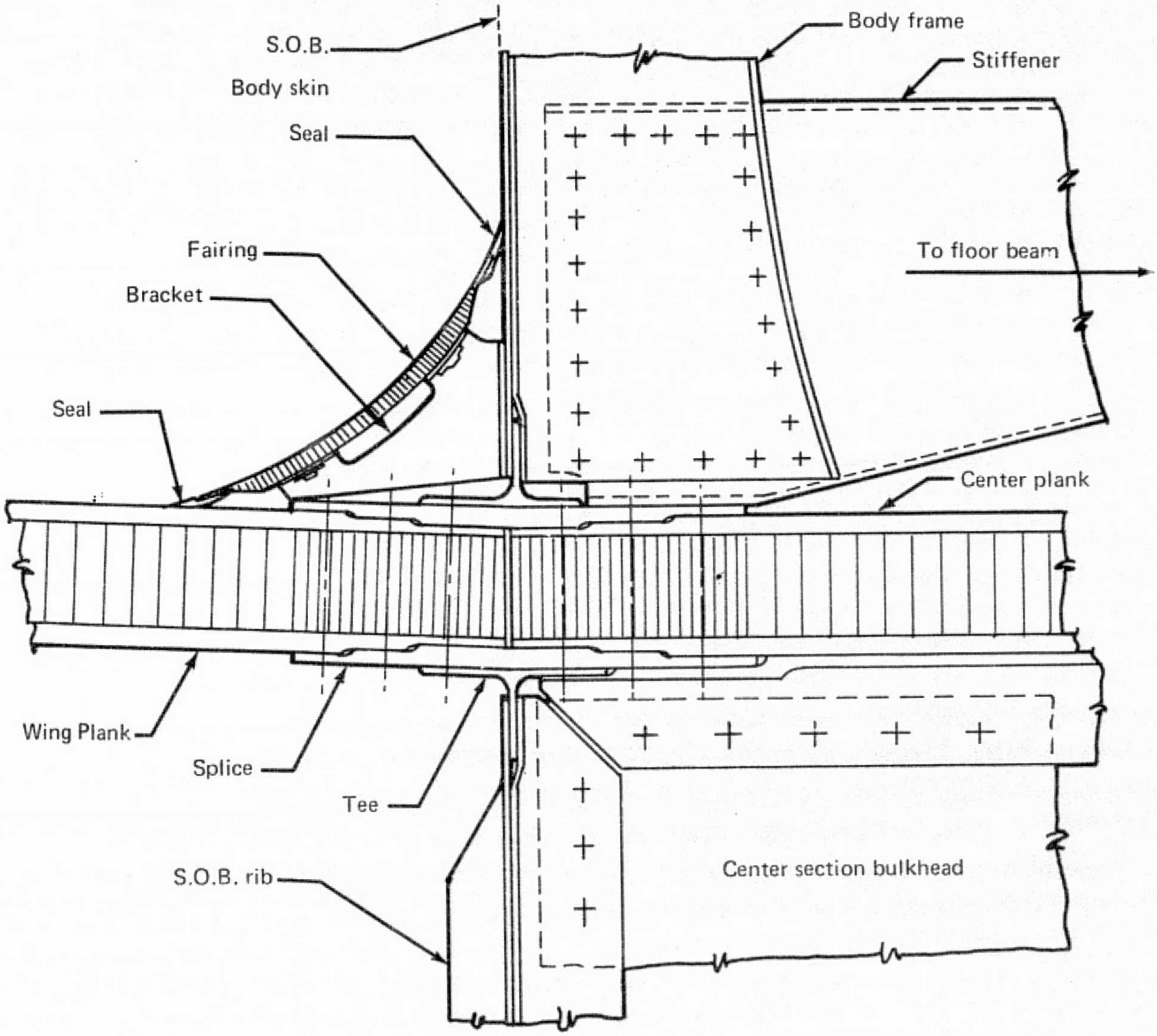


Figure 48. Side of Body Splice

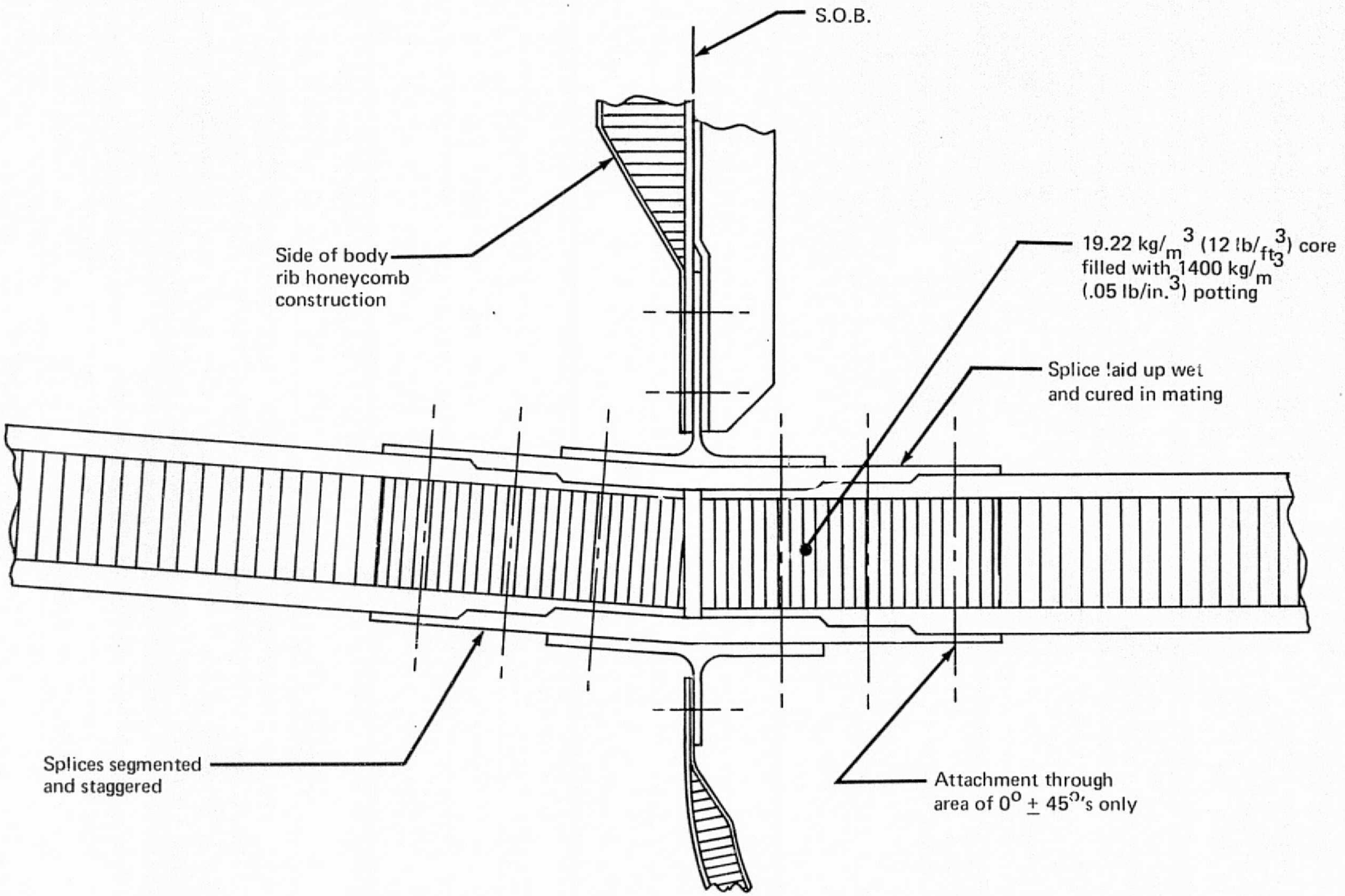


Figure 49. Side of Body Splice - Lower Wing Skin

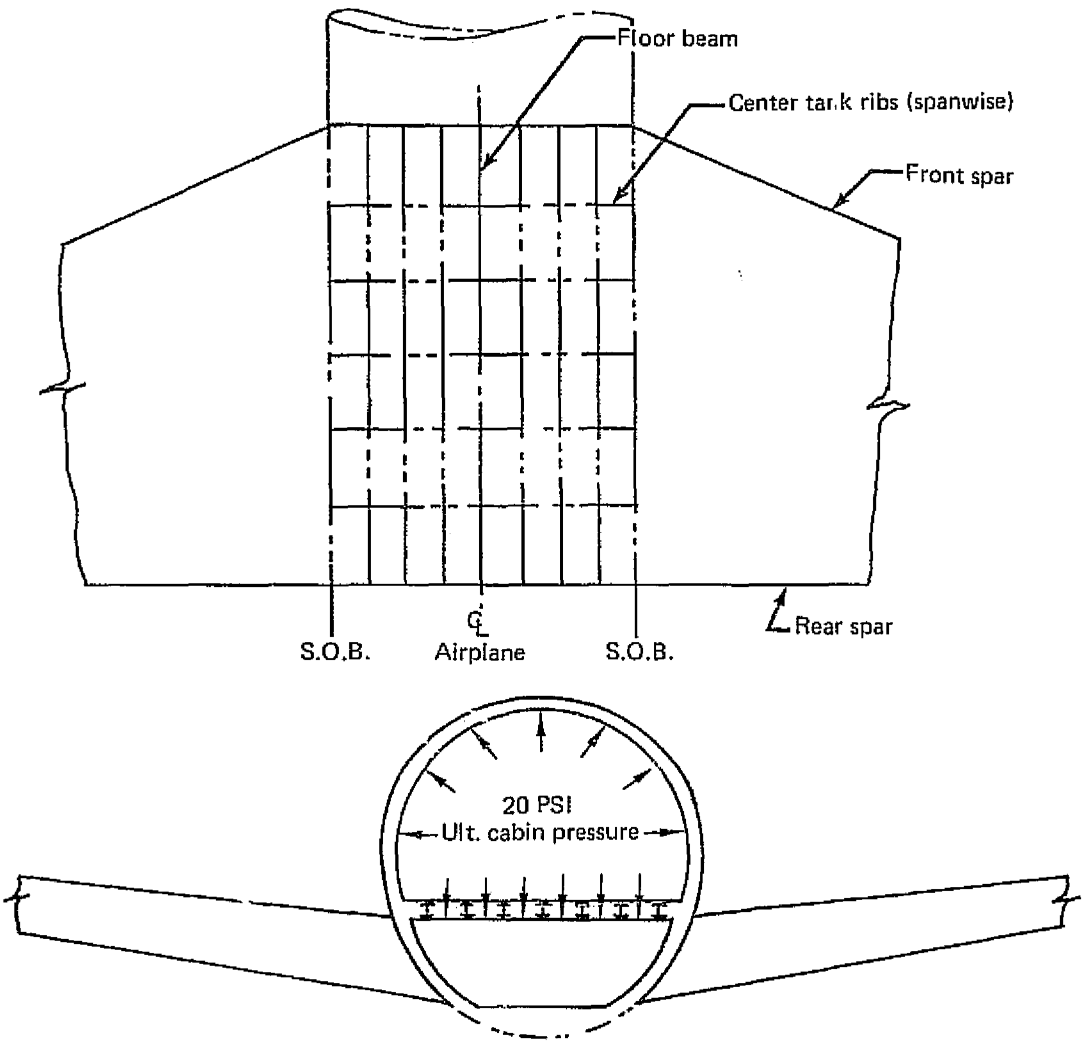


Figure 50. Wing Center Section

7.2.3 WING BODY JOINT

The wing to body attachment may be made by a pinned or rigid joint. Both concepts have a satisfactory history of service on current transport aircraft.

The pinned joint requires large machined fittings on the wing neutral axis at the side of body/spar intersections, and matching fittings on main body frames. Since wing deflections are not transmitted to the body through the pinned joints, pressure seals in this area tend to be complex. The rigid wing-body joint transfers shear load more directly and has proven to be lighter when executed in aluminum alloys. It was assumed that satisfactory methods of analysis will be developed to treat the thermal and mechanical problems of a composite to metal wing-body joint and that the weight advantage of a rigid joint will be realized. The selected rigid joint is shown in figure 51.

