N75-26997

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NASA CR-132668

A FUSELAGE/TANK STRUCTURE STUDY FOR ACTIVELY COOLED HYPERSONIC CRUISE VEHICLES

Aircraft Design Evaluation

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Prepared under Contract No. NAS1-12995
McDonnell Aircraft Company (MCAIR)
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for
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FOREWORD

This report summarizes the results of "A Fuselage/Tank Structure Study For Actively Cooled Hypersonic Cruise Vehicles" performed from 11 March 1974 through 30 June 1975 under National Aeronautics and Space Administration Contract NAS-1-12995 by McDonnell Aircraft Company (MCAIR), St. Louis, Missouri, a division of McDonnell Douglas Corporation.

The study was sponsored by the Structures and Dynamics Division with Dr. Paul A. Cooper as Study Monitor and Mr. Robert R. McWithey as Alternate Study Monitor.

Mr. Charles J. Pirrello was the MCAIR Study Manager with Mr. Allen H. Baker as Deputy Study Manager. The study was conducted within MCAIR Advanced Engineering which is managed by Mr. Harold D. Altis, Director, Advanced Engineering Division. The study team was an element of Advanced Systems Concepts, supervised by Mr. Dwight H. Bennett.

The basic purpose of this study was to evaluate the effects of fuselage cross section (circular and elliptical) and structural arrangement (integral and non-integral tanks) on the performance of actively cooled hypersonic cruise vehicles. The study was conducted in accordance with the requirements and instructions of NASA RFP 1-08-4129 and McDonnell Technical Proposal Report MDC A2510 with minor revisions mutually agreed upon by NASA and MCAIR. The study was conducted using customary units for the principal measurements and calculations. Results were converted to the International System of Units (S.I.) for the final report.

This is one of three reports detailing the technical results of the study. The other two reports are "Active Cooling System Analysis," Reference (1), and "Structural Analysis," Reference (2).

The primary contributor to the contents of this report was T. Nobe. Assistance was provided by D. A. Reddan, and C. Polleschultz. Other contributors were H. Landmann, K. Wilkison, W. Pekala, T. Broccard, C. Wilcox and H. Chase.

TABLE OF CONTENTS

TABLE OF CONTENTS

(Continued)

TABLE OF CONTENTS

(Continued)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Continued)

LIST OF TABLES

LIST OF ABBREVIATIONS AND SYMBOLS

Symbols	Definition
A_c	Capture Area
AR	Aspect Ratio
APU	Auxiliary Power Unit
b	wing span
c.g.	center of gravity
C_L	lift coefficient
C_M	Pitching moment coefficient
C_R	Root Chord
C_T	Tip Chord
CTR	Center
D	drag force
ΔE	differential energy
ECS	Environmental Control System
ft	feet
F_n	net thrust force
FS	fuselage station
FUS	fuselage
°F	Degrees Fahrenheit
FWD	forward
g	acceleration due to gravity
GPM	gallons per minute
I_{sp}	Specific Impulse
L	lift force
lbf	pounds force
lbm	pounds mass
LH_2	Liquid Hydrogen
M	Mach
MAC	Mean Aerodynamic Chord
N_{CR}	Critical flat panel edgewise compression load
NLG	nose landing gear
NM	nautical mile
O.W.E.	Operating weight empty
Pa	Pascal (Newton/m^2)
P_T	Total Pressure

Symbols	Definition
psf	pounds force per square foot
psi	pounds force per square inch
PTO	power takeoff
q	dynamic pressure
R_N	Reynold's Number
SFC	Specific fuel consumption
S.I.	International system
S.L.	sea level
Sp	Total planform area
S	theoretical wing area
S_{wet}	wetted wing area
t	time
T	thrust
T_i	adiabatic wall temperature
TOGW	Takeoff gross weight
TRJ	Turboramjet
T_W	Wall temperature
V	velocity
W	weight
Vol.	Volume
β	$\sqrt{M^2 - 1}$
α	angle of attack
λ	Taper ratio
Δ	change
Λ	sweep angle

SI Units

h	gram (weight)
K	kelvin (temperature)
m	meter (length)
N	Newton (force)
Pa	Pascal (pressure)
W	Watt (power)

SI Prefixes	Definition
c	Centi (10^{-2})
G	Giga (10^{9})
k	Kilo (10^{3})
m	Milli (10^{-3})
M	Mega (10^{6})

1. INTRODUCTION

The purpose of this study is to evaluate the effects of fuselage cross section (circular and elliptical) and structural arrangement on the performance of actively cooled Mach 6 cruise aircraft. The three aircraft shown in Figure 1 carry a constant fuel quantity and passenger payload. The aerodynamic characteristics of each aircraft were derived from the NASA HT-4 configuration. By using the same basis for configuration development, the effects of tank structural variations can be assessed independent of aerodynamic influences.

Representative fuselage/tank area structure was analyzed for strength, stability, fatigue and fracture mechanics. Various thermodynamic and structural trade-offs were conducted to refine the conceptual designs with the primary objective of minimizing weight and maximizing aircraft range.

This report presents the results of the aircraft design studies and evaluation. These results include aircraft design rationale, aircraft descriptions, performance comparisons and trade-off results. Many of the trade studies involved extensive interaction between the configuration design, structures and thermodynamics. We have presented the details of these studies in the particular technology area which had the greatest impact on the decision. However, for completeness this report highlights all studies conducted with reference to other reports (Reference 1 or 2) for more detail. This report is organized as follows:

- Sections 1 and 2 are introduction and summary.
- Section 3 presents the study ground rules and assumptions, design criteria and mission profile.
- Section 4 is a discussion of the aircraft configuration development with a rationale for each design.
- Section 5 discusses the design trade-offs completed for each aircraft.
- Section 6 presents a description of the three aircraft. Layout drawings are included for each of the major aircraft components. Also, qualitative assessment is presented in the major areas of producibility and maintainability.
- Section 7 presents the aerodynamic and propulsion performance as well as weight estimation techniques used to assure consistent comparisons.

o Section 8 summarizes comparison and evaluation of the studied aircraft. Included are quantitative evaluations of the aircraft performance, weight and volumetric efficiencies.

o Section 9 discusses the conclusions drawn from this study and offers MCAIR's recommendations for areas of future investigation.

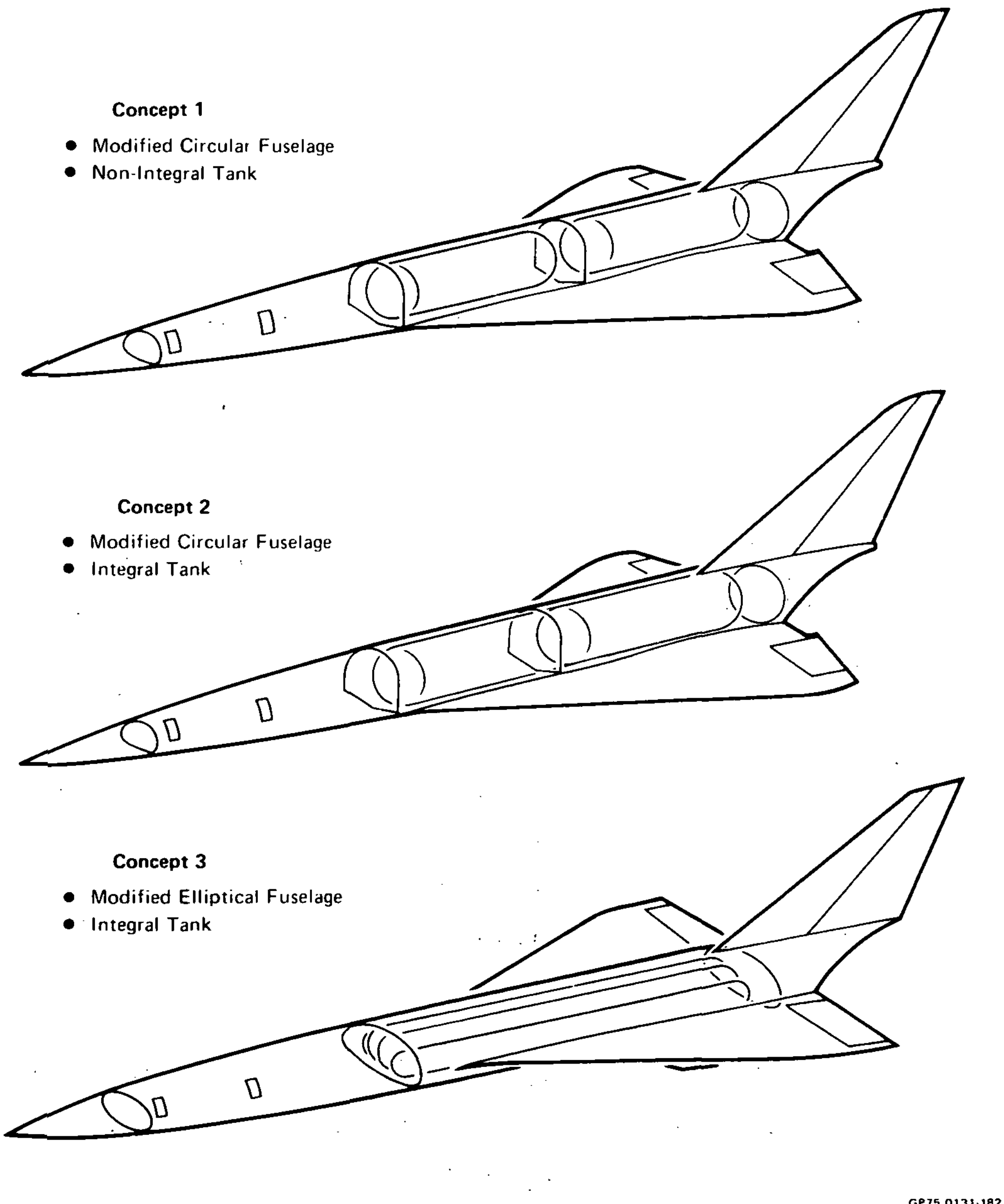

FIGURE 1
FUSELAGE/TANK CONFIGURATIONS

2. SUMMARY OF AIRCRAFT DESIGN AND PERFORMANCE CHARACTERISTICS

The combined effect of fuselage cross section and tank structure on actively cooled Mach 6 cruise vehicles was investigated. The three vehicle configurations studied were designed to reflect combinations of these effects and are shown in Figures 2 and 3. Concept 1 is a discrete wing-body configuration having a modified circular (Dee) fuselage cross section and incorporating a circular non-integral fuel tank structure. Concept 2 is a discrete wing-body configuration having a dee fuselage cross section and a circular integral fuel tank structure. Concept 3 is a blended wing-body configuration having an elliptical fuselage cross section incorporating an integral "bubble" fuel tank structure. Each aircraft carries 200 passengers and 108.9 Mg (240,000 lbm) of fuel. The external surface of each vehicle is maintained at a maximum temperature of 394 K (250°F).

Configuration development was primarily based on the NASA's HT-4 experimental model described in Reference (3). The aircraft is configured to provide comparable aerodynamic characteristics in each aircraft, so that the effect of the tankage structure and fuselage cross section can be isolated and the effect on aircraft performance can be evaluated. A qualitative evaluation of producibility and maintainability was also made to provide insight to initial investment cost and direct operating cost respectively.

Figure 4 summarizes the performance and design characteristics of the three concepts. The performance figure of merit for this study was designated to be range. It can be noted in Figure 4 that Concept 3 has a 5.9% greater range than Concept 1 and 5.4% greater than Concept 2. Concept 2 exhibits a small 0.5% increase over Concept 1. The range increases are due primarily to the increased volumetric efficiency of the integral tank configurations. Improved aerodynamics of the blended wing-body Concept 3 also contributes to its range superiority. The overall results, however, demonstrate volumetric efficiency to be the dominant factor in determining aircraft range.

A relative cost assessment, including the producibility and maintainability aspects, showed the non-integral tank aircraft Concept 1 to be the least-cost aircraft, as indicated in Figure 4.

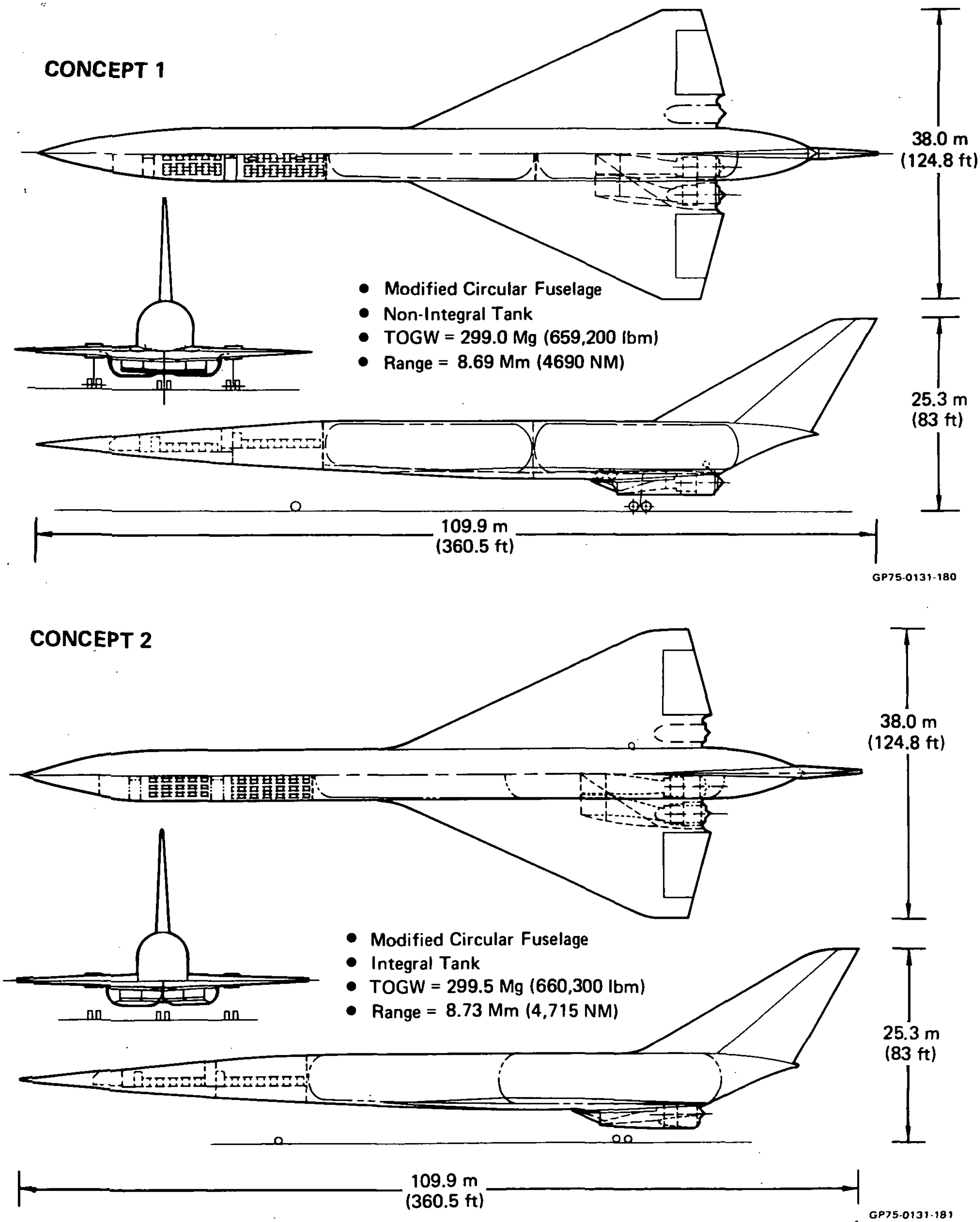

FIGURE 2

CIRCULAR TANK – GENERAL ARRANGEMENT

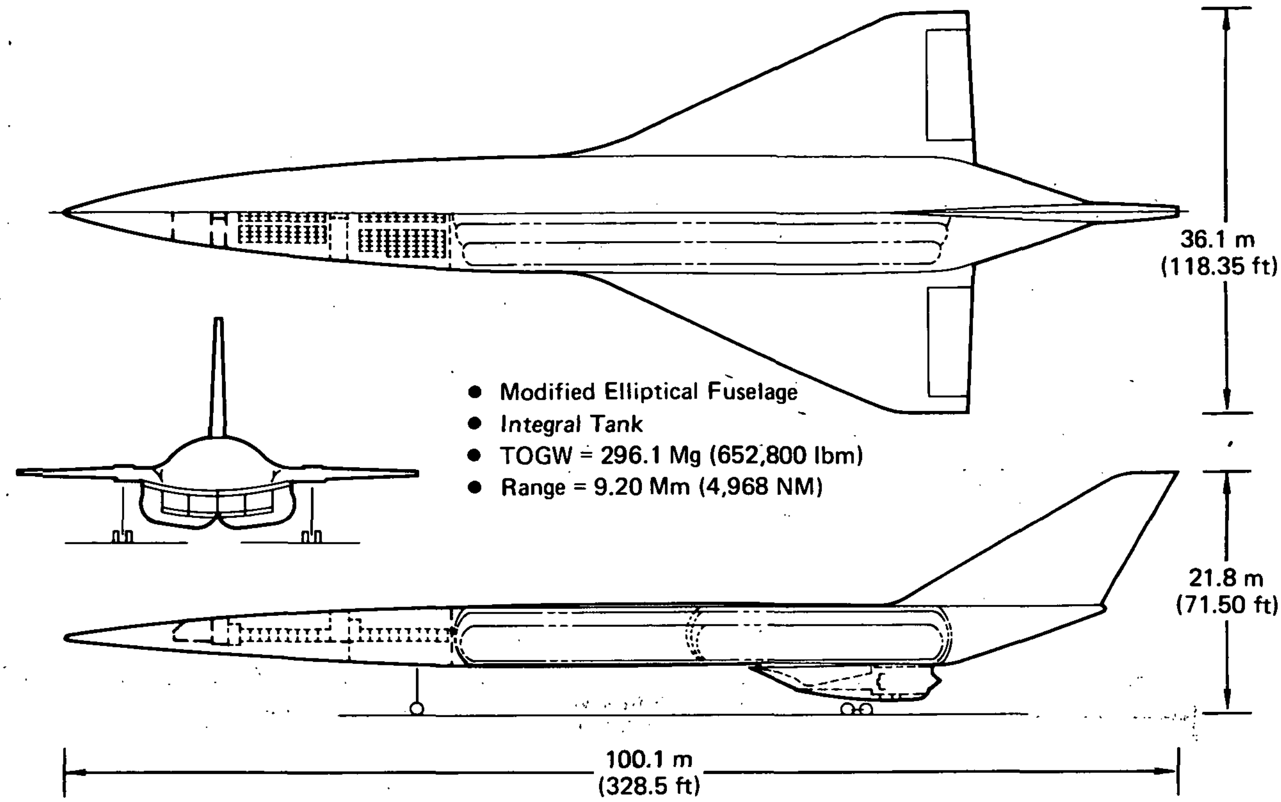

FIGURE 3
ELLIPTICAL TANK-GENERAL ARRANGEMENT

Characteristics	Concept 1	Concept 2	Concept 3
Fuselage Cross Section	Dee	Dee	Elliptical
Tank Shape	Circular	Circular	Bubble
Tank Structural Arrangement	Non-Integral	Integral	Integral
Body Length - m (ft)	101.8 (334)	101.8 (334)	93.9 (308)
Wing Area - m^2 (ft^2)	1,070 (11,530)	1,070 (11,530)	960 (10,377)
TOGW - Mg (lbm)	299.0 (659,200)	299.5 (600,300)	296.1 (652,800)
Fuel Weight Usable - Mg (lbm)	106.27 (234,300)	106.30 (234,400)	106.27 (234,300)
O.W.E. - Mg (lbm)	190.14 (419,200)	190.64 (420,300)	187.24 (412,800)
W/S_{theo} - kg/m^2 (lbm/ft^2)	279.5 (57.2)	279.9 (57.3)	308.5 (62.9)
$T/W_{T.O.}$ Installed	4.90 (0.495)	4.90 (0.495)	4.89 (0.500)
Range Mm (NM)	8.69 (4,690)	8.73 (4,715)	9.20 (4,968)
Volumetric Efficiency (Fuel Volume/Center Fuselage Tank Volume)	67%	71%	88%
Maintainability Complexity Factor	1	1.2	1.3
Production Cost Factor	1	3.5	3

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FIGURE 4
DESIGN/PERFORMANCE SUMMARY

The relative manufacturing costs of welding, forming, machining, and assembling of non-integral tank structures is the lowest because the tank fuselage transitional structure and wing support structure is the least complex. Less time is required to maintain the non-integral tank because of greater access provisions and a relatively less complex installation.

3. MISSION PROFILE AND DESIGN REQUIREMENTS

3.1 MISSION PROFILE

Concept 1, used as the baseline aircraft for this study, was designed to cruise at a Mach number of 6 and attain a range goal of 9.26 Mm (5000 NM). A mission profile was generated which took advantage of aerodynamic and structural concepts derived from previous hypersonic aircraft related studies. The critical sections of the mission profile are the ascent and descent paths. These were established based on aerodynamic performance, propulsion system performance, and structural design considerations, with the objective of providing minimum TOGW and maximum range. The ascent and descent path established for the mission is presented in Figure 5. Each of the study aircraft followed these paths as a part of the performance calculations and the resulting range was used as the primary evaluation criterion.

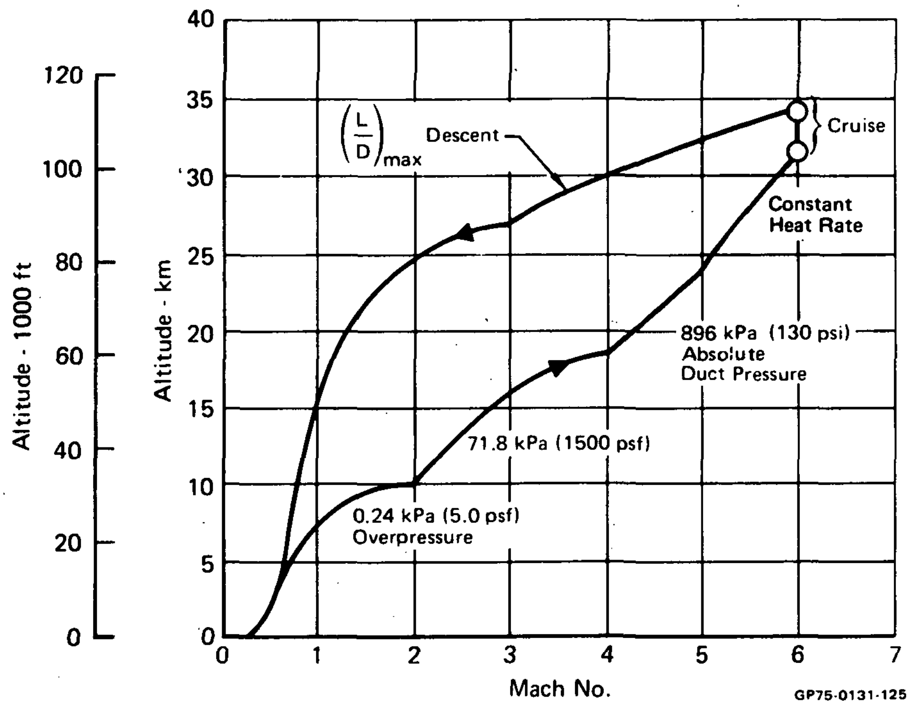

FIGURE 5
MISSION TRAJECTORY

The ascent path is subdivided into four segments. The first segment, ending at Mach 2 and 9.75 km (32,000 feet), is designed to limit sonic boom overpressure on the ground to 0.24 kPa (5.0 lbf/ft^2). Although Figure 6 shows that this pressure would result in some damage to glass windows, implying that special climb corridors may be required for these aircraft, this overpressure level was selected as a result of a trade study, described in Section 5.1 of this volume. The study showed that a higher rate of climb results in significant size and weight penalties to the aircraft.

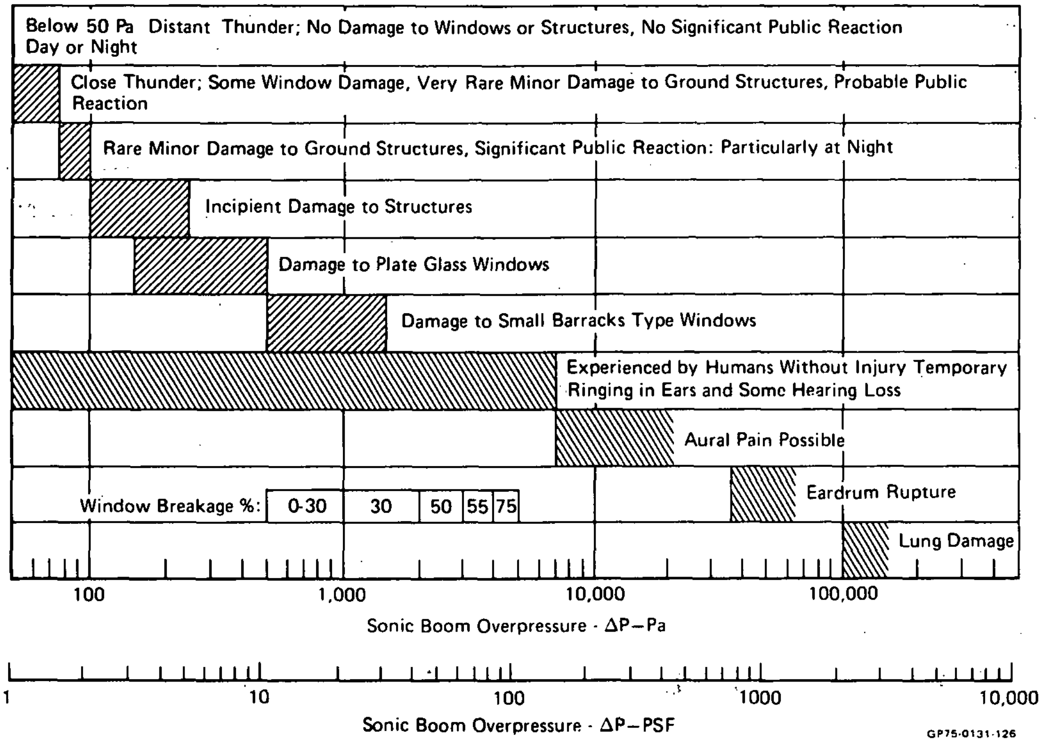

FIGURE 6
SONIC BOOM PHYSIOLOGICAL AND STRUCTURAL EFFECTS

The second segment of the ascent path is a structural consideration holding the maximum dynamic pressure to 71.8 kPa (1500 lbf/ft^2). The third segment is also a structural consideration and conforms to an inlet diffuser pressure limit of 896 kPa (130 psi) absolute. This pressure was established as a result of a previous study on inlet diffuser structure, Reference (4). The final ascent is made on a path which results in a constant heating rate to the structure. A trade-off, which is discussed in Section 4.2 of Reference (1), showed a significant reduction in the cooling system size and weight by following this constant heating rate path from Mach 5 to Mach 6 rather than continuing on the inlet diffuser pressure limit line.

The cruise leg was flown at a maximum range factor $(V\frac{L}{D})/I_{SP}$. This calculation included centrifugal relief, which at a velocity of approximately 1829 m/sec (6000 ft/sec), was equal to 6% of the weight.

Descent was accomplished at the maximum lift-to-drag ratio. This path provides a maximum time, maximum range descent.

The mission reserves consist of sufficient fuel to loiter 20 minutes at M = 0.8 and 12.2 km (40,000 ft), plus sufficient fuel for one "go around" [(5 minutes) at M = 0.4 at sea level].

3.2 DESIGN REQUIREMENTS

A common set of design requirements was established at the beginning of the study, for comparing the selected aircraft. The design requirements that are common to all three concepts are:

- o Cruise at Mach 6
- o External surfaces to be actively cooled to a maximum temperature of 394 K (250°F)
- o Payload = 21.8 Mg (48,000 lbm) with 200 passengers
- o Fuel weight = 108.9 Mg (240,000 lbm) (established on the Concept 1 baseline aircraft)
- o Propulsion system

  Four GE5-JZ6 TRJ wraparound turboramjet engines

  2 dimensional external compression inlets with 3 horizontal ramps
- o Volume Parameter: $V^{2/3} \div Sp$ approximately the same as NASA HT-4 configuration
- o Limit Tank Pressurization: 138 kPa (20 psi) gage

4. AIRCRAFT DESIGN RATIONALE AND CONFIGURATION DEVELOPMENT

This section describes the design parameters and interactive design concepts driving the aircraft configuration development. Each configuration is discussed separately to focus on specific differences.

4.1 CONFIGURATION DEVELOPMENT RATIONALE

As previously noted, the baseline configuration for the fuselage tank study was derived from the NASA HT-4 tailless delta configuration shown in Figure 7 and described in greater detail in Reference (3). The modifications to the fuselage cross section and the structural arrangement of the liquid hydrogen tankage, incorporated in all of the aircraft concepts, were successfully integrated into the HT-4 without sacrificing basic aerodynamic efficiency. These evolved into the baseline cross sections shown in Figure 8. The planform shape (wing sweep, geometry, etc.) was kept essentially constant for all three concepts. Therefore the cross sectional shape changes from the baseline had only a small aerodynamic effect on performance. This consistency was maintained throughout the configuration refinement phase.

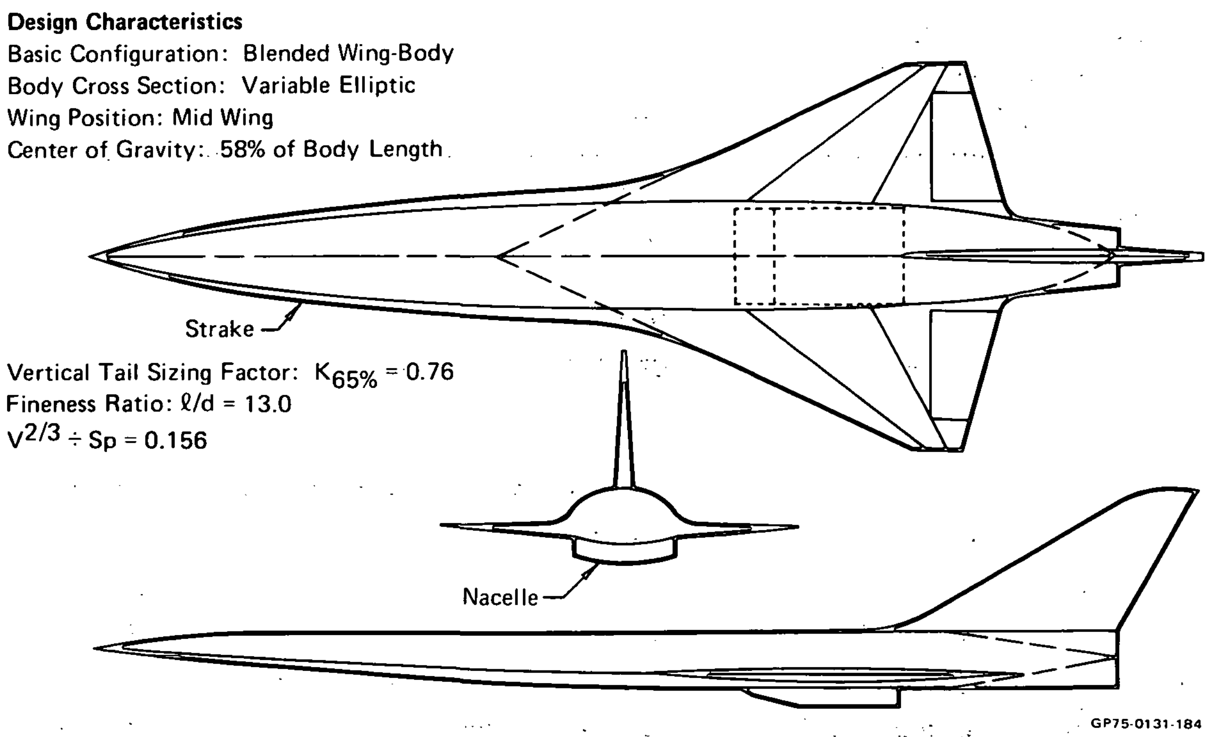

FIGURE 7
NASA HT-4 CONFIGURATION

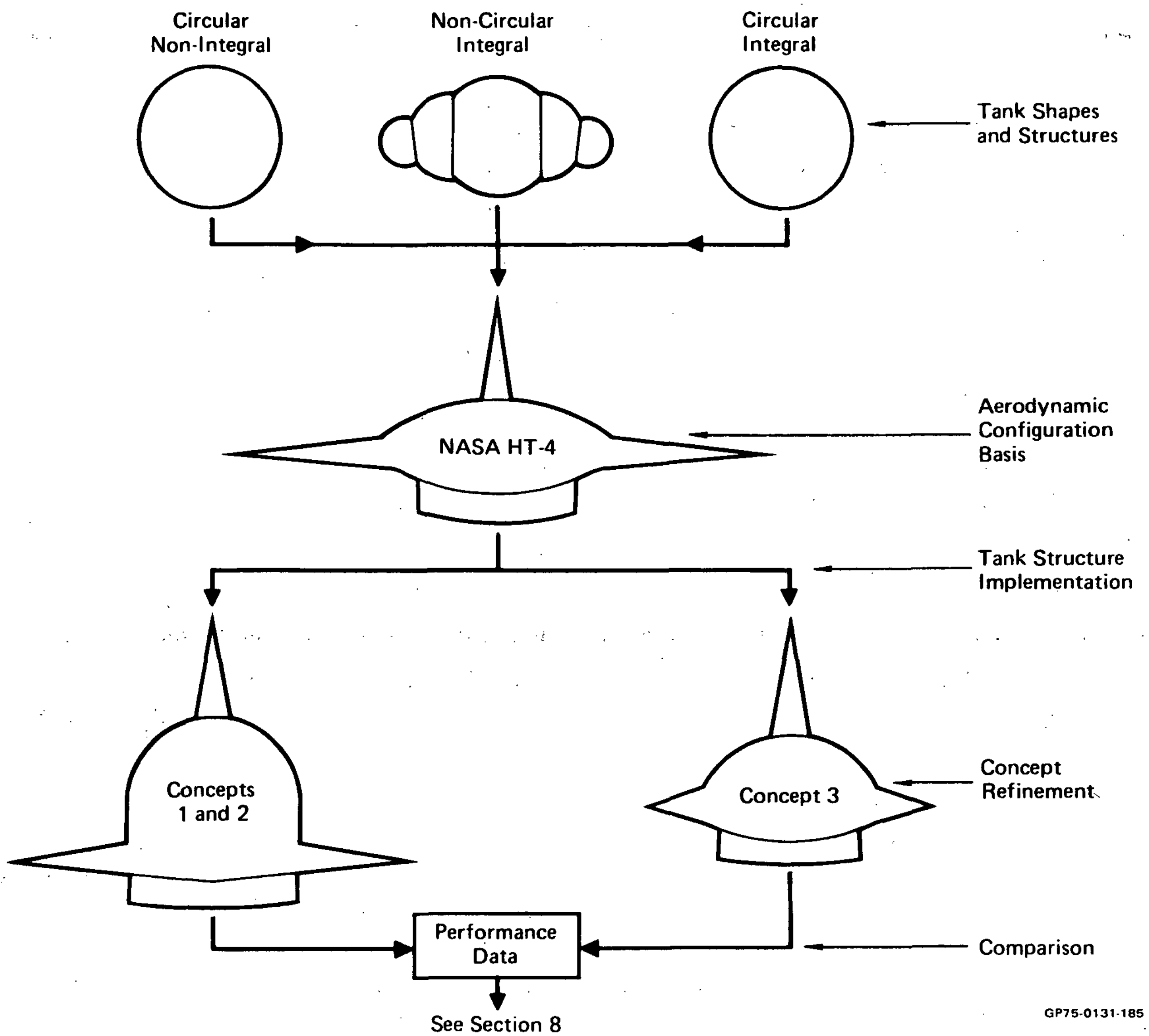

FIGURE 8
CONFIGURATION DEVELOPMENT APPROACH

A number of design options were considered in establishing each concept. These are discussed in the following sections.

4.2 CONCEPT 1

By definition, Concept 1 is a discrete wing-body configuration with a dee shaped fuselage cross section and non-integral fuel tanks. A non-integral tank must support fuel inertia loads and internal pressure loads. The primary aircraft load is carried in the fuselage shell.

Two wing positions were examined in transforming the baseline HT-4 blended wing-body shape into a circular fuselage cross section for Concept 1. The options are shown in Figure 9. The mid wing concept offers a classical circular cross section and also greater tank-to-fuselage volumetric efficiency, but it was discarded for the following reasons:

o Based on previous studies, straight carry-through wing structures are more efficient and result in lower weight.

o The fuselage cross sectional area would increase to accomplish the greater frame depth required to carry wing loads. This would increase aerodynamic wave drag.

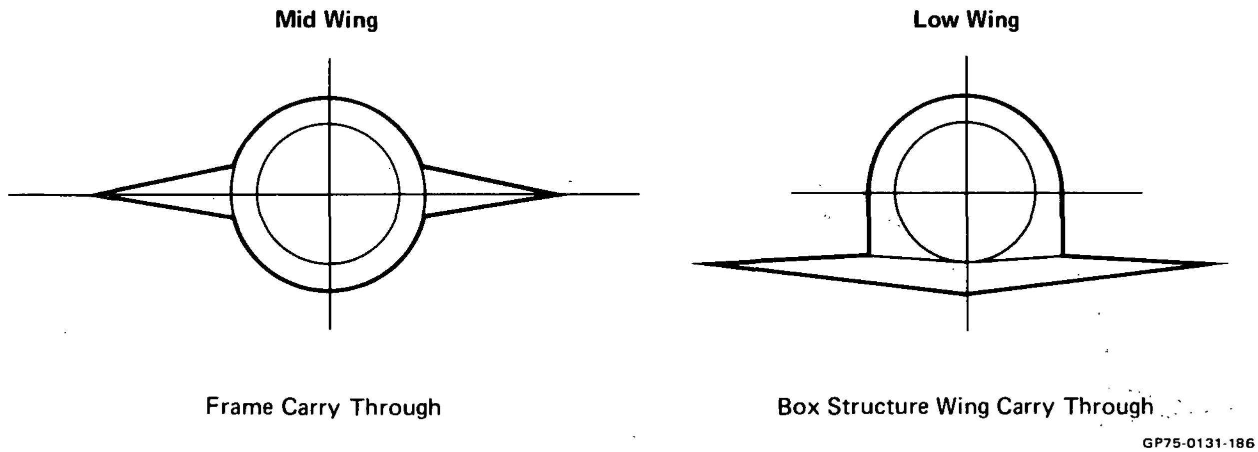

FIGURE 9
ALTERNATE WING POSITIONS

The low wing carry-through concept provides wing shielding for the inlet and acts as a precompression surface reducing the inlet capture area requirement.