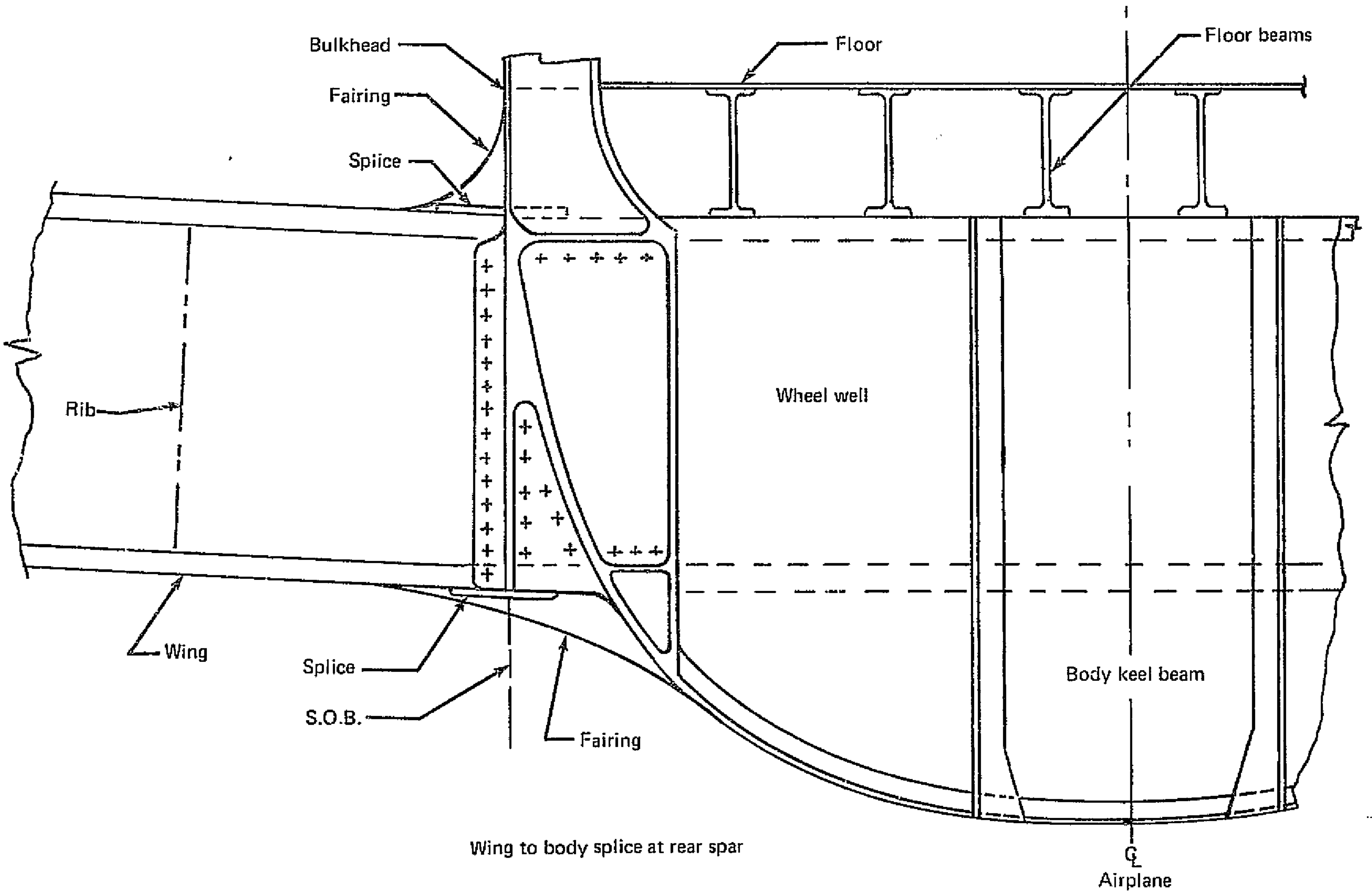


Figure 51. Wing to Body Splice

7.2.4 CONTROL SURFACES

Establishment of laminar flow across control surface hinge lines and over the surfaces was demonstrated as part of the X-21A program (Ref. 1.) In this study, a design solution similar to the X-21A has been adopted. Figures 52 through 56, showing the outboard or low speed flaperon, illustrate the general approach. The flaperon is divided by ribs into relatively short panels, each served by a chordwise duct. In this manner the required chordwise duct area and consequent flaperon spar penetrations are kept small. The tapered tributary ducts and chordwise ducts are molded assemblies, with the tributary ducts serving to stiffen the skin panels. Hinge and actuator fittings are part of the "I" section aileron spar and intermediate "fishmouth" shear fittings help maintain alignment of fixed trailing edge and control surface in the faired (cruise) condition. Use of duplicated irreversible actuators without manual reversion eliminates the requirement for flaperon mass balance.

Suction air is passed across the hinge line either by flexible ducting or by rotating or sliding fittings. Suction inflow is reduced in several slots just forward of the hinge line by deletion of some plenum bleed holes. This allows the laminar boundary layer to increase in thickness, reducing the boundary layer's sensitivity to disturbance by gaps and steps on the surface.

It is recommended that trade studies be performed to weigh the difficulties and complexities of full chord laminarization and the large increment of duct volume and pumping capacity required (figure 4) against the performance improvement provided. If full chord laminarization is found preferrable, a number of design problems should be studied. Among these are:

- Optimum integration of plenums and tributary ducts into the control surface skin, to minimize skin thickness and weight.
- Improvement of hinge and alignment fittings.
- Improvement of aerodynamic sealing of the control surface hinge gap.
- Refinement of flexible duct connections to insure adequate trouble free service life.

Development of a flexible skinned trailing edge control surface, analogous to variable camber leading edge flaps now in use, would eliminate a number of the problems of laminarizing hinged surfaces, and presents a promising opportunity for use of composites.

7.2.5 ACCESS PANELS

In a practical aircraft of this size, intended for commercial service, external access panels are a necessity. Since these doors are to be removable for flight line maintenance they will not be part of the load carrying structure. Gaps and steps between doors and adjacent skins must be minimized, and the boundary layer effects of such excresences as fastener heads must be small. Normal removal and replacement of such panels should not degrade the aerodynamic qualities of the wing surface to any significant degree.

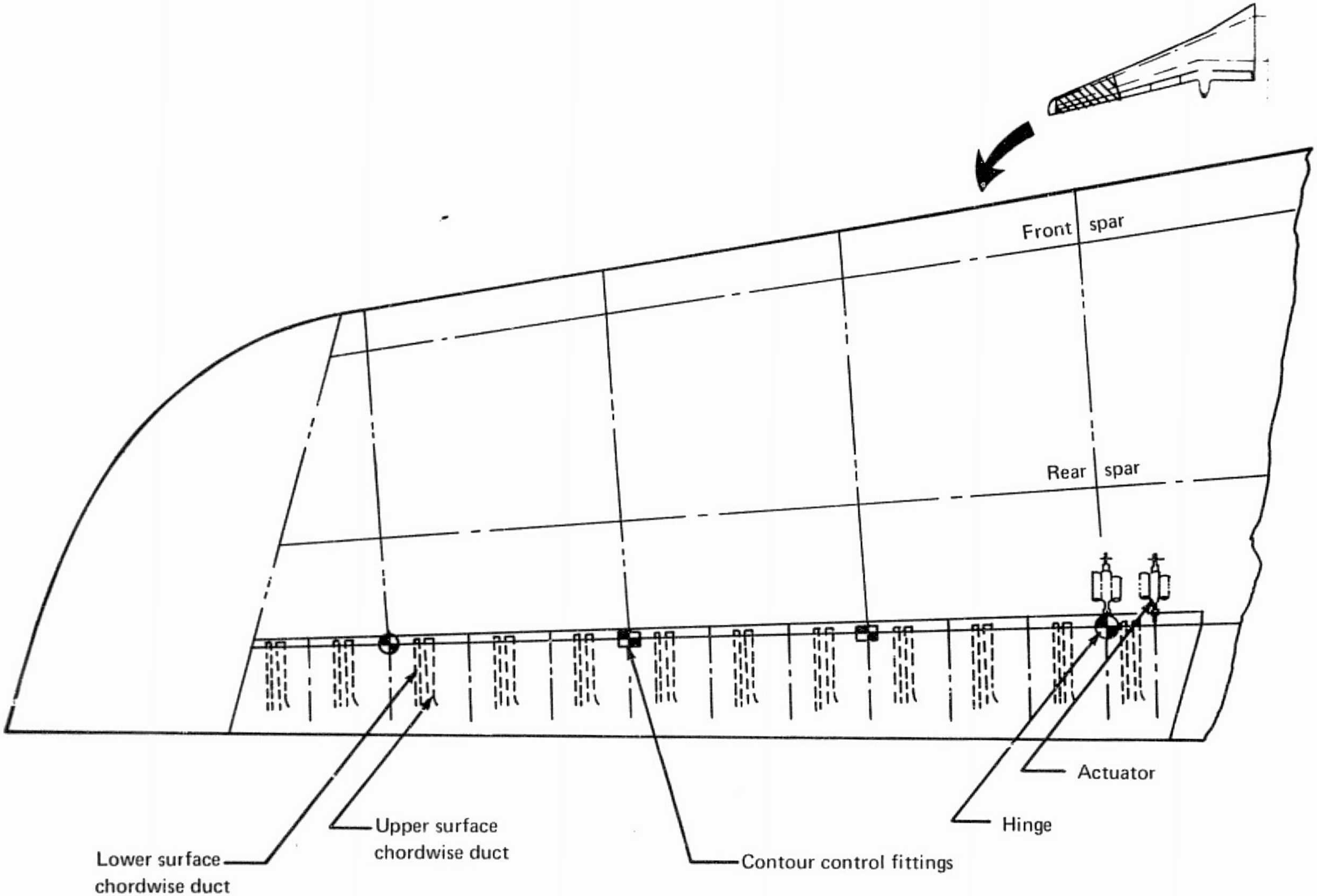


Figure 52. Flaperon Structure and Duct Arrangement

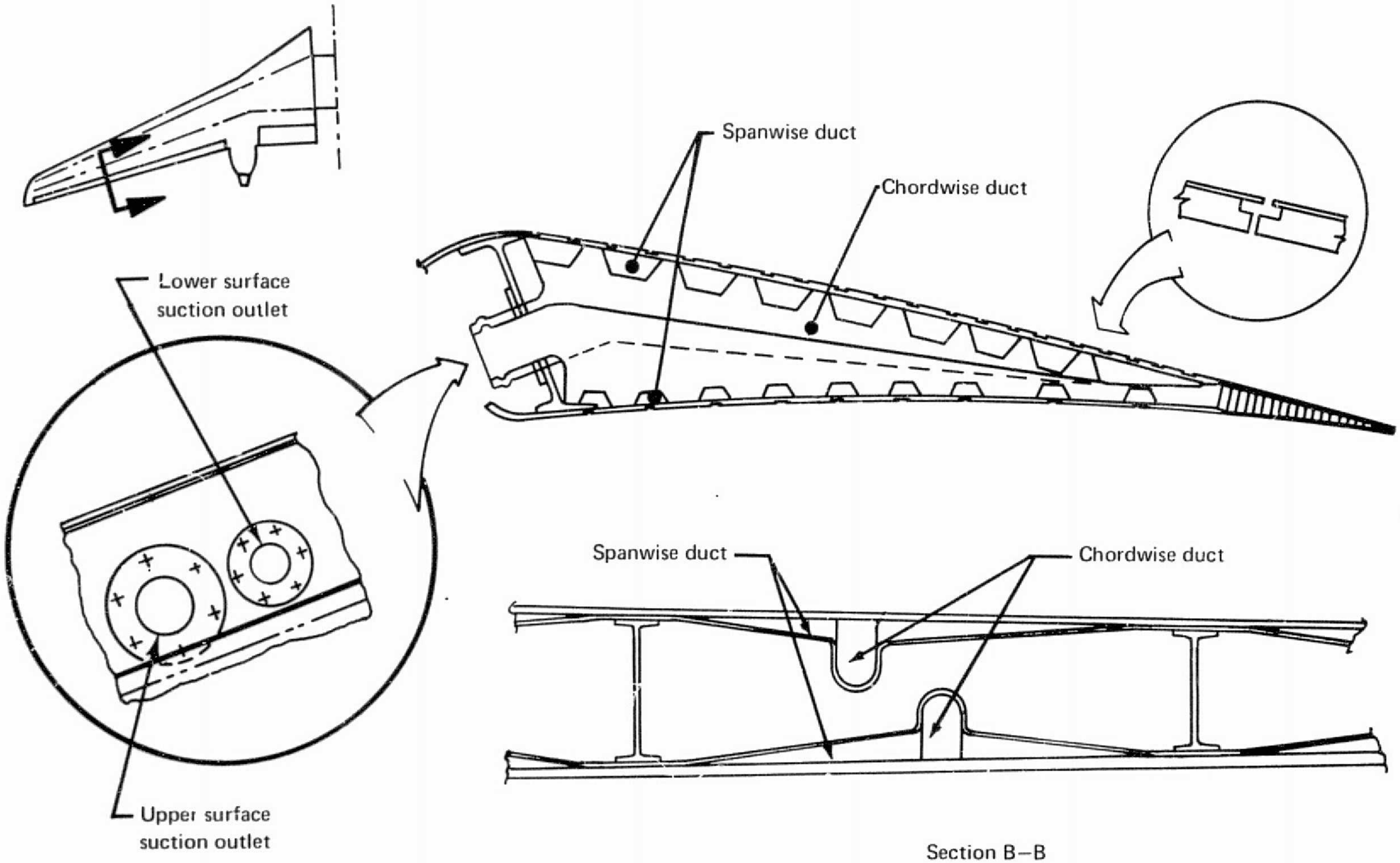
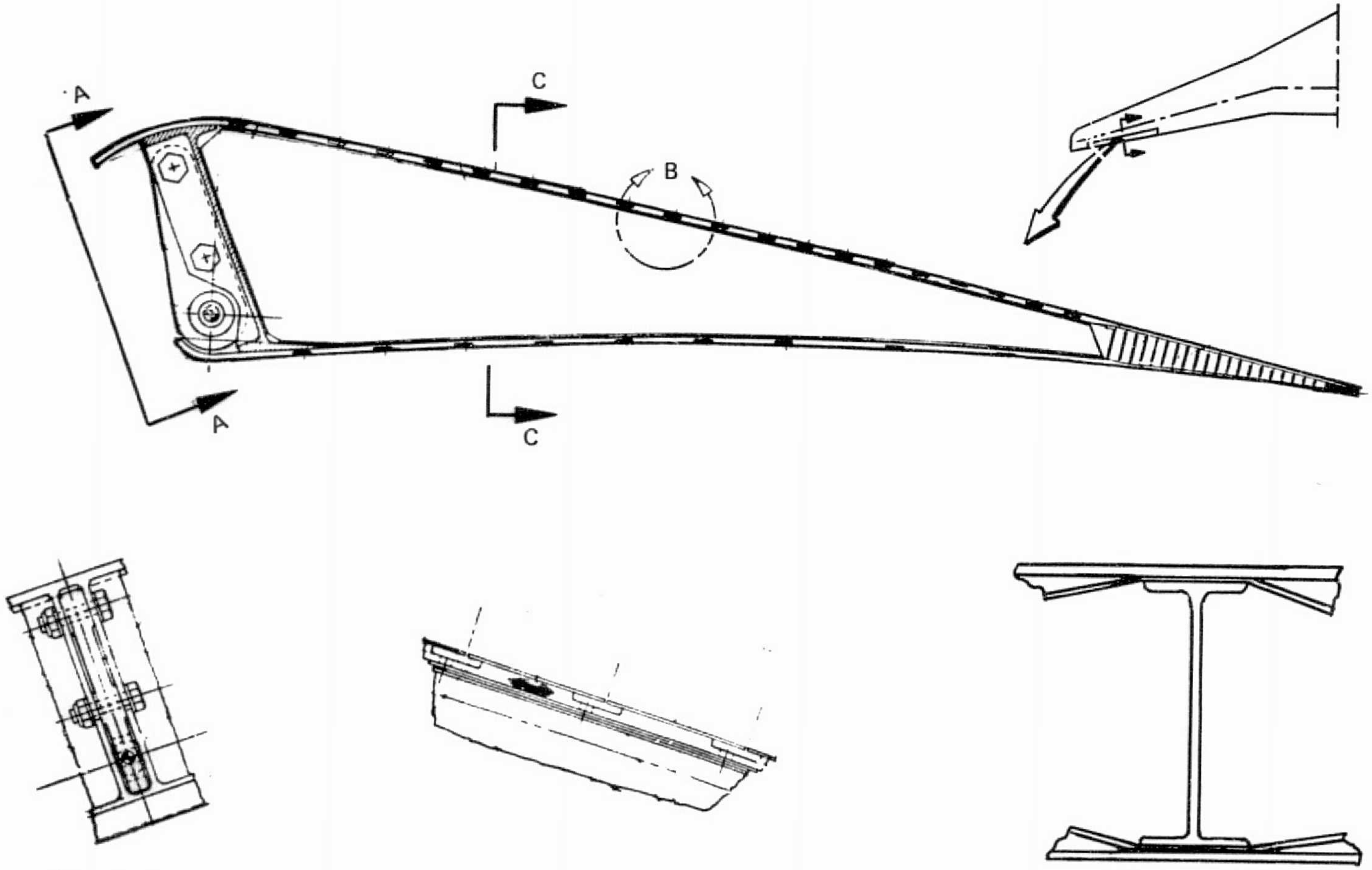