The fuselage fineness ratio (ℓ/d) was calculated at 13.45, to match the HT-4 configuration. This established the fuselage body length and the cross sectional area for a given fuel volume. Figure 10 shows a typical fuselage cross section developed for Concept 1. As shown, the wing is positioned below the circular tank to allow the carry-through spars and skins to be continuous. To maintain a minimum fuselage depth, the wing carry-through is essentially full depth where the maximum wing bending occurs. From this point forward the spar height in the wing torque box is reduced corresponding to the design wing loads to permit a lower position of the tank and reduce the overall profile.

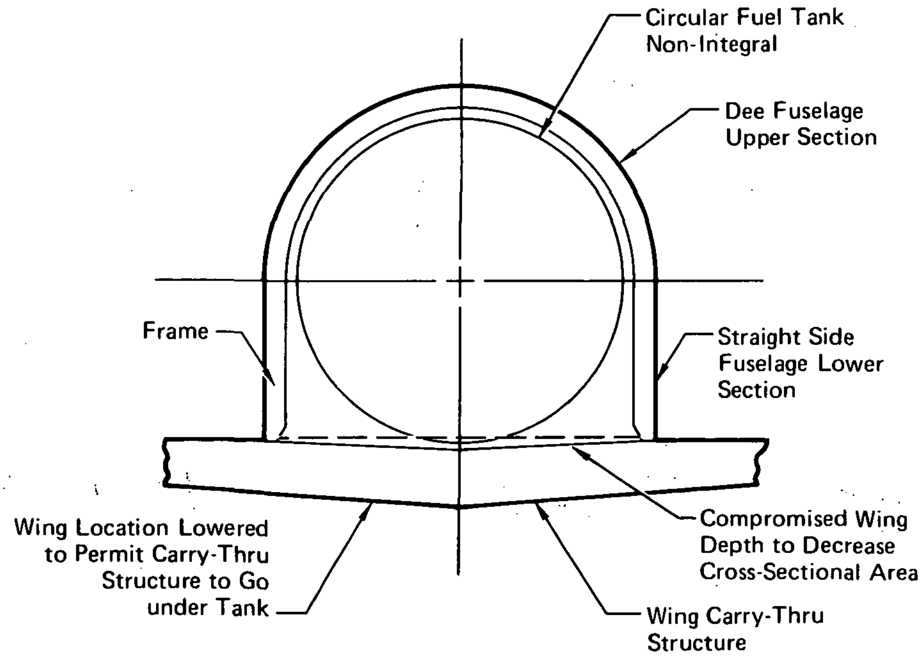

FIGURE 10

CENTER FUSELAGE SHAPING, CONCEPT 1

The wing size is a function of fuselage length and was developed by using the HT-4 wing/fuselage ratio as specified in Reference (3). The wing is located at 65% of fuselage body length with respect to 31% MAC of the wing. External fairings were added on the wing upper and lower surface to obtain sufficient depth to stow the main landing gear.

The forward fuselage upper and lower shear lines are essentially the same as HT-4 through the cockpit area. These angles were held essentially constant for all concepts. The forward passenger section was developed by providing sufficient volume for the nose landing gear, baggage compartment, and subsystems as well as the required passenger volume and still maintaining the HT-4 fineness ratio as close as possible.

a. Aft Fuselage Shaping - The aft fuselage shape was modified from the conical aft fuselage shape of the Model HT-4 as shown in Figure 11. The up-swept aft fuselage minimizes the engine exhaust plume impingement on this structure.

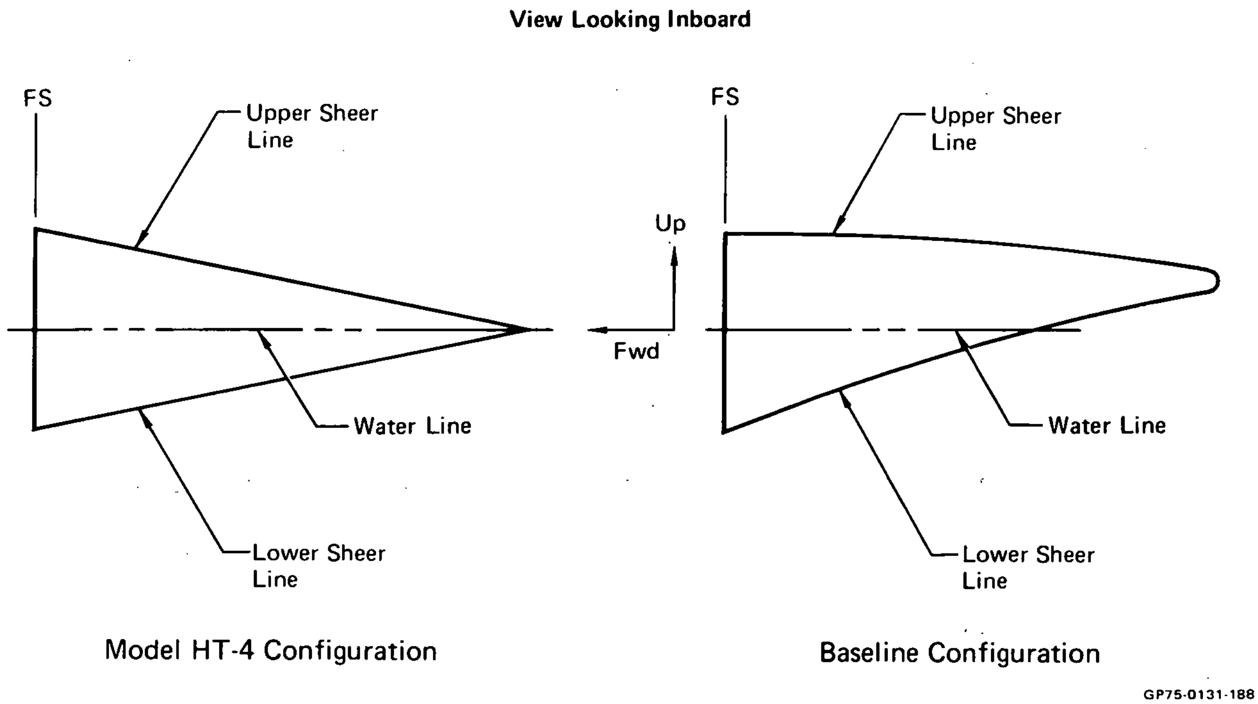

FIGURE 11
AFT FUSELAGE SHAPING

b. Vertical Tail Shaping and Position - The vertical tail shape was made similar to the Model HT-4 M = 1.50 to 5.00 tail, and sized as a function of fuselage length. The double wedge airfoil section has a 2° slope on each side in the fore and aft directions. The leading edge sweep is 60° and the trailing edge sweep is 30°. The root chord of the vertical tail is positioned on the fuselage upper shear line, with the trailing edge even with the aft end of the body section.

The surface area was determined using a balancing factor, K = 0.68. This factor is defined as a ratio of the vertical tail area moment to the fuselage area moment forward of the airplane center of gravity.

c. Nacelle Shaping - The nacelle shape utilized is a short external compression inlet. Wing shielding resulted in reduced capture area thus minimizing the nacelle weight.

From a preliminary "as drawn" vehicle, cross sectional area and wetted area distribution were measured and plotted and a preliminary weight estimation was made. This preliminary vehicle was used for the passenger/tank location study. The baseline weight, volume, and performance data were inputs to the computer sizing program which established the required vehicle size to meet the aircraft mission.

o <u>Baseline Weights</u> - The Concept 1 weights resulting from preliminary analysis of the aircraft submitted in the proposal are summarized below. These were submitted to and accepted for the Concept 1 baseline by NASA.

- o OWE = 190 Mg (419,200 lbm)
- o W_{fuel} = 108.9 Mg (240,000 lbm) Total
  = 106.3 Mg (234,300 lbm) Usable
- o Range = 8.69 Mm (4,690 NM)

4.3 CONCEPT 2

Concept 2 is almost identical to Concept 1 except that it has an integral tank. The integral tank carries all the primary aircraft load in the center fuselage and redistributes loads from all the appurtenant aircraft members. Although a mid wing position on the Concept 2 structural arrangement presents no adverse aerodynamic wave drag effect, the lower wing position of Concept 1 was chosen to maintain configuration commonality between Concepts 1 and 2.

The outer moldline covering is made of actively cooled panels similar to Concept 1. In Concept 2 they carry secondary fuselage bending loads as well as airload. The major effect is that less space is required between the tank moldline and the external moldline for frame structure. Therefore, the fuselage diameter can be reduced as shown in Figure 12. This resulted in

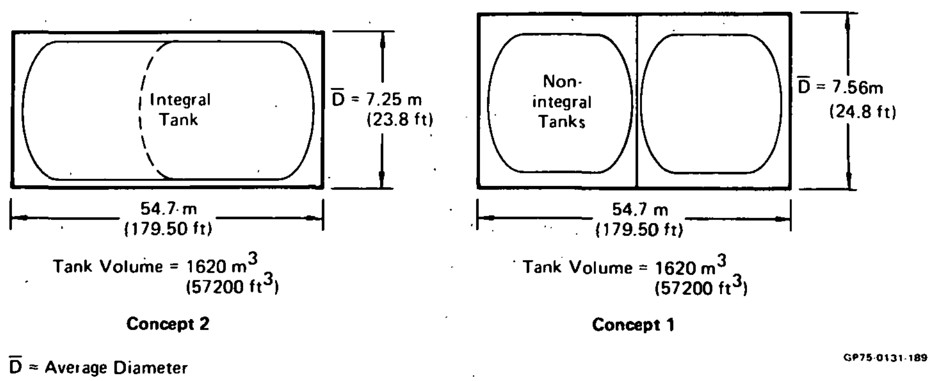

FIGURE 12
FUSELAGE DEVELOPMENT, CONCEPT 2

some modification to the forward passenger compartment, wing planform area and the overall airplane volume, and also increased the body fineness ratio from 13.45 for Concept 1 to 14 for Concept 2.

An optional fuselage approach, shown as Option 2 in Figure 13, was considered in which the outer moldline of Concept 1 was used and the average fuel tank diameter increased. This tank did not have to be as long for equal fuel volume, and resulted in a shorter body length. The aerodynamic drag of this version, however, was greater because of the decreased fineness ratio, and negated any advantages of the slightly smaller aircraft.

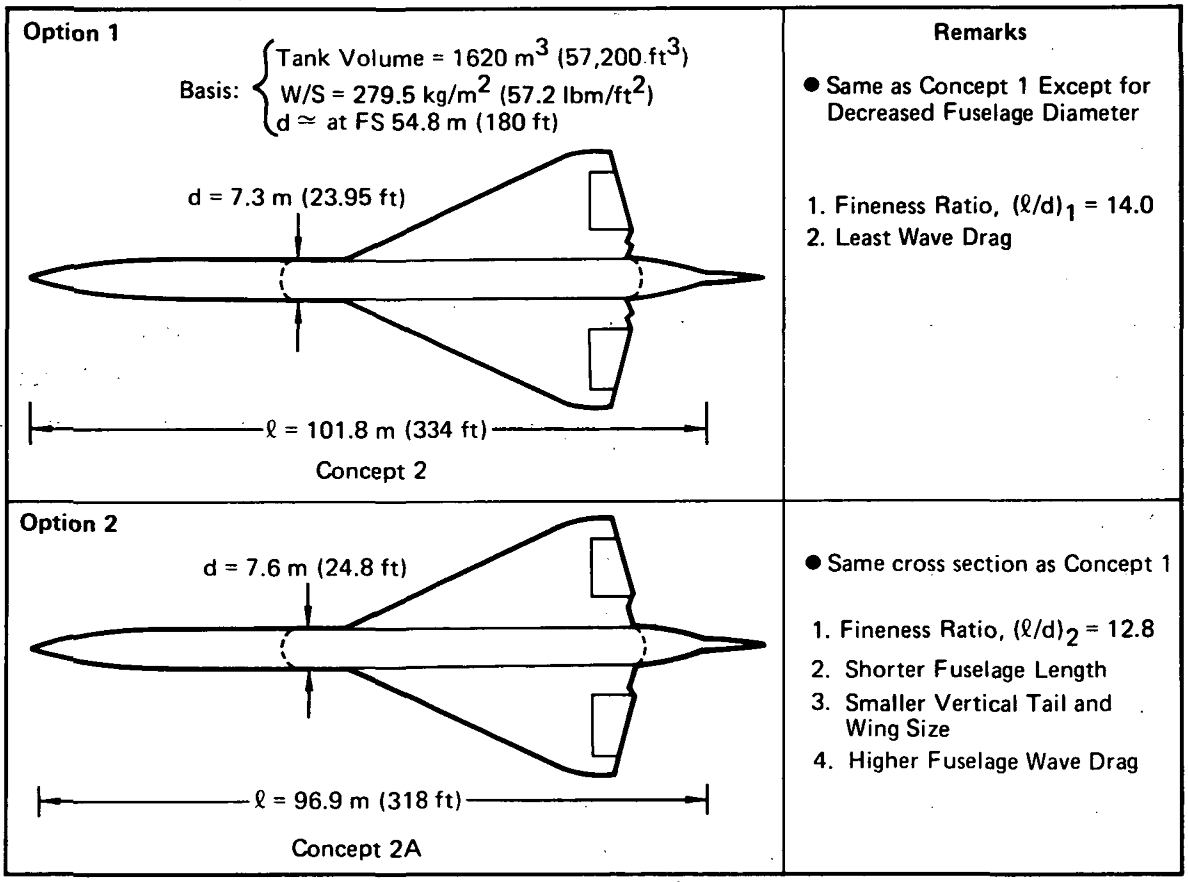

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FIGURE 13
CONFIGURATION OPTION, CONCEPT 2

4.4 CONCEPT 3

This concept also features an integral fuel tank as the primary center fuselage structure. The configuration is very similar to HT-4. The shape of the tank, however, is made to conform to a 2:1 elliptical cross section.

Figure 14 shows the relatively high volume utilization and the interaction of structural components of this concept. Unlike Concepts 1 and 2, the tank rings act as the wing carry-through.

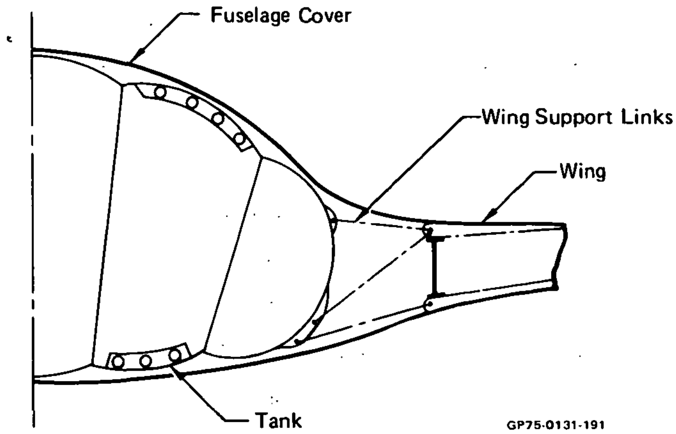

FIGURE 14
CENTER FUSELAGE SHAPING, CONCEPT 3

The tank shape was designed to maximize the aircraft range. A trade-off on tank shape is discussed in Section 5. A constant cross section was maintained in the center fuselage to simplify fabrication of the multi-bubble tank. Some rearrangement of the passenger seating was required from Concept 1 because of the elliptical shape.

a. Wing Shaping and Position - Wing shape and size are kept basically similar to NASA's HT-4. The strake on the HT-4 was removed since, when included, Concept 3 was longitudinally unstable. Section 7.1.4 of this volume provides an explanation of the longitudinal stability.

The wing is positioned near the middle of the fuselage cross section for the following reasons:

o Distributing the wing loads through the fuselage frame resulted in the highest volume utilization.

o There is adequate volume for subsystem, controls, and equipment at the wing root.

b. Nacelle Shape - The smaller wing and constant shape fuselage section on Concept 3 created a problem in trying to retain a nacelle that was common with Concept 1. The width of the baseline nacelle relative to the smaller wing interfered with the landing gear well and decreased the elevon span. It was necessary to increase the inlet aspect ratio (inlet capture height divided by inlet width) and decrease the engine spacing, so as to decrease the width of the nacelle. The width was decreased until the entire nacelle could be mounted without interfering with interfacing components.

Because the fuselage and tank cross section was kept constant for weight and manufacturing purposes, it was necessary to make the entire nacelle external to the fuselage. This caused the exposed nacelle volume to be greater than that of the baseline concept and resulted in an increase in nacelle drag. However it was felt that the benefits of reduced weight and lower manufacturing cost offset the small loss in aircraft performance.

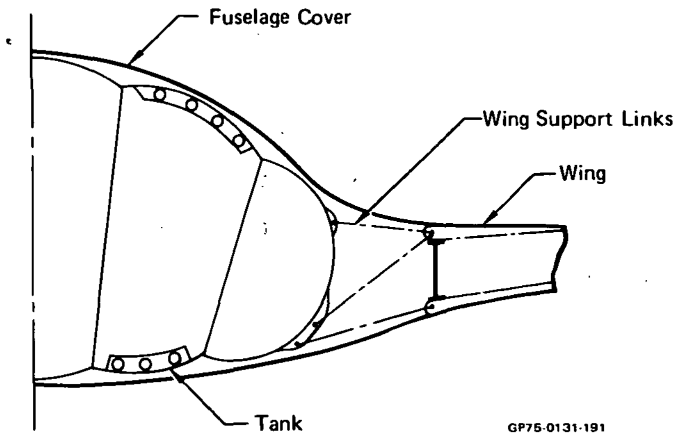

FIGURE 14
CENTER FUSELAGE SHAPING, CONCEPT 3

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5. TRADE STUDIES

Trade-off and design studies were conducted to achieve maximum aircraft range. Range sensitivity to fuel and dead weight were developed, as discussed in Section 7, and used as a basis of evaluating the various options studied. Trade-off studies ranged from operational considerations such as the aircraft trajectory to details of tank structure considering effects on weight, volumetric efficiency, and design practicality. A list of the trade-offs and design studies conducted is provided in Figure 15. Each study is summarized in this section and references to more detailed discussions are made where applicable.

Trade Study	Concept 1	Concept 2	Concept 3	Reference Index
Payload/Fuel Location Study	●	●	●	Section 5.1 of this Report
Tank Length and Dome Shape	●	●	●	Section 6.2.2 Reference 2
Ascent Trajectory	●	●	●	Section 4.2 Reference 1
Nacelle Cooling	●	●	●	Section 4.3 Reference 1
Sonic Boom Overpressure	●	●	●	Section 5.1 of this Report
Actively Cooled Fuselage Covering	●	●	●	Section 6.2.4 Reference 2
Tank Construction	●			Section 6.2.3 Reference 2
Tank Construction		●	●	Section 6.3.1 Reference 2
Thermal Protection System Selection		●	●	Section 8.1-8.3 Reference 1
Semi-Structural vs Non-Structural Tank Covering		●		Section 6.3.2 Reference 2
Actively Cooled Cover Structure Design		●		Section 5.5 of this Report
Tank Cross Section Optimization			●	Section 6.4.2 Reference 2
Semi-Structural vs Non-Structural Tank Covering			●	Section 6.4.1 Reference 2
Design Study				
Actively Cooled Panel Arrangement	●	●	●	Section 5.5 of this Report
Wing/Fuselage Attach Development		●		Section 5.5 of this Report

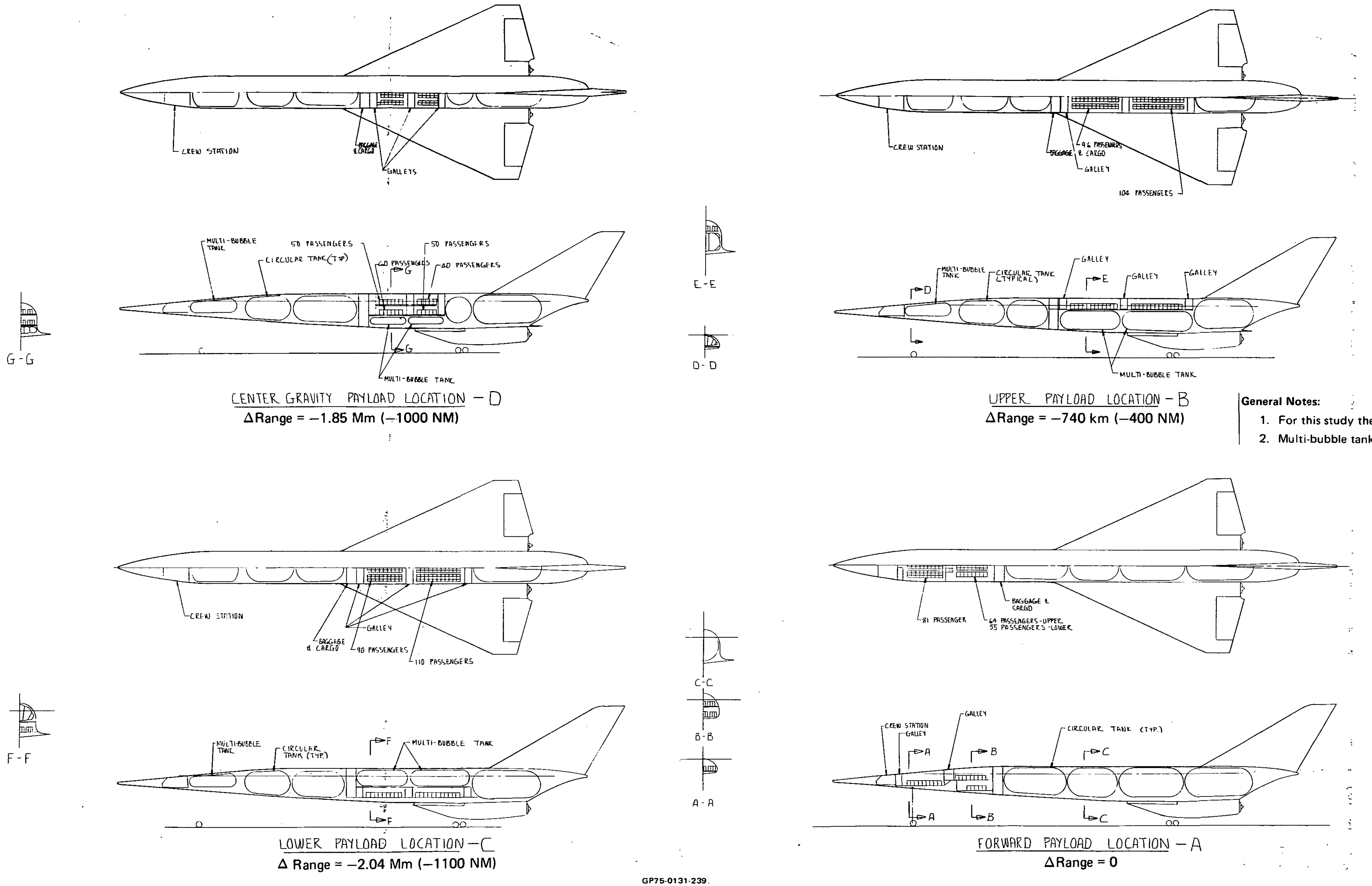
● Indicates study applicable to the concept

GP75-0131-192

FIGURE 15
TRADE STUDY INDEX

5.1 TRADE STUDIES (ALL DESIGN CONCEPTS)

The results of the following trade studies apply to all three study aircraft even though they were conducted only with the Concept 1 baseline.



General Notes:

1. For this study the length of the tanks was constrained to a maximum of 50 feet
2. Multi-bubble tank shape is used to optimize the volume utilization

FIGURE 16
EFFECT OF PAYLOAD LOCATION

5.1.1 Payload/Fuel Location Study - Four different payload/fuel arrangements were studied to determine the effect on volumetric efficiency. The configurations are shown in Figure 16 with an evaluation summary presented in Figure 17. The conventional "Forward Payload Location - A" was selected. This study had a significant effect on aircraft range and illustrates the effect of volumetric efficiency on aircraft performance.

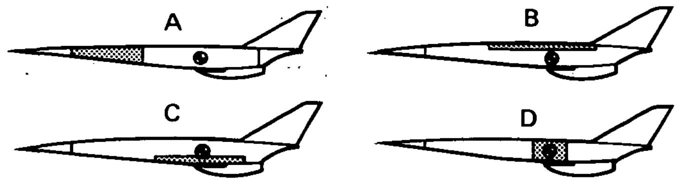


Figure of Merit	Payload Location			
	A	B	C	D
Passenger Compartment Volume m^3 ($10^3 ft^3$)	398 (14.05)	324 (11.45)	372 (13.13)	428 (15.12)
Crew Station Compartment Volume m^3 ($10^3 ft^3$)	30 (1.05)	40 (1.44)	40 (1.44)	40 (1.44)
Baggage Compartment Volume m^3 ($10^3 ft^3$)	73 (2.59)	89 (3.15)	89 (3.15)	89 (3.15)
Max Fuel Tank Volume Attainable m^3 ($10^3 ft^3$)	1,330 (47.01)	1,310 (46.42)	1,130 (39.79)	1,150 (40.60)
$\frac{\text{(Payload + Fuel) Volume}}{\text{Total Fuselage Volume}}$ (%)	56.9	55.0	50.7	53.1
Passenger Deboarding (Emergency Landing)	Easiest	Difficult	Difficult	Difficult
Passenger Boarding	Easier	Difficult	Difficult	Difficult
Aircraft Servicing	Easiest	Difficult	Difficult	Difficult
Aircraft Fabrication Complexity	Low	Moderate	High	Moderate
ΔRange Mm (NM)	0	−0.74 (−400)	−2.04 (−1100)	−1.85 (−1000)

Study Basis: 1 Available volume is constant (fixed size aircraft)
2 Fixed number of passengers

* Effect of fuel on C.G. control is assumed manageable

GP75-0131-193

FIGURE 17
PAYLOAD/FUEL LOCATION STUDY

5.1.2 Tank Length and Dome Shape Studies - Three tank arrangements were considered for the baseline. Pertinent evaluation factors are summarized in Figure 18. While use of one continuous tank normally would result in high volumetric efficiency; in this case, tank deflection clearance requirements limited tank volume. This concept was never seriously considered, in any event, because: (a) it would not allow adequate c.g. control, (b) crash pressure heads would exceed normal burst pressure design, (c) tank bending deflection due to inertia would have made the fuselage diameter requirements too large.

An arrangement of four tanks was used on the first preliminary layout with the assumption that a tank length of 15.2 m (50 feet) could be more easily handled. This consideration was offset, however, by volumetric efficiency

Figure of Merit	Full Length Tank	1/2 Length Tank	1/4 Length Tank
Tank Volume/Available Volume - Center Fuselage	65.7%	67%	62.1%
Total No. of Tank Supports	6	12	24
Tank Fabrication Cost	High*	Low	High
Tank Installation Cost	High	Low	Medium
Tank Servicing Difficulty	Low	Medium	High

GP75-0131-194

Study Basis: 1 Fixed Fuselage Compartment Volume Based on Concept 1 Cross Section
2 Considered Tank Deflection Due to Inertia
3 Used Ellipsoidal Tank Dome Ends (a/b = 1.4)

*Isogrid Construction

FIGURE 18
TANK LENGTH COMPARISON

effects. Thus, a two-tank configuration with elliptical domed ends was ultimately selected for Concept 1 on the basis of maximized range. Detailed discussion and the results of this study are presented in Section 6.2.2 of Reference (2).

Three fuel tank dome shapes were studied to determine which had the lowest weight to volume efficiency. These were the hemispherical, torispherical, and ellipsoidal. The ellipsoidal fuel tank dome shape having an a/b = 1.4 ratio was selected on the basis of having the best range potential for the aircraft. The analysis and evaluation of the study is given in Section 6.2.2, Reference (2).

5.1.3 Ascent Trajectory - A trade study was conducted to reduce the weight of the active cooling system by minimizing the design heating rates. This was accomplished by departing from the original trajectory, which adhered to a constant 896 kPa (130 psi) absolute duct pressure line above Mach 4. Instead, a constant heating rate line was followed from Mach 5 to Mach 6, as shown in Figure 19. A net gain of 289 km (156 NM) in range resulted. The 32 km (105,000 feet) start of cruise altitude was then selected as the design point for the active cooling system. Further evaluation is given in Section 4.2, Reference (1).

5.1.4 Nacelle Cooling - A trade study was conducted which showed that range was increased 137 km (74 NM) by eliminating the requirement for nacelle surface cooling. The nacelle represents 9.4% of the total wetted surface area

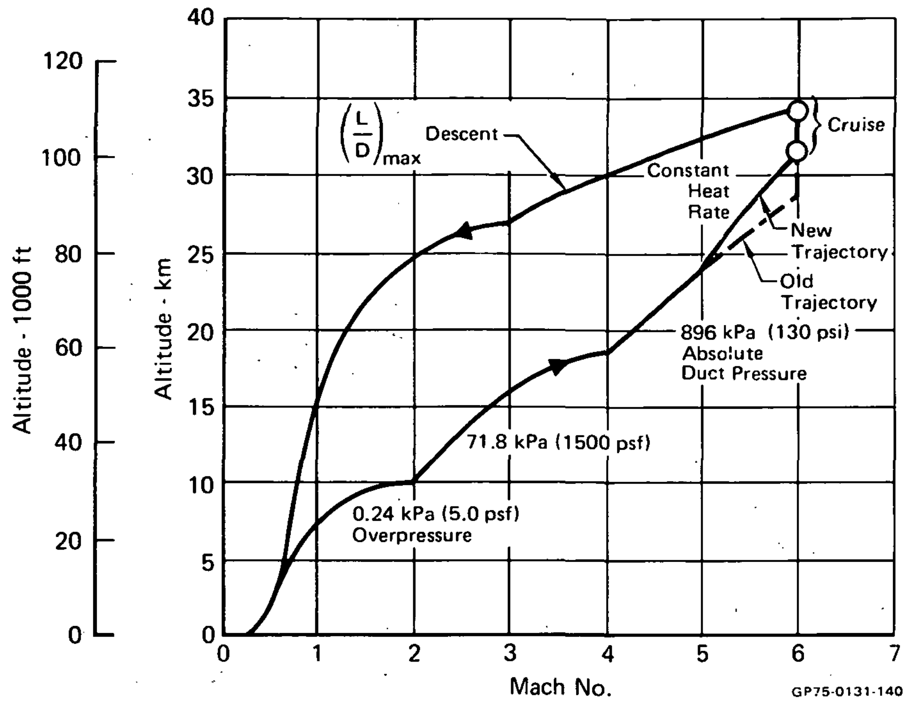


FIGURE 19
ASCENT TRAJECTORY TRADE-OFF
Altitude vs Mach Number

on the aircraft, but contributed 23.8% of the heat load to the original cooling system. Thus, hot nacelle structure was selected for the refined design. Details on this trade-off are found in Section 4.3, Reference (1).

5.1.5 Sonic Boom Overpressure - A tradeoff was conducted with Concept 1, on the effect of the sonic boom overpressure limit on the fuel and range used during ascent. The ascent paths considered are shown in Figure 20. These are the 0.10 kPa (3.0 lbf/ft^2) and 0.24 kPa (5.0 lbf/ft^2) climb paths described in Reference (5). The overpressure generated by Concept 1 will be nearly equal to the reference configuration, since both designs are about the same gross weight.

The tradeoff showed that the range of Concept 1 would be increased by over 741 km (400 NM) by following the 0.24 kPa (5.0 lbf/ft^2) climb path due to a fuel savings during climb of 7.7 Mg (17,000 lbm).

5.2 CONCEPT 1 TRADEOFF STUDIES

5.2.1 Tank Construction - Based on previous studies integral stiffening schemes were initially considered for the non-integral fuel tank. Strength analysis, based on the structural design criteria presented in Section 3 and

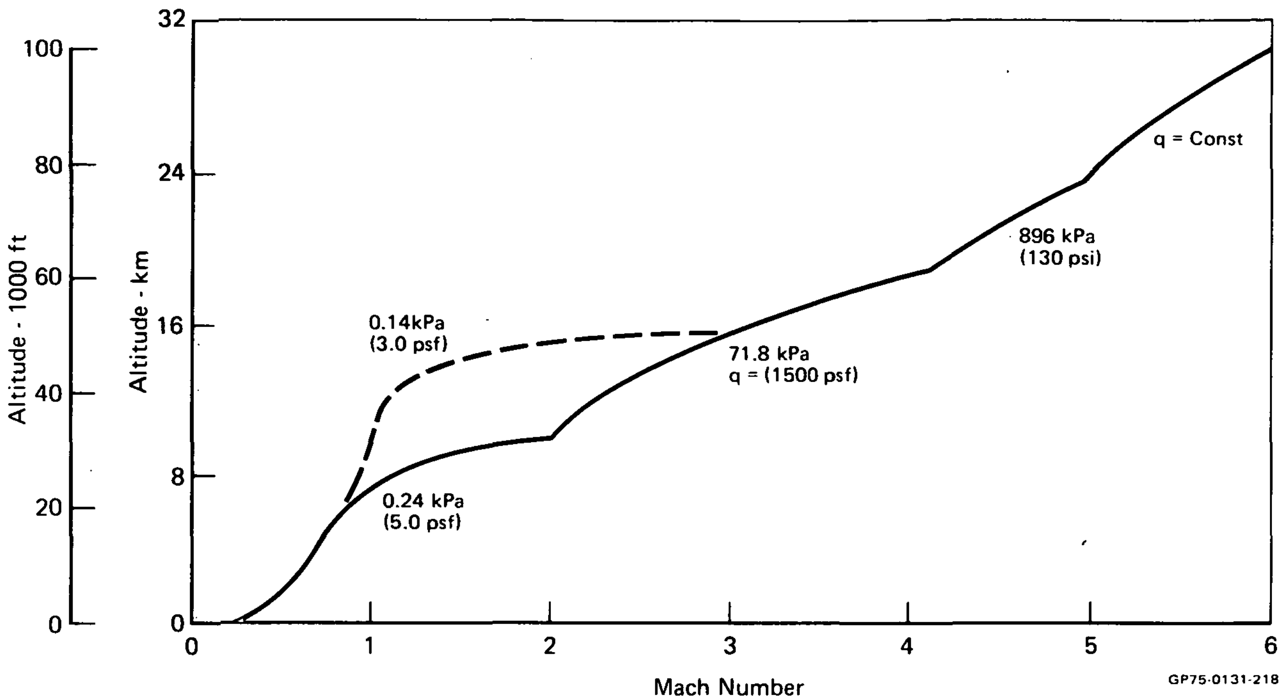


FIGURE 20
COMPARISON OF 0.14 kPa (3.0 PSF) AND 0.24 kPa (5.0 PSF) SONIC BOOM OVERPRESSURE CLIMB PATHS

the tank geometry described above, showed stiffening to be necessary. Once the burst pressure analysis established the tank thickness, the tank had adequate margins of safety in bending for the emergency landing condition and good margins of safety for all other conditions. Further discussion of this study may be found in Section 6.2.3 of Reference (2).

5.2.2 Actively Cooled Fuselage Covering - Two actively cooled structural concepts were studied: a beaded panel and a honeycomb sandwich panel. The beaded panel structure is composed of external actively cooled skin reinforced by a beaded panel containing coolant passages, stringer, and fuselage frame. The honeycomb sandwich panel structure contains the coolant passages inbedded in the core. Figure 21 summarizes the actively cooled structural concepts evaluated. The honeycomb sandwich panel concept was selected because its use would result in lighter aircraft. See Section 6.2.4, Reference (2), for a detailed discussion.