Figure 53. Flaperon Ducting



View A–A

Detail B

Section C–C

Figure 54. Flaperon Hinge Rib

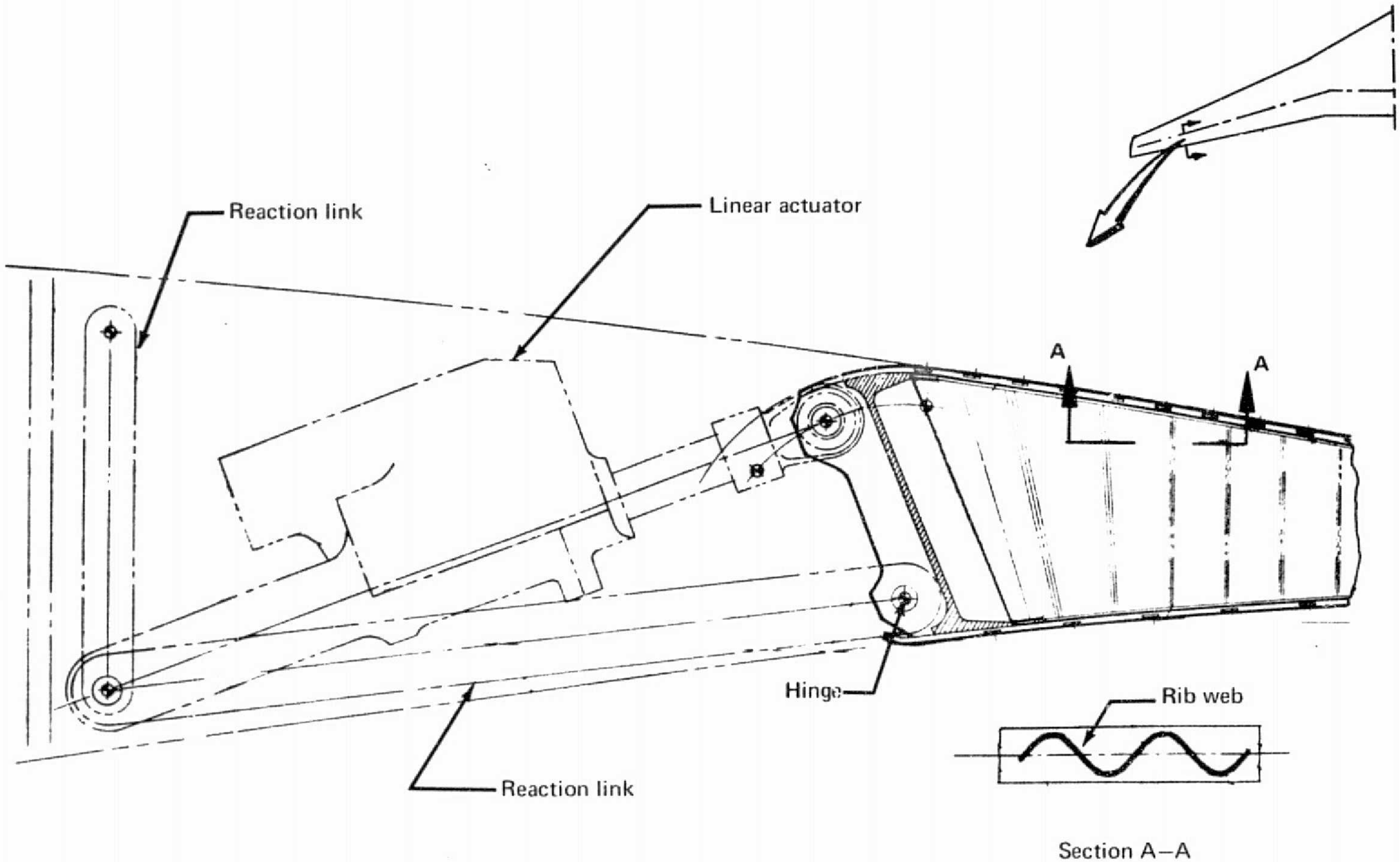
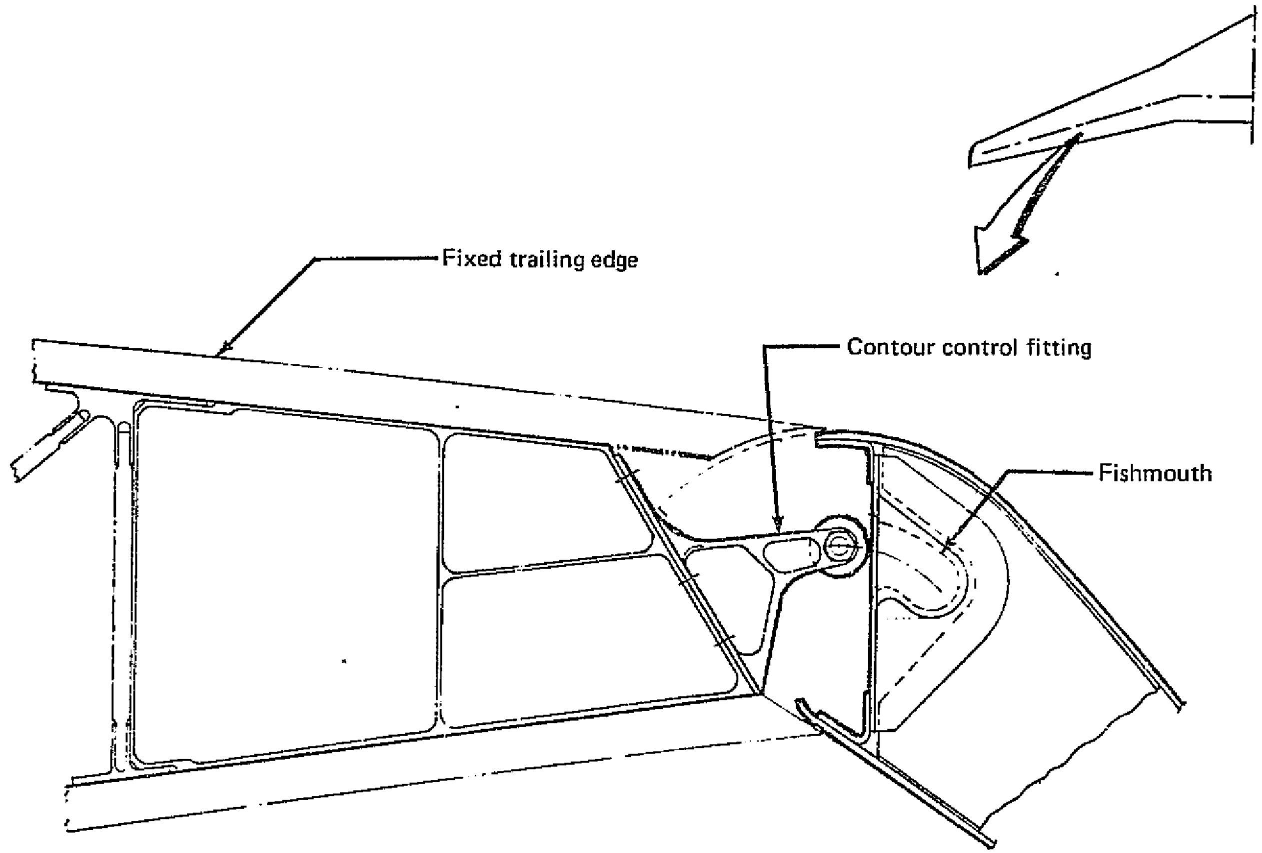


Figure 55. Flaperon Actuation



For control of relative deflections between control surface to wing box

Figure 56. Flap Contour Control Fittings

The access panel design solution adopted for this study is shown in figure 38. Suction slots are provided in the panel surface and are manifolded to a common outlet, connected to the lower surface chordwise duct serving the affected skin panel. The access panel may be largely molded from chopped graphite or glass fibers and epoxy, or may be a sandwich panel with formed ducts added.

Suction flow rate will be locally reduced immediately forward of the access opening and at the aft edge of the access panel to minimize the boundary layer effect of steps and gaps around the panel.

Access panels are of two different sizes, the larger intended to admit a mechanic's body or head and shoulders to the wing box, the smaller serving as access for head and one arm in bays too shallow to permit full entry. In deeper areas of the wing box, some bays may be entered by access panels in the rib webs, eliminating external access in these bays. Selection of thicker airfoil sections makes more bays accessible in this fashion. Drag increase due to the greater wing thickness may be traded against the reduction of complexity and improved laminarization in service.

Location of wing access panels is shown on figure 47.

The main landing gear is shown in figure 47 and the trunnion axis is defined. The landing gear trunnions are supported by a box structure which is cantilevered off the rear spar of the wing box. Large concentrated loads on the order of 680 390 kg (1.5 million lb) are introduced into the upper and lower wing skin panels which are locally padded up with substantial inclusions of 90° fibers in this area. While these loads can in fact be reacted by the wing skins, the trailing edge as defined by the recommended arrangement is too shallow to accommodate strut, flap support, and suction ducts. Design revisions to solve this and other problems are discussed in section 7.4.

7.2.6 RIBS

A typical rib is shown in figure 57. The rib web is a single edge honeycomb panel with laminated graphite epoxy composite face sheets and aluminum honeycomb core. Face sheet fiber orientation is primarily ±45°, with 0° (vertical) fibers provided to resist compression due to wing bending. In more lightly loaded outboard ribs, a single laminate web with honeycomb cored stiffeners or with bonded pultruded stiffeners may be employed. Rib chords and rib to spar attach members are pultruded graphite epoxy composite tee sections. The rib chords are bonded to the wing skin panels during skin panel assembly. It is assumed that the use of primary structural bonds subjected to such tension as generated by refueling pressure will have been validated in the 1985 time period. Spar attach tees are mechanically fastened to the rib web, as are the rib chord members. In ribs other than tank end ribs the lower rib chord will not be sealed at the depression in the bottom skin panel, permitting fuel to flow through. Similar provisions in the upper skin panel serve as tank vents. At tank end rib stations these openings are sealed. In some inboard rib webs, reinforced access openings are provided. Access panels serve to close these openings.

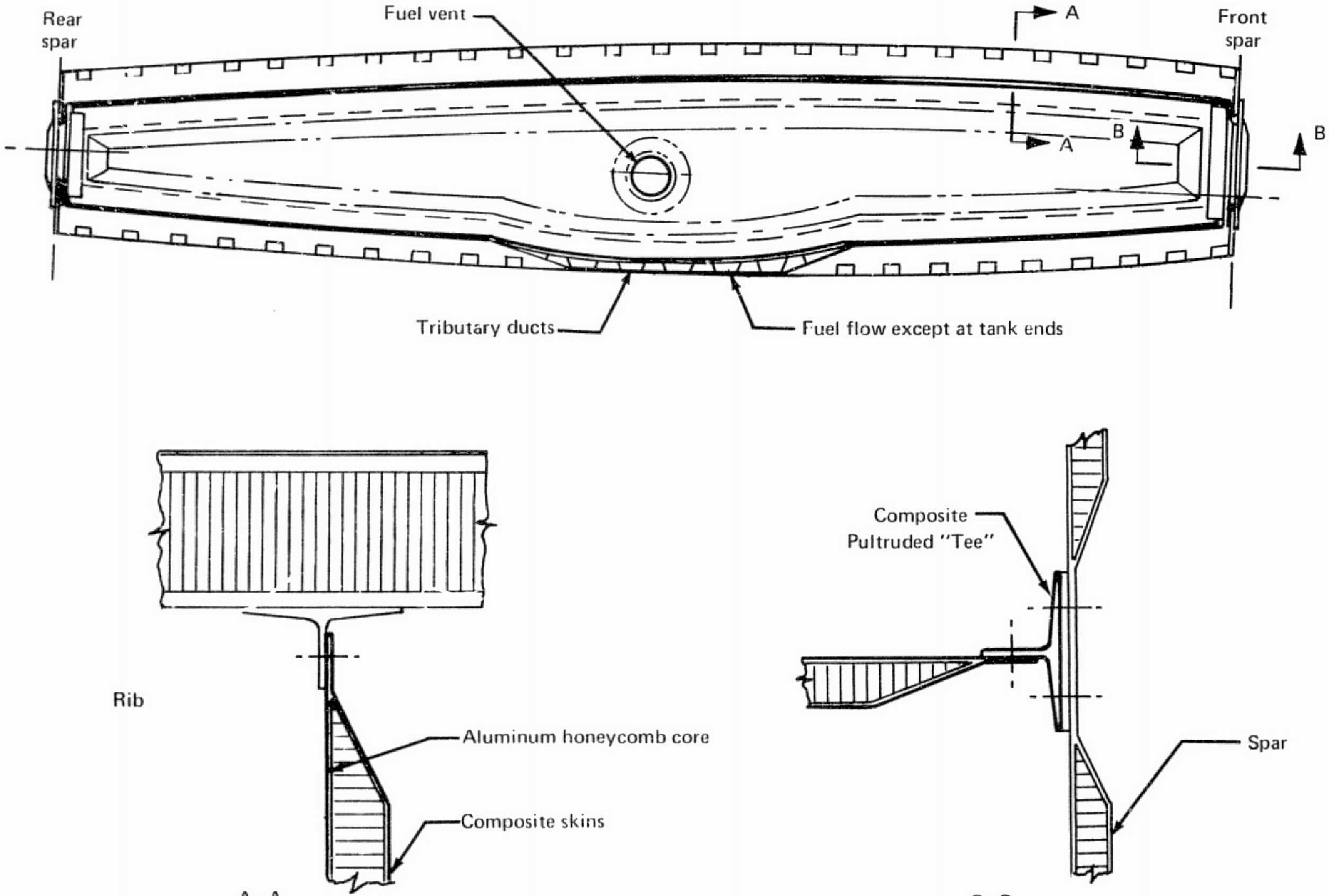


Figure 57. Typical Rib

7.2.7 SPAR PENETRATION

The front and rear spar web are penetrated by a major chordwise duct along the centerline of the suction engine. Approximately 80% of the web is removed in the affected bay. Continuity of wing skins is not disrupted. The spar web is replaced with a machined titanium fitting which serves to beam the loads across the cutout. The front spar fitting is shown in figure 58. Side of body and engine support ribs have increased sections in chords and stiffeners to accommodate the concentrated loads introduced.