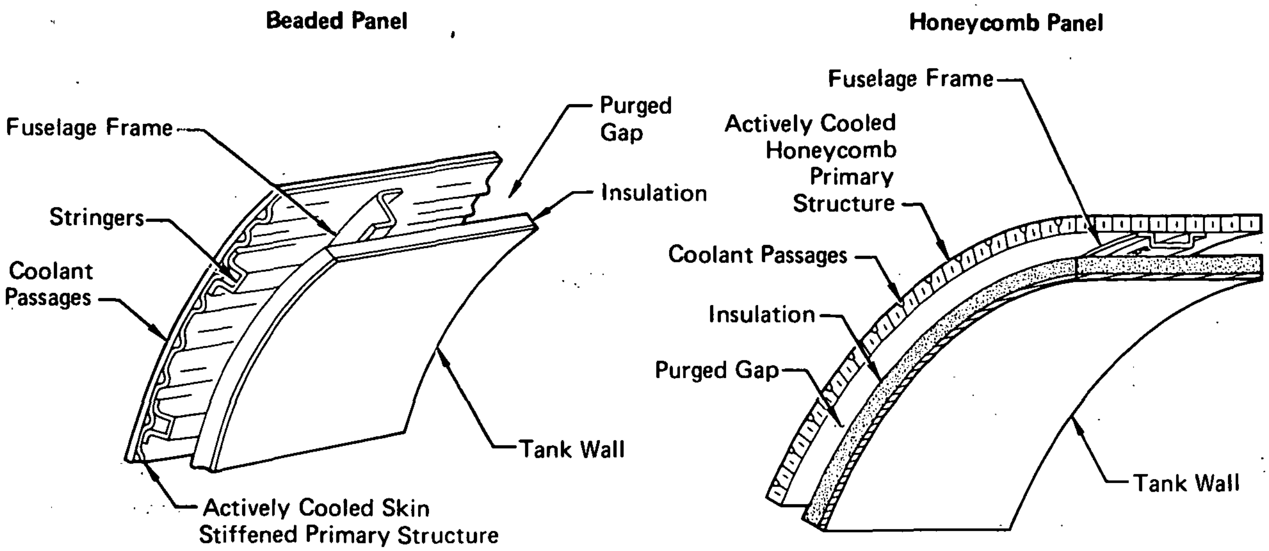


Figure of Merit	Beaded Panel	Honeycomb Panel
Weight* kg/m^2 (lbm/ft^2)	13.42 (2.75)	12.26 (2.51)
Inner Surface Interface with Substructure	Irregular	Smooth
Ability to Sustain Damage	Lower	Higher
Leakage Detectable	Yes	No
Number of Parts Interfacing	Three	Two

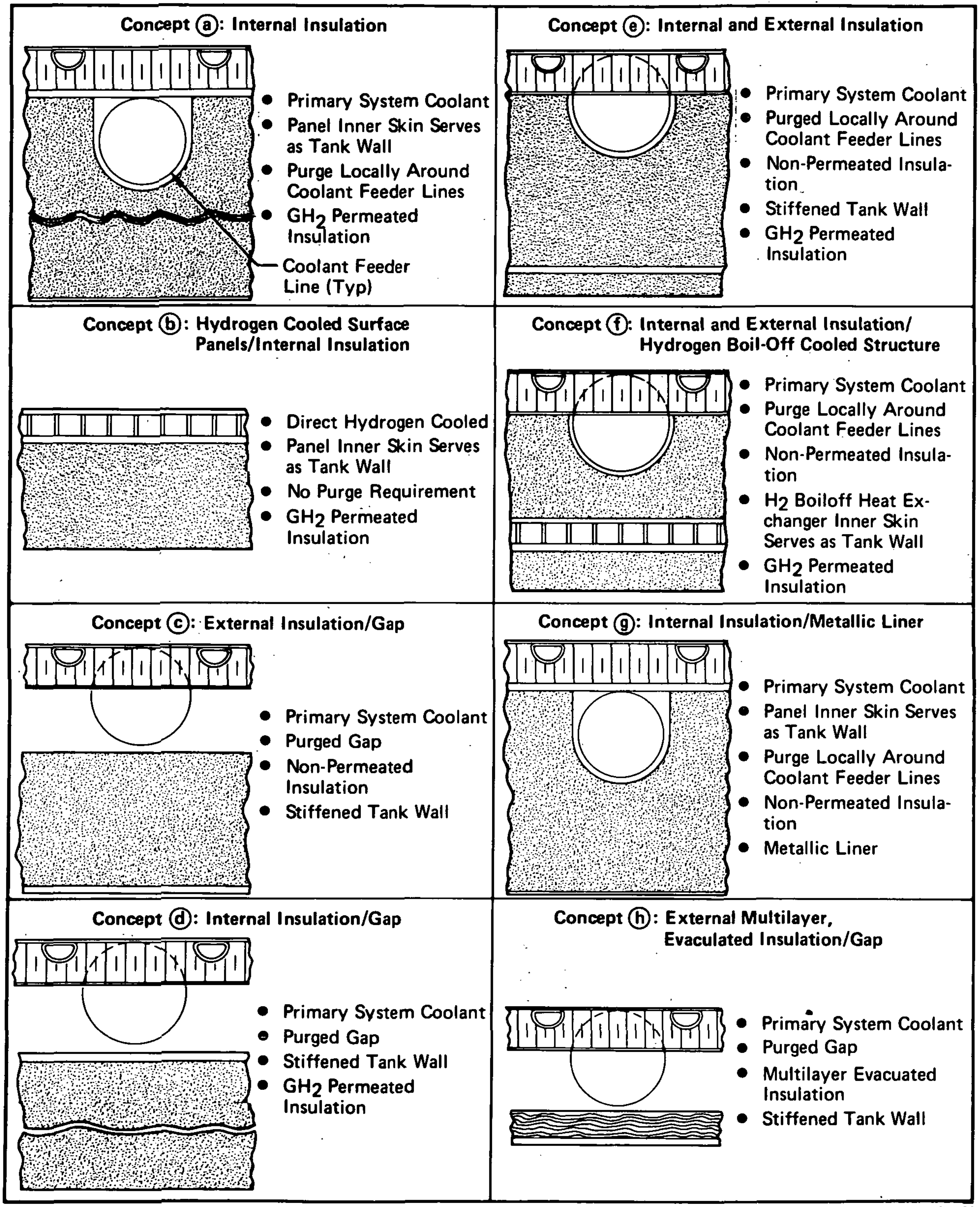
*Based on N_{cr} = 262.7 kN/m (1500 lbf/in.)

GP75-0131-216

FIGURE 21
ACTIVELY COOLED FUSELAGE COVERING

5.3 CONCEPT 2 TRADEOFFS

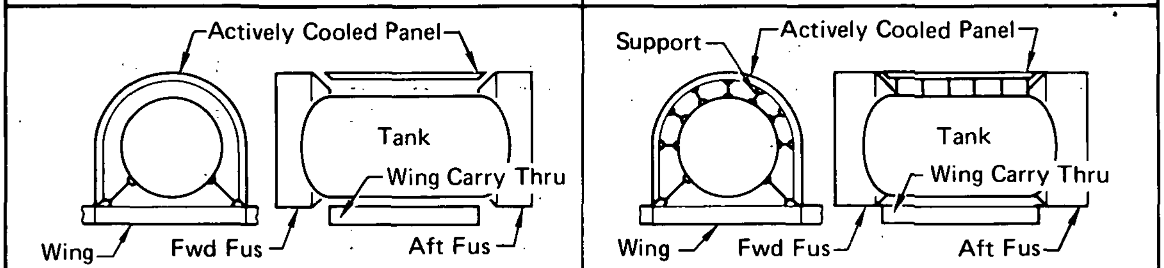
5.3.1 Thermal Protection System Selection - The primary tradeoff study conducted during the development of Concept 2 was the selection of a thermal protection system for the integral tankage. Eight thermal protection system concepts were evaluated, per Reference (6), as shown in Figure 22. The range differences among the concepts were found to be small enough, in most cases, to permit other considerations in the selection, including the fabricability, inspectability and maintainability. As a result, concept C was adopted for both integral tank Concepts 2 and 3. Further evaluation and analysis is provided in Sections 8.1, 8.2, and 8.3, of Reference (1).



GP75-0131-120

FIGURE 22
THERMAL PROTECTION SYSTEM SELECTION

5.3.2 Semi-Structural Versus Non-Structural Tank Covering - A trade study was conducted to compare non-structural actively cooled panels with panels that are partially effective (semi-structural). Two concepts were considered: (1) a semi-structural fuselage cover (formed of actively cooled panels) which was independently supported from the upper wing surface, and (2) a non-structural cover with the actively cooled panels supported individually from the tank. Figure 23 compares the two concepts. The semi-structural cover was selected for Concept 2. Analysis showed that the semi-structural cover, by acting as secondary bending structure, reduced tank bending loads enough that approximately 998 kg (2,200 lbm) of tank weight could be eliminated. Detail evaluation and analysis is provided in Section 6.3.2 of Reference (2).



Semi-Structural Wing Supported		Non-Structural Tank Supported	
PRO	**CON**	**PRO**	**CON**
Can Provide Semi-Structural Fuselage Cover Simple Assembly	A Large Slip Joint Required	A Large Slip Joint Eliminated	Slip-Joints Required Around Perimeter of Each Actively Cooled Panel Fabrication Cost High Due to Many Links and Complicated Assembly

FIGURE 23
SEMI-STRUCTURAL vs NON-STRUCTURAL
INTEGRAL TANK COVER - CONCEPT 2

5.3.3 Tank Construction - To carry the Concept 2 fuselage loads a stiffened tank structure was chosen. Isogrid construction with stiffening elements on the external surface was selected for Concepts 2 and 3, rather than the 0°-90° waffles, because of the potential weight saving, illustrated by Figure 24. Further discussion of this study is presented in Section 6.3.1 of Reference (2).

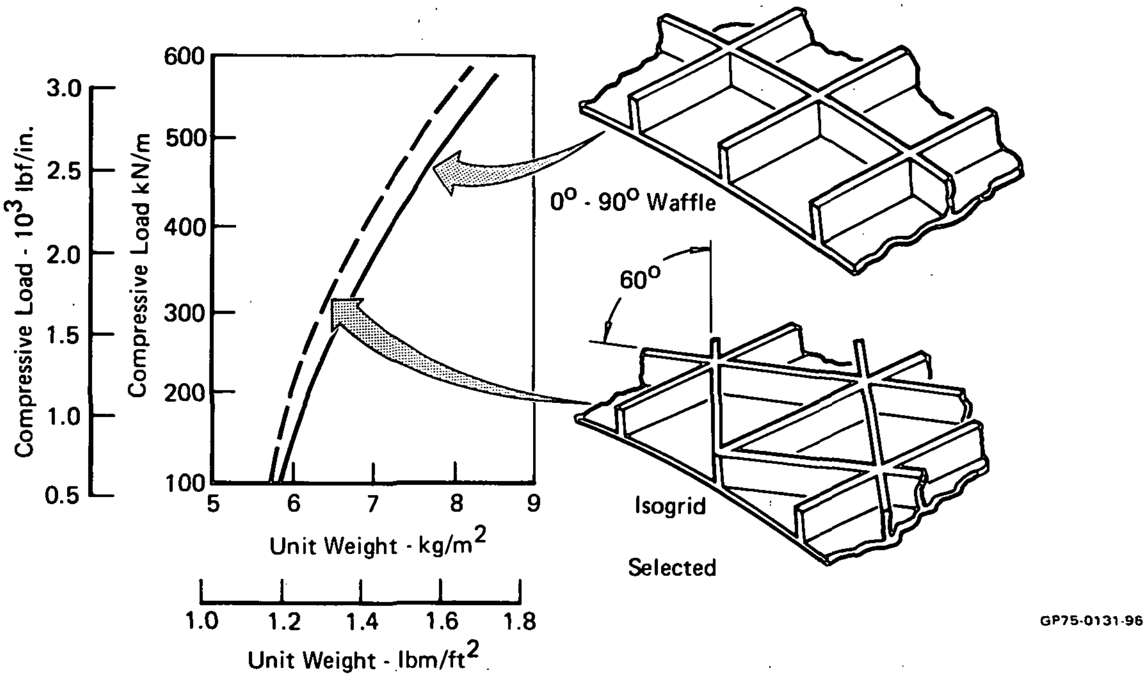


FIGURE 24
INTEGRAL TANK WALL CONSTRUCTION

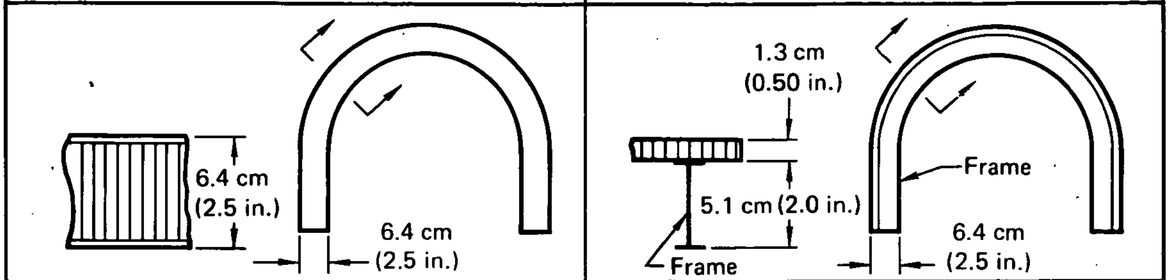
5.3.4 Actively Cooled Cover Structure Design - A full depth honeycomb concept was qualitatively compared, for Concept 2, with a combined honeycomb/frame concept, as shown in Figure 25. The reason for selection of the honeycomb/frame concept was primarily prevention of the potential decrease in volumetric efficiency inherent in the full depth honeycomb design.

5.4 CONCEPT 3 TRADEOFFS

5.4.1 Semi-Structural Versus Non-Structural Fuselage Covering Study - An investigation of the Concept 2 semi-structural panels for Concept 3 revealed that structural support arrangement would either induce excessive thermal stresses or would require such large frames that the net effect on weight and aircraft size would degrade range capability. Therefore, the panels were designed to be non-structural. The trade is discussed in Section 6.4.1 of Reference (2).

5.4.2 Tank Cross Section Optimization - A trade study was conducted to ascertain the most efficient fuel tank shape for the elliptical fuselage cross section. A number of multi-bubble tank configurations, ranging from three to

All Honeycomb *		Honeycomb with Frames	



Pro	Con	Pro	Con
Simple Design No Attachment Required Except at Panel Perimeter	Machine Fittings Required for Each Panel Joint Can not Provide Space for Active Cooling Feeder Lines, thus Reducing Volumetric Efficiency of Fuel	Provides Concentrated Load Path Provides Adequate Space Between Frames for Feeder Lines	Additional Fasteners to Attach Frame

*Assumption of panel supported every 3.05 m (10 ft)

GP75-0131-199

FIGURE 25

COVER STRUCTURE COMPARISON

seven bubbles, were studied, based on the ground rules stated in Figure 26. The five bubble tank shows the best overall efficiency. Detailed discussion can be found in Section 6.4.2 of Reference (2).

5.5 DESIGN SYNTHESIS STUDIES

Qualitative design studies were conducted along with the tradeoff studies. The more pertinent studies relating to the development of the aircraft concepts are discussed below.

5.5.1 Actively Cooled Panel Arrangement - The two actively cooled panel arrangements considered are shown in Figure 27. The staggered panel scheme locates the front edge of alternating panels at the mid points of the panels alongside it. The other scheme aligns the front and aft ends of the panels. The staggered arrangement for the 1.2 m x 6.1 m (4 foot by 20 foot) actively cooled panels was selected because the aligned panel arrangement requires larger coolant feeder lines, since spacing would be 6.1 m (20 ft) rather than 3.05 m (10 ft). This would decrease aircraft volumetric efficiency for fuel containment. In other areas such as the wings and vertical tail where the

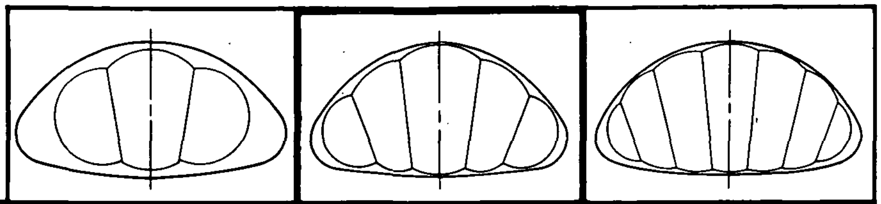


Figure of Merit	Three Bubble	Five Bubble	Seven Bubble
Tank Cross Sectional Area / Fuselage Cross Sectional Area	73%	90%	91%
Weight Efficiency lbm Fuel / lbm Structure	17.6	19.6	18.2
Fabrication Cost	Low	Moderate	High

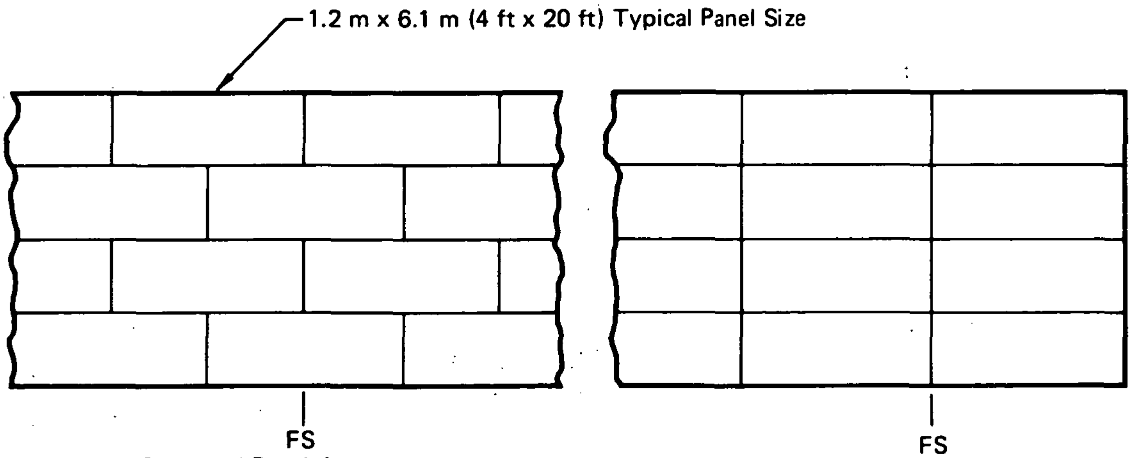
Fuselage Cross Sectional Area = 40.69 m^2 (438 ft^2)

GP75-0131-200

Tank Geometry Construction Guide Lines

1. Non-Warping Web Planes
2. Straight Line Element Intersection
3. Minimum Clearance of 0.09m (3.5 in.) between the Tank Structure and the External Mold Line.
4. Common Volume Allowed for Control and Subsystem Line Routing at Each Side of the Tank.

FIGURE 26
TANK CROSS SECTION OPTIMIZATION



Staggered Panel Arrangement (Used on Fuselage)

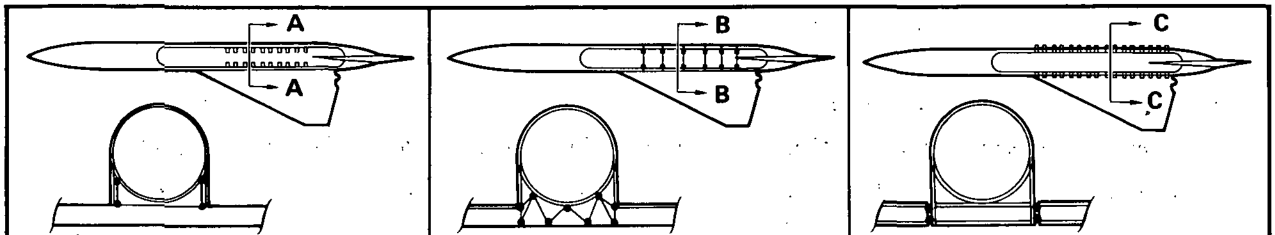
Aligned Panel Arrangement (Used on Wing and Vertical Tail)

GP75-0131-195

FIGURE 27
ACTIVELY COOLED PANEL ARRANGEMENT

volume is of lesser importance the panels were aligned in order to permit the weight saving afforded by the larger line size.

5.5.2 Wing/Fuselage Attach Development - Three wing/fuselage structural integration design concepts were evaluated for the integral tank in Concept 2. Figure 28 presents a qualitative comparison of the concepts. The spar carry-through configuration was selected because all thermal deflections can be accommodated while maintaining a stable load path for primary aircraft loads. Further description of the selected concept may be found in Section 2.1 of Reference (2).



A-A Spar Carry Through		B-B Truss Carry Through		C-C Side-Mounted Wing	
Pro	**Con**	**Pro**	**Con**	**Pro**	**Con**
1. Compensates for Thermal Growth 2. Least Number of Major Components to be Assembled	1. Complicated Link Assembly Due to Close Tolerance Required	1. Inboard and Outboard Wing Deflection Compensated	1. Difficult to Compensate Thermal Growth in Fore-Aft Direction 2. Excessive Concentrated Loads on Tank Rings	1. Simple Wing Attach	1. Difficult to Provide for Thermal Growth in Multiple Lugs 2. Wing Weight Increases with Segmented Carry-Through

GP75-0131-197

FIGURE 28
WING/FUSELAGE ATTACH STUDY, CONCEPT 2

6. AIRCRAFT DESCRIPTION

A detailed description of the selected aircraft configuration for each of the three basic concepts is presented in this section, including the propulsion system and aircraft subsystems as well as the thermo-structural arrangements. Each description is followed by layout drawings.

In addition, the relative producibility of each tank concept in the major areas of manufacturing such as assembly, forming, machining and welding is presented. Also, the relative degree-of-difficulty to accomplish maintenance is provided so as to derive some insight into the operational costs associated with different tank concepts.

6.1 CONCEPT 1 (NON-INTEGRAL TANKS)

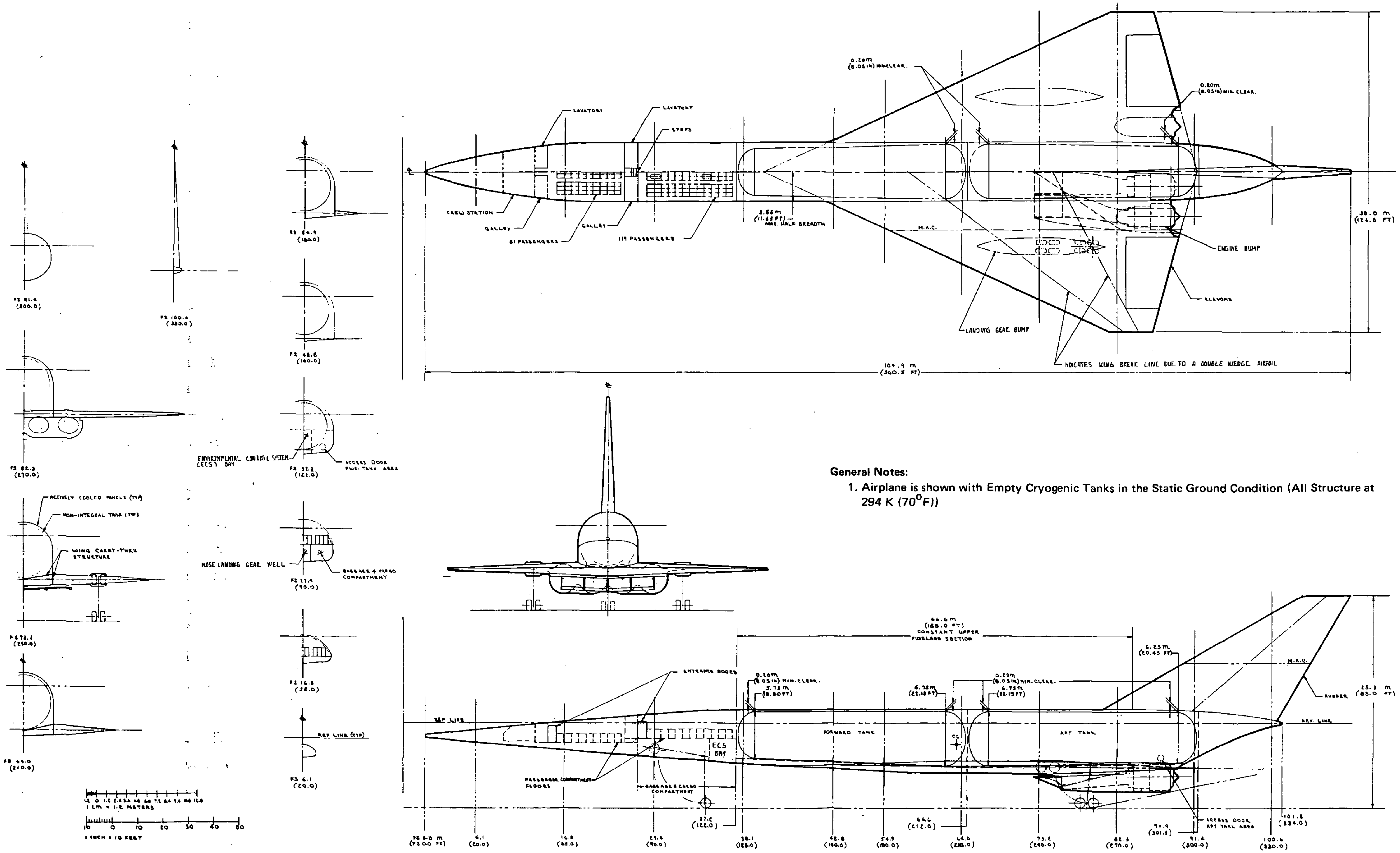
The Concept 1 general arrangement is presented in Figure 29. The delta wing has a 3 percent thickness-to-chord ratio and a modified double wedge airfoil with a fixed 65° leading edge sweep. Wing incidence is set at 1/2°.

Basic flight control surfaces are conventional elevons, and a fixed single vertical fin with a split rudder which doubles as a speed brake. The aircraft is powered by four hydrogen-fueled turboramjets located in an integrated engine nacelle module underneath the fuselage.

Figure 30 depicts the major structural assemblies. The fuselage consists of a forward, center and aft section. The forward fuselage includes the crew station, passenger cabins, cargo and baggage areas, and nose landing gear well. The center fuselage consists of the primary external shell structure and contains the two non-integral LH_2 fuel tanks. The center fuselage structure is split into two sections separated by a bulkhead for ease of manufacturing. The aft fuselage consists of the vertical tail and aft aerodynamic fairing. The wing is subdivided at the center fuselage bulkhead into a forward and aft wing.

Actively cooled surface panels maintain the structural temperature to a maximum of 394 K (250°F). This low temperature allows the use of aluminum for all external airplane structure.

6.1.1 Structural Arrangement - The fuselage is a full monocoque structure. The structural arrangement is presented in Figure 31. The primary load carrying structure consists of a series of interconnected 1.2 m x 6.1 m (4 ft x 20 ft) panels which cover the entire external surface except for the nacelle module. These panels are constructed of aluminum honeycomb sandwich.



General Notes:

1. Airplane is shown with Empty Cryogenic Tanks in the Static Ground Condition (All Structure at 294 K (70°F))

FIGURE 29
CONFIGURATION, CONCEPT 1

GP75-0131-14

Volume Summary

Forward Fuselage FS 0.00-37.2m (FS 0.00-122.0 ft)	680 m^3 (24,100 ft^3)
Center Fuselage FS 37.2-91.9m (FS 122.0-301.5 ft)	2,420 m^3 (85,600 ft^3)
Aft Fuselage FS 91.9-101.8m (FS 301.5-334.0 ft)	110 m^3 (3,800 ft^3)
Total Fuselage	3,210 m^3 (113,500 ft^3)
Tank Volume	1,620 m^3 (57,200 ft^3)
Tank Volume/Center Fuselage Volume*	76%
Tank Volume/Total Fuselage Volume	50%

*Minus 306 m^3 (10,800 ft^3) wing carry-thru structure volume

Physical Characteristics

Item	Wing		Vertical Tail	
S_{theo}	1,070 m^2	(11,530 ft^2)	180 m^2	(1,970 ft^2)
AR	1.35	–	2.00	–
λ	0.10	–	0.27	–
b	38.0m	(124.8 ft)	–	–
b/2	19.0m	(62.4 ft)	13.5m	(44.4 ft)
C_R	51.2m	(168.0 ft)	21.4m	(70.2 ft)
C_T	5.1m	(16.7 ft)	5.6 m	(18.5 ft)
MAC	34.2m	(112.3 ft)	15.1m	(49.5 ft)
ΛLE (deg)	65	–	60	–
ΛTE (deg)	–15	–	30	–
Incidence (deg)	+1/2	–	–	–
Dihedral	0	–	–	–
Thickness Ratio	0.03	–	0.03	–

Performance Summary

Range	8.69 Mm (4,690 NM)
Payload (200 Passengers)	21.8 Mg (48,000 lbm)
Operating Weight Empty	190 Mg (419,234 lbm)
Takeoff Gross Weight	299 Mg (659,234 lbm)

Propulsion

(4) GE5/JZ6-C, 400 kN (90,000 lbf) T_{SLS} per Engine Uninstalled

Total Inlet Capture Area ($A_{C_{total}}$) = 15.8 m^2 (170 ft^2)

	Tire Size
Main Gear	1.27m x 0.51m (50 in.x20 in.)
Nose Gear	1.27m x 0.51m (50 in.x20 in.)

Wetted Area

Fuselage	1,910 m^2 (20,600 ft^2)
Nacelle	340 m^2 (3,640 ft^2)
Wing	1,370 m^2 (14,800 ft^2)
Vertical Tail	390 m^2 (4,150 ft^2)
Total	4,010 m^2 (43,190 ft^2)

Fuel Distribution

Tank Section	Type	*Usable Volume	Fuel Weight
Forward	Nonintegral	740 m^3 (26,225 ft^3)	52.6 Mg (115,900 lb)
Aft	Nonintegral	800 m^3 (28,075 ft^3)	56.3 Mg (124,100 lb)
Total		1,540 m^3 (54,300 ft^3)	108.9 Mg (240,000 lb)

*5% of tank volume allowed for ullage, rings, etc. ∴ usable volume = 0.95 tank volume
Fuel-liquid hydrogen at 20.3 K (–423°F) ρ (density) = 70.8 kg/m^3 (4.42 lbm/ft^3)

Fineness Ratio	13.45	–
Total Aircraft Volume	4,300 m^3	(152,000 ft^3)
Planform Area	1,380 m^2	(14,825 ft^2)
Max Cross Sectional Area	99.9 m^2	(1,075 ft^2)
Less Capture Area	15.8 m^2	(170 ft^2)
Net Cross Sectional Area	84.1 m^2	(905 ft^2)
Mach No. (Cruise)	6	–
$V^{2/3} \div S_p$ Factor	0.178	–

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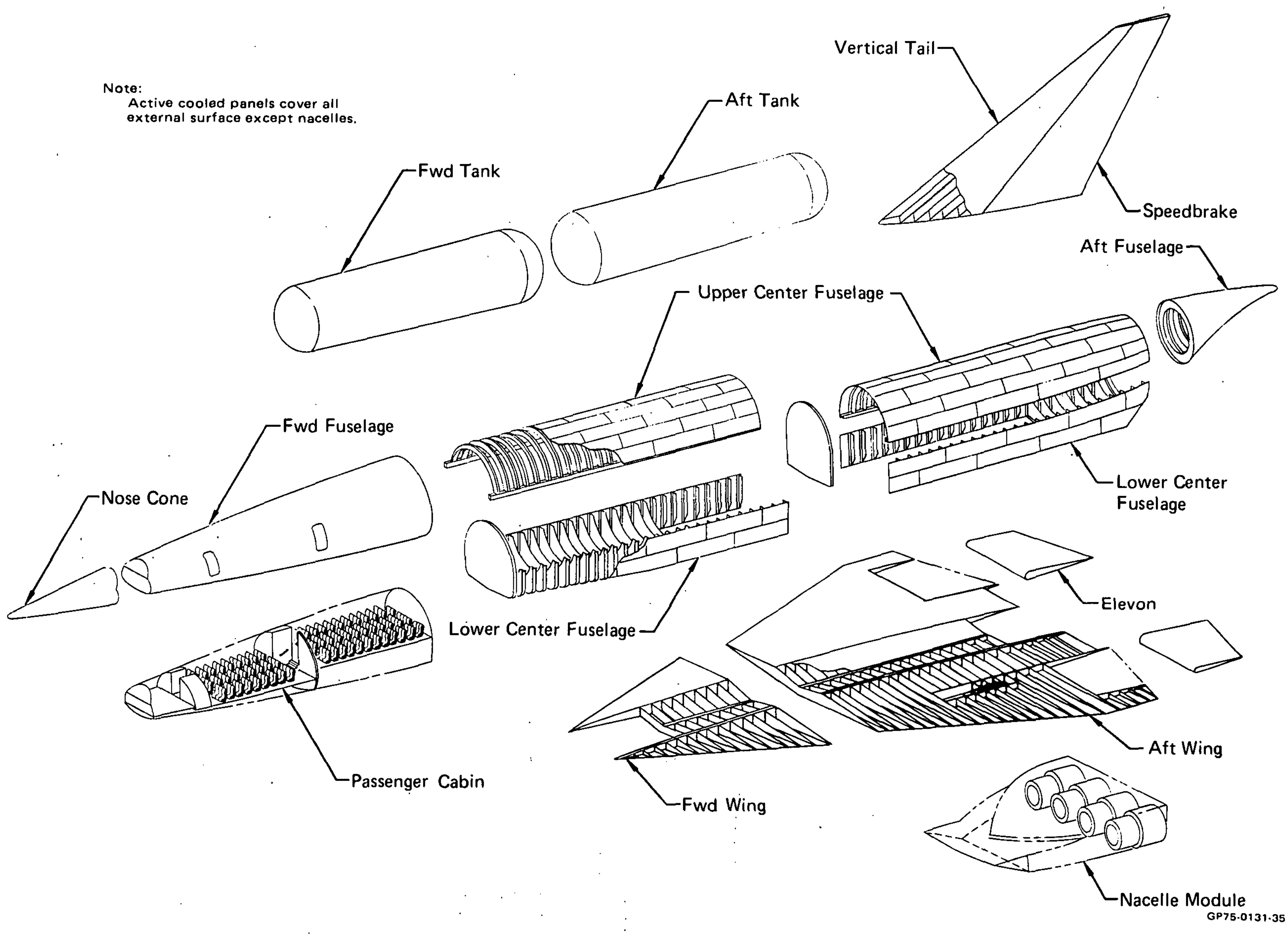
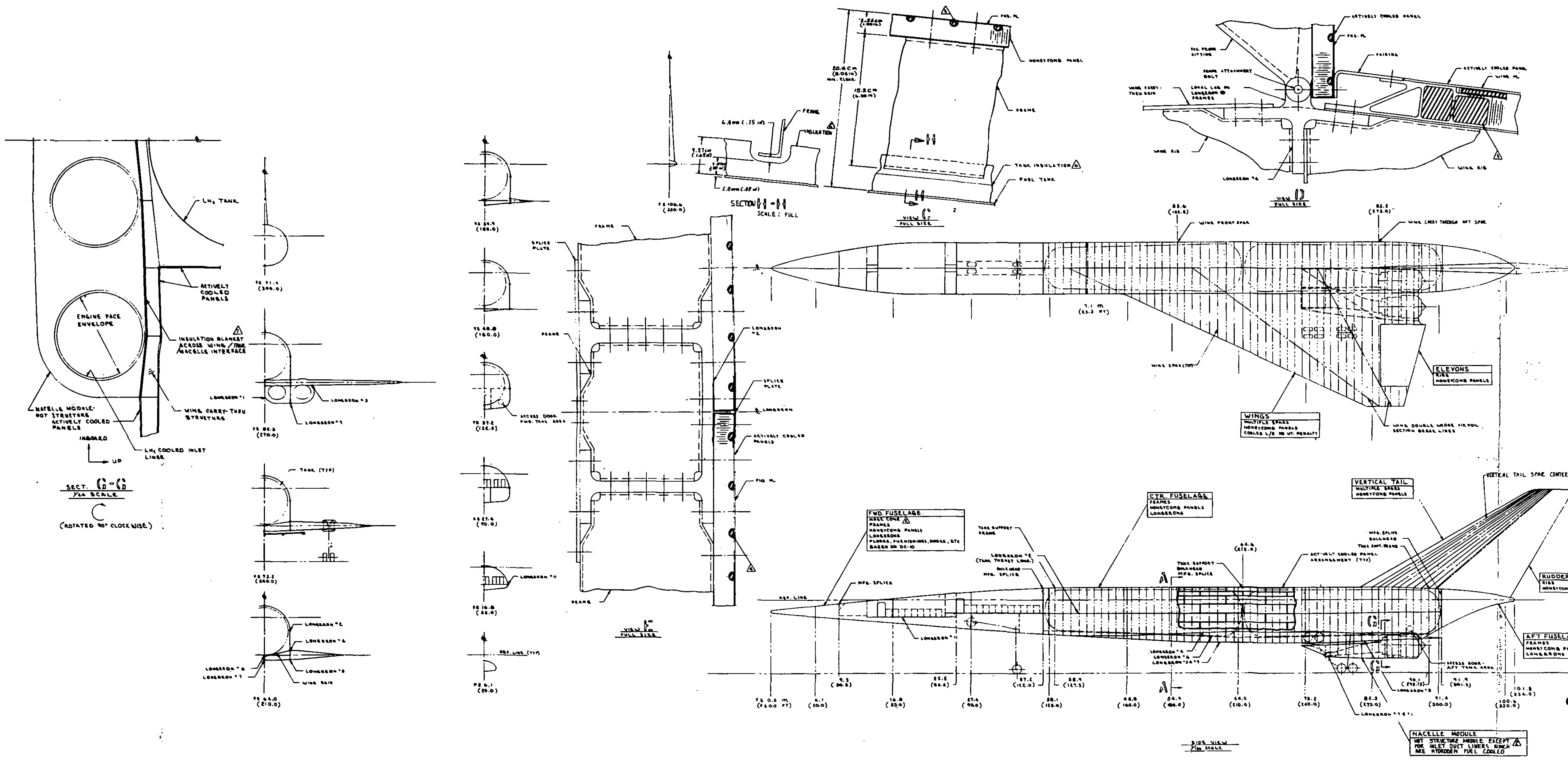
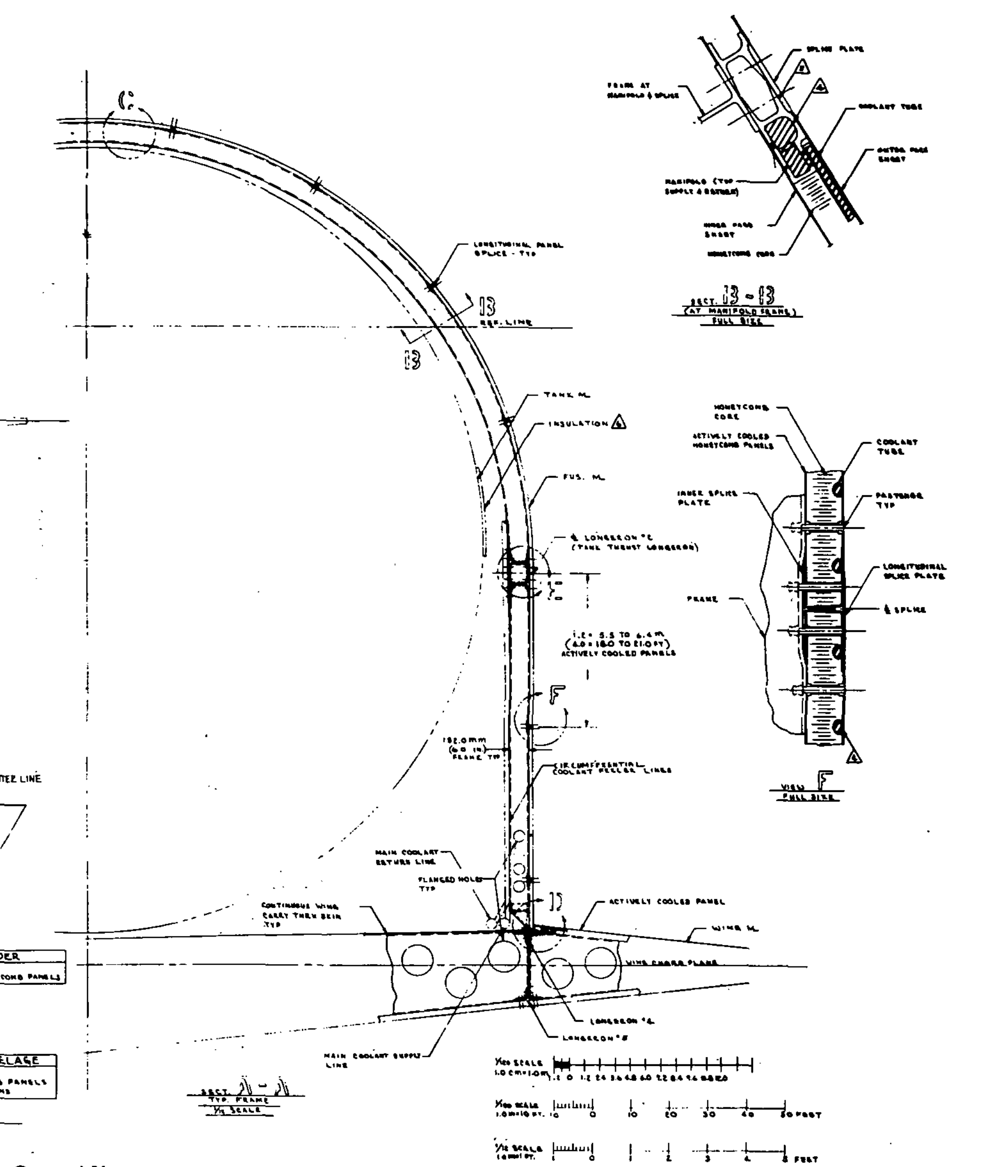


FIGURE 30
CONCEPT 1 ASSEMBLY BREAKDOWN





General Notes:

1 Honeycomb panels and support structure are not shown on the forward fuselage, aft fuselage, elevons, vertical tail, and rudder

2 Special consideration was given to the panels in this area due to the higher heat load, which makes these panels weigh 2.0 times the average panel weight in the center fuselage

3 Airplane is shown with the empty cryogenic tanks in the static ground position (all structure at 294 K (70°F))

4 Indicates methanol/water coolant

5 A high thermal conductive adhesive is applied between surfaces

6 Insulation material, 64.1 kg/m^3 (4 lbm/ft^3) closed cell fiberglass reinforced polyurethane with an aluminized mylar covering

7 Insulation material, 56.0 kg/m^3 (3.5 lbm/ft^3) high temperature glass fiber

FIGURE 31
STRUCTURAL ASSEMBLY, CONCEPT 1

GP75-0131-3

Dee-shaped coolant tubes, for the circulation of methanol/water coolant, are imbedded in the core and are bonded to the inside of the external face sheet.

a. Forward Fuselage, Wing, Vertical Tail, and Nacelle Structure - The forward fuselage contains the passenger compartments, crew station, and landing gear/ECS/baggage bays. The internal bulkheads, floors and webs are distributed to separate these compartments and redistribute primary loads.