7.2.8 SUCTION ENGINE INSTALLATION

The suction engine and compressor are suspended in the suction nacelle by vibration absorbing mounts. The main mounts are near the center of gravity of the installation, resisting vertical, lateral, and rolling loads. The aft mount stabilizes the engine/compressor assembly and accommodates thermal expansion. The upper half shell of the suction nacelle is fixed structure, serving to beam the engine/compressor loads forward to engine mount structure which is attached at four points to the rear spar web, and stabilized by angled members also attached to the rear spar. The lower pair of engine mount attachments are designed to break away under crash conditions, protecting the integrity of the wing box fuel tanks. Below the engine centerline suction nacelle panels are hinged or removable to permit servicing. The major suction ducts aft of the rear spar pass through the engine mount. Figure 59 shows the suction nacelle and engine mount on the wing.

7.3 SUCTION SYSTEM

Suction system sizing was accomplished through the use of the computer programs "LFC", "LFCI" and "LFCDUC". For purposes of preliminary design suction system sizing the wing was divided into 24 sections per side, each section corresponding approximately to one rib bay. Within each section the wing was divided into 10 equal chordwise zones.

Initially the slot velocity was limited to 15.24 m/sec (50 ft/sec) and slot-to-slot Reynolds number was held constant at 100. As a consequence of the low slot velocity, combined with relatively low cruising altitude conditions and an assumed pressure distribution corresponding to a positive maneuver case, the slot widths were unrealistically large.

A second computer run was made with appropriate adjustments to the pressure distribution. Slot velocity was limited to 30.48 m/sec (100 ft/sec.) and a minimum slot-to-slot Reynolds number of 70 was maintained.

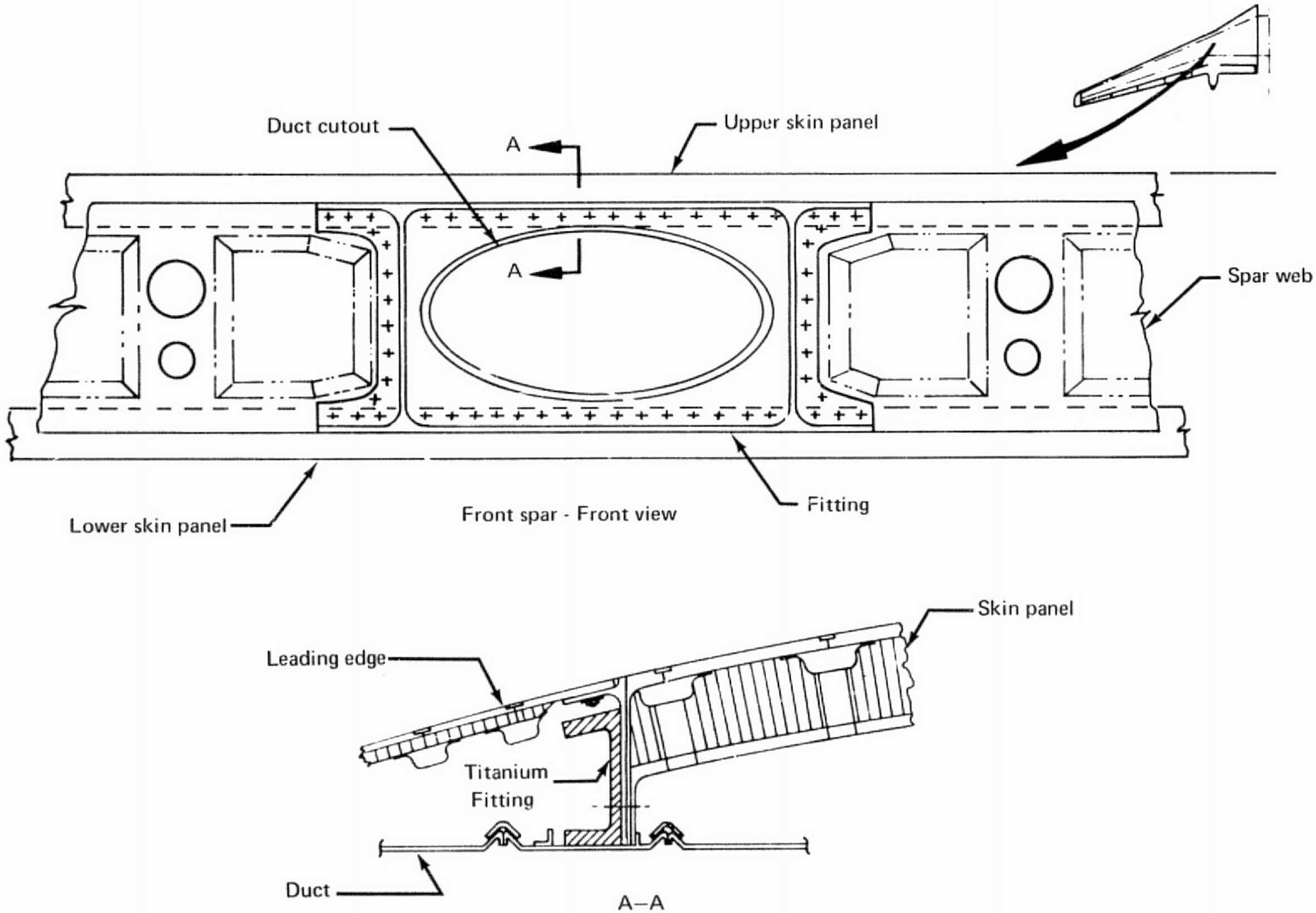


Figure 58. Front Spar Penetration

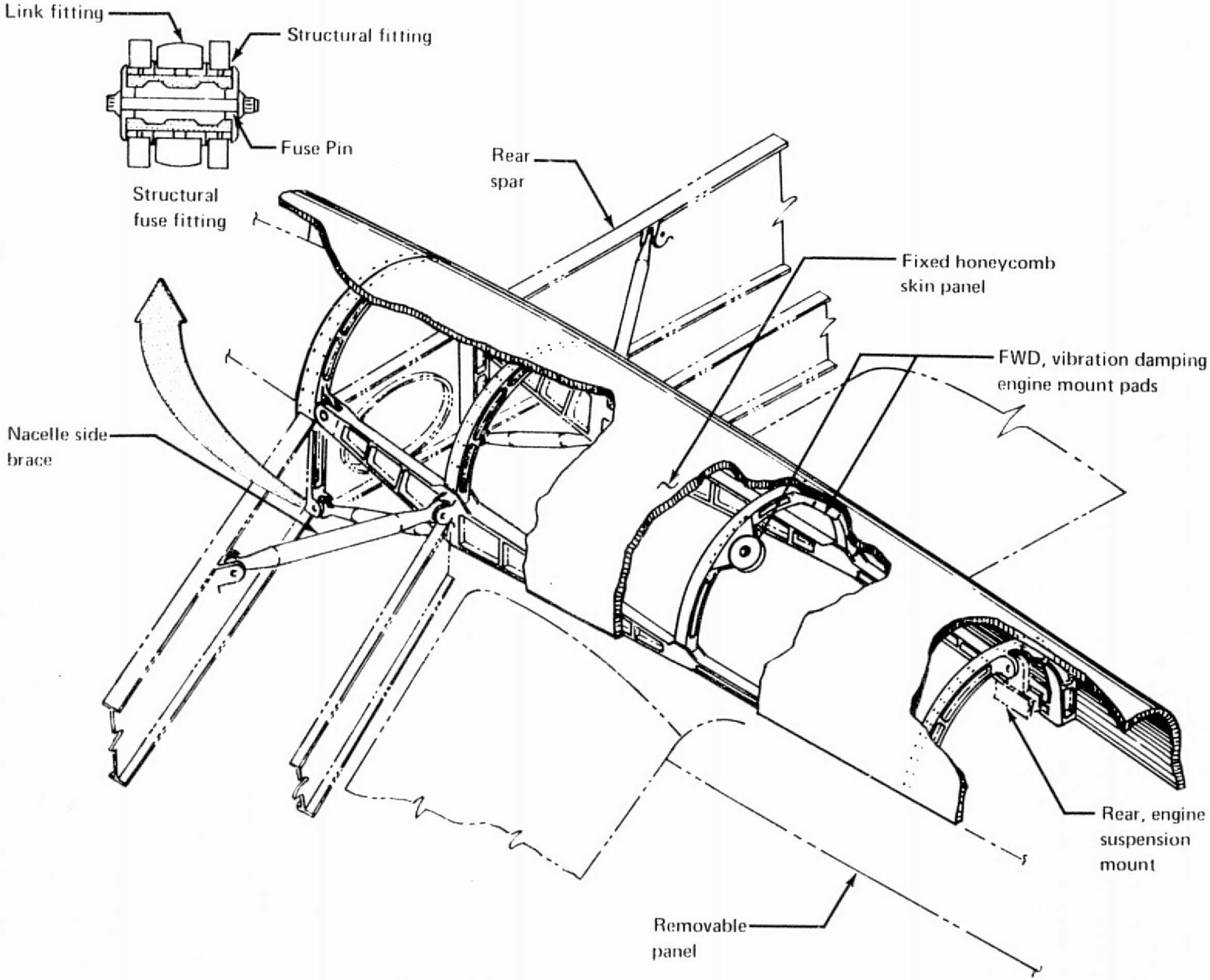


Figure 59. Suction Engine Installation

An additional condition, that the width of any individual slot should remain constant from root-to-tip, was imposed. This condition simplifies slot manufacture. The results of the second run are considered to be representative of a practical LFC suction system.

Free stream and program boundary conditions for both computer runs are listed below. The slot widths and slot spacings determined in the second run are shown in figures 60 through 63.

The suction duct sizing program, "LFCDUC", results in main duct cross section requirements as shown by figure 54. Duct velocity is limited to m = 0.2.

CONDITION	INITIAL RUN	SECOND RUN
Free Stream Velocity	234.112 m/sec (775.718 ft/sec)	Same
Free Stream Dynamic Pressure	9266.55 Pa (1.334 lb/in^2)	Same
Ambient Density	0.33269 kg/m^2 (0.0207692 lb/ft^3)	Same
Absolute Viscosity	15.457758 $N\text{-}Sec/m^2$ (0.322842 x 10^{-6} $Lb\text{-}Sec/ft^2$)	Same
Max. Slot Velocity	15.24 m/sec (50 ft/sec)	30.48 m/sec (100 ft/sec)
Max. slot Reynolds Number	100	Not restricted - (max. resulting 110)

Figure 65 is a cross section view of the suction engine and three stage suction compressor assumed for this study. The suction engine is based on the TF34, with two additional compressor stages replacing the deleted fan section. The three suction compressor stages include variable stators and variable pitch rotor blading.