The actively cooled panels are the primary fuselage load carrying structure. These panels are supported on 15 cm (6 inch) frames spaced approximately 0.91 m (3 ft) apart. The compartments are pressurized to a minimum of 75.2 kPa (10.9 psi) absolute. Also, two bulkheads are used to subdivide the compartments into three sections, one for the crew station and two for the passenger compartment. Internal furnishings and accommodations such as galleys and lavatories are patterned after the DC-10.

The wing is a multispar continuous carry-through structure. The theoretical wing area is 1,071 m^2 (11,530 ft^2). The fuselage formers are pinned to the top of the wing spars as shown in View D of Figure 31. Maximum spar bending load occurs at the fuselage sidewall under the 2 g taxi condition.

The exposed surface of the wing is covered with actively cooled structural panels. The upper wing cover within the wing carry-through structure is a conventional skin/stringer design.

The vertical tail has a double wedge airfoil section and a projected area of 183 m^2 (1,970 ft^2) with a maximum thickness ratio of 3%. It also incorporates spar construction with actively cooled panel covers, and is supported off the fuselage structure.

The nacelle module is the only major aircraft component that is not actively cooled. The rationale for this decision has been presented earlier in this volume and is discussed in more depth in Reference (1). Because structural temperatures would approach 1144 K (1600°F), superalloys are used. The module is attached to the fuselage structure through links which allow for thermal deflections.

b. Center Fuselage/Tank Structure - Most of the design effort was concentrated in the fuselage/tank area. This section is a full monocoque structure enclosing two non-integral, circular cross section, hydrogen tanks. The surface consists of actively cooled panels supported on 15 cm (6 in.) deep frames spaced approximately 0.91 m (3 ft) apart. The frames are pinned to the upper wing surface as previously stated.

The center section shell is divided into two sections by a bulkhead which provides part of the support for the two non-integral fuel tanks. The two tanks hold 108.9 Mg (240,000 lbm) of liquid hydrogen. The tank structural design details are presented in Figure 32. The circular tank consists of welded non-stiffened plain skin and machined elliptically domed ends.

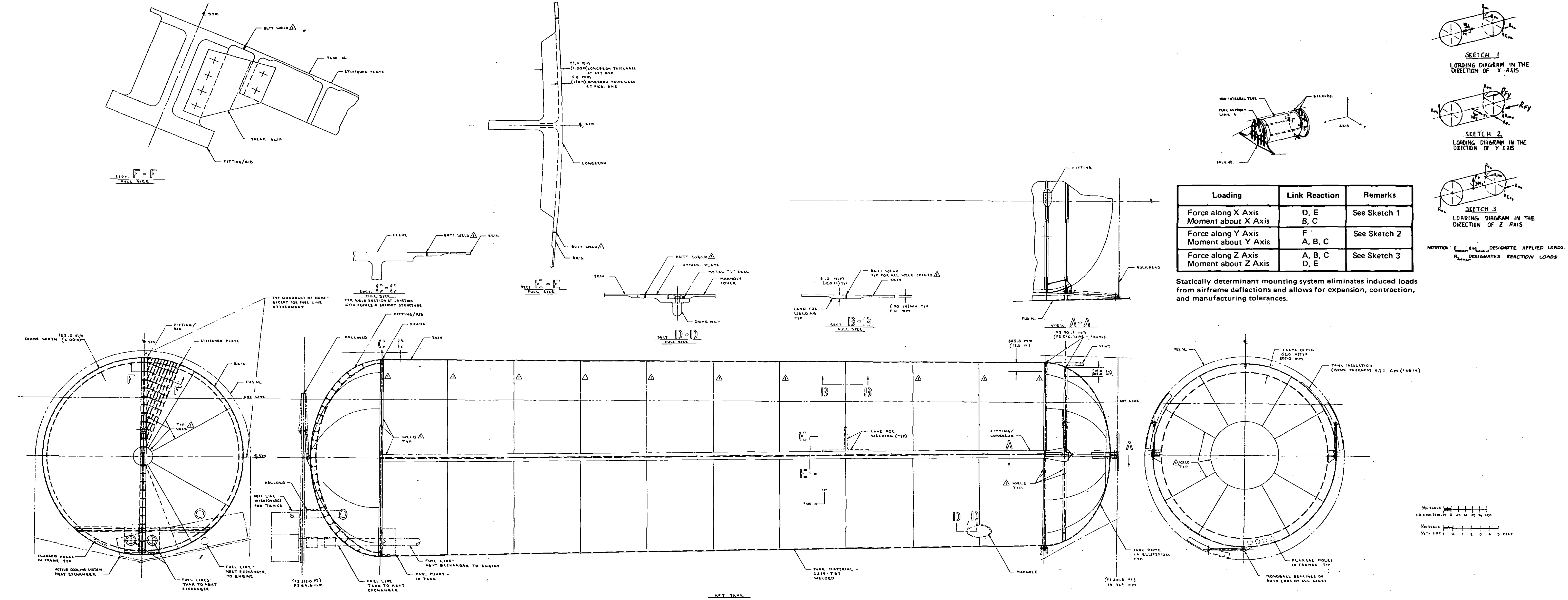
Each tank is supported, as illustrated in Figure 32, at four points in such a manner that the tank support is statically determinate. Thus, no loads are induced into the tanks by fuselage bending or thermally induced relative motion.

6.1.2 Thermal Protection - The active cooling system maintains the external structural temperatures at an average of 366 K (200°F). Thermal protection in the center fuselage area basically consists of the actively cooled external panels and insulation packages over the tankage external surfaces. The insulation is sized to minimize the range penalty caused by the weight of insulation/fuel boiloff. Thermal protection in the forward fuselage consists of the actively cooled external panels and insulation positioned around the crew station and passenger compartment walls to maintain the average internal wall temperature at 305 K (90°F). The wing, vertical tail, and aft fuselage areas are protected by the actively cooled panels only.

Insulation thickness in the center fuselage varies from 2.03 cm (0.80 inch) at the fuselage frames to 4.27 cm (1.68 inches) between frames. The selected material is 64.1 kg/m^3 (4 lbm/ft^3) closed-cell, fiberglass reinforced, polyurethane foam with an aluminized mylar covering. The void between the surface panels and the tank wall is purged with dry nitrogen gas to a constant 3.45 kPa (0.5 psi) gage to prevent the build-up of gaseous hydrogen and water vapor condensation. The nitrogen is stored in a bottle in the equipment compartment, aft of the nose landing gear compartment and connected to the tank compartment by supply lines.

a. Active Cooling System - The heat exchanger for the active cooling system is located between the fuel tanks, as shown in Figure 33. The basic panel coolant distribution system scheme is also shown. The heat exchanger transfers the heat absorbed by methanol/water coolant directly to the fuel.

Coolant is distributed to the surface panels by a set of supply and return lines on either side of the fuselage extending forward from FS 64 m



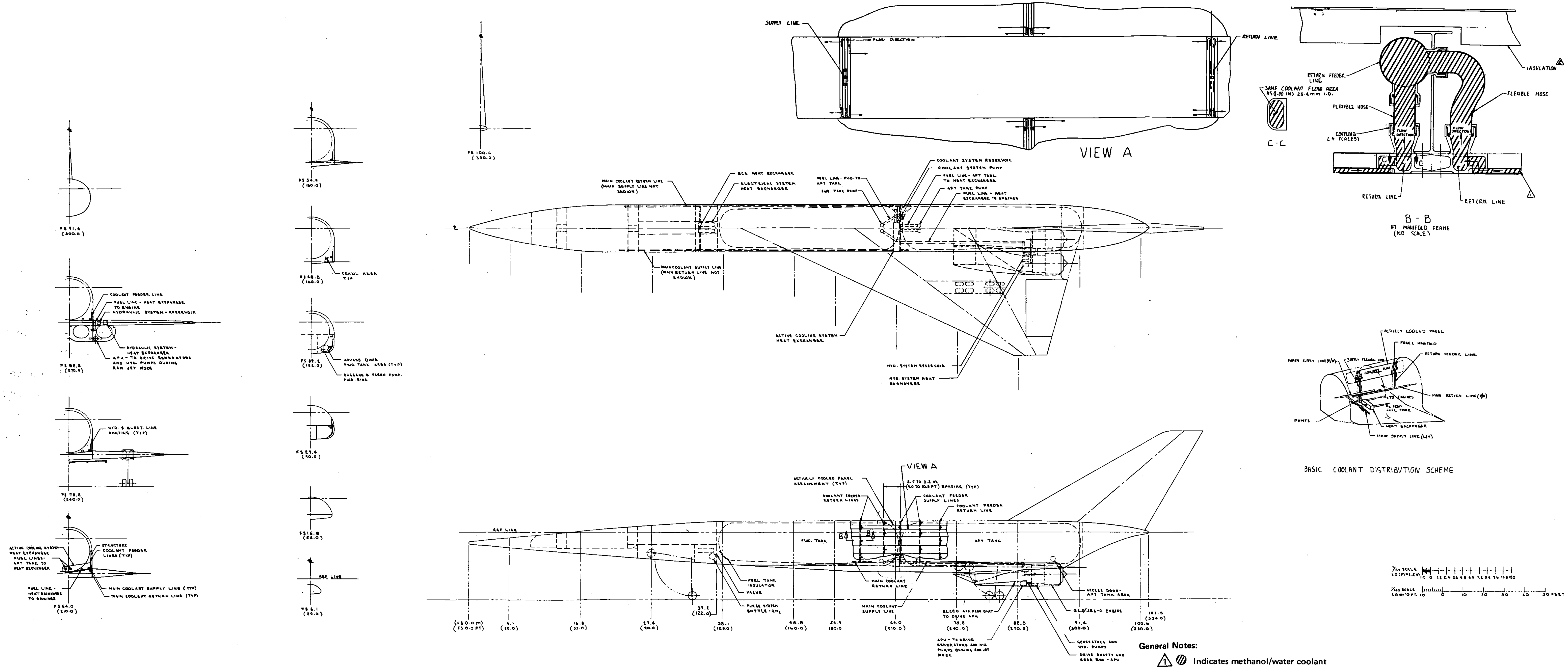
Loading	Link Reaction	Remarks
Force along X Axis Moment about X Axis	D, E B, C	See Sketch 1
Force along Y Axis Moment about Y Axis	F A, B, C	See Sketch 2
Force along Z Axis Moment about Z Axis	A, B, C D, E	See Sketch 3

Statically determinant mounting system eliminates induced loads from airframe deflections and allows for expansion, contraction, and manufacturing tolerances.

FIGURE 32
TANK STRUCTURE, CONCEPT 1

GP75-0131-4

General Notes:
1. Tank Material: Aluminum 2219-T87
2. Tank is assembled primarily by automatic tungsten insert gas welding



General Notes:

1. Indicates methanol/water coolant
2. Insulation material, 64 kg/m^3 (4 lbm/ft^3) closed cell fiberglass reinforced polyurethane with an aluminized mylar covering

FIGURE 33
ACTIVE COOLING SYSTEM, CONCEPT 1

GP75-0131-6

(210 ft); a set of lines on either side of the fuselage extending aft from FS 64 m (210 ft) to the aft fuselage area; a set of lines branching out of this aft fuselage ducting up into the vertical tail area; and a set of lines extending directly outboard into each wing area. Feeder lines (both supply and return) to the manifold region of each panel are spaced approximately 3.05 m (10 feet) apart, based on 6.1 m (20 ft) panels in a staggered arrangement. Each feeder line services two adjacent panels such that the flow in adjacent panels is in opposite directions.

The main distribution lines for the forward fuselage also provide coolant for the ECS and the electrical system heat exchangers. The main distribution lines returning from the aft section of the center fuselage supply coolant for the hydraulic system heat exchanger. Pumps adjacent to the main heat exchanger are sized to deliver the required coolant flow at a 1.03 MPa (150 psi) absolute head.

b. Actively Cooled Panel Joint and Manifold Design - Panel coolant manifolds are located at the ends of each panel to distribute flow to and from each tube. The manifolds are supplied through flexible connections.

The panel joint provides a basic airframe loadpath and is designed to minimize leakage of the nitrogen purge gas. The space between the tank and the external panel is adequate to allow inspection of the joint for leakage from inside the airplane. Section B-B of Figure 33 shows the joint design.

6.1.3 Propulsion System - The propulsion system consists of four variable cycled General Electric GE5/JZ6-Study C turboramjets rated at 400 kN (90,000 lbf) thrust each. A two-dimensional external compression inlet with a vertical ramp is provided for each engine. The engines are cantilevered and flange mounted at the engine face to the diffuser section of the inlets. The four engines, inlets, and exhaust ducts are integrated into the fuselage body to provide low nacelle drag on the total vehicle.

6.1.4 Landing Gear - The landing gear is a conventional tricycle type arrangement. The cantilever-type main landing gear is a four-wheel bogey with 1.27 m (50 inch) by 0.51 m (20 inch) tires. It is hydraulically retracted forward into the wing. Free-fall emergency extension of the main gear occurs after the uplock and door mechanism are released by an emergency hydraulic accumulator. The nose landing gear is hydraulically retracted forward into the fuselage and the emergency extension is the same as for the main gear. The dual nose gear wheels also have 1.27 m (50 inch) by 0.51 m (20 inch) tires. The nose wheel strut is extended pneumatically to achieve a 6° wing incidence for takeoff.

6.1.5 Airplane Subsystems - To compensate for the thermal expansion and contraction, bellows or mechanical compensators are required in the control, fuel, cooling, and hydraulic systems.

a. Fuel System and Fuel Pressurization System - The two cryogenic tanks are interconnected by fuel lines, with the forward tank feeding into the aft main feed tank as shown in Figure 33. The aft tank fuel lines extend forward to the active cooling system heat exchanger, which is located between the tanks. The fuel lines are routed aft along the bottom outboard side of the aft tank in the center fuselage crawl areas to the nacelle area and drop down to the engine pumps. Fuel transfer is accomplished with electrically driven boost pumps which can operate at or near zero suction head.

An autogenous fuel pressurization system has been implemented in the overall system design. The system, shown conceptually in Figure 34, provides a constant 138 kPa (20 psi) internal tank gage pressure throughout the flight profile. This provides adequate net positive suction pressure from the engine bleed GH_2 at the boost pump inlets.

For servicing, the tank is vented to ambient at a slight positive pressure, and established chilldown and fill procedures are employed. After servicing, the LH_2 is essentially at NBP (normal boiling point) equilibrium conditions 101 kPa (14.7 psi) absolute, 20.3 K (-423°F). With these fluid conditions and the selected insulation system, approximately one hour of unattended ground hold is available prior to venting.

Preflight fuel system pressurization is accomplished by starting the electrically driven submerged boost pumps, which are capable of low speed operation at a moderate level of cavitation. A small portion of boost pump flowrate is vaporized in a heat exchanger and returned to the tank at 138 kPa (20 psi) gage, permitting normal fuel flow rates.

Two APU's, with gear box and power take off (PTO) shaft, are used to drive the engine-mounted hydraulic pumps and electrical generators during the ramjet mode. The APU's are located between the outboard and inboard engines on either side.

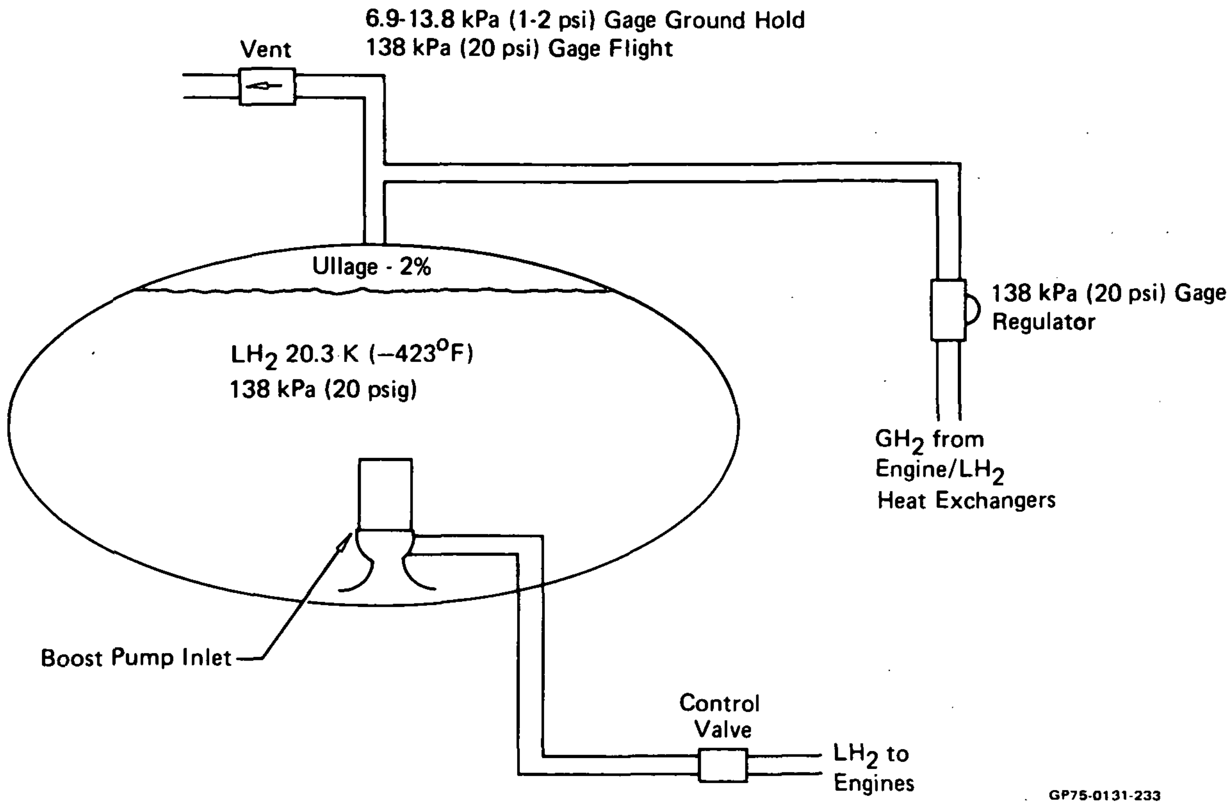


FIGURE 34
FUEL/PRESSURIZATION SCHEMATIC

b. Electrical System - KVA requirements for the electrical power generation system were based on scaling a DC-10 aircraft system. Four engine-mounted generators (one per engine) provide a total of 420 KVA. An Auxiliary Power Unit provides energy to drive generators during ramjet mode.

c. Avionics System - Space for avionics equipment is provided in the crew station compartment at the front end of the aircraft and in the equipment compartment located aft of the NLG compartment in the forward fuselage.

d. Controls Systems - Lateral and longitudinal control is provided by elevons on the wing trailing edges. Directional control is accomplished with a split rudder which also acts as a speed brake during high Mach No. cruise.

e. Environmental Control System - The ECS provides a suitable temperature, pressure and humidity environment for the crew, passengers, and equipment throughout all modes of flight. To minimize the length of the ECS duct routings, the ECS heat exchanger is located below the cabin area and aft of the NLG compartment.

f. Hydraulic System - The Concept 1 hydraulic system requirements were based on scaling a DC-10 aircraft system. A 27.6 MPa (4000 psi) absolute system was selected. The required flow rate was established to be 0.01 m^3/sec (163 gpm) with an input power of 283.5 kW (380 HP). Eight engine-mounted hydraulic pumps (two per engine) supply power to drive the elevons, rudders, landing gear, air inlet ramps, fuselage nose droop, radar, etc. The hydraulic system heat exchanger, reservoir, and accumulator are located in the nacelle/center fuselage area. The hydraulic lines are mostly routed through the fuselage crawl area outboard of the tanks. An Auxiliary Power Unit provides energy to drive hydraulic pumps during ramjet mode.

6.2 CONCEPT 2 (INTEGRAL TANK)

The Concept 2 general arrangement is presented in Figure 35. The external configuration and aircraft subsystems are almost identical to Concept 1. The major difference is that Concept 2 features an integral fuel tank.

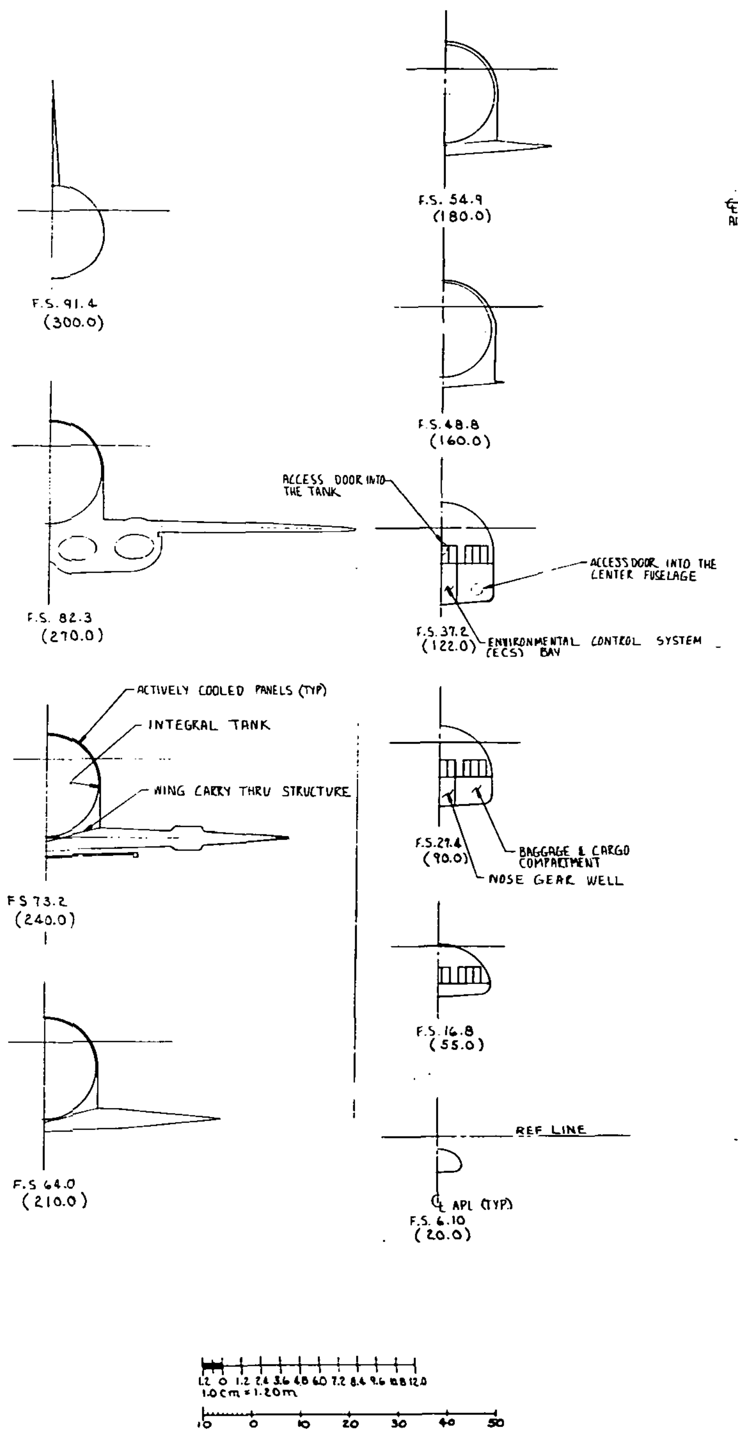
The fuel tank comprises the center fuselage section and is the backbone of the aircraft structure. It distributes the aircraft primary loads through truss links attaching it to the forward and aft fuselage, and inter-supports the wing, nacelle module, and the vertical tail.

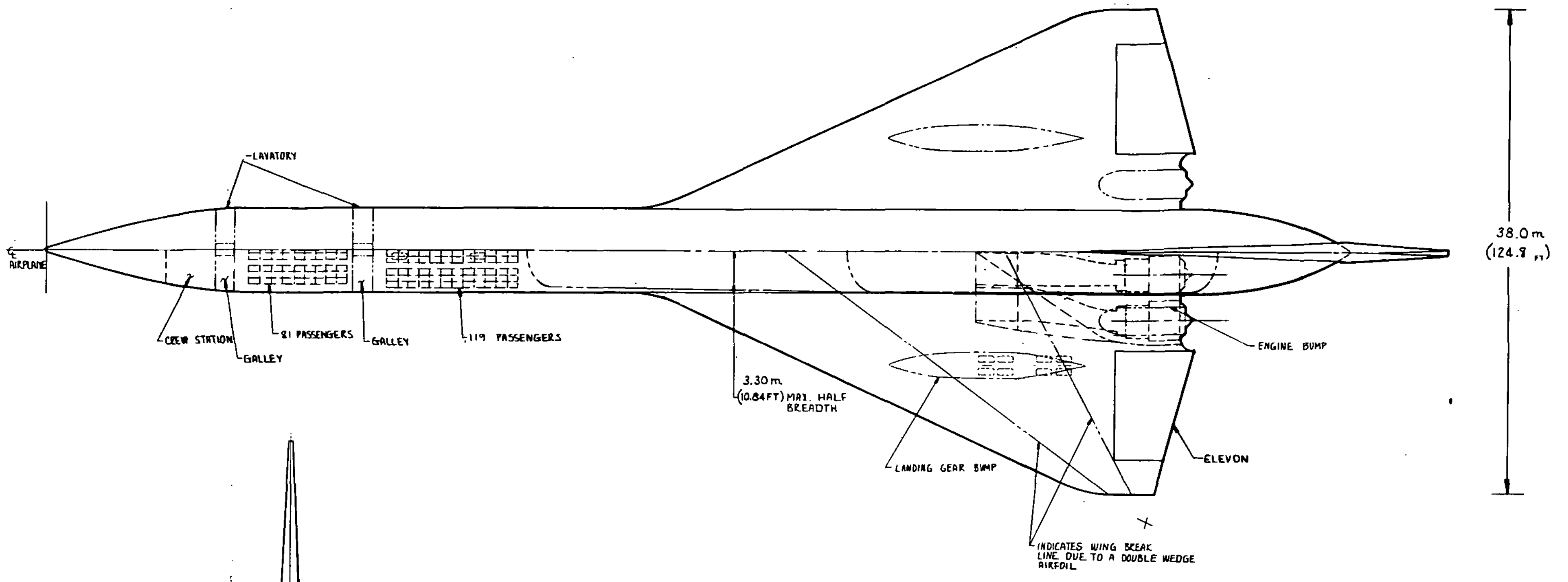
The thermal protection of the tank consists of external actively cooled panels and tank insulation. For the forward fuselage, aft fuselage and nacelle, the thermal protection is the same as for Concept 1.

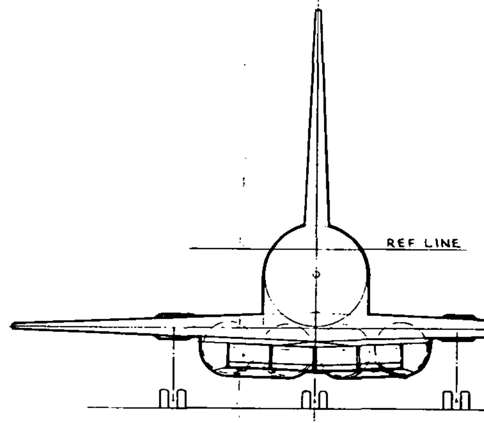
Major slip joints are located at the forward and aft ends of the tank to allow for contraction and expansion of the tank.

6.2.1 Structural Arrangement - The assembly breakdown, shown in Figure 36, is similar to Concept 1 except for the center fuselage. Concept 2 has a single tank/fuselage section with truss links forward and aft. The structure of the forward fuselage, empennage, and wing is identical to Concept 1 except at the splice interface with the center fuselage section.

- Center Fuselage/Tank Structure - The single integral fuel tank is the main load carrying member of the center fuselage. The large difference in temperature between the tank and interfacing structure was an important design consideration. The tank structure has external insulation and the wall is maintained at the same low temperature as the liquid hydrogen. The remaining structure, composed of actively cooled panels, is kept at an average temperature of 366 K (200°F).







General Notes:

1. Airplane is shown with Empty Cryogenic Tanks in the Static Ground Position (All Structure at 294 K (70°F))

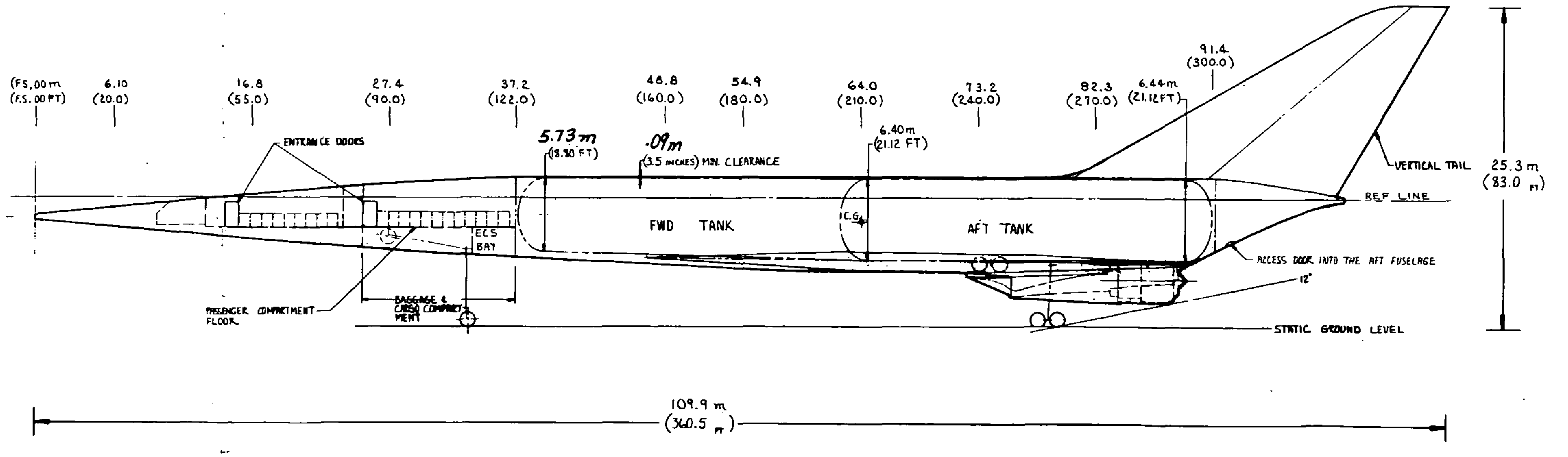


FIGURE 35
CONFIGURATION, CONCEPT 2

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Volume Summary

Forward Fuselage FS 0.00-37.2m (FS 0.00-122.00 ft)	670 m^3 (23,700 ft^3)
Center Fuselage FS 37.2-91.9m (FS 122.0-301.50 ft)	2,270 m^3 (80,300 ft^3)
Aft Fuselage FS 91.9-101.8m (FS 301.50-334.00 ft)	110 m^3 (3,800 ft^3)
Total Fuselage	3,050 m^3 (107,800 ft^3)
Tank Volume	1,620 m^3 (57,200 ft^3)
Tank Volume/Center Fuselage Volume *	81%
Tank Volume/Total Fuselage Volume	53%

*Minus 283 m^3 (10,000 ft^3) wing carry-thru structure volume

Physical Characteristics

Item	Wing		Vertical Tail	
S_{theo}	1,070 m^2	(11,530 ft^2)	180 m^2	(1,970 ft^2)
Æ	1.35	–	2.00	–
λ	0.10	–	0.27	–
b	38.0m	(124.8 ft)	–	–
b/2	19.0m	(62.4 ft)	13.5m	(44.4 ft)
C_R	51.2m	(168.0 ft)	21.4m	(70.2 ft)
C_T	5.09m	(16.7 ft)	5.6m	(18.5 ft)
MAC	34.2m	(112.3 ft)	15.1m	(49.5 ft)
ΛLE (deg)	65	–	60	–
ΛTE (deg)	–15	–	30	–
Incidence (deg)	+1/2	–	–	–
Dihedral	0	–	–	–
Thickness Ratio	0.03	–	0.03	–

Performance Summary

Range	8.73 Mm (4,715 NM)
Payload (200 Passengers)	21.8 Mg (48,000 lbm)
Operating Weight Empty	190.6 Mg (420,252 lbm)
Takeoff Gross Weight	299.5 Mg (660,252 lbm)

Propulsion

(4) GE5/JZ6-C 400 kN (90,000 lbf) T_{SLS} per Engine Uninstalled

Total Inlet Capture Area ($A_{C_{total}}$) = 15.8 m^2 (170 ft^2)

	Tire Size
Main Gear	1.27m x 0.51m (50 in.x20 in.)
Nose Gear	1.27m x 0.51m (50 in.x20 in.)

Wetted Area

Fuselage	1,820 m^2 (19,600 ft^2)
Nacelle	340 m^2 (3,640 ft^2)
Wing	1,440 m^2 (15,512 ft^2)
Vertical Tail	390 m^2 (4,150 ft^2)
Total	3,990 m^2 (42,902 ft^2)

Fuel Distribution

Tank Section	Type	*Usable Volume	Fuel Weight
Forward	Integral	680 m^3 (24,000 ft^3)	48.1 Mg (106,000 lb)
Aft	Integral	860 m^3 (30,300 ft^3)	60.8 Mg (134,000 lb)
Total		1,540 m^3 (54,300 ft^3)	108.9 Mg (240,000 lb)

*5% of tank volume allowed for ullage, rings, etc. ∴ usable volume = 0.95 tank volume
Fuel: Liquid hydrogen at 20.3 K (–423°F) ρ (density) = 70.8 kg/m^3 (4.42 lbm/ft^3)

Fineness Ratio	14.00	–
Total Aircraft Volume	4,150 m^3	(146,500 ft^3)
Planform Area	1,360 m^2	(14,596 ft^2)
Max Cross Sectional Area	97.6 m^2	(1,050 ft^2)
Less Capture Area	15.8 m^2	(170 ft^2)
Net Cross Sectional Area	81.8 m^2	(880 ft^2)
Mach No. (Cruise)	6	–
$V^{2/3} \div S_p$	0.177	–

GP75-0131-58

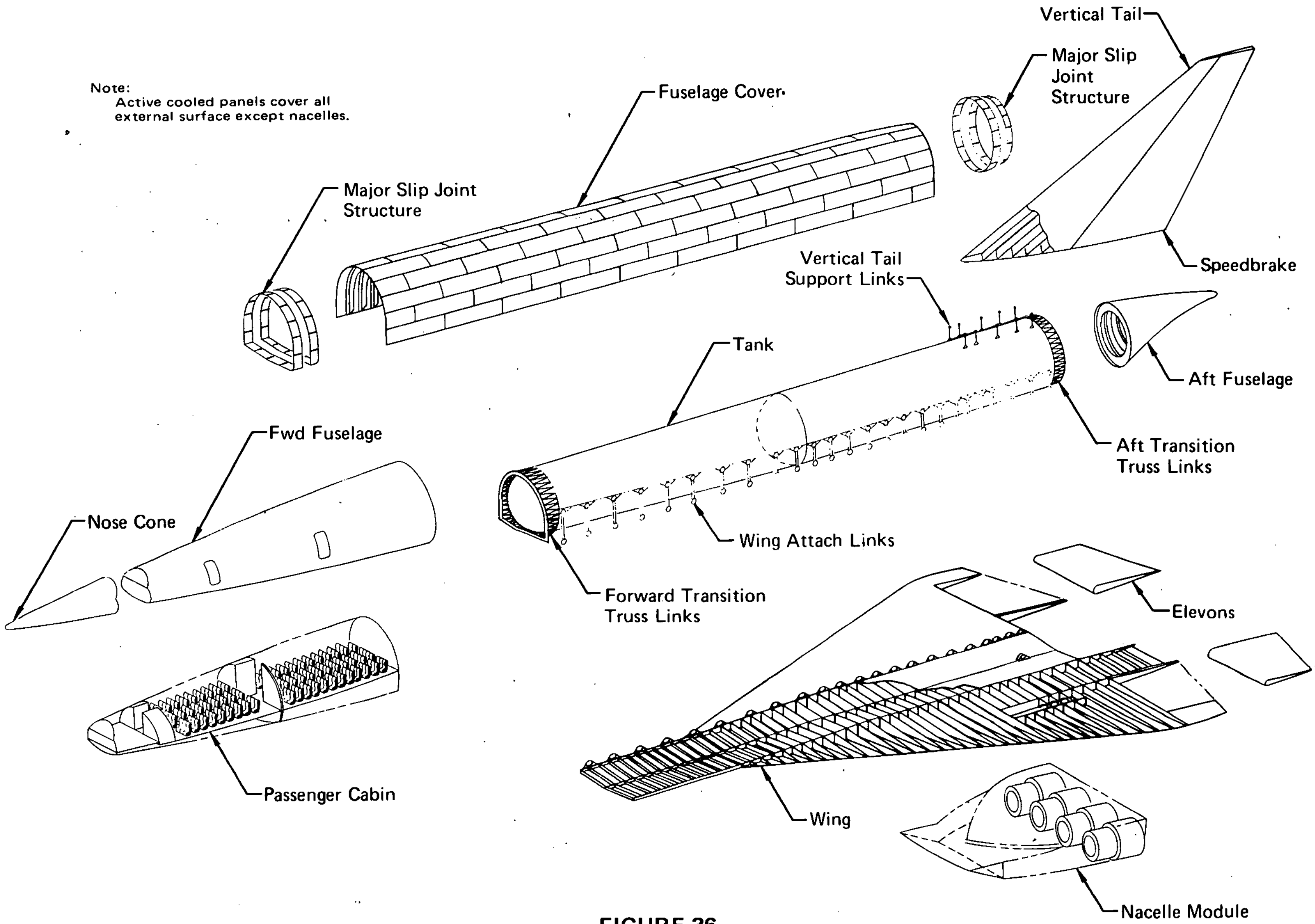


FIGURE 36
CONCEPT 2 ASSEMBLY BREAKDOWN

This large temperature differential (and the associated thermal stresses) is accommodated by a system of interconnected links which allow for the thermal movement while maintaining primary structural load paths.

The wing is supported by the tank with a series of links which have monoball bearings at each end to allow the links to move with respect to each other. The shear view shows the longitudinal distribution of the vertical links with Section A-A and View B in Figure 37 defining details on these and the transverse links. The transverse links prevent sideward motion with respect to the wing. A single longitudinal link, attached to the wing at an aft location on the centerline of the carry-through, prevents fore and aft movement.

Details of the forward to center, aft to center and vertical tail to center splice trusses are shown on view M-M and D-D of Figure 37. These truss networks are formed of links and are used to relieve the thermal strain at the splice joints.

Major slip joints are used to allow thermally induced relative motion between the tank and fuselage cover while prohibiting introduction of airplane loads as shown in the Section view, U-U.

The aluminum tank is made from welded isogrid panels. The tank structure details are shown in Figure 38. View K-K is an example of the integral isogrid tank structure pattern. This method of stiffening resulted in the lightest structural arrangement, as discussed in Reference (2).

The tank is divided into two sections by a centrally located internal dome bulkhead. The tank ends are elliptically domed the same as on Concept 1. The divider helps control aircraft center of gravity and keeps crash pressure heads below the burst pressure of the tank. Internal frames are located at each wing support link to redistribute wing loads. These frames are welded between adjacent isogrid panels as depicted in Section B-B of Figure 38.

The tank mold line cover consists of actively cooled panels similar to Concept 1. These panels are attached to each other and the wing in the same manner as Concept 1 but the forward and aft ends of the cover are discontinuous and, through a slip joint, allow relative motion at the fuselage splices. These panels are semi-structural in that loads are induced into the

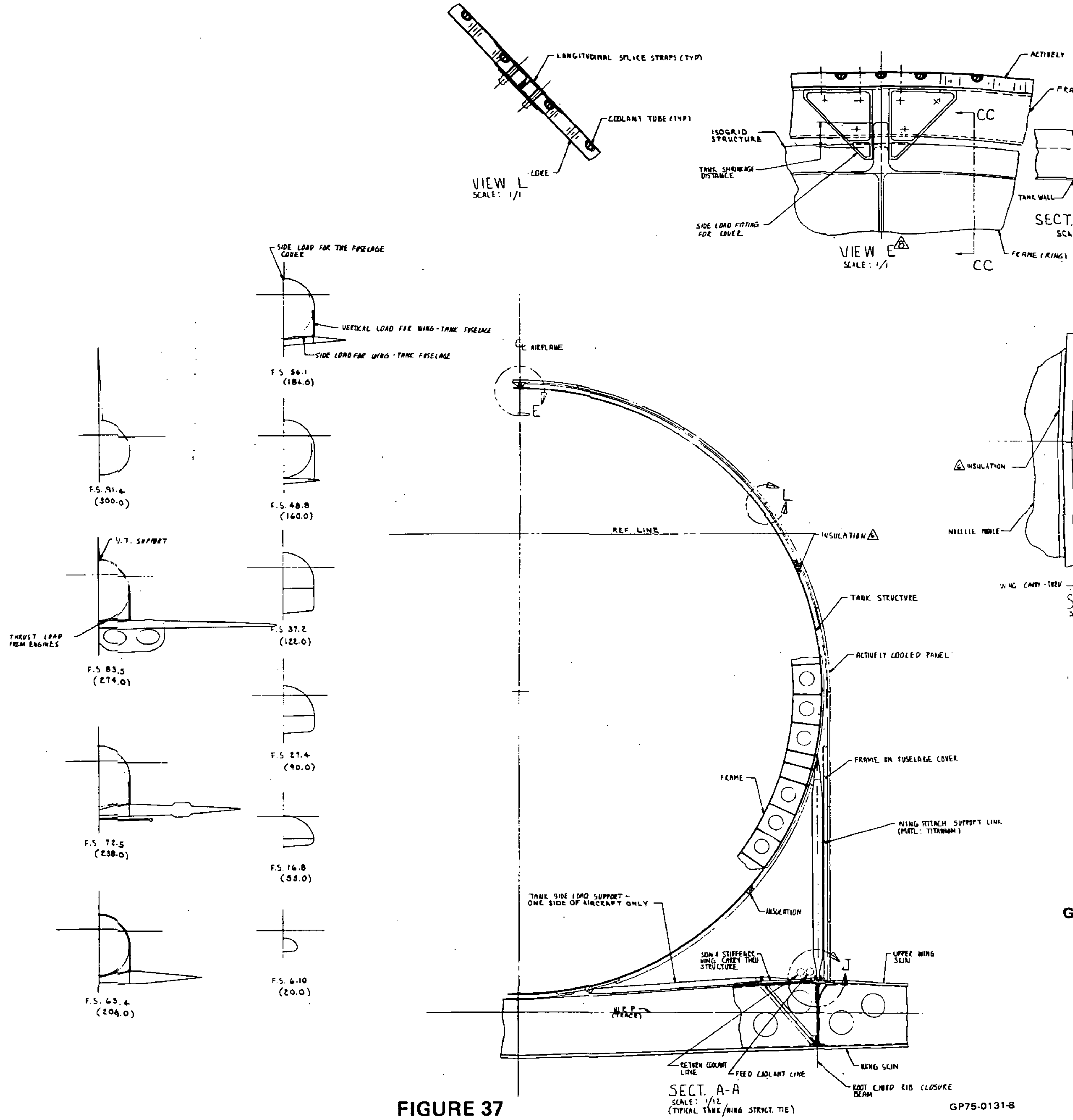
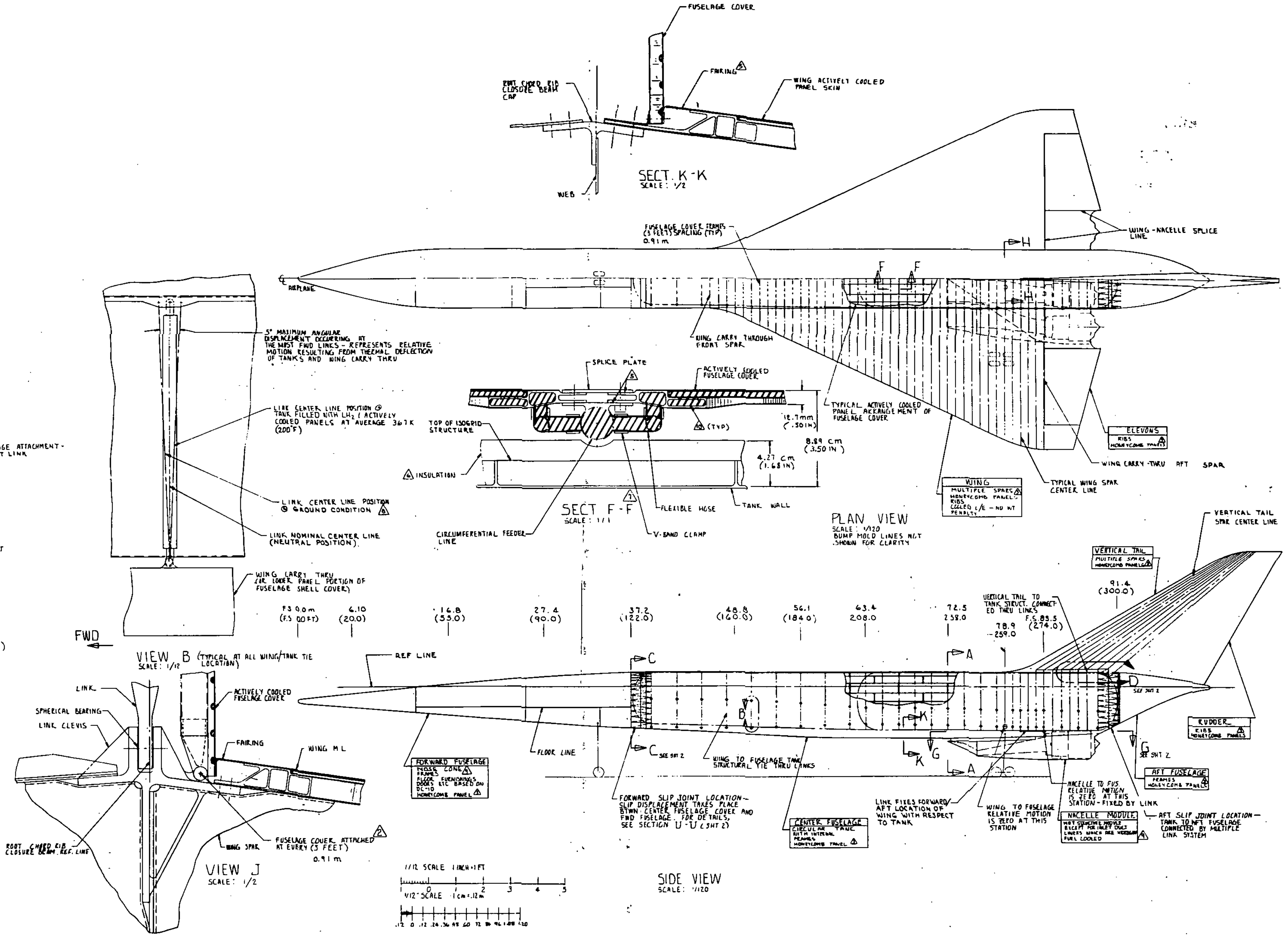


FIGURE 37
STRUCTURAL ASSEMBLY, CONCEPT 2



General Notes:

1. Operating weight empty estimation of forward fuselage FS 0.00 - 37.2 m (FS 0.00 - 122.0 ft), aft fuselage FS 91.9 - 101.8 m (FS 301.5 - 334.0 ft), wing, elevons, vertical tail, and nacelle module structure are analyzed on the same basis as the Concept 1 airplane
2. Actively cooled panels acting as a heat shield cover over the fuselage/tank structure is removable in 5.49 m (18 ft) quarter shell segments
3. Honeycomb panel structure is actively cooled
4. Primary structural material used is Al 2219-T87
5. High thermal conductive adhesive applied between surfaces
6. Insulation material, 64.1 kg/m^3 (4 lbm/ft^3) closed cell fiberglass reinforced polyurethane with an aluminized mylar covering.
7. Spherical bearings on all link joints
8. Airplane is shown in the state of the tank empty and the airplane position (all structure at 294 K (70^{o}F))
9. Special consideration was given to the panels in this area due to the higher heat load, which makes these panels weigh 2.0 times the average panel weight in the center fuselage
10. ⊘ Indicates methanol/water coolant
11. Insulation material, 56.0 kg/m^3 (3.5 lbm/ft^3) high temperature glass fiber

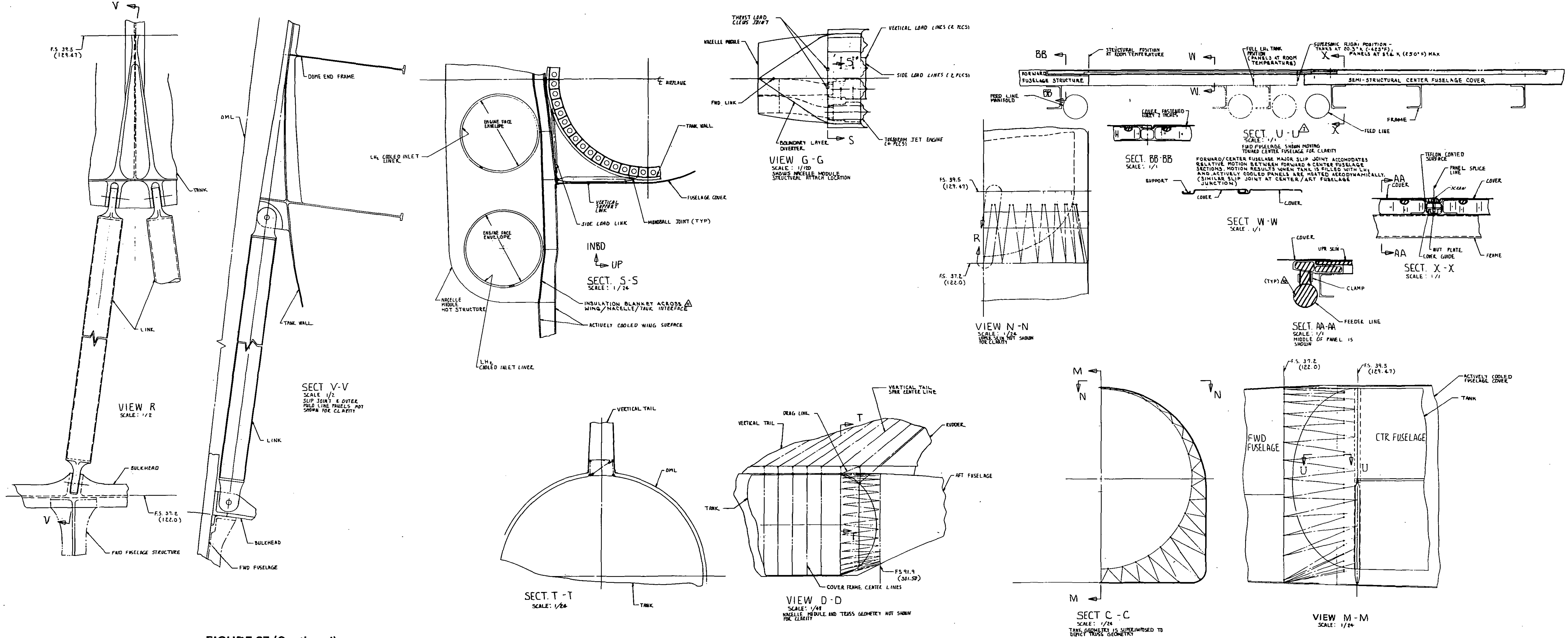


FIGURE 37 (Continued)
STRUCTURAL ASSEMBLY, CONCEPT 2

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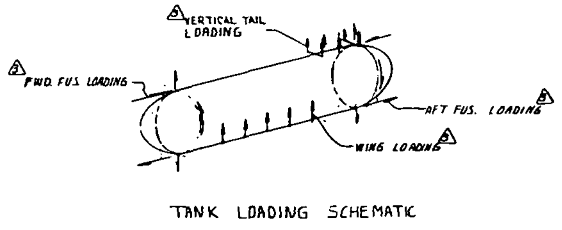
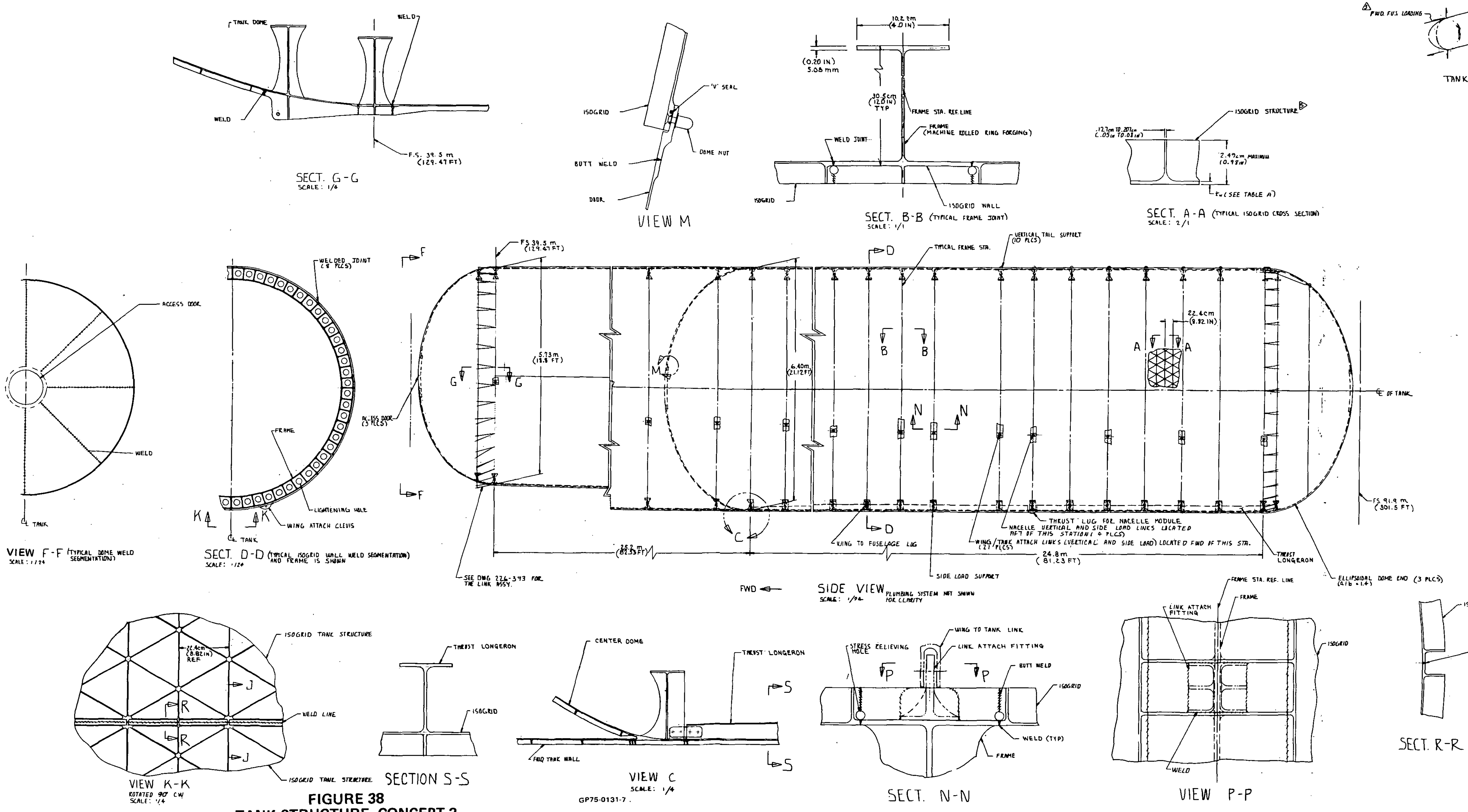


Table A
Fuel Tank Thicknesses

Fuselage Stations		t_w Web Thickness		$\bar{t}$ Average Equivalent Weight Thickness	
m	ft	cm	in.	cm	in.
39.6 - 48.8	130 - 160	0.152	0.060	0.254	0.100
48.8 - 56.4	160 - 185	0.165	0.065	0.229	0.090
56.4 - 62.5	185 - 205	0.178	0.070	0.229	0.090
62.5 - 68.6	205 - 225	0.178	0.070	0.279	0.110
68.6 - 73.2	225 - 240	0.165	0.070	0.216	0.085
73.2 - 76.0	240 - 246	0.165	0.065	0.206	0.081
75.0 - 79.2	246 - 260	0.165	0.065	0.251	0.099
79.2 - 83.8	260 - 275	0.152	0.060	0.262	0.103
83.8 - 89.6	275 - 294	0.152	0.060	0.234	0.092

General Notes:

1. Tank Material: Aluminum 2219-T87
2. Tank is assembled primarily by automatic tungsten inert gas welding
3. Representative reaction points are shown for clarity
4. Provisions for fueling are not shown for clarity
5. Isogrid tank structure is machined and roll formed
6. Frames are roll formed and machined
7. No heat treatment required after welding



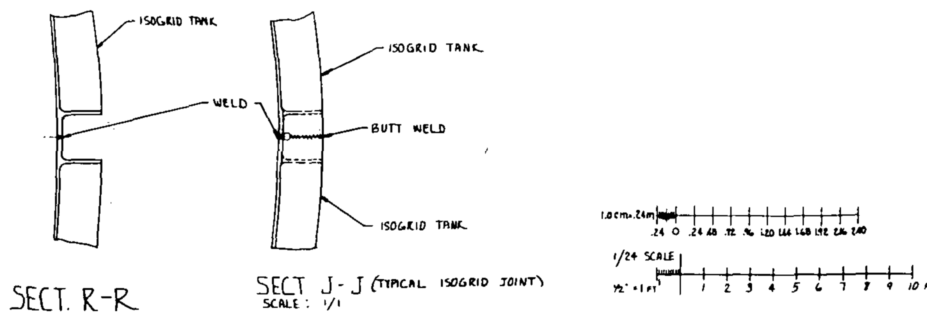


FIGURE 38
TANK STRUCTURE, CONCEPT 2

wing cover combination by fuselage deflection, but no shear, bending moment or axial load is introduced at either end.

6.2.2 Thermal Protection - The active cooling system for this concept is similar to Concept 1, as discussed in Section 6.1.2. The fuselage cover is composed of 1.2 m x 6.1 m (4 ft x 20 ft) actively cooled panels (average size). The panel manifold distribution point is located at the ends of each panel. The coolant is introduced (or returned) into the panels through flexible connections. Section B-B on Figure 39 shows the joint design. Since the panel assembly is a semi-structural member, the joint is designed to minimize purge gas leakage, to have a water-tight surface, and have an adequately inspectable joint.

Individual insulation packages are used to fill each isogrid cavity of the tank wall. A solid layer of insulation is also used on top of the isogrid surface.

6.3 CONCEPT 3 (INTEGRAL TANK)

The general arrangement of Concept 3 is presented in Figure 40. Unlike the modified circular sections of Concepts 1 and 2, this concept is an elliptical wing-body configuration. Like Concept 2, however, it features an integral fuel tank. The tank cross section consists of multiple bubbles, so as to achieve maximum volumetric efficiency within the elliptical fuselage cross section. Because this concept has the best volumetric efficiency it is physically smaller than Concepts 1 and 2 for the same total fuel weight of 108.9 Mg (240,000 lbm).

The structural arrangement of the forward and aft fuselage and nacelle is similar to that in Concept 2, except for the cross sectional shape. The center section is considerably different. The integral tank is the primary load-carrying member, as in Concept 2, but in addition it acts as the wing carry-through structure. The relation of the major components is shown in the assembly breakdown on Figure 41.

The aircraft controls, subsystems, structural materials and propulsion system are similar to Concepts 1 and 2.

6.3.1 Structural Arrangement - The multi-bubble tank is the backbone of the center fuselage structure and the wing carry-through. The tank relationship to moldline and typical wing connection is depicted in Section B-B of the

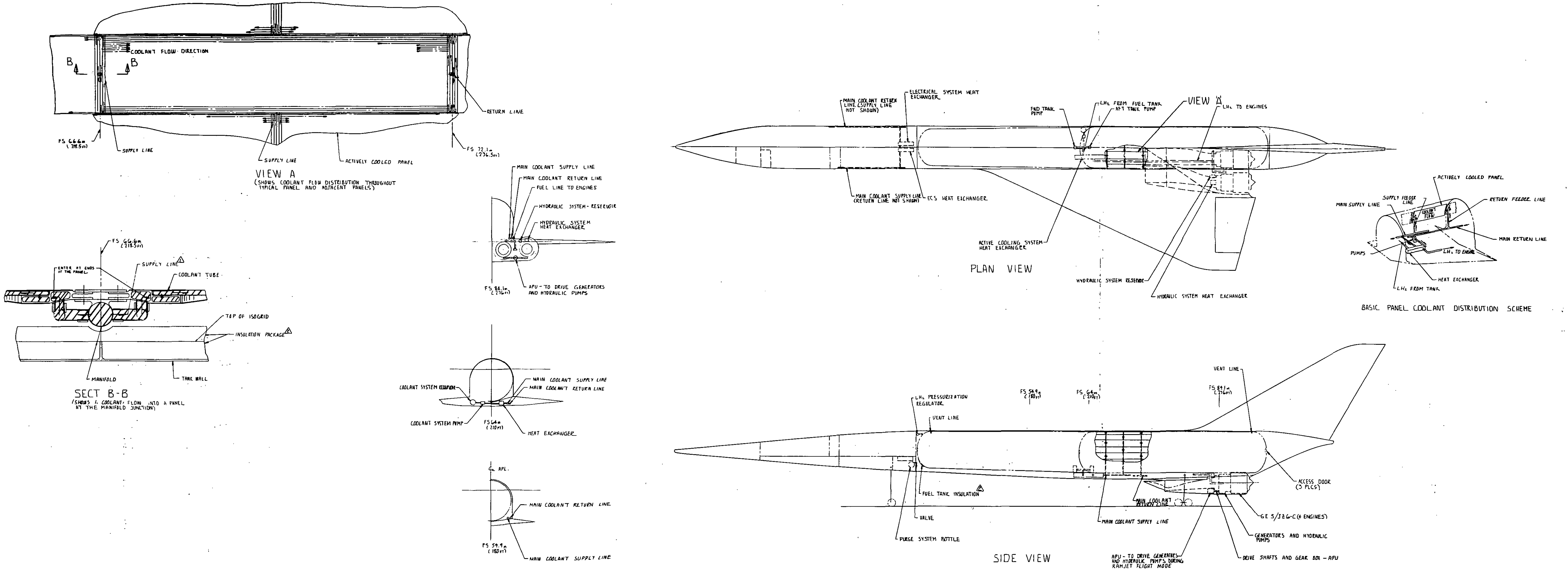


FIGURE 39
ACTIVE COOLING SYSTEM, CONCEPT 2

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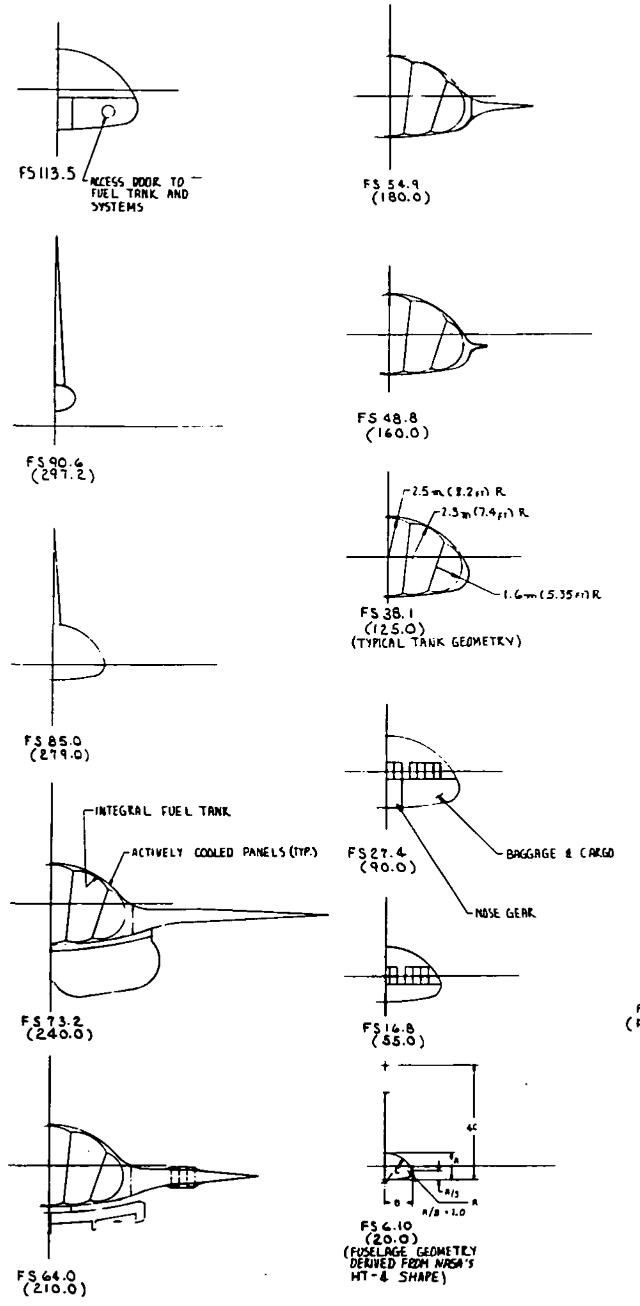
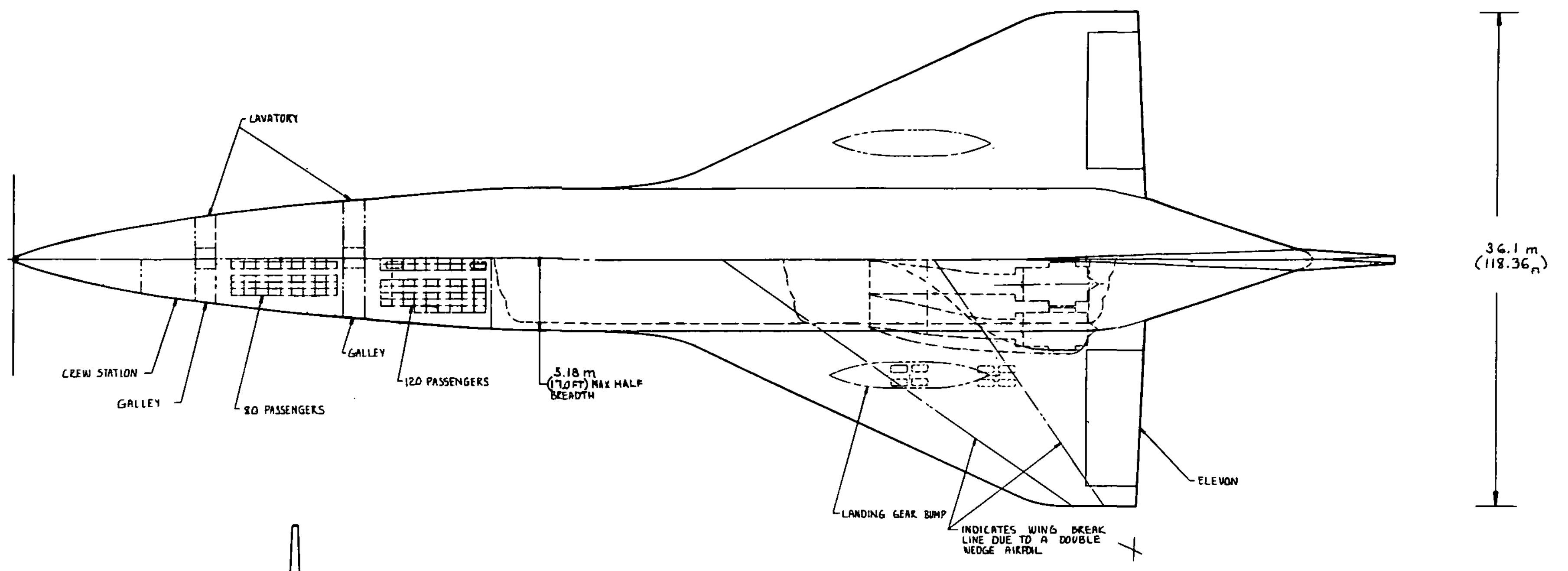
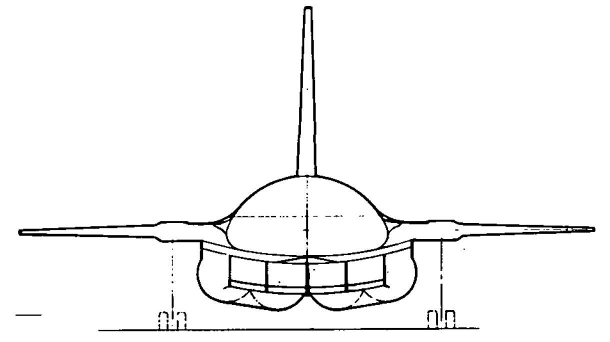
General Notes:

1. Indicates methanol/water coolant
2. Insulation material, 64.1 kg/m^3 (4 lbm/ft^3) closed cell fiberglass reinforced polyurethane with an aluminized mylar covering







General Notes:

1. Airplane is shown with Empty Cryogenic Tanks in the Static Ground Position (All Structure at 294 K (70°F))

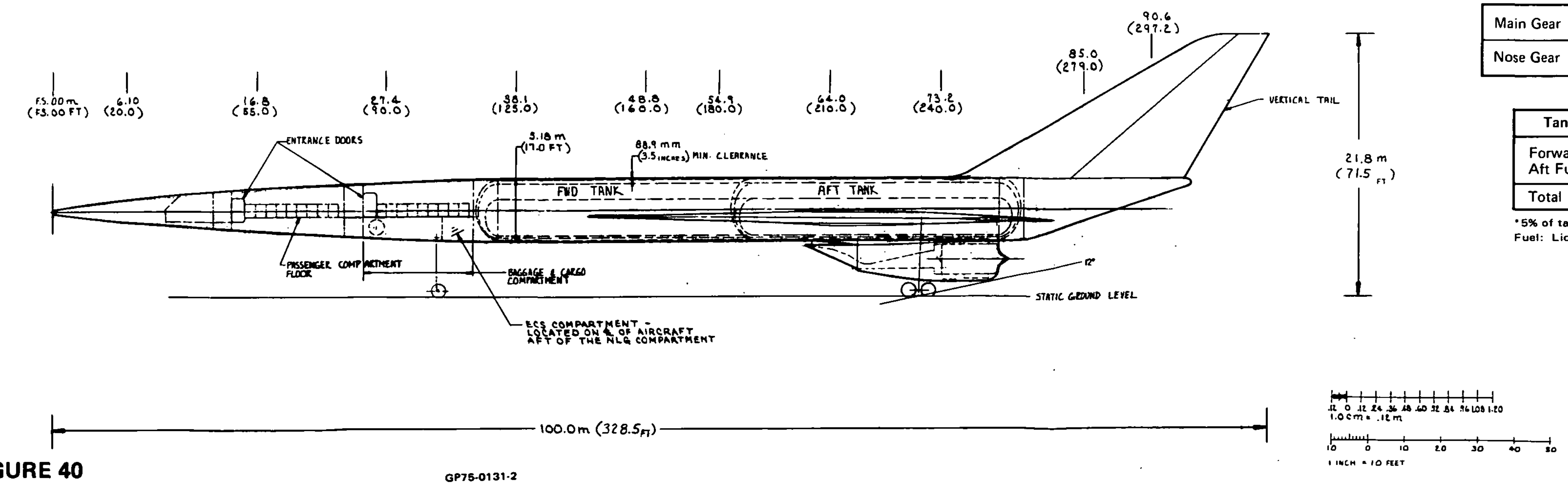


FIGURE 40
CONFIGURATION, CONCEPT 3

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Forward Fuselage	FS 0.00–34m	(FS 0.00–113.5 ft)	620 m^3	(22,000 ft^3)
Center Fuselage	FS 34.6–80.0m	(FS 113.5–262.52 ft)	1,840 m^3	(65,100 ft^3)
Aft Fuselage	FS 80.0–93.9m	(FS 262.52–308 ft)	110 m^3	(3,800 ft^3)
Total Fuselage			2,570 m^3	(90,900 ft^3)
Tank Volume			1,620 m^3	(57,200 ft^3)
Tank Volume/Center Fuselage Volume			87.9%	
Tank Volume/Total Fuselage Volume			63%	

Physical Characteristics

Item	Wing		Vertical Tail	
S_{theo}	960 m^2	(10,377 ft^2)	140 m^2	(1,535 ft^2)
AR	1.35	–	2.00	–
λ	0.15	–	0.27	–
b	36.1 m	(118.36 ft)	11.9 m	(39.18 ft)
b/2	18.0 m	(59.18 ft)	–	–
C_R	46.5 m	(152.47 ft)	18.6 m	(61.01 ft)
C_T	7.0 m	(22.87 ft)	5.1 m	(16.66 ft)
MAC	31.6 m	(103.64 ft)	13.3 m	(43.49 ft)
Λ LE (deg)	65	–	60	–
Λ TE (deg)	–3	–	30	–
Incidence (deg)	+1/2	–	–	–
Dihedral	0	–	–	–
Thickness Ratio	0.03	–	0.03	–

Performance Summary

Range	9.20 Mm (4,968 NM)
Payload (200 Passengers)	21.8 Mg (48,000 lbm)
Operating Weight Empty	187.3 Mg(412,816 lbm)
Takeoff Gross Weight	296.1 Mg(652,816 lbm)

Propulsion

(4) GE5/JZ6-C 400 kN (90,000 lbf) T_{SLS} per Engine Uninstalled
Total Inlet Capture Area ($A_{C_{total}}$) = 15.8 m^2 (170 ft^2)

	Tire Size
Main Gear	1.27 m x 0.51 m (50 in. x 20 in.)
Nose Gear	1.27 m x 0.51 m (50 in. x 20 in.)