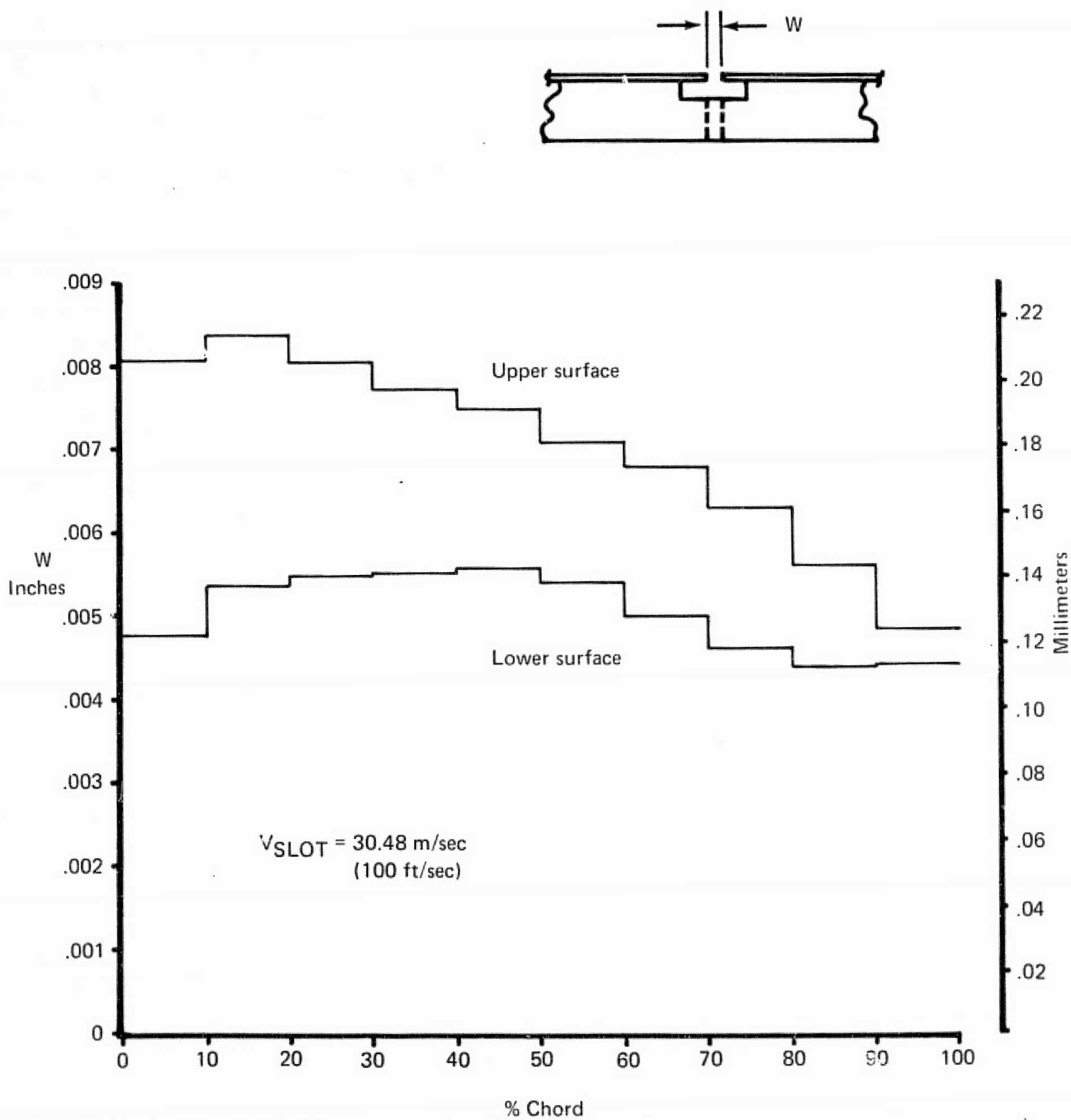


Figure 60. Slot Widths

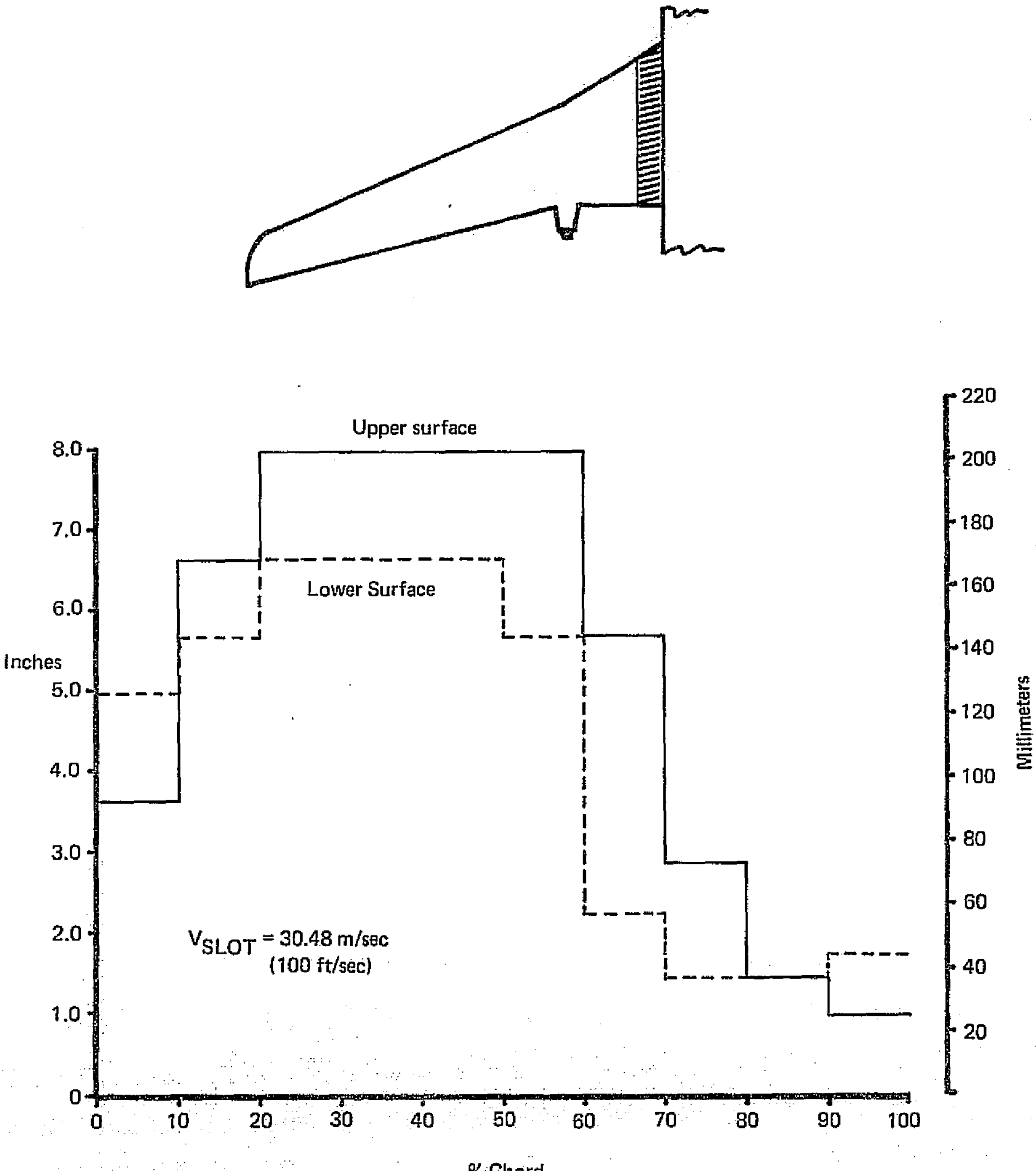
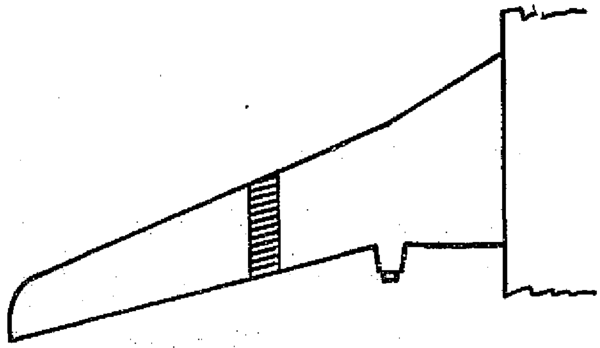


Figure 61. Slot Spacing — Inboard

C, 2



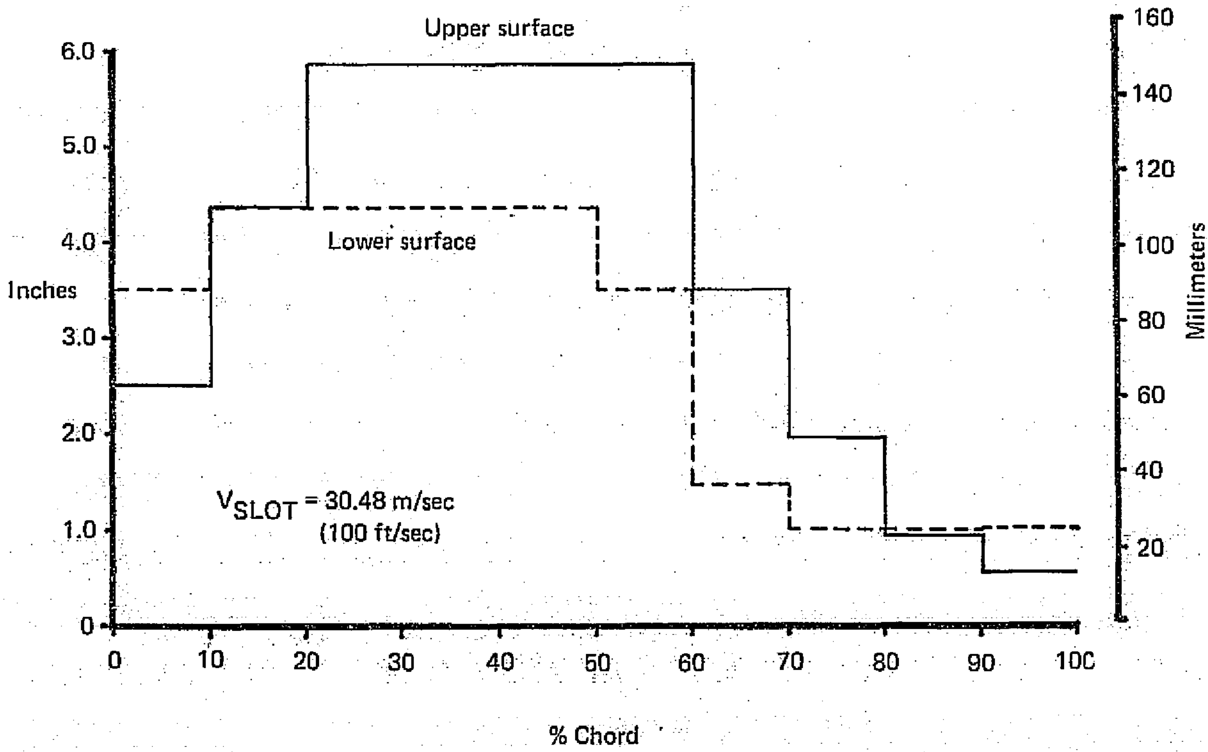
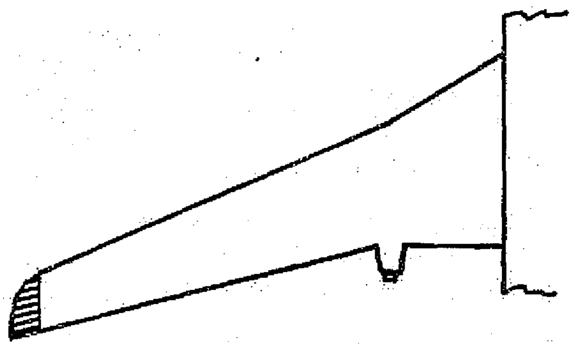


Figure 62. Slot Spacing — Mid-span



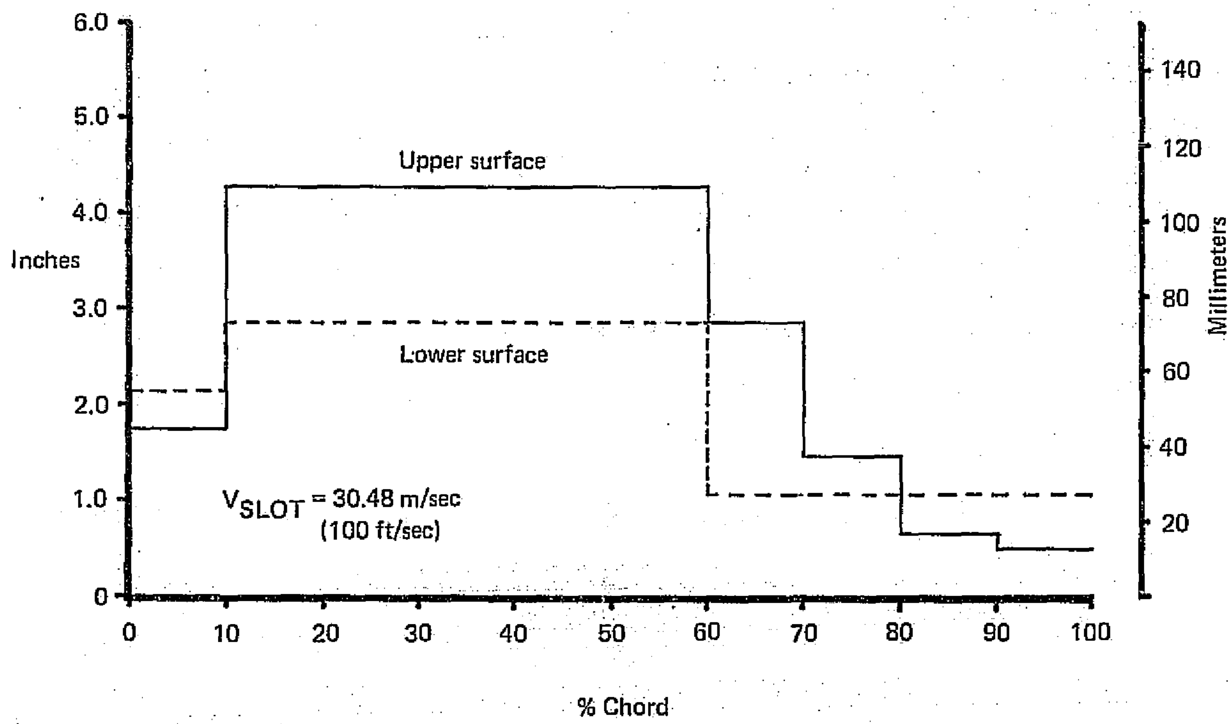


Figure 63. Slot Spacing — Outboard

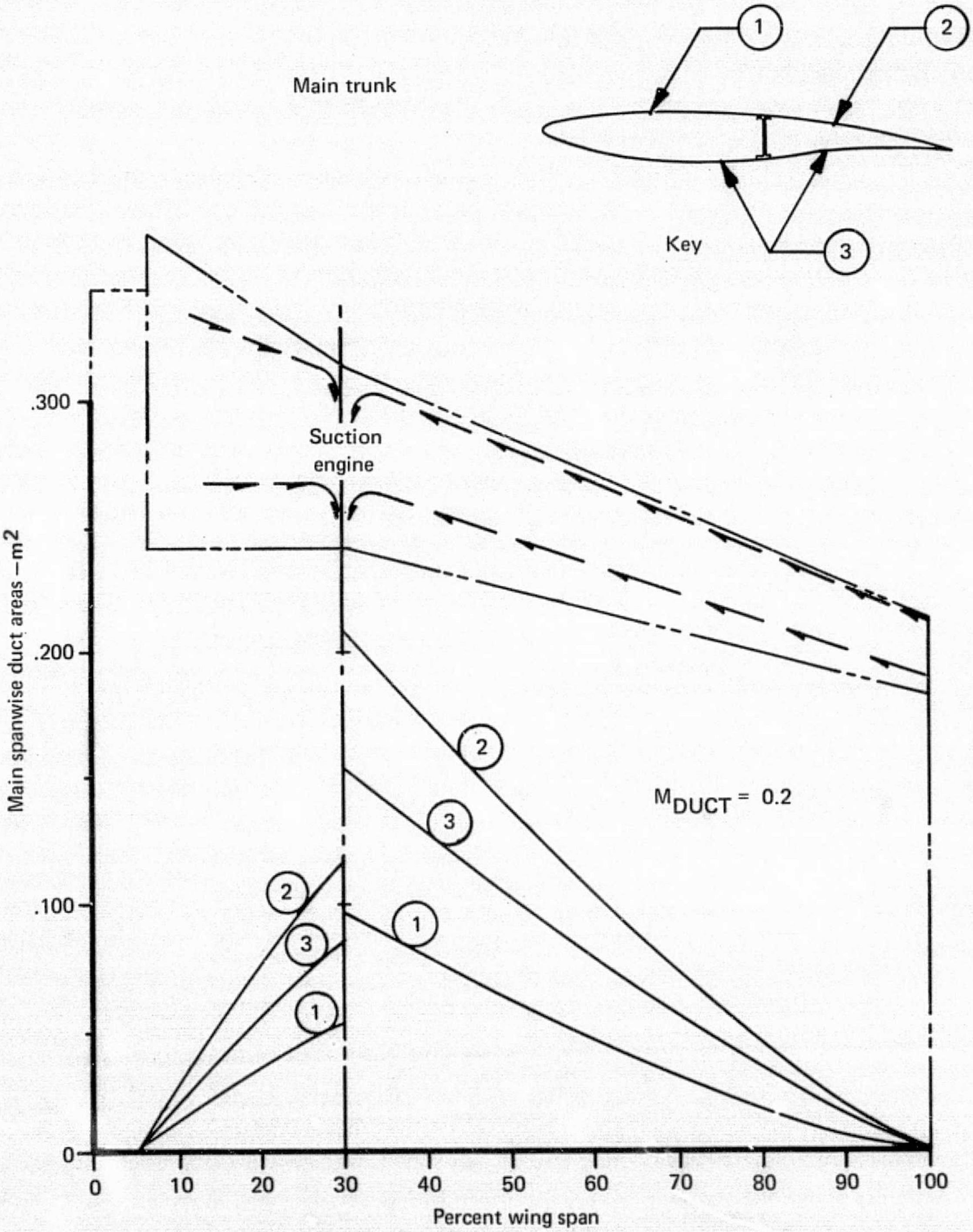


Figure 64. Duct Area

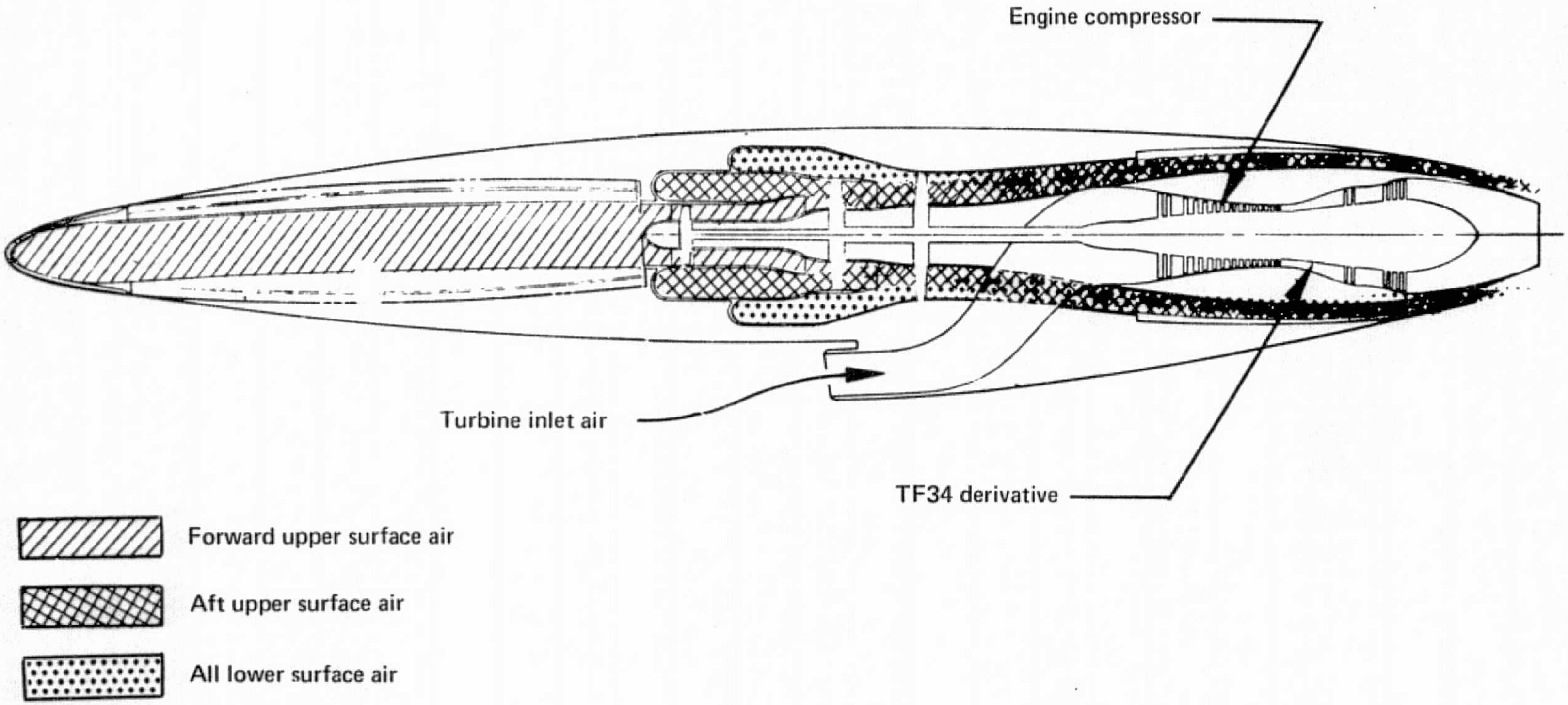


Figure 65. Suction Engine and Compressor

8.0 DESIGN APPRAISAL

8.1 MANUFACTURING APPRAISAL

Manufacturing cost of the composite LFC wing design was appraised on a comparative basis, using a conventional wide-body wing and associated systems as a baseline. The wide-body wing was selected because it most nearly matches the size of the LFC wing, and is the most recently designed of Boeing's production wings. It was assumed that the LFC wing would enter production in 1985 and that the required manufacturing technology will have been developed by that date. A production run of 200 LFC airplanes was assumed.
The cost assessment was made by a number of individuals from different manufacturing areas who collectively established complexity factors for each major component of LFC wing structure and systems, using the complexity of the conventional wing and systems as a baseline, with a value of 1.00. The complexity factors thus developed were then applied to the baseline manufacturing labor distribution for the components evaluated, to obtain a weighted labor cost factor for the entire LFC wing design. The results of the appraisal are shown in figures 66 through 68 and table 12. Figure 66 shows the comparative manufacturing labor cost of all the equivalent components of the conventional baseline and LFC wing. Components related to the wing mounted engines of the baseline were excluded. The complexity of the LFC wing was assessed as twice that of the conventional wing.

Figure 67 shows the percentage of total airplane manufacturing labor cost which is attributed to the structure and systems of the wing. Figure 68 makes a similar comparison for wing structure alone.

Areas in which future trade studies can lead to reductions in production complexity and cost include the front and rear spar skin panel joints and attachment of leading and trailing edge structure; the productibility of slots and collection ducts in fixed skins and control surfaces; the attachment of body skins to wing panels in the side of body area; provision of manufacturing joints in the fixed leading edge; and simplification of laminarized wing access panels.

The preliminary LFC wing design was judged to represent a producible wing which will be more costly than a conventional wing of similar size. The increased cost will be due primarily to the added complexity of the LFC provisions, rather than the use of composite structure.

8.2 WEIGHT APPRAISAL

Estimates of the weight of the LFC airplane were made by extension of conventional preliminary design weights techniques to account for the use of composite structure and the incorporation of LFC features.

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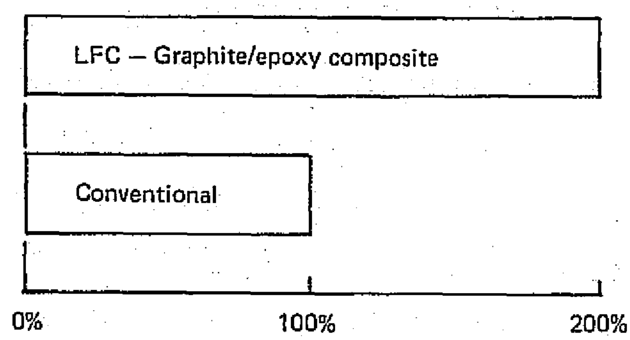

Figure 66. Wing Manufacturing Labor Cost

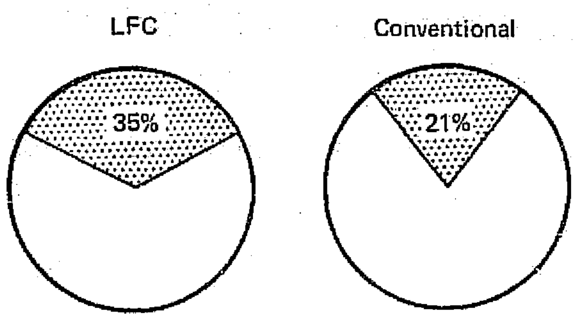

Figure 67. Wing Structure and Systems — % of Total Airplane Labor Cost

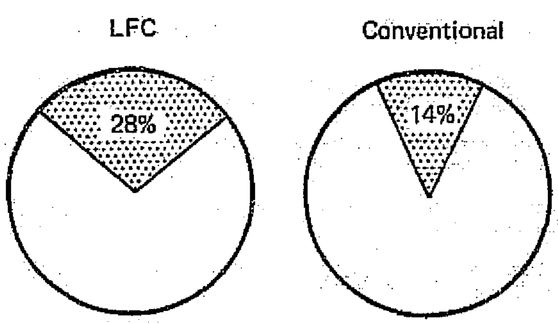

Figure 68. Wing Structure Only, % of Total Airplane Labor Cost

Table 12. LFC Wing Manufacturing Labor Complexity

	LFC wing complexity factor
I. Structure	2.4
A. Fabrication	4.7
Stub	2.0
Fixed leading edge	8.0
Fixed trailing edge	7.4
Spars	1.0
Ribs	1.0
Upper panels	8.0
Lower panels	8.0
L.E. control surfaces	0
Landing gear doors	4.0
Tip	1.0
Spoilers	2.0
Trailing edge flaps	1.0
Ailerons	4.0
B. Minor assembly	1.5
C. Major assembly	0.8
II. Systems	1.1
A. Hydraulics	0.7
B. Fuel	1.3
C. Electrical	1.0
D. Electronics	1.0
E. Controls	0.7
F. LFC systems	1.3
III. Total LFC structure/systems	2.0

8.2.1 METHOD OF ANALYSIS

Initial estimates of takeoff gross weight (TOGW), fuel weight, and landing weight were combined with the configuration geometry (Section 4.1.3) and engine data, and an initial operating empty weight (OEW) was estimated. This OEW estimate was established by Class I (preliminary design) parametric methods. With the OEW estimate as a base the following adjustments were made:

Wing weight was adjusted for high aspect ratio and low thickness to chord ratio, using results of previous studies on another of Boeing's designs.

The weight of leading edge devices was removed. A 20% reduction in wing box weight was made. accounting for the use of composite structure in this area. This reduction was based on recent Boeing studies of redesigning a current production wing in composites. A unit weight penalty of 4.149 kg/m^2 (0.85 lb/ft^2) was applied to the total laminarized area, 585.3m^2 (6300 ft^2). This unit weight penalty was developed as follows:

Weight of titanium, applied as slot tapes (section 5.4.1, method 3) 0.38 mm (0.015 in) thick by 38 mm (1.5 in.) wide	0.88 kg/m^2 (0.18 lb/ft^2)
Weight of tributary ducts at 0.043 kg/m (0.029 lb/ft)	1.71 kg/m^2 (0.35 lb/ft^2)
Weight of chordwise ducts	0.342 kg/m^2 (0.07 lb/ft^2)
Weight of main spanwise ducts	0.976 kg/m^2 (0.20 lb/ft^2)
Allowance for ineffective material near slots	0.244 kg/m^2 (0.05 lb/ft^2)
Total	4.149 kg/m^2 (0.85 lb/ft^2)

The weights of chordwise and main spanwise ducts were estimated from parametric weight data and are conservative in that no weight credit is taken for integration of main ducts and structure. Additional conservatism is introduced by the allowance for ineffective material near the slots.

Applying the same set of assumptions to the baseline skin concept (section 5.4.1, method 1 or 2), with the same skin thickness, the weight of titanium is increased to 1.717 kg/m^2 (0.35 lb/ft^2). The unit weight penalty is increased to 4.98 kg/m^2 (1.02 lb/ft^2).