Wetted Area

Fuselage	1,630 m^2 (17,600 ft^2)
Nacelle	380 m^2 (4,080 ft^2)
Wing	1,070 m^2 (11,464 ft^2)
Vertical Tail	280 m^2 (3,070 ft^2)
Total	3,360 m^2 (36,214 ft^2)

Fuel Distribution

Tank Section	Type	Usable Volume*	Fuel Weight
Forward Fuselage	Integral	760 m^3 (26,778 ft^3)	53.7 Mg (118,400 lb)
Aft Fuselage	Integral	780 m^3 (27,522 ft^3)	55.2 Mg (121,600 lb)
Total		1,540 m^3 (54,300 ft^3)	108.9 Mg (240,000 lb)

*5% of tank volume allowed for ullage, rings, etc. ∴ Usable volume = 0.95 tank volume
Fuel: Liquid hydrogen @ 20.3 K (–423°F) ρ (density) = 70.8 Kg/m^3 (4.42 lbm/ft^3)

Fineness Ratio	13.10	–
Total Aircraft Volume	3,500 m^3	(123,800 ft^3)
Planform Area	1,280 m^2	(13,756 ft^2)
Max Cross Section Area	98.5 m^2	(1,060 ft^2)
Less Capture Area	15.8 m^2	(170 ft^2)
Net Cross Sectional Area	82.7 m^2	(890 ft^2)
Mach No. (Cruise)	6	–
$V^{2/3} \div S_p$ Factor	0.156	–

GP75-0131-124

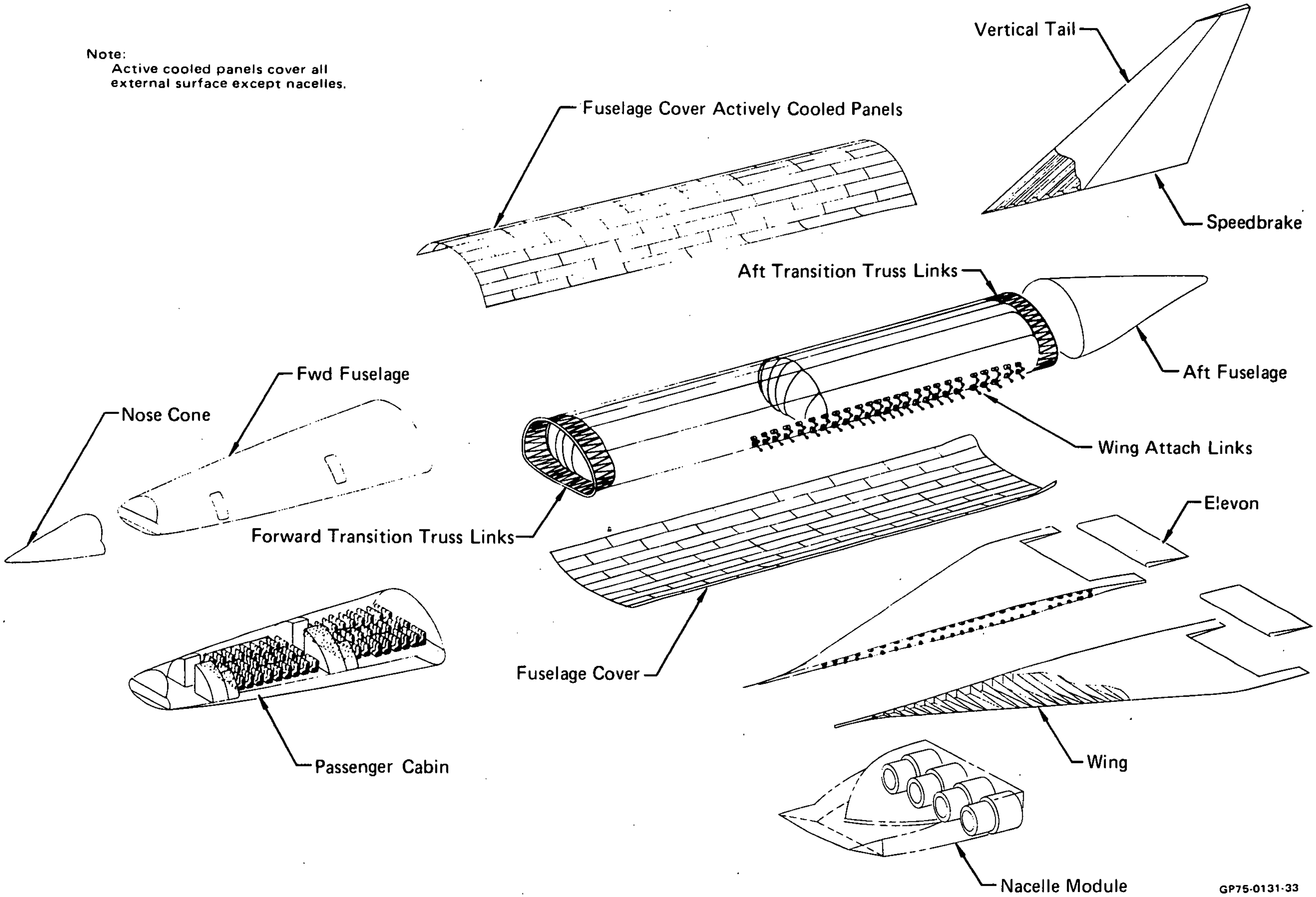


FIGURE 41
CONCEPT 3 ASSEMBLY BREAKDOWN

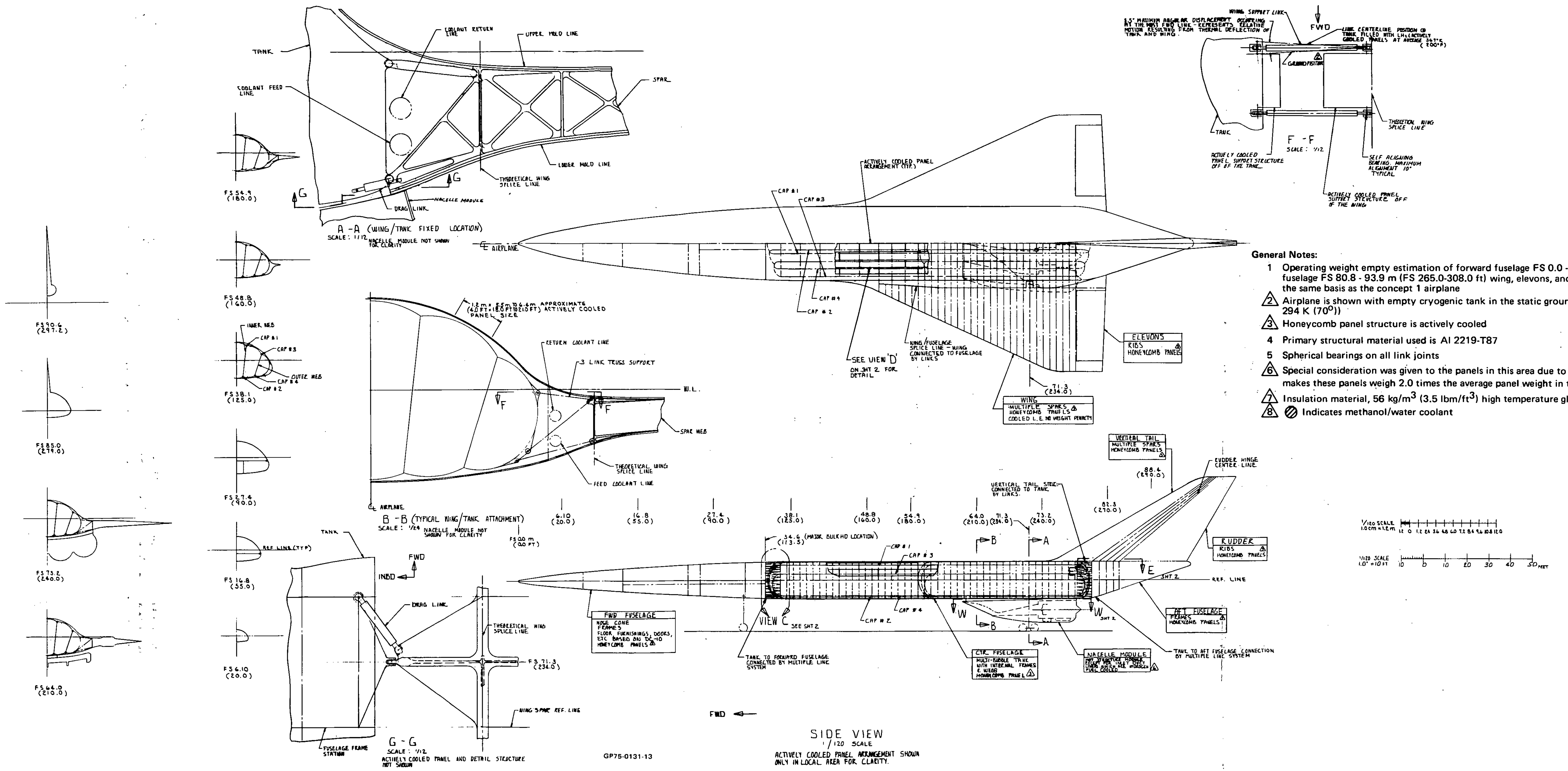
structural arrangement drawing presented in Figure 42. The tank cross section consists of segments of five intersecting circles, with longitudinal shear webs attached at the intersections. Tank walls are stiffened in an isogrid pattern and the fuel tank divider and end domes are elliptically shaped, similar to Concept 2.

The wing is located roughly at the centroid of the fuselage cross section. It has a multi-spar construction, but the spars are not continuous across the fuselage. Each wing is supported off the side of the tank by a tri-link truss system with the links spaced 0.91 m to 2.74 m (3 ft to 9 ft) apart. The wing load is distributed across the tank through internal tank frames. These wing support links accommodate thermal deflection in a similar manner to the wing attach links of Concept 2, as shown in View F-F of Figure 42. The wing is rigidly attached to the tank structure at one aftmost point by a drag link (Section G-G, Figure 42). The wing and tank are free to move relative to each other fore and aft of this point. Details of the isogrid pattern and frames, and the welding method for the tank, are presented in Figure 43.

The tank cover consists of non-structural actively cooled panels. The panel reacts only airloads and is supported from the coolant feeder lines attached to the tank, as shown in Section R-R, Figure 42. Slip joints are provided around the perimeter of each panel. Section K-K of Figure 42 shows how the support links of the center fuselage panels compensate for the irregular shaped tank mold line to achieve a smooth external shape.

Forward fuselage and aft fuselage load-carrying splice joints incorporate link trusses similar to Concept 2. The nacelle is attached to the tank directly through a series of links, typically shown in Section X-X of Figure 42. The vertical tail is supported by the aft fuselage.

6.3.2 Thermal Protection - The cooling system schematic is shown in Figure 44. Each actively cooled panel is allowed to displace independently to compensate for the thermal growth and contraction of the tank. The major difference in the Concept 3 active cooling system is the location of the heat exchanger and system equipment. The heat exchanger equipment is located forward of the tank because no space is available in the center section of the compact bubble tank design.



General Notes:

1 Operating weight empty estimation of forward fuselage FS 0.0 - 34.6 m (FS 0.0 - 113.5 ft), aft fuselage FS 80.8 - 93.9 m (FS 265.0-308.0 ft) wing, elevons, and vertical tail are analyzed on the same basis as the concept 1 airplane

2 Airplane is shown with empty cryogenic tank in the static ground position (all structure at 294 K (70°))

3 Honeycomb panel structure is actively cooled

4 Primary structural material used is Al 2219-T87

5 Spherical bearings on all link joints

6 Special consideration was given to the panels in this area due to the higher heat load, which makes these panels weigh 2.0 times the average panel weight in the center fuselage

7 Insulation material, 56 kg/m^3 (3.5 lbm/ft^3) high temperature glass fiber

8 ▨ Indicates methanol/water coolant

FIGURE 42
STRUCTURAL ASSEMBLY, CONCEPT 3

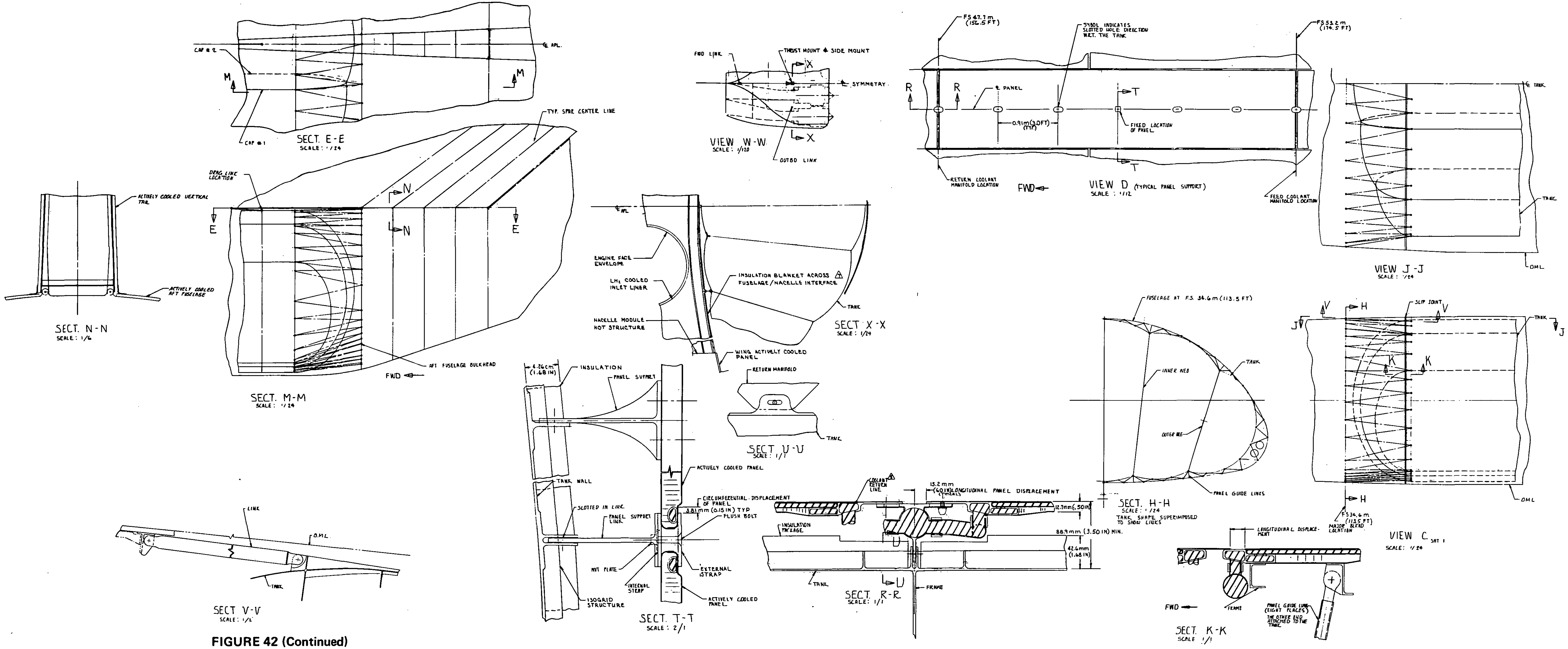
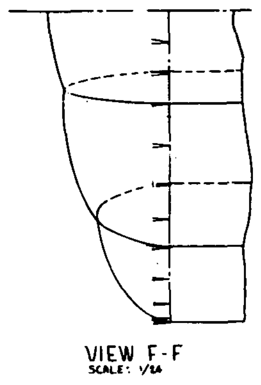


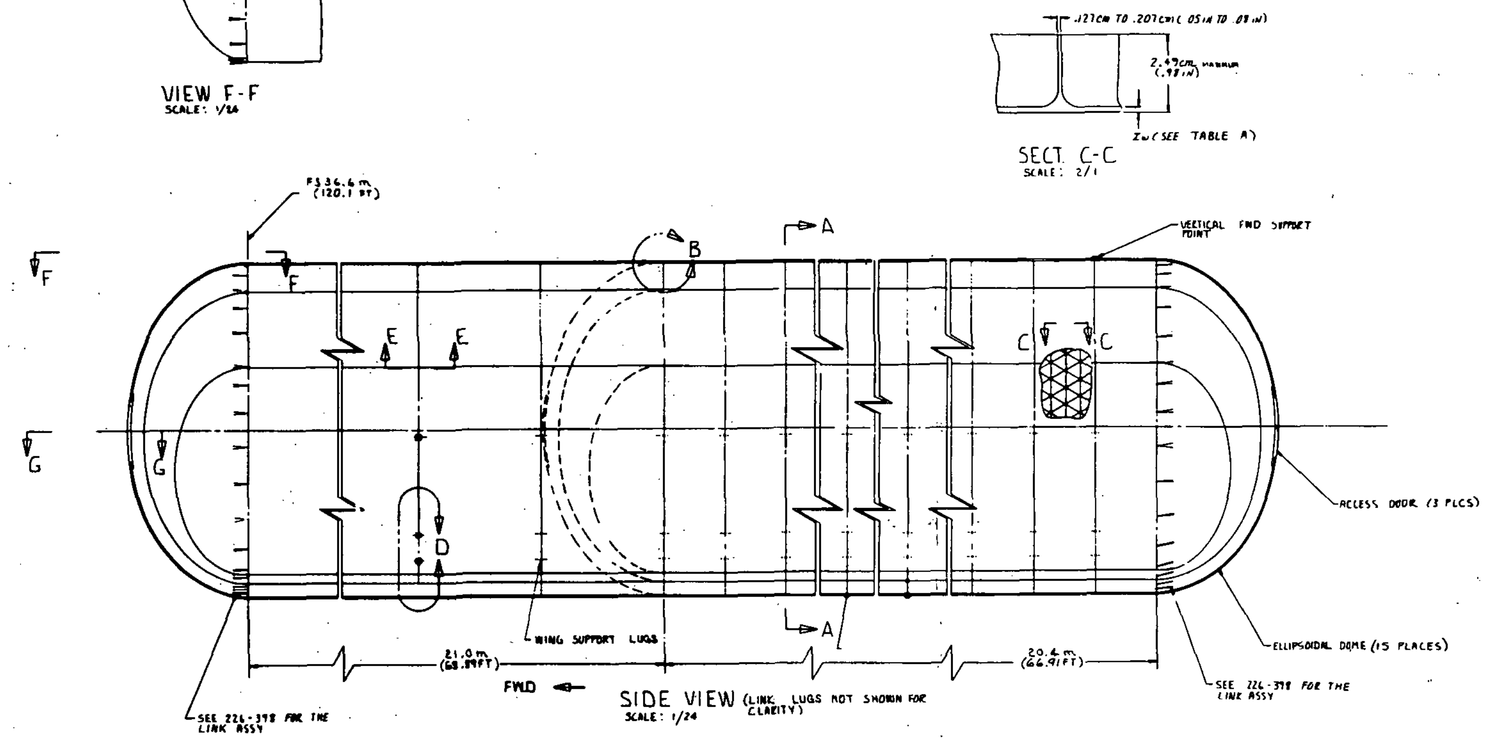
FIGURE 42 (Continued)
STRUCTURAL ASSEMBLY, CONCEPT 3

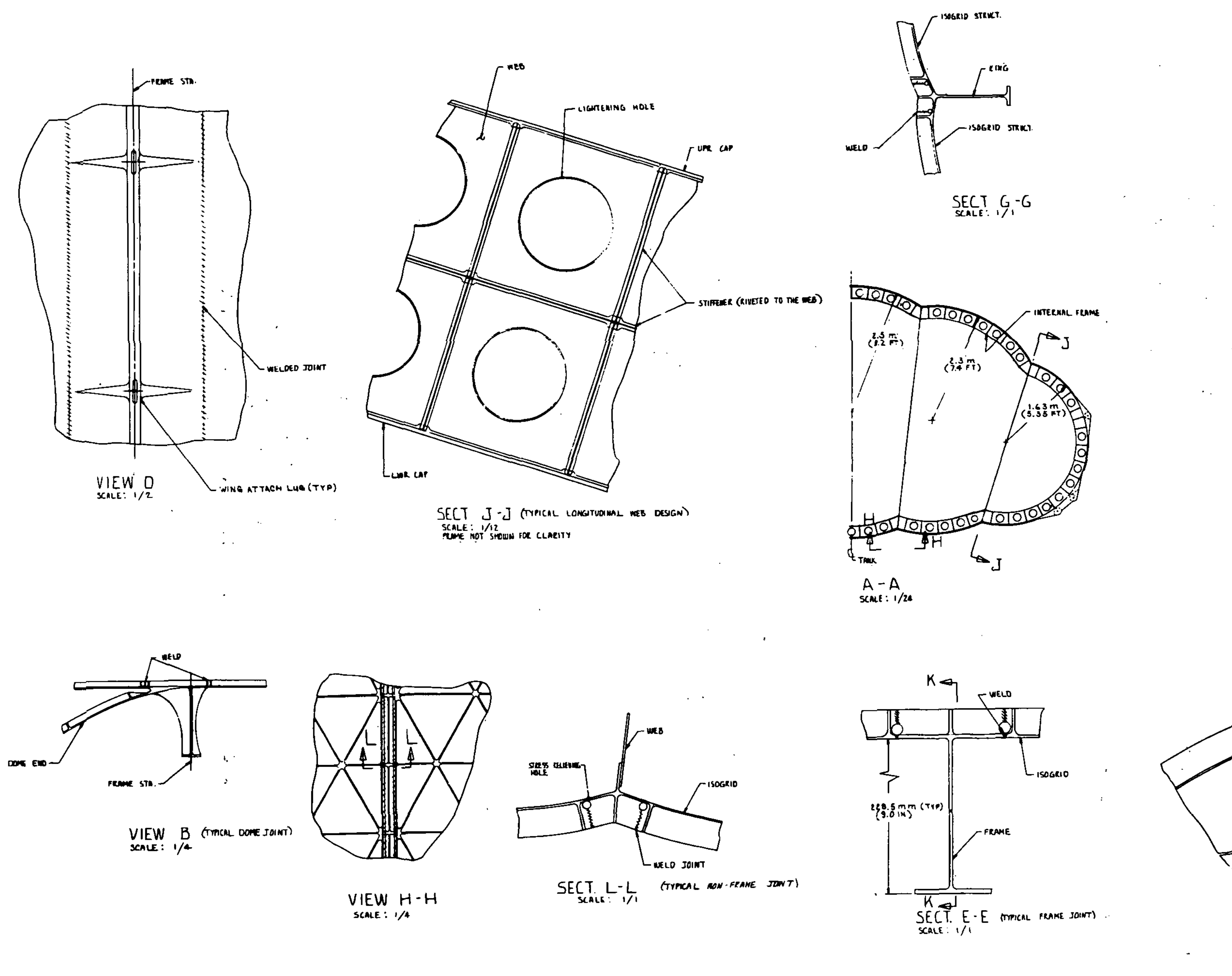
GP75-0131-11

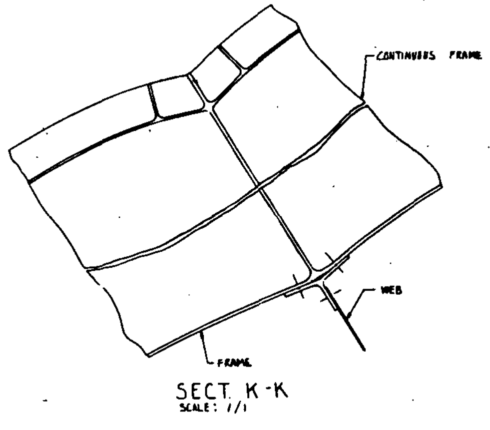
General Notes:

1. Tank material: Aluminum 2219-T87
2. Tank is assembled primarily by automatic tungsten inert gas (TIG) welding technique
3. Isogrid tank structure is machined and roll formed
4. Representative reaction points are shown
5. Frames are roll formed and machined
6. No heat treatment required after welding









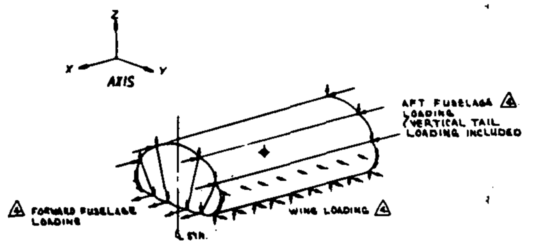


TABLE A
FUEL TANK THICKNESS

Fuselage Stations		Average Equivalent Weight Thickness - $\bar{t}$					
		℄ Bubble		Middle Bubble		Outboard Bubble	
m	ft	cm	in.	cm	in.	cm	in.
37.5 - 46.6	123 - 153	0.269	0.106	0.216	0.085	0.178	0.070
46.6 - 59.4	153 - 195	0.224	0.088	0.206	0.081	0.178	0.070
59.4 - 62.8	195 - 206	0.213	0.084	0.191	0.075	0.178	0.070
62.8 - 70.1	206 - 230	0.208	0.082	0.191	0.075	0.178	0.070
70.1 - 76.2	230 - 250	0.229	0.090	0.201	0.079	0.178	0.070
Web Thicknesses:		0.178	.070	0.160	0.063	0.147	0.058

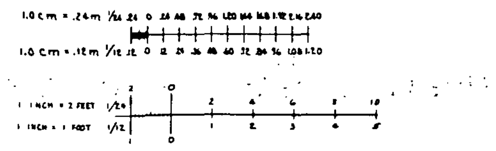


FIGURE 43
TANK STRUCTURE, CONCEPT 3

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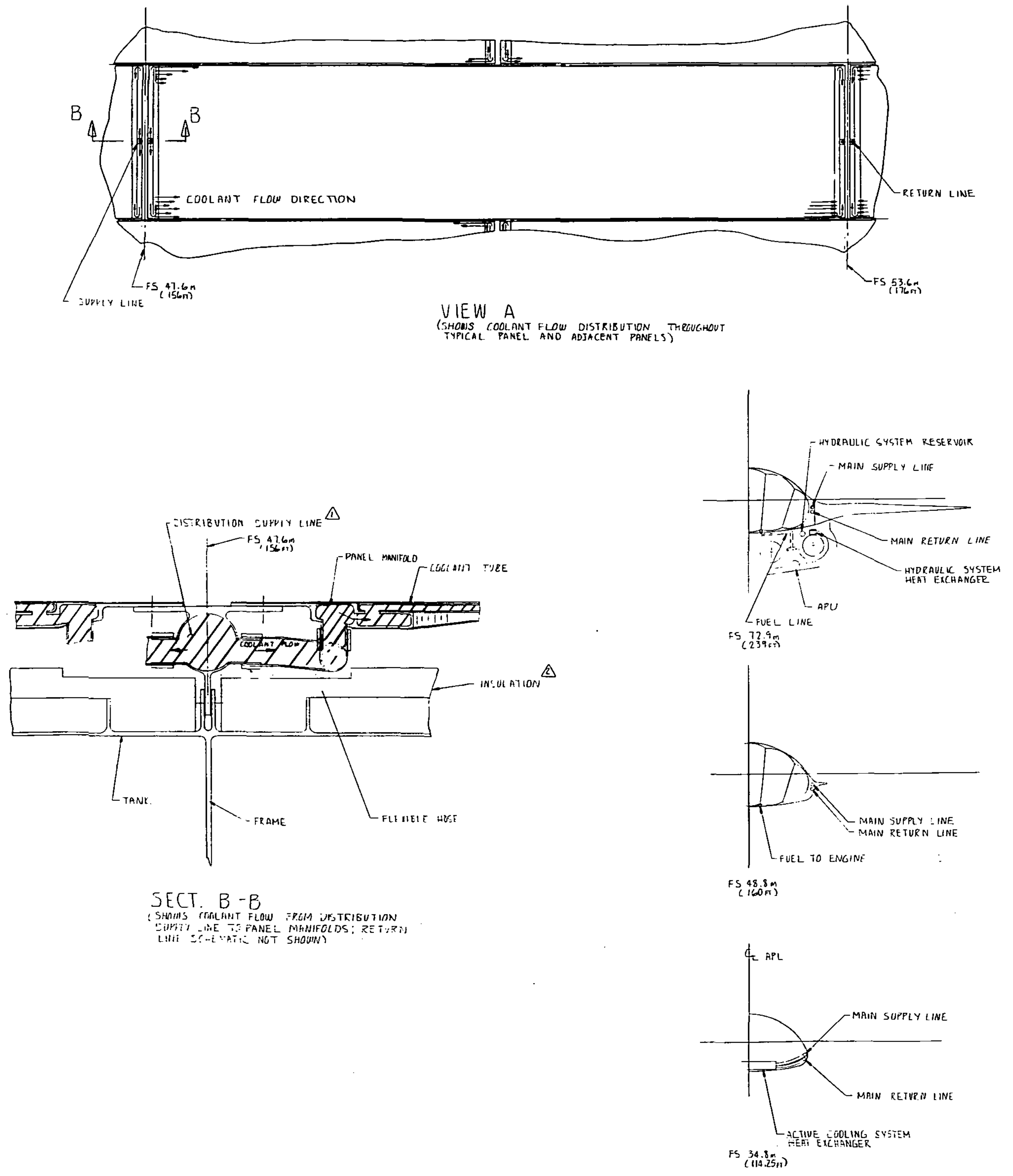
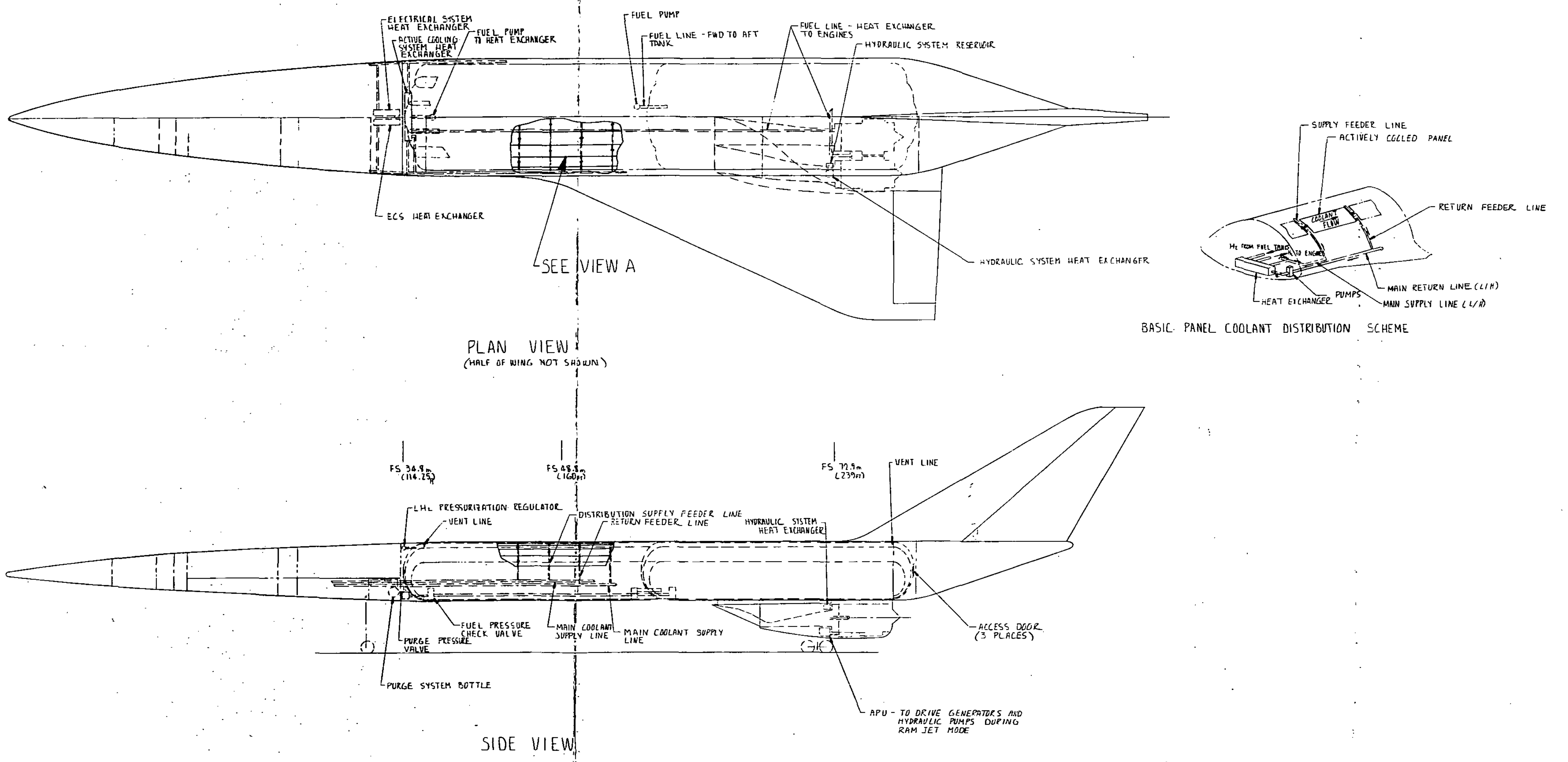




General Notes:

1. ▨ Indicates methanol/water coolant
2. Insulation material, 64.1 kg/m^3 (4 lbm/ft^3) closed cell fiber reinforced polyurethane with an aluminized mylar covering

FIGURE 44
ACTIVE COOLING SYSTEM, CONCEPT 3

GP75-0131-15

6.4 QUALITATIVE ASSESSMENT OF PRODUCIBILITY ASPECTS

A detailed cost analysis was beyond the scope of this study. However, for purposes of comparison, relative costs are a valuable guide to supplement performance evaluation. These costs are a function of the producibility aspects of each airplane. Relative costs were determined by examining each producibility factor and applying a relative value of complexity, machinability, tooling, etc., to each peculiar element in the three concepts.

Producibility factors include both material and labor. In this producibility assessment, welding is treated as a separate factor to give it greater visibility because of its extended usage.

This producibility comparison concentrates on the structural items which differ among the three concepts. Common structural items such as the wing, vertical tail, active cooled panels etc. are not included in detail. Also, tooling costs are not treated in depth because this non-recurring cost is a function of production rate. These were considered equivalent for all concepts. The impact of this factor on the relative total airplane cost was examined in relation to DC-10 production costs to provide an understandable basis of comparison.

Concept 1 was determined to be the "most producible" airplane. It was assigned the unit value of 1.0 in the relative cost comparisons.

6.4.1 Comparison of Concepts

a. Commonality Among Concepts - Common structural items which have little impact on the relative cost of the aircraft concepts, and involve 70% to 80% of the total vehicle initial investment cost, include:

- o Wing structure
- o Nacelle module and supports
- o Vertical fin structure and supports
- o Forward and aft fuselage structure
- o Actively cooled skin panel construction, attachment, and plumbing
- o Systems installation

b. Concept 1 Analysis - Concept 1 is considerably more producible in all areas of welding, forming, machining, and assembly than either Concept 2 or 3 and is significantly less expensive from a material cost viewpoint.

The non-integral fuel tank design of Concept 1 is preferable from a producibility view point because it permits complete tank assembly independent of the fuselage. This simplifies the fabrication of both tank and fuselage as well as inspection and rework that may be required.

The single bead type weld used on Concept 1 is readily welded and inspected. Similarly, the dome construction presents no unusual welding problems. However, care would be required to assure adequate fit-up and mating of the welded cylinders to each other to take into account possible problems arising from weld distortion and tolerances.

The potential for using forgings on Concept 1, rather than bar or plate to achieve a better raw material utilization is considered very good, especially on applications like the tank dome stiffening rib and pie shaped plates. The use of numerous and efficient small forgings in applications such as frame fittings is also feasible.

Relatively little machining is required on Concept 1, due principally to the extensive use of conventional airframe (sheet metal) design; only three frames per tank and one out of every three fuselage frames is a machining or extrusion.

c. Concept 2 Analysis - Using relative cost as a producibility yardstick, Concept 2 is about 3-1/2 times as difficult to produce as Concept 1.

Producibility complexities arising out of the use of the integral tank design, coupled with the extensive use of intersecting integral stiffeners account for higher relative costs (compared to Concept 1) in the major areas of manufacturing as follows:

- Assembly (3:1)
- Forming (2.5:1)
- Machining (20:1)
- Welding (5:1)

The material costs of Concept 2 are five times as high as for Concept 1. Concept 1 utilizes sheet stock which is procured in near finished thicknesses. However, the material for Concept 2 is procured as thick plate and much of the material is lost due to machining required to produce a finished part.

While forming flat machined panels of intersecting integral structure is feasible, and in fact is discussed in current literature, a certain amount of development effort is foreseen to identify the various forming parameters for use by design personnel.

Although the tank frames of Concept 2 are considerably more costly than those for Concept 1 (Machined forgings vs. mostly sheet metal), their fabrication is well within the state of the art. Two approaches are possible for these frames; one, frames machined from rolled ring forgings and; two, frames fabricated from several segments machined from curved die forgings. The forging cost for Concept 2 is estimated to be about 30 times that of Concept 1.

The high overall machining cost for Concept 2 is due to the multiplicity of machining (tooling, setups, operations) required to produce the integrally stiffened tanks. Included are 58 tank frame rings of various shapes and diameters and about 200 support links.

The higher welding cost of Concept 2 is due not only to direct welding but also to other considerations associated with the impact on the facilities and equipment required for automatic welding and inspection of the Concept 2 tank, which is twice as long as the Concept 1 tank. Since the tank sections are too large to relieve the residual stresses thermally after welding, other means need to be considered. One promising method that could be considered for this application is shot peening.

d. <u>Concept 3 Analysis</u> - Concept 3 is somewhat less difficult (about 85% as expensive) to produce than Concept 2 but still approximately three times the cost of Concept 1.

As with Concept 2, producibility considerations include the use of an integral tank design employing intersecting integral stiffeners. In addition, the elliptical cross section of the fuselage/integral tank of Concept 3 introduces production problems. Relative costs of Concept 3 (to Concept 1) in the major areas of manufacturing are as follows:

- o assembly (5:1)
- o forming (2.5:1)
- o machining (15:1)
- o welding (7:1).

The material cost for Concept 3 is about three times that for Concept 1.

The forming complexity of Concept 3 is about 2.5 times that of Concept 1.

Forgings are projected for the outer curved segment at each fuselage station of the integral tank, in order to obtain a raw stock form having integral attach lugs. No problems are anticipated for this forging application. Forging costs for Concept 3 are estimated to be about five times those for Concept 1 but considerably less than for Concept 2.

The Concept 3 design lends itself readily to the use of net extrusions which, impacts favorably on production by lowering overall machining cost and improving material utilization.

The higher cost of welding Concept 3 (as well as Concept 2) is due to the tentative use of a double weld bead at the attachment of all integral stiffener structure at joints and to frames.

e. Relative Comparison With DC-10 - In order to provide a frame of reference an estimate was made to relate the cost of Concept 1 to that of a DC-10. Results showed that Concept 1 would be approximately 2.3 times the cost of a McDonnell Douglas DC-10 for a comparable economic time period and production quantity. Figure 45 shows a cost trend with respect to the DC-10 in terms of quantity. The point of interest is the 100 quantity, where this evaluation was based. Concept 3 cost crosses over Concept 2 at a certain quantity in the figure. This occurs because the initial non-recurring cost of Concept 3 is higher. Also, the initial learning curve is higher because it has a larger complexity factor.

6.4.2 Cost Comparison Summary - A summary comparison of the relative cost of the three concepts is given in Table 1. The table indicates the center fuselage factors as well as the total aircraft factors. It should be noted that all of the values above are ratios of relative cost for common items and, are not directly additive.