The installed weight of the suction engines and compressors is added to the unit weight penalty. The installed weight of the TF34 derivative power plants and the 3 stage suction compressors is 2767 kg (6100 lbs). An additional 45.4 kg (100 lbs) is added for transition ducts and plenums in the suction compressor area.

An allowance of 272 kg (600 lb) is added to the airplane weight to account for the empennage suction compressor, driven by the center (main) engine.

The airplane drag was estimated using skin friction coefficients developed in a previous study for the case of "optimum uniform suction" of the laminarized areas.

The drag and OEW estimates were used as inputs to the range equation. The resulting range was compared to the required range. Based on the difference between the resulting and required ranges, a new cycle of calculation was started. The process was continued until the calculated range agreed with the range required for the LFC mission.

8.2.2 WEIGHT ANALYSIS RESULTS

The preliminary design weight estimate, for the completed cycle of airplane sizing, yielded the following results:

OEW.	86 183 kg	(190 000 lb)
TOGW.	156 489 kg	(345 000 lb)
Wing Weight.	21 772 kg	(48 000 lb)

(Wing weight does not include installed weight of suction powerplants and compressors.)

8.2.3 STRUCTURAL WEIGHT SUMMARY

Weights of the structural elements of the initial LFC wing are summarized in table 13. The summary includes wing structure outboard of the side of body splice, the side of body rib, and the wing splice members. LFC system and flight control system weights are included. Weights of other systems in the wing were not considered to be sensitive to the incorporation of LFC features, and are not accounted for in this summary.

8.3 OPERATIONAL APPRAISAL

Operational aspects of the LFC wing preliminary design are considered below. With the exception of the LFC features, the selected airplane layout was conventional. The laminarized tailplanes were assumed to be similar in design of suction features to the wing. For this reason, operational appraisal is limited to features of the wing.

8.3.1 ACCESSIBILITY

The LFC wing was designed to have the same degree of accessibility as a conventional metal wing. The leading edge may be removed, in segments, to expose the forward surface of the front spar web, the chordwise upper surface duct outlets, the forward upper surface duct walls and the interior surface of the leading edge structure. There are no movable leading edge components to be inspected or maintained. Access to the large chordwise duct at the suction engine is gained by removing a leading edge segment. Access to all of the interior of the wing box is gained by removing the lower surface access panels and the rib access panels, if required.

Table 13. Wing Structural Weight Summary — 138 078 kg (300 000 lb) T.O.G.W. Airplane

Item	Weight		Weight / airplane	
	kg	lb	kg	lb
Wing box	7127.7	15 717	14255.3	31 434
Upper skin panel	2654.8	5854		
Lower skin panel	2362.7	5210		
Spar webs	157.8	348		
Ribs	462.1	1019		
Fasteners	1034.9	2282		
Titanium cover	228.6	504		
Sealant, protective coating and misc.	226.8	500		
Leading edge	440.4	971	880.7	1942
Honeycomb, L.E. panels	269.4	594		
Titanium cover	70.3	155		
Internal duct provisions	45.4	100		
Fasteners	55.3	122		
Fixed trailing edge	837.2	1846	1674	3692
Upper skin panels	176.4	389		
Lower skin panels	165.1	364		
Chordwise stiffeners	23.6	52		
Titanium cover	101.1	223		
Duct wall panels	104.8	231		
Hinge ribs	29.5	65		
Fasteners	236.7	522		
Control surfaces	826.4	1822	1652.6	3644
Outboard (low speed) flaperon	46.3	102		
Inboard (high speed) flaperon	103.9	229		
Flaps	433.1	955		
Flaperon actuation system	137.0	302		
Flap actuation system	106.1	234		
Total structure	9231.6	20 356	18462.9	40 712
Suction engine and compressor	1895.6	4180	3791.3	8360
Engine and compressor - installed on wing	1850.3	4080		
non-structural ducts in nacelle area	45.4	100		
Total structure and LFC installation	11 127.2	24 536	22254.2	49 072

The lower skin surface of the fixed trailing edge consists of removable panels between each rib bay. When these panels are removed the interior of the main lower surface suction ducts and the rear surface of the rear spar web are exposed. The interior of the aft-upper surface main duct is exposed by removing access panels in the duct wall. Control system wiring and hydraulic runs, actuators, hinges, and suction connections also are made accessible by removing the wing lower surface panels.

The interior of the chordwise duct system may be inspected from its outlets in the main duct system.

Inspection of tributary ducts and plenums, during major inspection periods, may be accomplished by borescope, with access gained at panel ends, and at special access plugs in nonlaminarized areas of the basic skin panel (under cowl fillets, side of body fillet, etc.).

The engine and compressor may be serviced through access doors on the lower half of the suction nacelle. The engine and compressor are removed by removing the entire lower cowl assembly.

The main landing gear is accessible through the open wheel well as in conventional transport aircraft.

The provision of access to all surfaces of the structure and all items of equipment is equivalent to access provisions on current conventional wings.

8.3.2 SIMPLICITY

The LFC wing has far fewer structural parts than a similar wing in conventional metal construction. The suction ducting is integrated into the basic structure to a high degree. The two suction engine and compressor installations are little more complex than modern high bypass ratio fan engines of equivalent size.

The simple control surfaces have no linkage systems other than actuator reaction links, so few bearings need be provided. If reliable non lubricated control bearings exist when the LFC airplane is built they may be used.

Otherwise, the few lubrication points required will mean a reduction in routine maintenance. The minimization of external skin joints will mean minimum inspection and maintenance of the aerodynamic quality of the LFC surface.

8.3.3 INSPECTION AND REPAIR

All structural surfaces are accessible to visual inspection. Plenums and tributary ducts may be inspected by borescope techniques already in use. The integrity of honeycomb panels is currently verified by ultrasonic techniques which will be applicable to the LFC airplane. It is anticipated that other inspection techniques will evolve from composite structure development programs.

Repair techniques for composite skins and honeycomb panels have been developed, and are applicable to internal surfaces of the wing structure. A satisfactory repair technique for the exterior face sheet of the wing skin panel must be realized by development and testing of thick panels of high strength composite.

Local repair to suction slots may be accomplished by removing a circular patch of the titanium surface and bonding in a previously prepared slotted circular patch which is carried on a removable backing strip. This method is comparable to the slot tape concept of section 5.4.

Leading and trailing edge panels which are most vulnerable to handling, damage, ground vehicle strikes, etc. are easily removed, in segments, for repair or replacement.

8.3.4 DAMAGE RESISTANCE AND DURABILITY

The thick composite face sheets made possible by the integrated skin panel design will be resistant to damage by minor impact. The titanium outer skin provides protection against ultraviolet light and other environmental agents. The titanium is in itself resistant to attack by commonly encountered fluids and atmospheric aerosols. In suction plenums and ducts, where composite surfaces are exposed, suitable protective treatment must be developed and applied.

The high strength composite structure does not experience cyclic fatigue damage in the same sense that conventional metal structures do. The effects of cyclic loading on panels incorporating perforated softening strips should be assessed by testing, however.

Advanced composite test specimens are being subjected to environmental exposure in several NASA flight service programs, as well as in ground-based outdoor exposure and laboratory immersion tests. After more than two years in flight service and over a year of ground exposure and immersion testing it has been concluded that there is no significant effect of environment on composites for subsonic commercial aircraft applications (Reference 8).

8.4 DESIGN ITERATION

Following the weight appraisal, a second iteration of wing layout was initiated to resize the wing for higher gross weight. This iteration, normal to a continuing preliminary design study, was not carried to completion but was sufficiently exercised to expose differences from the original layout.

For the second iteration, the original wing plan form and loading were maintained. The resulting wing has a reference area of 332.5 m^2 (3 579 ft^2) and a span of 63.15 m (207.2 ft). Wing geometric twist was eliminated from the inboard 30% of half span. Aerodynamic twist would exist in this area, due to the necessary tailoring of the wing root profile.

In light of the LFC consultant's conclusion that the original wing's thickness ratio might be increased with small aerodynamic penalty, a 10% increase in t/c was assumed. Figure 69 is a structural arrangement of the revised wing. Characteristics of the original and second iteration wings are compared below:

TAKEOFF GROSS WEIGHT	136 078 kg (300 000 lb)		156 489 kg (345 000 lb)	
SPAN	59.44 m	195 ft	63.15 m	207.2 ft
M.A.C.	5.5 m	18.08 ft	5.87 m	19.27 ft
TAPER RATIO	.25		.25	
SWEEP @ .25 c	22°		22°	
$AREA_{ref}$	293.4 m^2	3158 ft^2	332.5 m^2	3579 ft^2
$AREA_{gross}$	313.2 m^2	3371 ft^2	354.8 m^2	3819 ft^2
FUEL QUANTITY (NET)	46 135 kg	101 711 lb	62 758 kg	138 358 lb
t/c @ η = .3	.11		.121	

Fuel and duct volumes were assumed to be adequate, in view of conservative assumptions originally made. For a first estimate of fuel volume, skin panel thicknesses were assumed to be the same as for the original wing, at corresponding spanwise stations. The beneficial structural effects of increased wing box depth were assumed to approximately equal the increased bending material requirements of a larger wing carrying greater weight. With tank ends located at the same percent of span, the resulting fuel capacity is summarized below. Gross internal volume is assumed 85% usable.

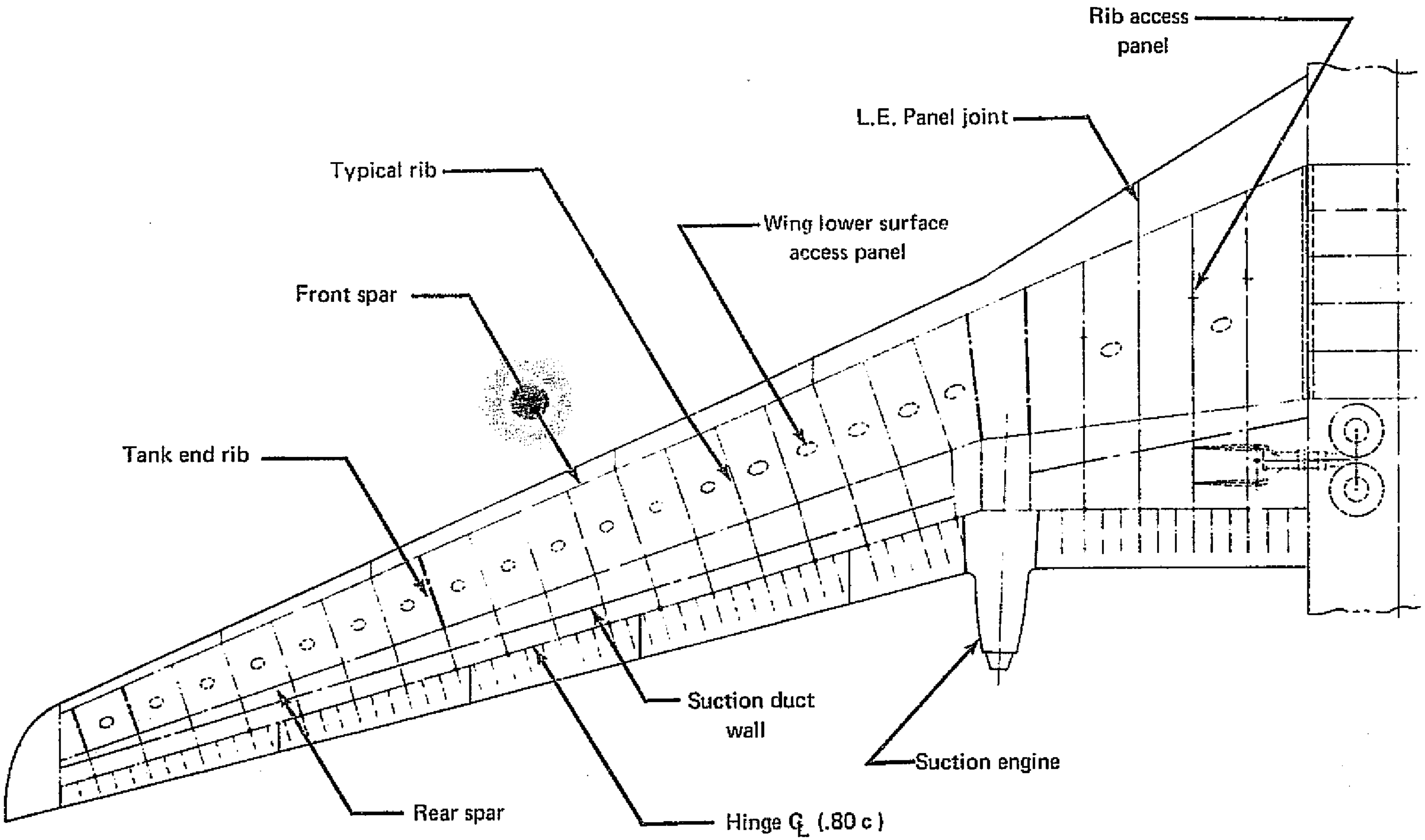

Figure 69. Structural Arrangement – Second Interation

TANK	QUANTITY			
	KG	POUNDS	METERS^3	U. S. GALLONS
Center	16 741	36 908	21.5	5 678
Inboard	34 750	76 609	44.6	11 786
Outboard	19 483	42 953	25.0	6 608
Total	70 978	156 470	91.1	24 072

The fraction of fuel to airplane takeoff gross weight is increased from .34 to .40 by the second iteration of wing size.

The inboard rear spar is angled forward permitting the main landing gear to retract inward at right angles to the body, which facilitates emergency free fall extension as well as placing the retracted strut in a deeper portion of the wing profile.

The main landing gear installation is shown in figure 70. Details of the landing gear support structure are shown in figure 71.

The suction engine nacelle is not changed in size, therefore it is more nearly buried in the larger deeper wing. The suction engine center line remains at approximately the same spanwise percent station, and the linear distance from the rear spar to the section compressor face is unchanged.

Rib spacing in the outboard panel is unchanged, resulting in the addition of one rib bay. Inboard ribs are respaced, the number of bays remaining the same. The new center section is actually shorter than the original, due to the angled inner rear spar, however, the number of center section bays is unchanged. Outboard, the front and rear spars remain on 10% and 55% chord respectively.

If the preliminary design process were carried to completion, several more iterations of airplane weight estimation and wing resizing would be carried out. Other portions of the airplane would also be revised. However, the structural techniques and methods of analysis shown for the original wing retain their validity, although they would be refined in detail.

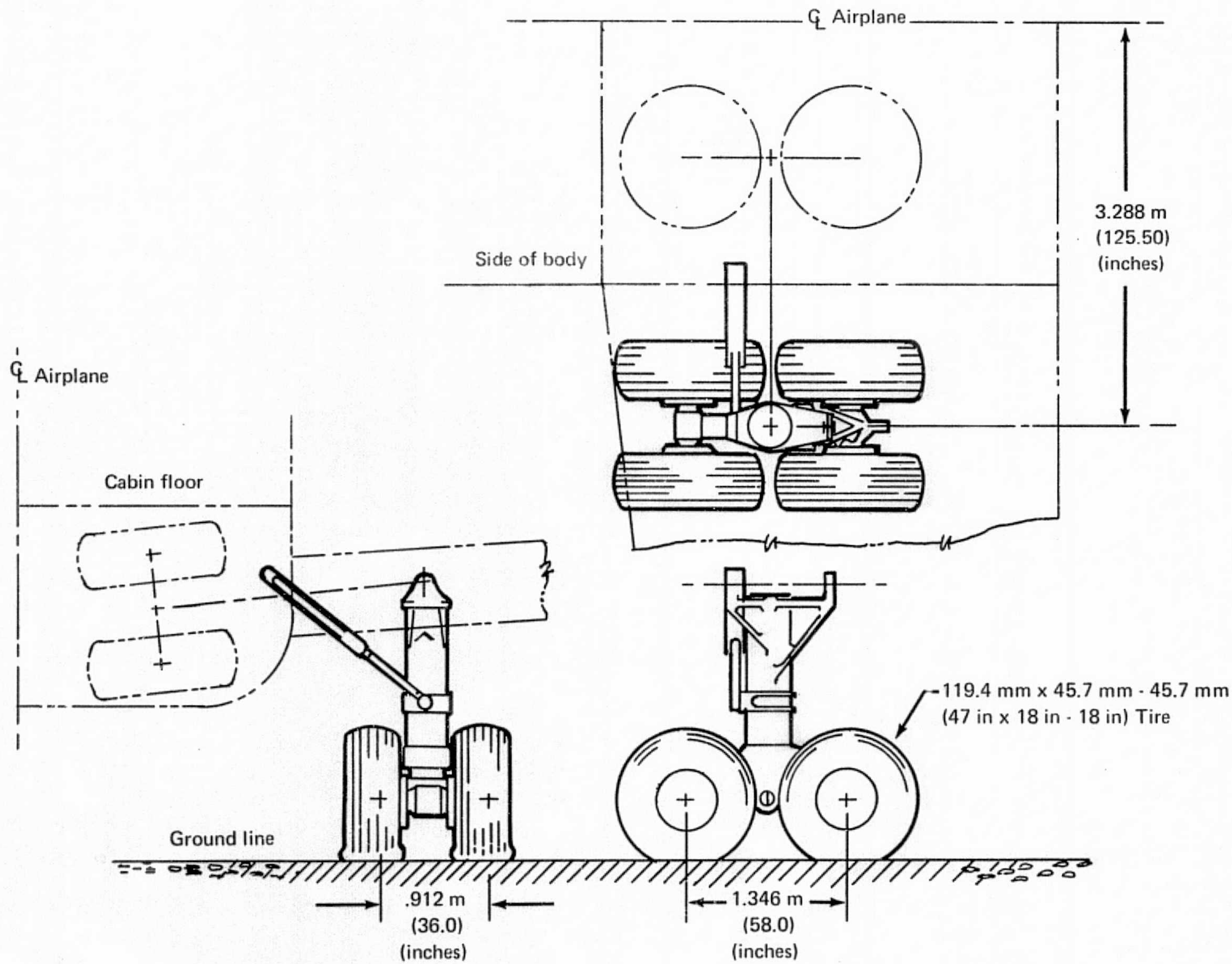

Figure 70. Landing Gear Arrangement

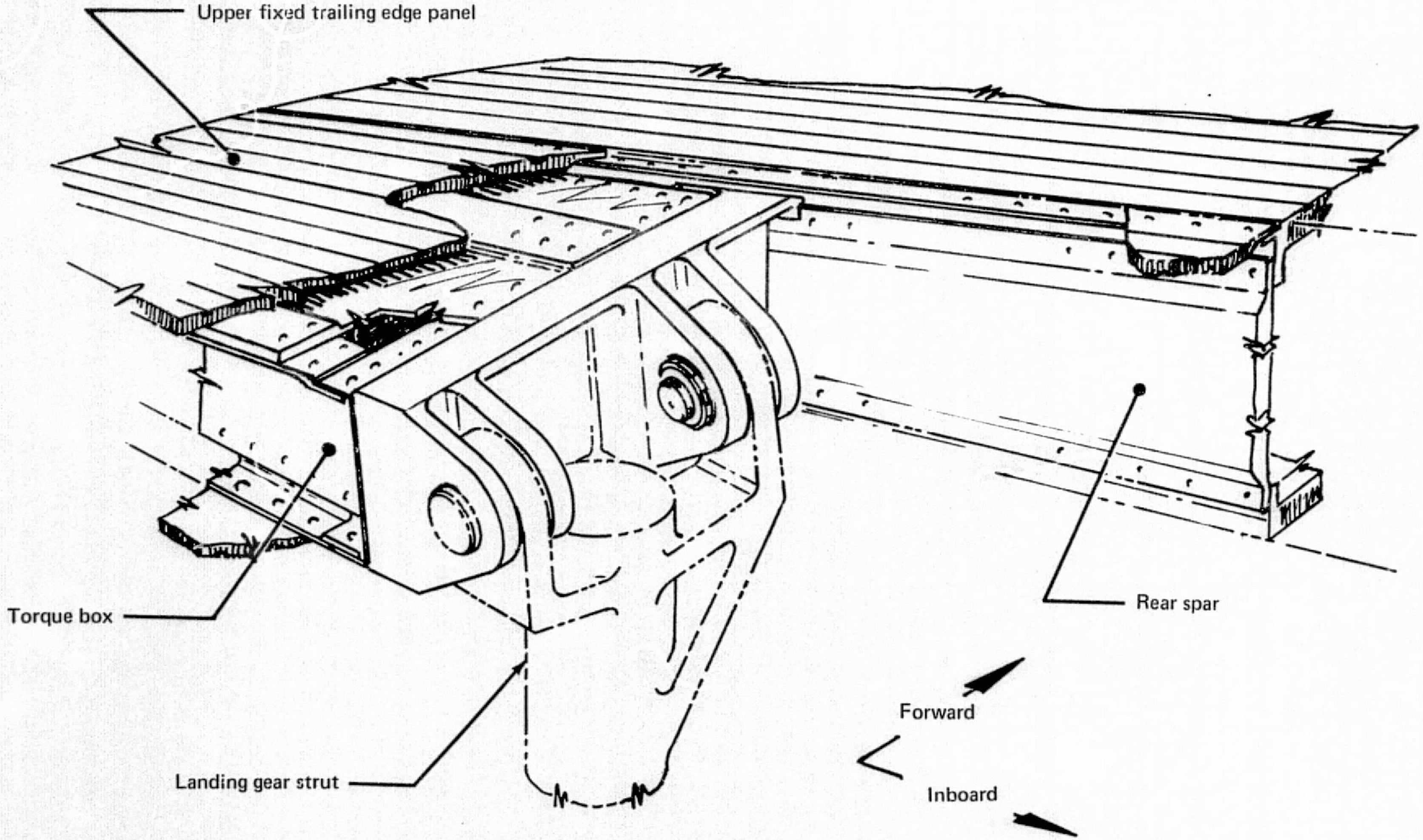

Figure 71. Landing Gear Torque Box

9.0 CONCLUSIONS

As a result of the preliminary design and subsequent design assessment conducted under this contract, the following conclusions have been drawn:

1. There is no technological barrier to the production of laminar flow wings of composite materials for long range transport aircraft. The technical problems that were found to exist are problems of the sort which major aircraft manufacturers are well equipped to resolve.

2. High strength composite primary structure has advantages over conventional metallic structure in LFC wings. Some of these advantages are related to the improved stiffness and lower density of such composites, and can be also realized in the design of non-laminar large wings. Other advantages are peculiar to the incorporation of LFC features in the wing structure. Some of the latter are summarized below:

- Suction tributary ducting and suction plenums may be integrated in the wing skin panel layup with little increase in complexity.
- The metallic material which is necessary for electrodynamic protection can be made to serve as the LFC surface. Use of such an external metallic layer also protects the composite against erosion, impact damage, degradation by ultraviolet radiation, and the action of airborne contaminants.
- The close conformity to contour which is essential for the attainment of laminar flow can be achieved by use of precisely contoured layup tooling, rather than by filling and fairing an irregular wing surface following assembly.
- The feasibility of assembling thick faced honeycomb skin panels allows the use of large rib bays, minimizing the number of ribs and external access openings.
- Bonding such structural joints as wing skin to rib chords allows the elimination of many external fasteners which can serve to disrupt laminar flow. However, the integrity of such bonds, subjected to tension, must be verified.

3. Since it is essential to integrate the design of the suction system with the design of the wing structure, design effort in these two areas should proceed in parallel.

4. The high aspect ratio and moderate sweep characteristic of long range LFC transport airplanes tend to complicate landing gear structural provisions. It is important to configure landing gear in the wing preliminary design phase.

5. The aerodynamic shape of the LFC wing/turbulent body intersection should be developed with the problems of landing gear support and stowage being considered.

6. A slotted LFC wing of composite materials can be built with present technology. A perforated LFC wing requires advances which are readily projected from current technology.

7 A satisfactory combination of materials, structural methods, and manufacturing techniques for producing a porous LFC wing does not now exist.

8. A composite LFC wing will be lighter than an equivalent metal LFC wing. The major weight saving will be realized in the primary wing box structure. Lightweight fiberglass composite panels are already in use as fixed leading and trailing edge skins, reducing the potential for weight improvement by use of high strength composites in these areas.

9. The cost of the composite LFC wing will be greater, in production quantities, than the cost of an equivalent turbulent metal wing. Most of this additional cost is a consequence of incorporating suction provisions.

10. The composite wing should be spliced at the side of body rather than on the airplane centerline, for large pressurized airplanes.

11. It is feasible to incorporate suction provisions in wing trailing edge control surfaces.

10.0 RECOMMENDATIONS

As a result of this study, a number of recommendations for future study and development have been identified. These recommendations are summarized below.

10.1 SUCTION SURFACE SERVICE EVALUATION

Invaluable data may be obtained by testing various suction surfaces for clogging, contamination, flow degradation, and other environmental effects caused by the wide range of atmospheric pollutants found in the service environment of the world's airline fleet. It is nearly impossible to obtain this type of information by laboratory test or analysis. In order that a substantial history of operating exposure be developed in time to be used for designing the first LFC transports it is recommended that, as soon as possible, a number of small test panels of candidate LFC surfaces be placed in secondary structure areas on commercial air carrier aircraft, provided with controlled suction, and examined at regular intervals. With strict attention to simplicity of design and attention-free operation such a program can be completed at modest cost.

10.2 CONFIGURATION AND AERODYNAMIC DEVELOPMENT STUDIES

LFC configuration and aerodynamic studies should be continued or initiated to determine:

- If the effects of propulsion engine noise on laminar flow permit the use of wing mounted engines.
- To what extent spoilers and high lift devices, particularly leading edge devices, are required on LFC wings, and how they may best be configured for laminarization.
- Methods of optimization of LFC airfoils, aspect ratio, wing loading, sweep, and thickness.
- Realistic tolerances for smoothness, contour, gaps, and mismatch, and sealing requirements for the LFC surface.

10.3 STRUCTURAL AND MANUFACTURING STUDIES

Structural and manufacturing studies should be conducted to develop:

- Practical methods of attaining smoothness and contour, minimizing gaps and mismatch and sealing the LFC surface as required by studies of section 10.2.
- Application of suction to movable surfaces.
- Deicing and antiicing methods and provisions, for exterior surfaces and interior ducts and plenums of the LFC system.
- Inspection, maintenance, and repair methods and provisions.

- Satisfactory treatment of such necessary items as access panels, landing gear doors, refueling points, lights, etc.
- Assurance of integrity of primary structural bonds, particularly with respect to tension across the bond line.

In addition, extensive weight, serviceability, and producibility studies of glove versus integral suction surface configurations should be made to fully expose the advantages and shortcomings of both concepts.

10.4 SUCTION SYSTEM DEVELOPMENT

Suction system studies should provide design data on:

- Duct configuration requirements, including areas, maximum flow velocities, effects of non-circular cross sections, etc.
- Suction flow rates.
- Slot, perforation and pore geometries and tolerances.
- Optimum laminarized area (percent of chord length).
- Suction engine and compressor requirements and their effects on airplane configuration, including the consequences of partial loss of suction.

Suitable computer programs should be developed to facilitate design integration of suction requirements.

10.5 MATERIALS TECHNOLOGY

Efforts to identify and improve suction surface materials should be continued. Particular effort should be made to develop a suitable porous suction surface material and to define acceptable methods of integrating the porous surface material with the structure and duct system of the LFC wing.

The extensive development of advanced composite technology which is required to permit the use of these promising materials in the primary structure of commercial LFC aircraft can best be realized in the context of present and planned composite structure programs not now related to the LFC effort. Such programs should be structured to include the requirements of the LFC wing, and should be coordinated with a continuing LFC effort as outlined above.

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