6.4.3 Alternative Integral Tank Construction - A brief additional study was conducted to obtain trends for cost reduction in production of integral hydrogen tanks. In the first phase, two alternate methods of providing integral stiffening without the costly machining required for the original isogrid configuration were considered.

The first of these was to forge the isogrid pattern into 2219 aluminum plate and then weld assembled plates. It can be seen in Table 2 that tank wall machining costs are significantly reduced but that material costs are dramatically higher than those shown in Table 1. The resulting cost saving, although significant, was not of the magnitude that could make this method of

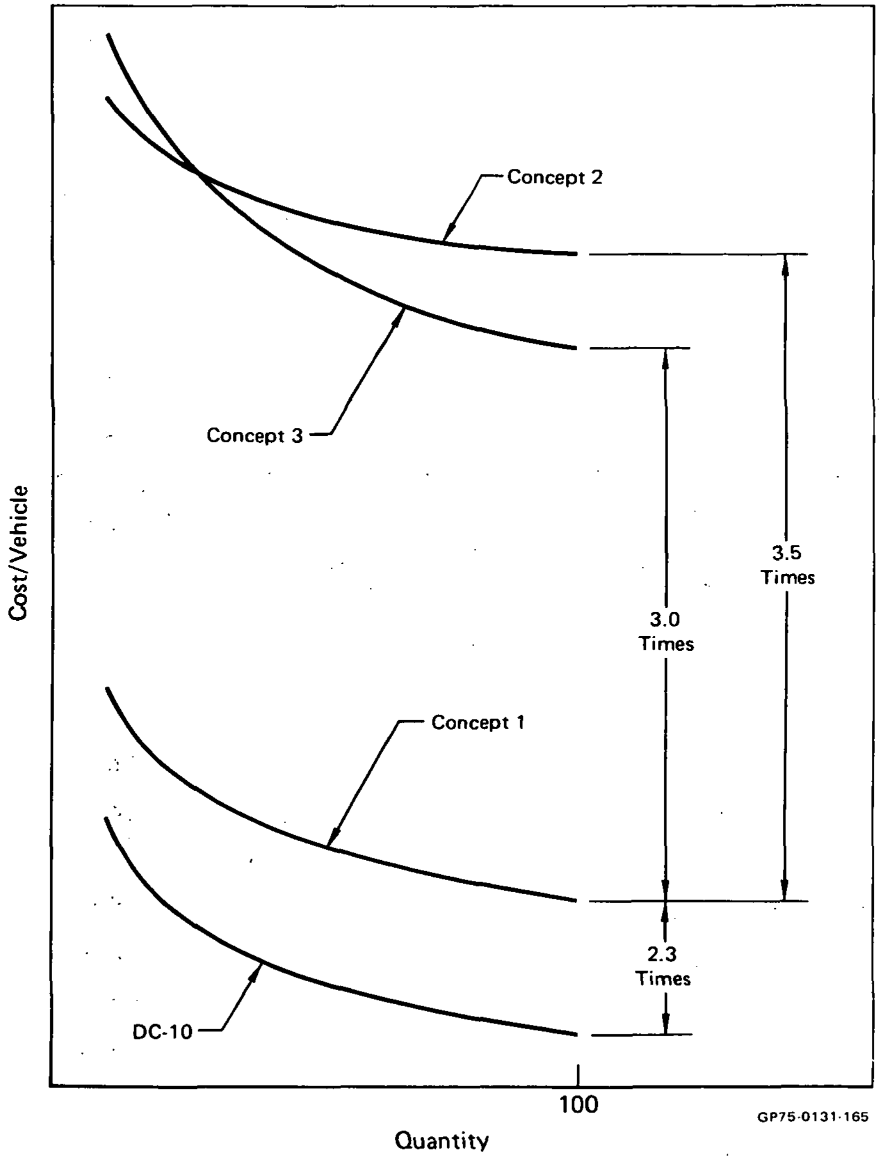


FIGURE 45
TREND OF VEHICLE COST

TABLE 1
RELATIVE COST RATIOS

Item	Concept 1	Concept 2	Concept 3
Welding	1	5	7
Forming	1	2.5	2.5
Material	1	5	3
Machining			
• Fuselage Frames and Bulkheads	1	2	0.4
• Tank to Fuselage Ties	1	15	32
• Tank Frames	1	24	15
• Tank Wall	1	31	35
• Tank Ends	1	9	9
• Wing Attachment	1	1	0.1
Overall Machining	1	20	15
Assembly	1	3	5
Center Fuselage*	1	10	8
Total Vehicle Cost	1	3.5	3

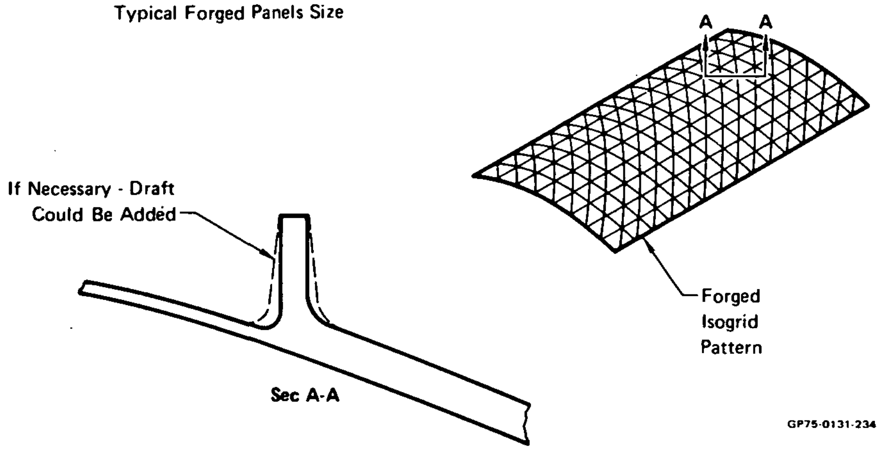
*Includes tank, wing supports, and fore and aft stress links

GP75-0131-143

TABLE 2
FORGINGS IN LIEU OF PLATE STOCK TANK WALLS

Item	Concept 1	Concept 2	Concept 3
Welding	1	10	11
Forming	1	1	1
Material	1	14	12
Machining			
(a) Fuselage Frames and Bulkheads	1	2	0.4
(b) Tank to Fuselage Ties	1	15	32
(c) Tank Frames	1	24	15
(d) Tank Wall	1	2	3
(e) Tank Ends	1	9	9
(f) Wing Attachment	1	1	0.1
(g) Overall Machining	1	13	9
Assembly	1	3	5
Total Vehicle Cost	1	3.1	2.9

Typical Forged Panels Size



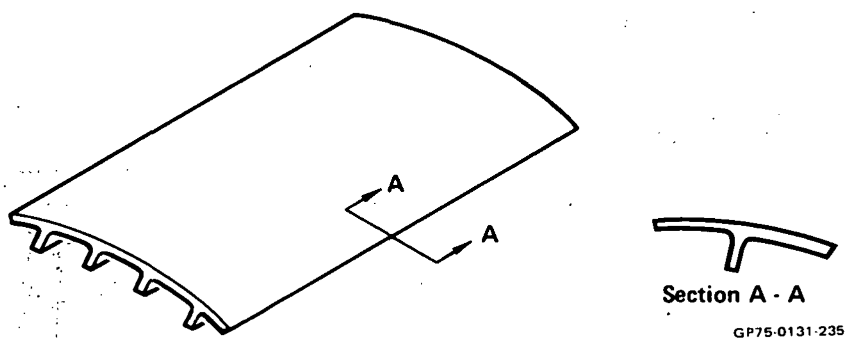
GP75-0131-234

construction competitive with the plain monocoque shells of Concept 1.

The second alternative method was to extrude longitudinal stiffener "planks", weld them into the tank assembly, and weld the tank rings to the stiffeners to stabilize them. Improved cost ratios resulted, as shown in Table 3. Again, however, they were not large enough to be competitive with the original non-integral tanks.

TABLE 3
EXTRUSIONS IN LIEU OF PLATE STOCK TANK WALLS
(7.6 cm x 102 cm, 3 in. x 40 in. Extrusion)

Item	Concept 1	Concept 2	Concept 3
Welding	1	6	8
Forming	1	1.1	1.1
Material	1	4	2
Machining			
(a) Fuselage Frames and Bulkheads	1	2	0.4
(b) Tank to Fuselage Ties	1	15	32
(c) Tank Frames	1	24	15
(d) Tank Wall	1	1	1
(e) Tank Ends	1	9	9
(f) Wing Attachment	1	1	0.1
(g) Overall Machining	1	12	8
Assembly	1	3	5
Total Vehicle Cost	1	2.7	2.4



A final investigation was conducted to assess a heavier, but less expensive, configuration for Concepts 2 and 3. That was to use plain skin monocoque tanks as had been used on the Concept 1 aircraft. The results of that study are illustrated in Table 4. In this instance there was a drastic reduction in overall cost. However, this modification would penalize the Con-

TABLE 4
UNIFORM THICKNESS TANK WALLS AND ENDS
IN LIEU OF INTEGRAL STIFFENERS

Item	Concept 1	Concept 2	Concept 3
Welding	1	4	5.5
Forming	1	1	1
Material	1	1.2	0.8
Machining			
(a) Fuselage Frames and Bulkheads	1	2	0.4
(b) Tank to Fuselage Ties	1	15	32
(c) Tank Frames	1	24	15
(d) Tank Wall	1	0.2	0.3
(e) Tank Ends	1	4	4
(f) Wing Attachment	1	1	0.1
(g) Overall Machining	1	7	8
Assembly	1	3	5
Total Vehicle Cost	1	1.6	1.8

GP75-0131-236

cept 2 aircraft over 7.85 Mg (17,300 lbm) and the Concept 3 aircraft over 2.27 Mg. (5,000 lbm) in tank weight alone. This construction would have penalized these aircraft 452 km (244 NM) and 117 km (63 NM) in range, respectively.

The conclusions reached during these studies were that the integral tanks could not be made competitive from a cost standpoint without both going to plain skin construction and eliminating the need for thermal strain compensation.

6.5 QUALITATIVE ASSESSMENT OF MAINTAINABILITY ASPECTS

A maintainability assessment of each concept has been performed. The type of assessment made was based on a comparative analysis using the concept with lowest mean time to complete maintenance action as a base and assigning it a unit value of 1.0.

The maintainability assessment was made in terms of relative merit values based on the anticipated degree of difficulty to accomplish maintenance of the various concepts. It provided a general understanding of relative maintenance complexity of the three aircraft. The relative merit values were then related to the DC-10 as a point of reference.

6.5.1 Comparison of Concepts

a. Commonality Among Concepts - The following systems were determined to be common from a maintenance standpoint, except where some of these systems interfaced with the fuel tankage.

- o Power Plants
- o Electrical System
- o Auxiliary Power Unit
- o Hydraulics and Pneumatics
- o Fuel Feed/Distribution System
- o Thermal Protection System
- o Flight Controls
- o Landing Gears

Only the differences in the above systems resulting from variations in fuel tank configurations were included in further evaluation. Special consideration was given to access doors and equipment spacing.

b. Concept 1, 2 and 3 Analysis - Fundamental differences exist between concepts in the center fuselage tankage areas. In Concept 1, the tank element is not subject to airframe structural loads. In Concepts 2 and 3, the tank element is subject to varying degrees of structural loads.

The maintainability assessment of each concept included access and repair of the tanks themselves and the equipment, lines, cables, etc. within the tank area. Significant factors considered in development of relative values for each concept are as follows: Numbers in parentheses designate Concept 1, 2 or 3.

Fuel Tanks:

(1) Entire tank is isolated from structure. Tank can be removed for repair.

(2) Tank is isogrid shell and is part of basic fuselage structure. It contains many links and lugs to provide support for the wing and outer fuselage cover. For this reason, tank repairs would be made with tank remaining in place.

(3) Tank is isogrid shell and is part of basic fuselage structure. Has many links and lugs supporting the actively cooled panels. Tank repair would be accomplished with tank remaining in place.

Actively Cooled Panel Manifolds and Controls:

(1) Wide separation (0.22 m (8.5 in.) minimum) for access and leak detection.

(2) Limited separation (0.09 m (3.5 in.) minimum) for access and leak detection.

(3) Limited separation (0.09 m (3.5 in.) minimum) for access and leak detection.

Actively Cooled Panel Removal:

(1) All panels are structural which makes use of large access doors more difficult.

(2) Center fuselage panels are semi-structural which limits the use of large access doors.

(3) All panels can be considered as removable doors.

Link and Lug Adjustment/Repair:

(1) Very few links, which provide tank support only.

(2) Substantial use of links and lugs, at many frames, to support tank to fuselage and tank to wing.

(3) Extensive use of links and lugs, at all frames, to provide actively cooled panel support and tank suspension compatible with expansion/contraction requirements.

Nitrogen Purge System:

(1) Extensive volume to be purged between tank and actively cooled panels.

(2) Smaller volume to be purged between tank and actively cooled panels.

(3) Small volume to be purged between tank and actively cooled panels.

Subsystem Line Routing:

(1) Very good access for repair and servicing functions from inside fuselage.

(2) Access through external doors required for many repair and servicing functions.

(3) Access through external doors required for most repair and servicing functions since much routing is in the wing root area.

Equipment in tank area included in evaluation were:

- o Coolant Supply/Return Lines
- o Heat Exchanger Unit
- o Fuel Feed Lines
- o Fuel Boost Pumps
- o Fuel Transfer Controls
- o Plumbing Repairs
- o Electrical Repairs

6.5.2 Concept Comparison - The relative values for the three concepts are shown in Table 5. These values are based on opinion as to the degree of difficulty of performing inspection and repair tasks.

As shown, if the subsystem with the lowest mean time to complete a maintenance action is taken as 1.0, the average value for all subsystems being compared is 1.04 for Concept 1. Using this value as baseline, Concept 2

TABLE 5
RELATIVE COMPARISON VALUES

Service or General Maintenance Action	Concept		
	1	2	3
Structural Tank Repairs	1.00	1.30	1.40
Actively Cooled Panel Leak Inspection	1.00	1.20	1.40
Actively Cooled Panel Removal	1.30	1.20	1.00
Actively Cooled Panel Manifolds and Controls	1.00	1.30	1.40
Link and Lug Adjust/Repair	1.00	1.60	1.80
Coolant Supply Lines	1.00	1.40	1.50
Coolant Return Lines	1.00	1.40	1.50
Heat Exchanger Unit	1.00	1.30	1.40
Nitrogen Purge System	1.40	1.30	1.00
Fuel Feed Lines	1.00	1.30	1.50
Fuel Boost Pumps	1.00	1.20	1.40
Fuel Transfer Controls	1.00	1.30	1.30
Plumbing Repairs	1.00	1.20	1.30
Electrical Repairs	1.00	1.20	1.30
Flight Control Cables	1.00	1.20	1.40
Average Level of Difficulty	1.04	1.28	1.37
Normalized Level of Difficulty	1	1.2	1.3

Comparative Ratings (Degree of Difficulty)

1.0 = Concept with Lowest Mean Time to Complete Maintenance Action (Used as Baseline)
1.5 = 50% Greater Time to Complete Action Compared to Baseline
1.8 = 80% Greater Time to Complete Action Compared to Baseline

GP75-0131-144

requires 24% more maintenance time than Concept 1, and Concept 3 requires 32% more than Concept 1. Note that these relative values are based on averages of all maintenance actions for subsystems in the tank area and all other factors are considered common to the three concepts.

Therefore, the concept having the circular, non-integral tankage (Concept 1) is most suitable for maintenance. The primary reason is that considerable space is available between the tank and fuselage structure to permit equipment installation, servicing, repair and inspection.

6.5.3 Relative Comparison with DC-10 - A comparison was made of Concept 1 with the DC-10. The maintainability differences considered in the two aircraft are as follows:

o Concept 1 employs a thermal protection system whereas the DC-10 does not.

o The thermal protection design dictates the use of honeycomb structure panels over a large percentage of Concept 1 surface area. The simpler skin type structure of the DC-10 is easier to maintain.

o On Concept 1, nitrogen purge is required between the fuel tanks and outer structure, whereas this is not required on the DC-10.

o Due to the much larger size of Concept 1 (overall length of 109.9 m (360.5 ft) vs. 55.53 m (182.17 ft), aircraft structure, tubing, and wire runs will require more maintenance man hours.

o Less time is required for fuel tank repair on the DC-10 because fuel tank access has fewer panel screws and no insulation.

o Fuel servicing of the DC-10 is much easier because of relative ease of handling JP fuel rather than cryogenic LH_2.

Table 6 provides a relative comparison of known difference between Concept 1 and the DC-10. Due to lack of definitive information on turboramjet engines, comparison of engine maintenance is not included. The comparison is based on airframe and installed equipment.

TABLE 6
GENERAL SERVICE AND MAINTENANCE ACTIONS

TASK	CONCEPT 1	DC-10
Access	1.5	1.0
Tank Repairs	1.2	1.0
Actively Cooled Panels	1.5	0
Nitrogen Purge	1.4	0
Structure Repairs	1.5	1.0
Control Lines	1.2	1.0
Fuel Service	1.3	1.0
Engine Controls	1.3	1.0
Average Level of Difficulty	1.36	.75

(assuming equal time for each of the above classifications)

Based on the above evaluation, Concept 1 will require 1.8 times the maintenance man hours required for the DC-10. Typical direct maintenance man hours for the DC-10 airframe and installed equipment (less engines) are approximately 3 MMH/FH plus slightly over 9 man hours per flight. The Concepts 2 and 3 man hours per flight are 24% and 32% higher than Concept 1, respectively.

7. PERFORMANCE ANALYSIS

7.1 AERODYNAMICS

The aerodynamic coefficients used to compute mission performance, takeoff and landing characteristics and longitudinal stability characteristics and the methods used to obtain them are described. These include the zero lift drag (C_{D_o}), the induced drag factor (L'), and the lift curve slope (C_{L_α}).

7.1.1 Zero Lift Drag - The MCAIR advanced design drag method was used to estimate zero lift drag. C_{D_o} consists of skin friction drag, base drag, protuberance drag, wave drag, boundary layer diverter drag, and cowl drag. Ram drag and spill drag are accounted for as propulsion drag. A detailed description of this method can be found in Reference (7). This method uses the Schoenherr flat plate friction coefficient to determine the incompressible skin friction coefficient. This is corrected for compressibility and temperature effects using the Sommer and Short T' method. Figure 46 presents this correction. Base drag is estimated using the data correlations of Figure 47.

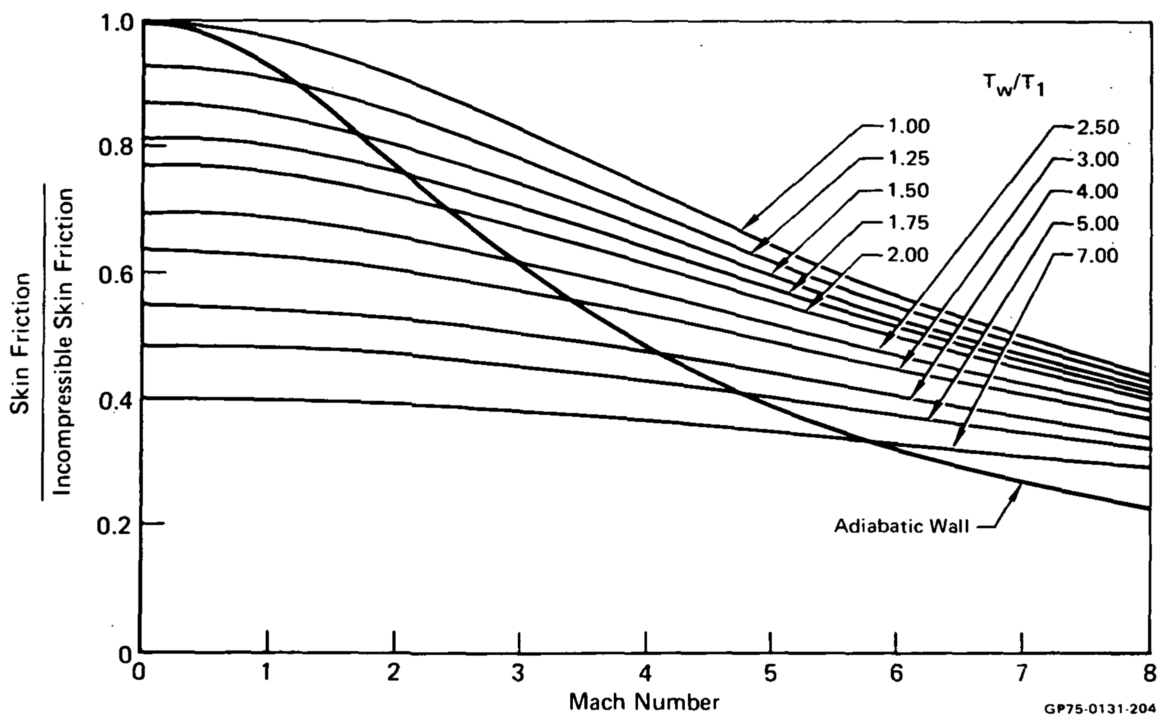


FIGURE 46
EFFECT OF MACH NUMBER AND TEMPERATURE ON SKIN-FRICTION RATIO

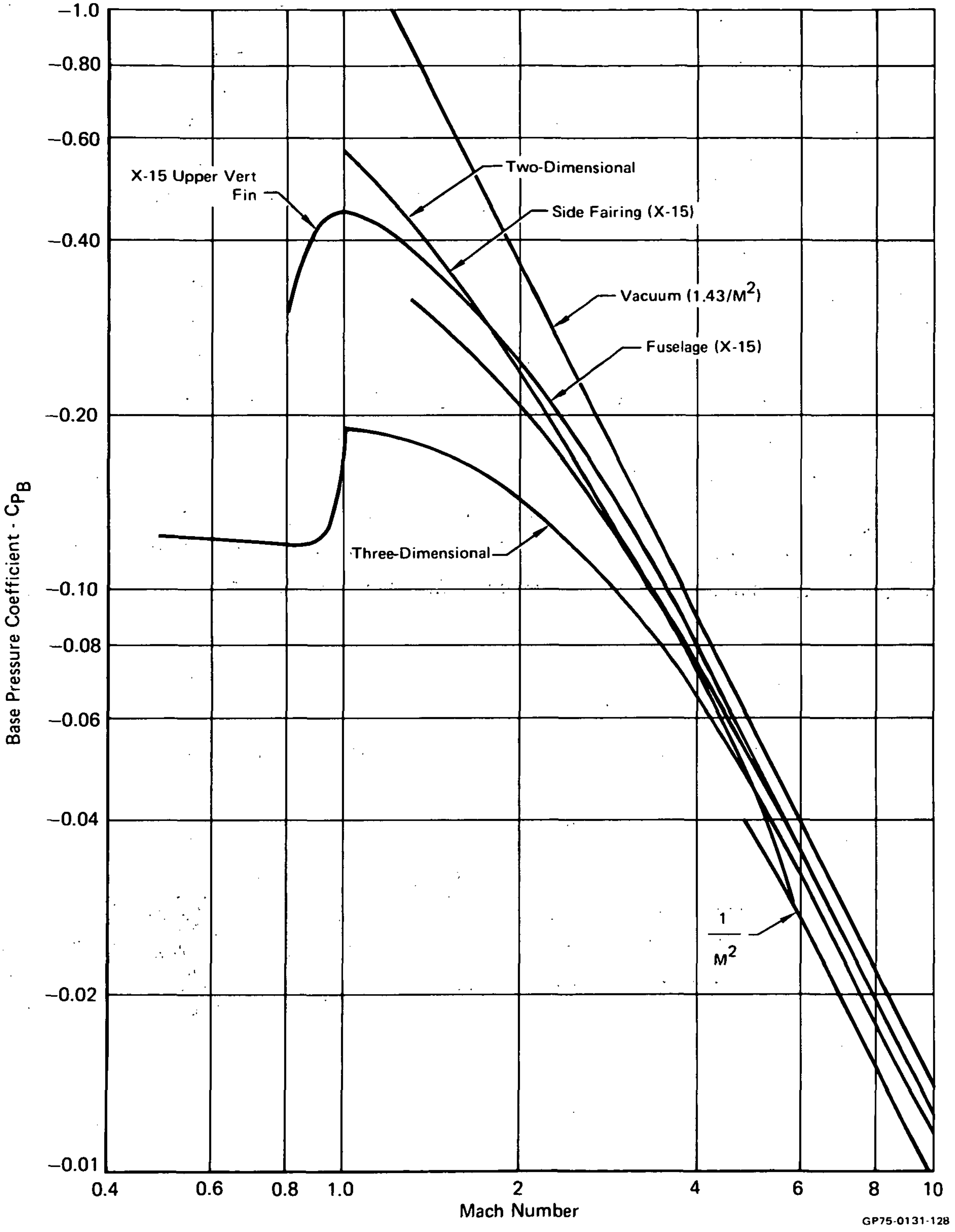


FIGURE 47

BASE DRAG TURBULENT BOUNDARY LAYER

Drag due to protuberances, such as rivet heads, gaps between plates, etc., can be estimated only with a detailed knowledge of the aircraft. A C_{D_F}= 0.00065 based on total wetted area subsonically and C_{D_F} = 0.00085 supersonically were used for protuberance drag in this study.

Wave drag terms are obtained by examining the individual aircraft components (fuselage, nacelle, wing, and vertical tail). This has the advantage that the components can be defined directly from the configuration area distribution, and the best available correlation for each component and Mach region can be used. At transonic speeds, where theoretical treatments are inadequate, data correlations are used: for simple shapes at supersonic speeds the method of characteristics is used, and for complex shapes linear theory is used.

The disadvantage of the component build-up method is its inability to account for mutual interference. In order to check the amount of interference present, the drag of the NASA HT-4 configuration, on which the study configurations are based, was estimated and compared to the drag measured by NASA in the wind tunnel. This comparison is shown in Figure 48. Since the measured and predicted drags agree so closely, interference effects appear to be minimal and are neglected.

Table 7 presents estimated drag coefficients for Concepts 1 and 2. The drag of these configurations is higher than that obtained by simply extrapolating the model drag to the full scale Reynolds Number. This is caused by the additional drag of the engine nacelle and the higher drag of the cooled skin. The drag of Concept 2 was obtained by incrementing the drag of Concept 1. The only difference is that the fuselage diameter of Concept 2 is 0.3 m (12 inches) less in average diameter than Concept 1. This resulted in a decrease in wave drag and a small decrease in skin friction drag due to a reduction of fuselage wetted area.

Table 8 presents the drag coefficients of Concept 3, the blended wing body configuration. The coefficients of Concept 3 are based on smaller reference area than Concepts 1 and 2, $S_{\text{theo concept 3}} = 0.90\ S_{\text{theo concepts 1\&2}}$, resulting in higher drag coefficient for Concept 3. The lower drag that results is primarily due to the smaller size of the vehicle. However, this is partially offset by a larger nacelle, although all three configurations have the same size engines. On Concepts 1 and 2, it was possible to partially submerge the

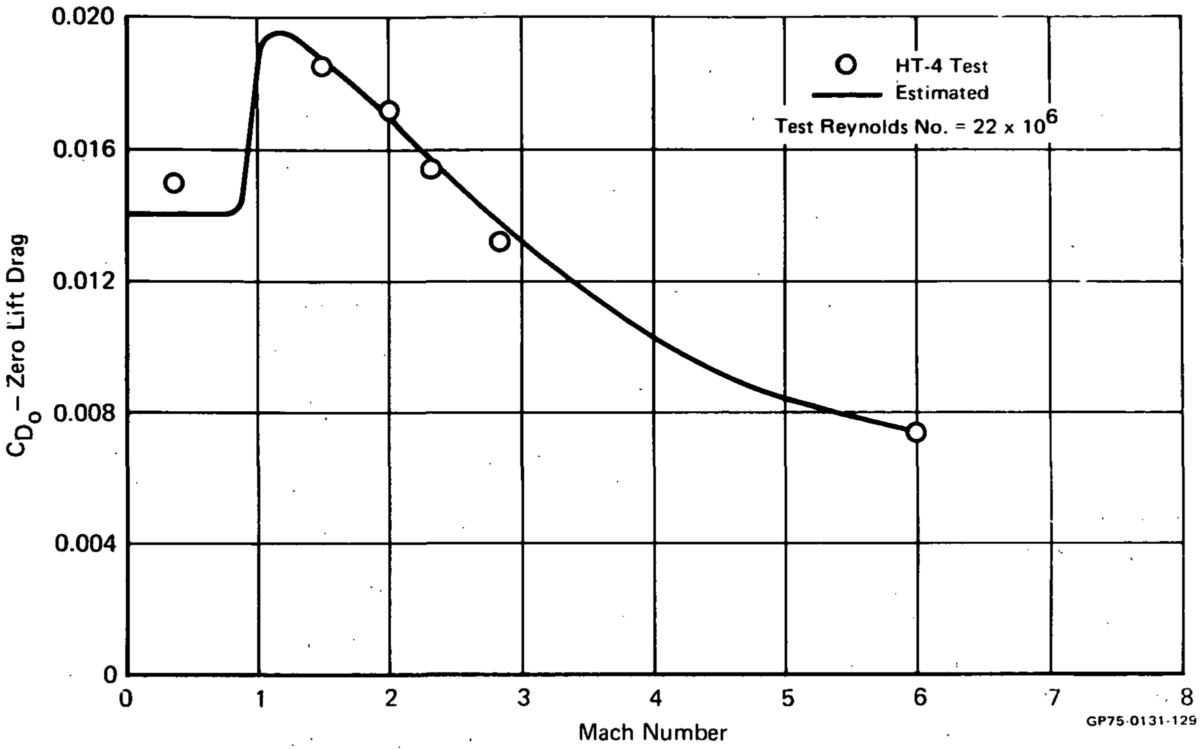


FIGURE 48
ZERO LIFT DRAG OF HT-4 MODEL vs MACH NUMBER

TABLE 7
DRAG COEFFICIENT ESTIMATE OF CONCEPTS 1 AND 2

Component	Mach Number								
	0.8	1.2	1.5	2.0	2.5	3.0	4.0	5.0	6.0
Nose		0.00134	0.00121	0.00109	0.00099	0.00094	0.00085	0.00078	0.00076
Forebody		0.00037	0.00032	0.00028	0.00026	0.00024	0.00021	0.00019	0.00018
Boattail		0.00460	0.00414	0.00352	0.00293	0.00257	0.00181	0.00138	0.00123
Wing		0.00230	0.00230	0.00187	0.00151	0.00120	0.00088	0.00080	0.00073
Vertical Tail		0.00059	0.00058	0.00048	0.00039	0.00031	0.00023	0.00021	0.00019
Wave Drag Sum		0.00920	0.00855	0.00724	0.00608	0.00526	0.00404	0.00336	0.00309
Skin Friction	0.00870	0.00859	0.00805	0.00712	0.00669	0.00633	0.00579	0.00546	0.00543
Nacelle Base	0.00104	0.00165	0.00152	0.00126	0.00101	0.00083	0.00056	0.00039	0.00029
Nacelle Wave		0.00082	0.00079	0.00059	0.00056	0.00053	0.00053	0.00053	0.00053
*Boundary Layer Diverter		0.00130	0.00200	0.00177	0.00142	0.00124/ 0.00000	0.00000	0.00000	0.00000
Concept 1	0.00974	0.02157	0.02091	0.01793	0.01564	0.01419/ 0.01295	0.01092	0.00974	0.00917
Δ Skin Friction (1 → 2)	−0.00008	−0.00008	−0.00007	−0.00006	−0.00006	−0.00006	−0.00006	−0.00005	−0.00005
Δ Fuselage Wave (1 → 2)		−0.00058	−0.00042	−0.00043	−0.00027	−0.00022	−0.00013	−0.00014	−0.00010
Concept 2	0.00966	0.02033	0.01992	0.01704	0.01501	0.01379/ 0.01255	0.01063	0.00945	0.00892

Note: Numbers shown are drag coefficients based on total planform area.

GP75-0131-132

TABLE 8
DRAG COEFFICIENT ESTIMATE OF CONCEPT 3

Component	Mach Number								
	0.8	1.2	1.5	2.0	2.5	3.0	4.0	5.0	6.0
Wave Drag - Nose		0.00148	0.00134	0.00120	0.00109	0.00104	0.00094	0.00086	0.00084
Forebody		0.00032	0.00029	0.00025	0.00024	0.00022	0.00020	0.00018	0.00017
Boattail		0.00390	0.00350	0.00300	0.00250	0.00210	0.00150	0.00116	0.00103
Wing		0.00184	0.00184	0.00150	0.00121	0.00096	0.00070	0.00064	0.00058
Vertical		0.00059	0.00058	0.00048	0.00039	0.00031	0.00023	0.00021	0.00019
Wave Drag Summation		0.00813	0.00755	0.00643	0.00543	0.00463	0.00357	0.00305	0.00281
Skin Friction	0.00865	0.00889	0.00751	0.00663	0.00623	0.00589	0.00542	0.00508	0.00512
Nacelle Wave		0.00152	0.00145	0.00118	0.00109	0.00098	0.00098	0.00098	0.00098
Nacelle Base	0.00243	0.00385	0.00355	0.00296	0.00236	0.00194	0.00132	0.00092	0.00068
Boundary Layer Diverter		0.00143	0.00220	0.00195	0.00156	0.00136/ 0.00000	0.00000	0.00000	0.00000
Concept 3	0.01108	0.02382	0.02226	0.01915	0.01867	0.01480/ 0.01344	0.01129	0.01003	0.00959

Note: Numbers shown are drag coefficients based on total planform area. GP75-0131-133

engines within the wing and fuselage. On Concept 3 the engines are completely external which mandated a larger nacelle as was explained in Section 4.4.

7.1.2 Lift Curve and Induced Drag - Figure 49 was used to estimate the lift curve slope, and its correlation was based on wind tunnel data. This figure shows that lift curve slope is primarily dependent on wing leading edge sweep angle and the ratio of fuselage diameter to wing span. When the wing leading edge is supersonic, the induced drag factor (L') is equal to the inverse of the lift curve slope (per radian).

For this study the effect of leading edge suction was neglected in computing subsonic performance since in the design mission the three configurations are at these Mach numbers for only a short time.

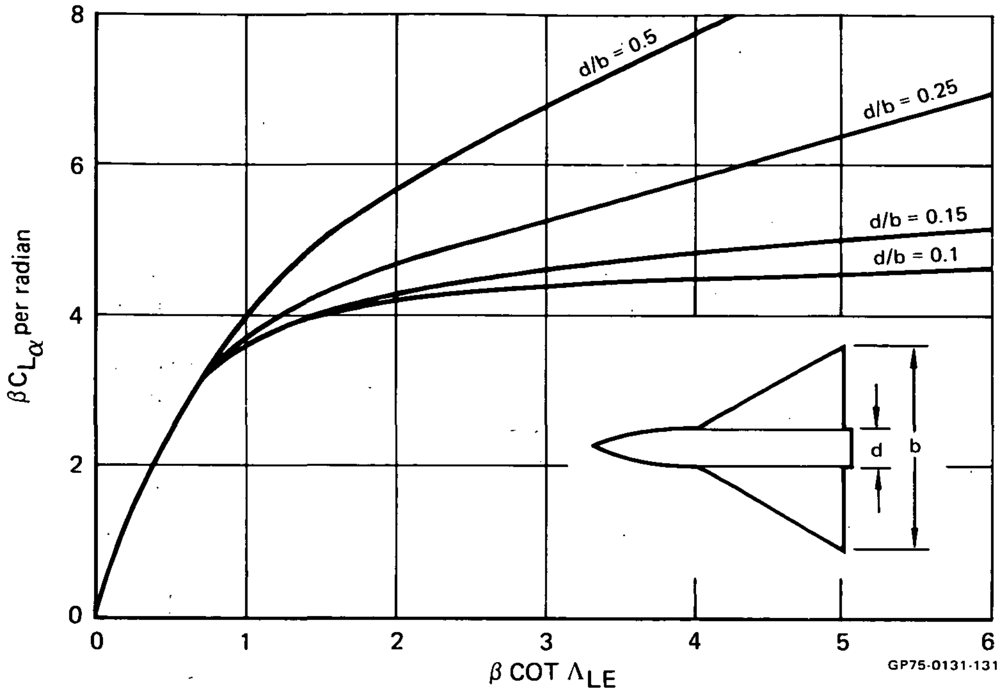


FIGURE 49
LIFT CURVE SLOPE CORRELATION

Figure 50 presents the lift curve slope and induced drag factor of the three study configurations and the HT-4.

7.1.3 Takeoff and Landing Characteristics - The landing gear location was selected to reduce bending loads and prevent taxiing bumps from designing most of the fuselage/tank section. The gear retracts into the thickest part of the wing, only a small amount of frontal area being added with a wing fairing. This location is about 15.2m (50 ft) aft of the center of gravity; therefore, the aircraft takes off and lands on all landing gear simultaneously, without rotation (similar to the B-52).

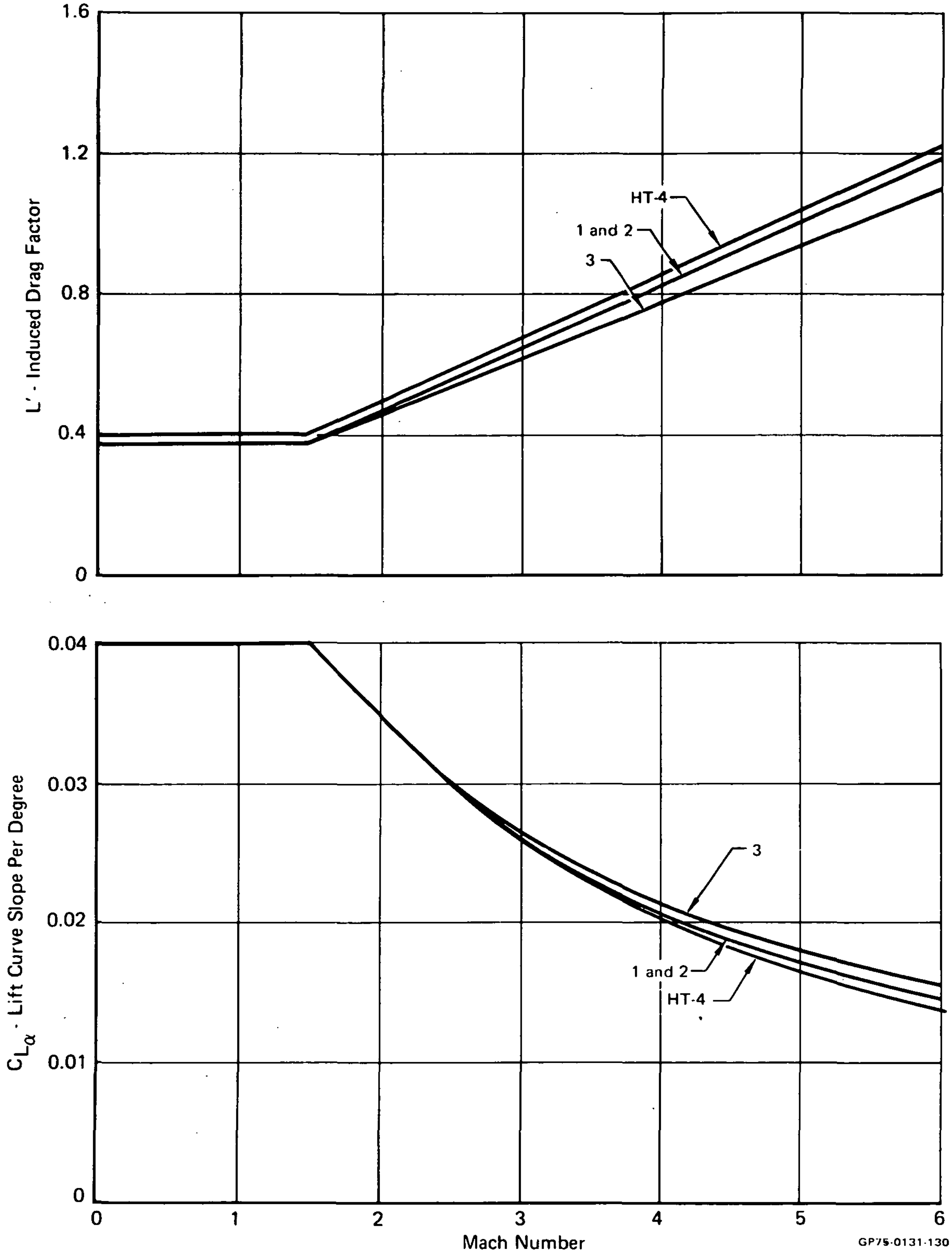


FIGURE 50
LIFT CURVE SLOPE AND INDUCED DRAG FACTOR vs MACH NUMBER

Figure 51 presents the results of a study on Concept 1 to determine the effect of ground incidence angle on takeoff velocity and distance. A 6° baseline incidence was selected. The takeoff distance (ground roll) is 1740 m (5700 ft) and the takeoff velocity is 441 km/hr (238 kts). Higher incidence angles require extremely long nose gear legs. Based on a 6° incidence and a landing weight of 190,500 kg (420,000 lbm) the landing velocity and ground roll are 352 km/hr (190 kt) and 981 m (3220 ft) respectively.

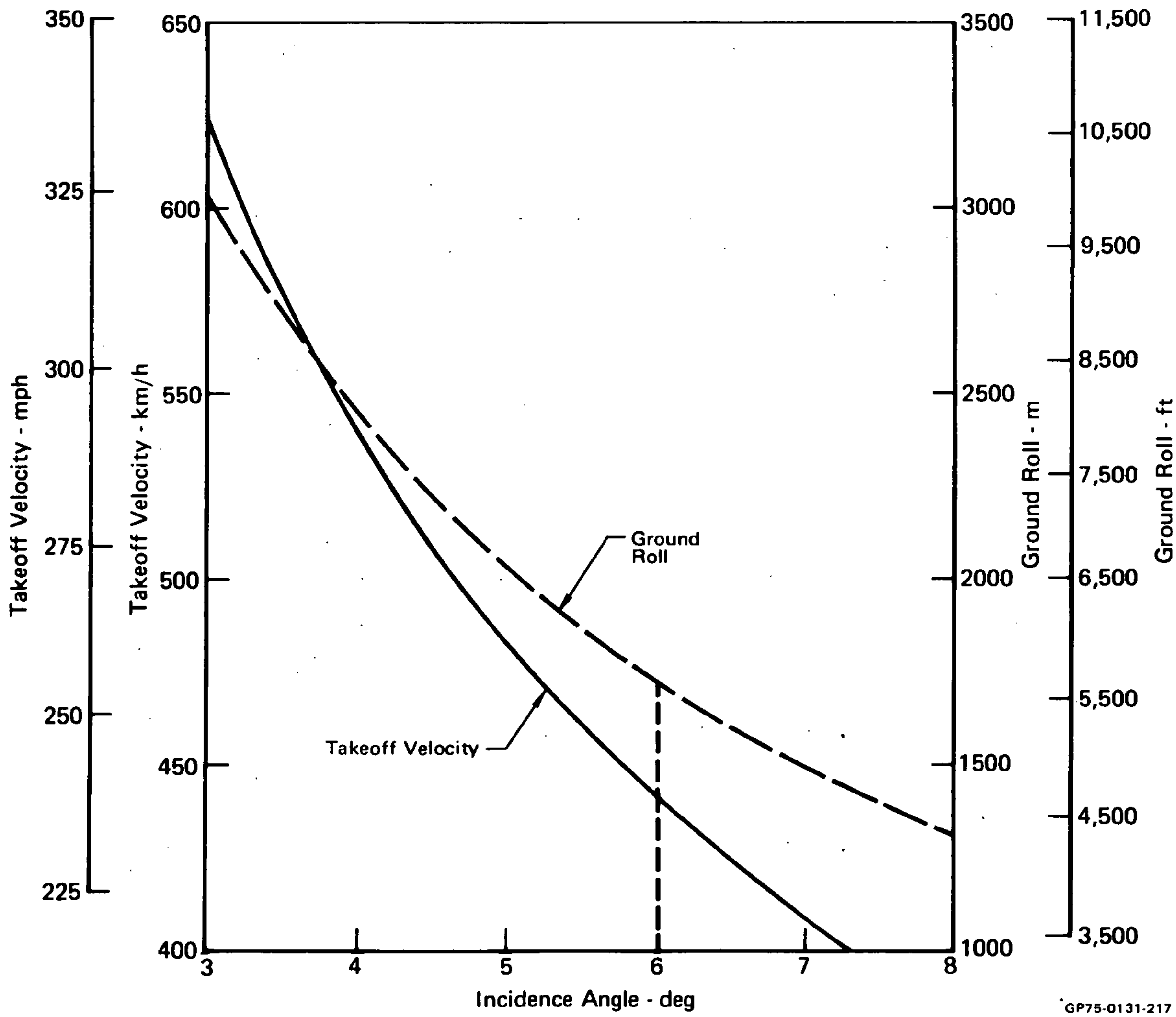


FIGURE 51
TAKEOFF VELOCITY AND GROUND ROLL vs INCIDENCE, CONCEPT 1
W = 290,300 kg (640,000 lbm)

All three concepts will have very similar takeoff and landing characteristics, since the thrust loading and wing loading are nearly equal.

7.1.4 Longitudinal Stability - The one significant change that was made to the HT-4 planform was removing the strake and adding a small amount of area to the wing trailing edge. Based on Reference (3) the aerodynamic center of the HT-4, at Mach 6, strake on, is at 56% body length. Removing the strake and adding wing area moves the aerodynamic center back to 60% body length. This will result in almost neutral stability for the estimated center of gravity (60% ℓ) with a small decrease in lift, as shown in Figure 52. These data were obtained using the Gentry Hypersonic Arbitrary Body computer program and a geometry definition supplied by NASA-Langley. The computer representation of the HT-4 is shown in Figure 53.

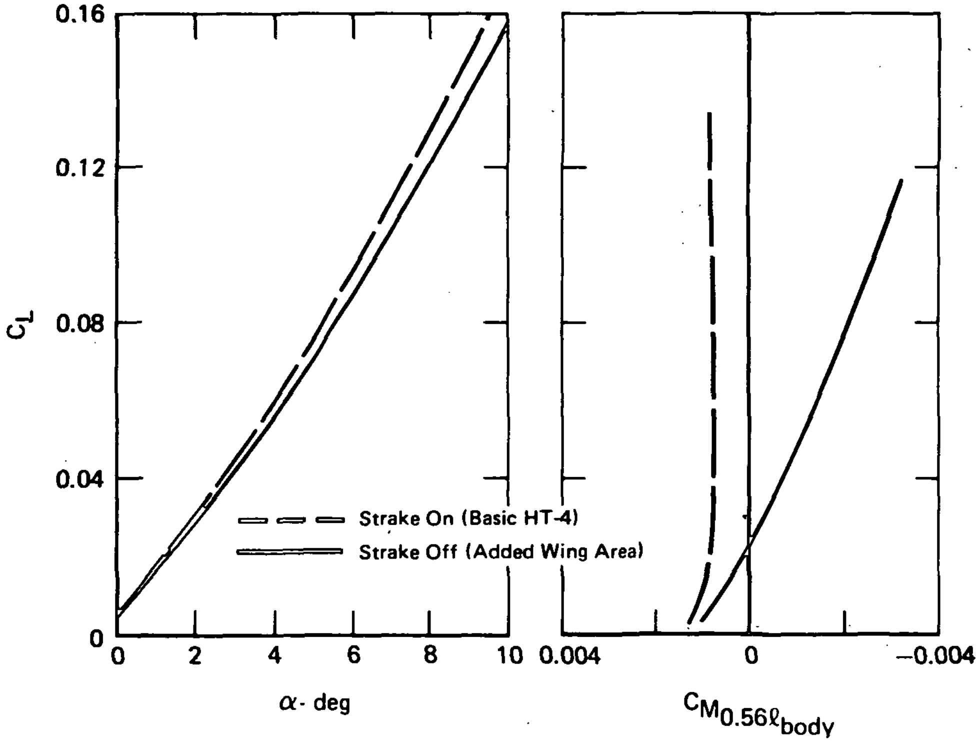


FIGURE 52
EFFECT OF STRAKE ON HT-4 LONGITUDINAL CHARACTERISTICS

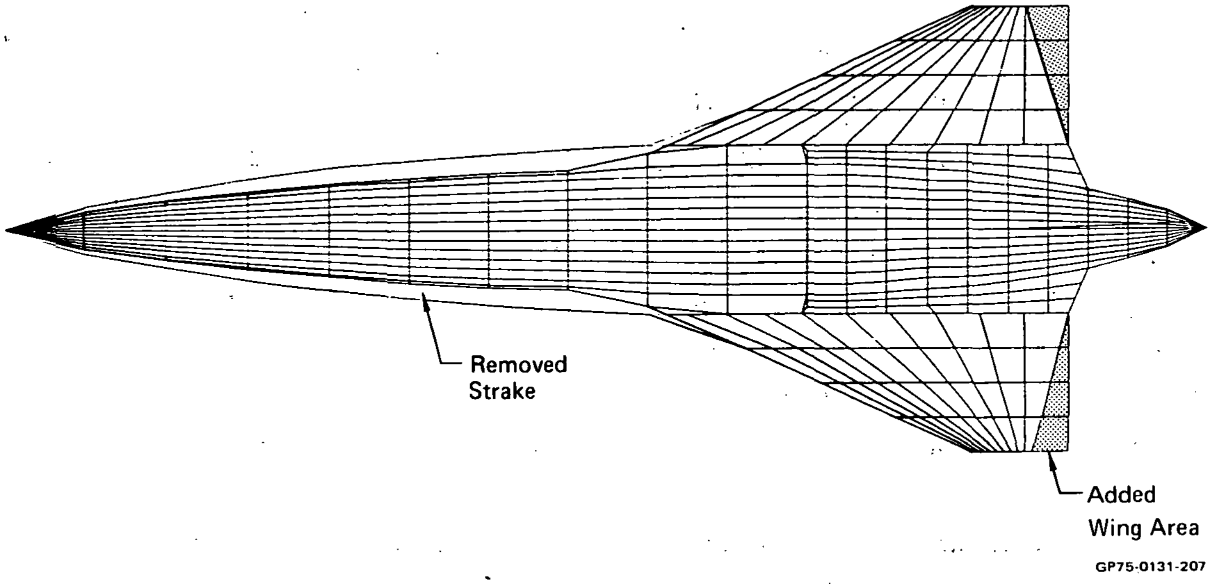


FIGURE 53
COMPUTER REPRESENTATION OF HT-4 PLANFORM

7.2 PROPULSION

The propulsion system consists of a General Electric advanced hydrogen-fueled turboramjet engine with an external compression inlet and a coannular sliding shroud nozzle. To maintain consistency in the study, the same basic propulsion system was incorporated on all three aircraft concepts, with only minor changes to inlet aspect ratio (inlet capture height divided by width) on Concept 3 to facilitate inlet/airframe integration.

It was anticipated during the proposal phase that a Mach 6.0 mixed compression inlet would be used in this study. The inlet pressure recovery to be assumed was MIL-E-5008C. However, further investigation indicated that a smaller, lighter-weight, lower drag air induction system would significantly improve aircraft performance. Therefore, a two-dimensional, three ramp, external compression inlet was designed, with variable capture area and a translating cowl to enhance the airflow capture characteristics and minimize inlet drag over the entire mission.

The inlet is located beneath the wing to obtain the benefits of the wing compression flowfield. The effect of the wing compression on the local Mach number upstream of the inlet is shown in Figure 54.

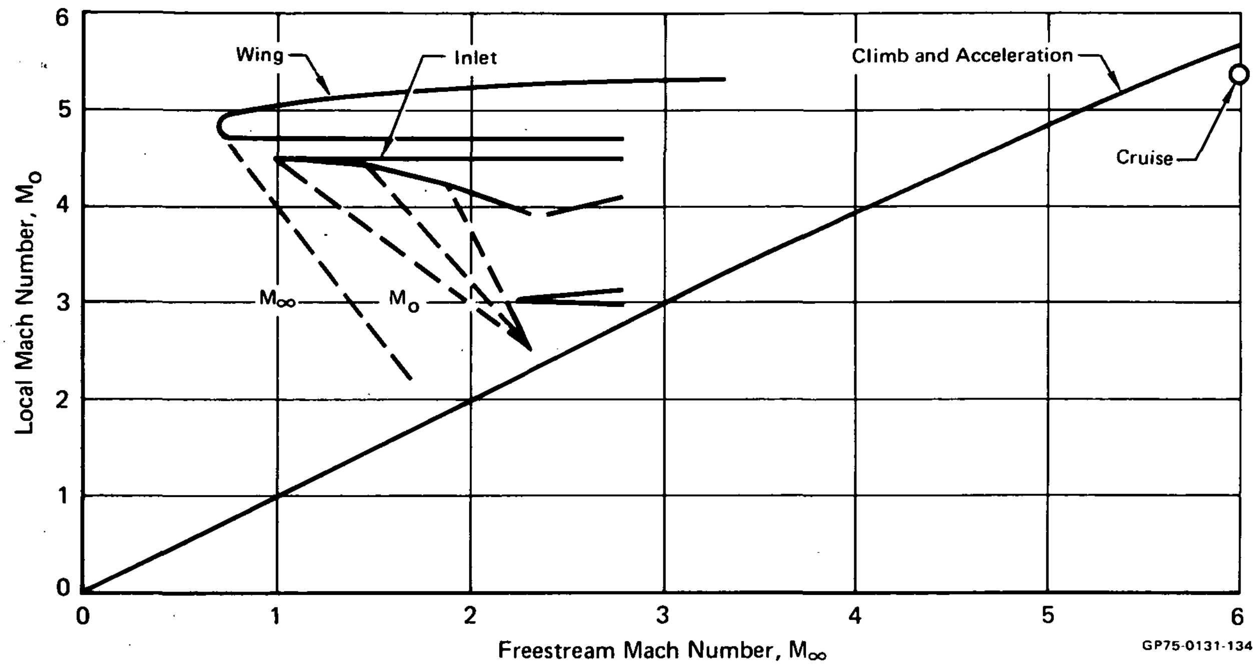


FIGURE 54
EFFECT OF WING FLOWFIELD
Level Flight Angle of Attack

The total pressure recovery of the selected inlet is compared to MIL-E-5008C in Figure 55. Even though the total pressure recovery is lower at high Mach numbers, a net increase in performance is achieved due to the improved capture characteristics and lower inlet drag. A comparison in inlet drag for a mixed compression inlet and the external compression inlet is presented in Figure 56.

A study was performed to evaluate candidate turbojet/ramjet engines. The performance characteristics of the two leading candidates, the P&W SWAT 201A and the GE5/JZ6-Study C, are compared in Figures 57 and 58. The General Electric GE5/JZ6-Study C advanced hydrogen fueled turboramjet engine was selected due to its superior climb/acceleration thrust performance and subsonic throttled specific fuel consumption. At maximum power the engine operates at near stoichiometric conditions in the turbojet combustor. The turbojet and ramjet operate simultaneously above Mach 1.0, until transition to full ramjet power occurs at Mach 3.0. Turbojet nozzle area is varied by means of a translating plug and ramjet nozzle area with a sliding shroud. The engine performance data is classified Reference (8) and therefore is not included in this report.

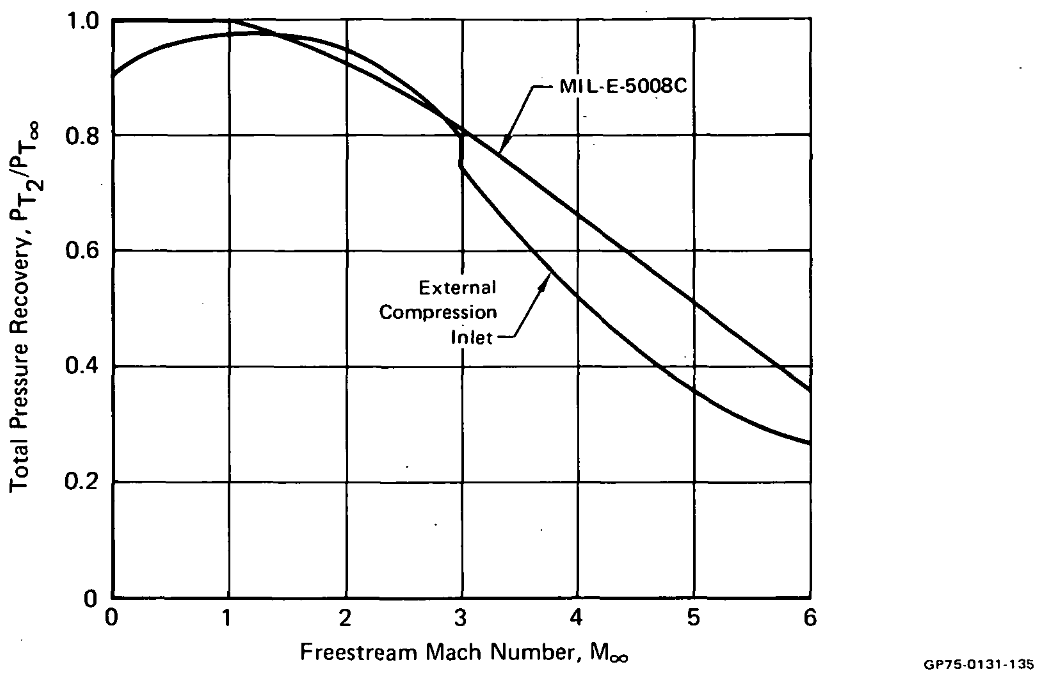


FIGURE 55
INLET TOTAL PRESSURE RECOVERY
Level Flight Angle of Attack

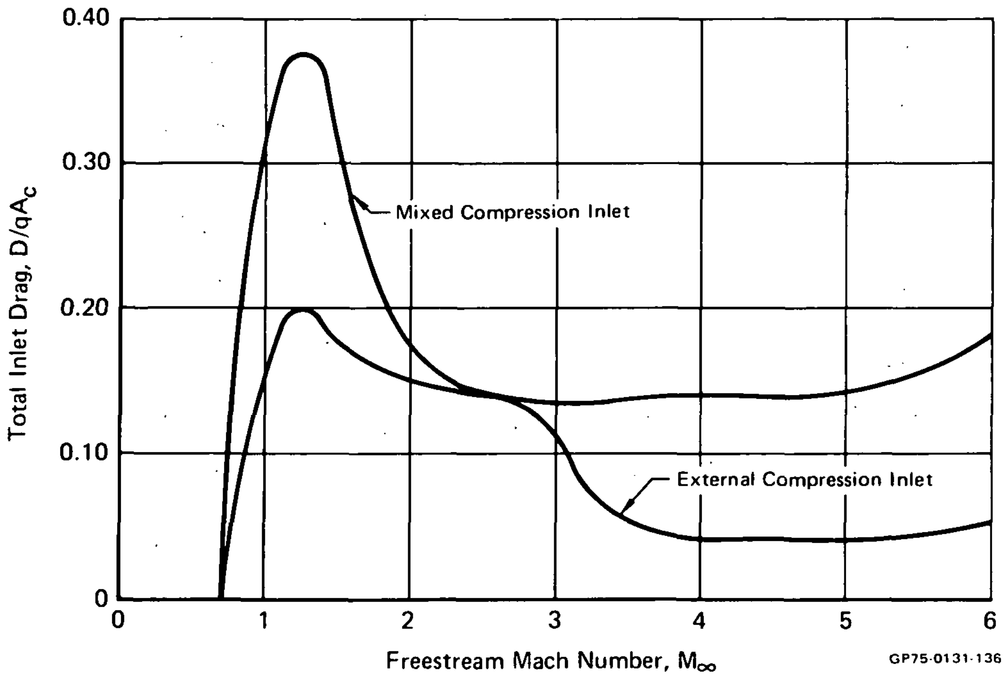


FIGURE 56
TOTAL INLET DRAG
Level Flight Angle of Attack

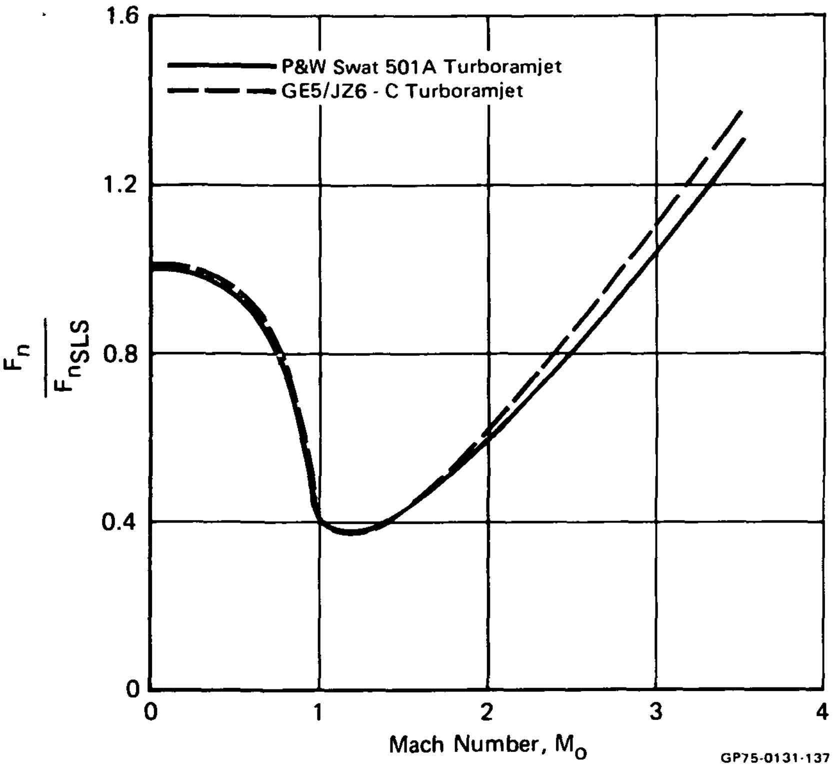


FIGURE 57

ACCELERATION THRUST COMPARISON TURBOJET MODE

Study Trajectory, Uninstalled Values, Maximum Values

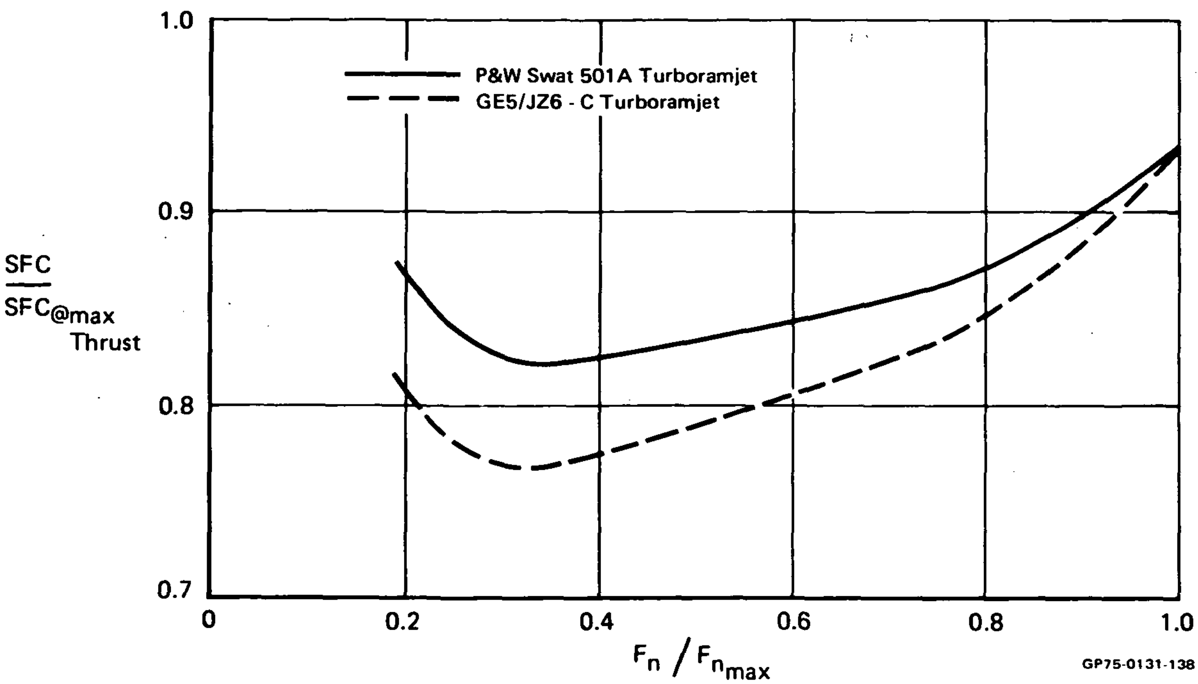


FIGURE 58

REDUCED POWER SFC COMPARISON

M_o = 0.95, 11.0 km (36,089 ft)

Uninstalled Values

7.3 WEIGHTS

The weights of the three configurations were analyzed to the same degree of detail so that consistent comparisons could be made. The weight analysis included evaluation in three categories:

- o Constant weight items
- o Non-tankage structure
- o Center fuselage tank structure

The constant weight items represented those components whose weight remained the same for all configurations. These were given minimal analysis and held constant so as not to impact the study results. Included in this group were propulsion-related items such as engines, gear boxes, and engine controls; systems such as hydraulics, electrical, and electronics; and useful load items including crew, payload, and miscellaneous residuals. In addition, two structural components, the landing gear and the air induction system were kept constant for this study.

The non-tankage structural items included the forward and aft fuselage, the wing, and vertical tail. The weights of these structural components were estimated weights responding to variations in the configuration geometry or wetted area. In all cases, these weights were determined by MCAIR estimation equations, with modifications to provide for the use of actively cooled panels.

The principal weights effort was focused on the center fuselage tank structure. This effort consisted of an initial weight estimate based on current MCAIR estimation techniques, followed by a refined detailed analysis in which each of the major components in the center fuselage was evaluated. These included the basic tank shell, domes, frames, actively cooled panels, longerons, bulkheads, tank support links, insulation, splices, access doors, and miscellaneous supports. In the refined analysis, each component was analyzed by using the detailed drawings presented in Section 6.

To insure consistency between configurations, the weight of Concept 3 was adjusted to account for the fact that the wing carry-through structure was included with the center fuselage tank structure.

Figure 59 presents the group weight statements.

	Concept 1		Concept 2		Concept 3	
	Mg	(lbm)	Mg	(lbm)	Mg	(lbm)
I Structure						
A. Fuselage						
1. Fwd	12.16	(26,800)	12.16	(26,800)	12.66	(27,900)
2. Center (Includes Fuel Tanks)	29.39	(64,800)	29.98	(66,100)	32.25	(71,100)
3. Aft	1.72	(3,800)	1.72	(3,800)	1.91	(4,200)
B. Remaining Structure	62.87	(138,600)	62.73	(138,300)	57.88	(127,600)
II Propulsion Group	27.76	(61,200)	27.76	(61,200)	27.76	(61,200)
III Systems						
A. Coolant Distribution System	15.15	(33,400)	15.20	(33,500)	13.74	(30,300)
B. Remaining Systems	15.42	(34,000)	15.42	(34,000)	15.37	(33,900)
IV Useful Load	25.67	(56,600)	25.67	(56,600)	25.67	(56,600)
V O.W.E.	190.14	(419,200)	190.64	(420,300)	187.24	(412,800)
VI Fuel	108.86	(240,000)	108.86	(240,000)	108.86	(240,000)
Usable	106.27	(234,300)	106.30	(234,400)	106.27	(234,300)
Boil-off	2.59	(5,700)	2.56	(5,600)	2.59	(5,700)
VII TOGW	299.0	(659,200)	299.5	(660,300)	296.1	(652,800)

GP75-0131-139

FIGURE 59
WEIGHT SUMMARY

7.4 PERFORMANCE CALCULATION

The MCAIR generalized mission performance program, KC6G, was used to compute vehicle performance. This is a Fortran IV program which operates on an IBM 360 computer. The program iterates on energy state to determine the time, fuel and distance required to travel from one energy state to another.

Input to the program consists of aerodynamic characteristics, propulsion system characteristics, the climb and descent paths, and the vehicle description. The aerodynamic characteristics consist of the zero lift drag (C_{D_O}), induced drag factor (L'), and lift curve slope (C_{L_α}). The propulsion system characteristics consist of net thrust and fuel flow versus Mach number and altitude, the climb and descent paths are input as Mach number versus altitude. The vehicle description consists of total planform area, takeoff gross weight, fuel weight, engine scale factor and fuel flow and a safety factor.

Climbs and descents are computed by first dividing the path into numerous segments. The program calculates the energy level at the end points of the first segment. The weight of the vehicle is known at the first point and the weight at the second point is estimated. Based on these weights the average specific excess power (Ps = (T-D) V/W) between the two points is computed. This provides the time required (t = ΔE/ΔPs), which is used to compute the distance, fuel used, and weight. The computed weight is compared to the estimated weight. If they agree within a small tolerance the next segment is computed; if not, the computed weight is used as an estimate for the next iteration.

The cruise calculation consists of determining the maximum range factor at the average cruise weight. This is accomplished by computing the range, factor at several cruise altitudes and searching for a maximum. The fuel flow at this point then determines the time and range during cruise.

Program output consists of time, fuel, and distance during climb and descent, and a mission summary consisting of the fuel expended and range obtained during each mission segment. Range sensitivity curves were developed by varying the OWE, TOGW and fuel weight and allowing the range to be a fall-out. These curves are presented in Figures 60 and 61.

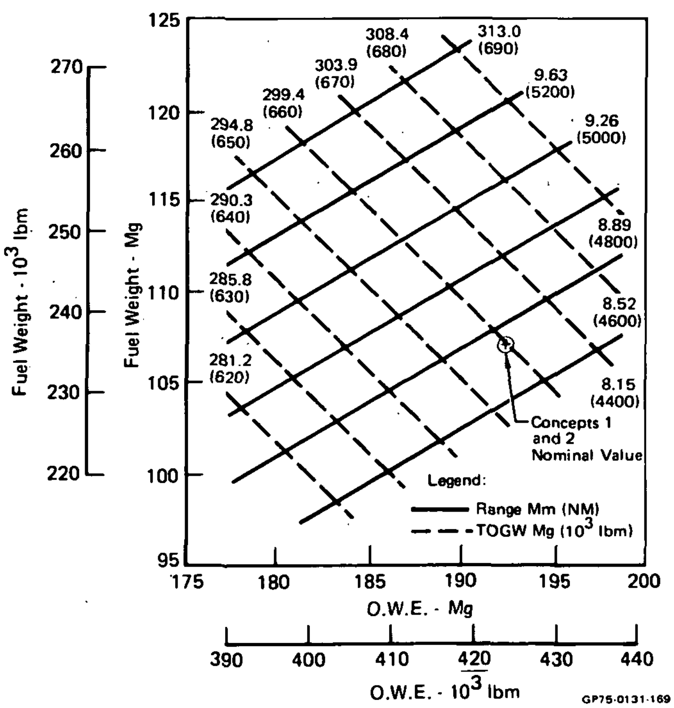


FIGURE 60
RANGE SENSITIVITY, CONCEPTS 1 AND 2

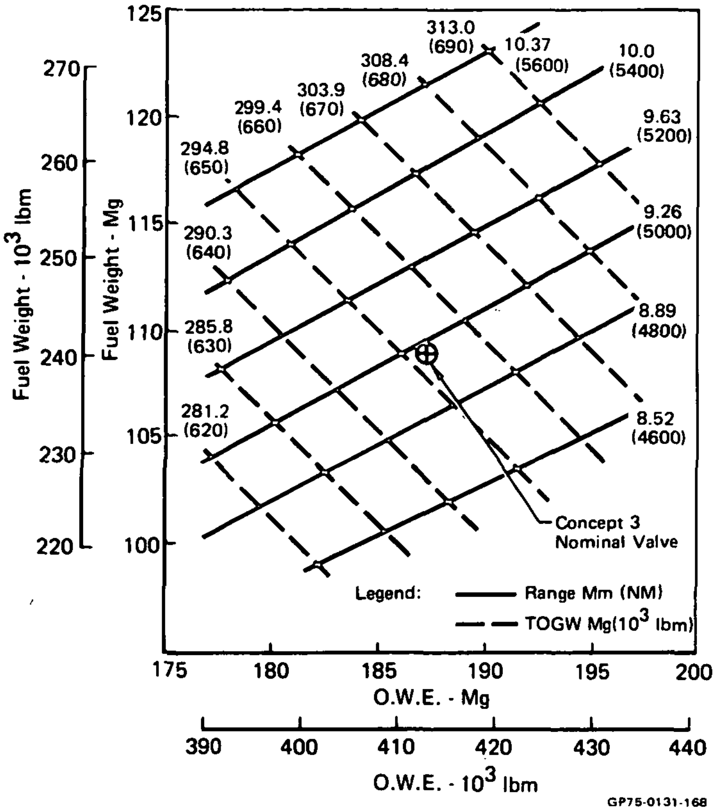


FIGURE 61
RANGE SENSITIVITY, CONCEPT 3

The effect of OWE on range can be assessed with these curves. They can also be used for assessing changes in fuel weight and deadweight if the 9% growth factor is accounted for. This growth factor was assigned to each aircraft to account for modification of structural components such as wings and landing gear. The curves are used in the following manner: (1) for deadweight changes multiply ΔOWE by 1.09. Enter the chart at new OWE; (2) for a change in total fuel weight multiply Δfuel weight by 0.09 to obtain the ΔOWE. Enter the chart with the new fuel weight and the new OWE; (3) for changes in both OWE and the total fuel weight perform steps 1 and 2 combined.

Range sensitivities for Concept 3 to variations in drag are presented in Figure 62. To generate the curve, the C_{D_O} and induced drag factor (L') was increased and decreased 10% at all Mach numbers. The configuration is nearly twice as sensitive to C_{D_O} as L' because C_{D_O} affects both the lift/drag ratio during cruise and the fuel required to climb. Whereas L' effects cruise and is only of secondary importance during climb because lower lift coefficients are used more during this mode than during cruise.

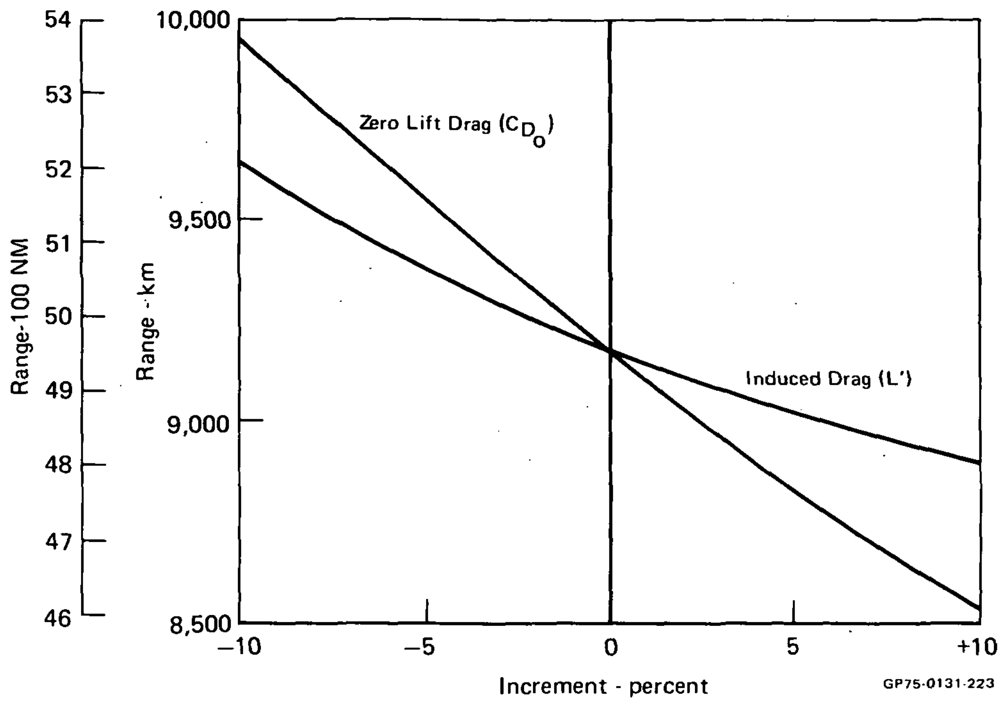


FIGURE 62
EFFECT OF DRAG VARIATION ON RANGE, CONCEPT 3

The effect of structural weight and cooling system weight on range can be assessed with Figures 63 and 64 respectively. These sensitivity curves were generated from the range sensitivity curves of Figures 60 and 61.

The airplane performance for Concept 3 is presented as a time history in terms of the Mach number and altitude in Figure 65.

Concepts 1 and 2 have similar time histories to Figure 65 except for the cruise time, which reveals the difference in range between all concepts. There are small differences in acceleration time and descent time, but these are less than one minute.

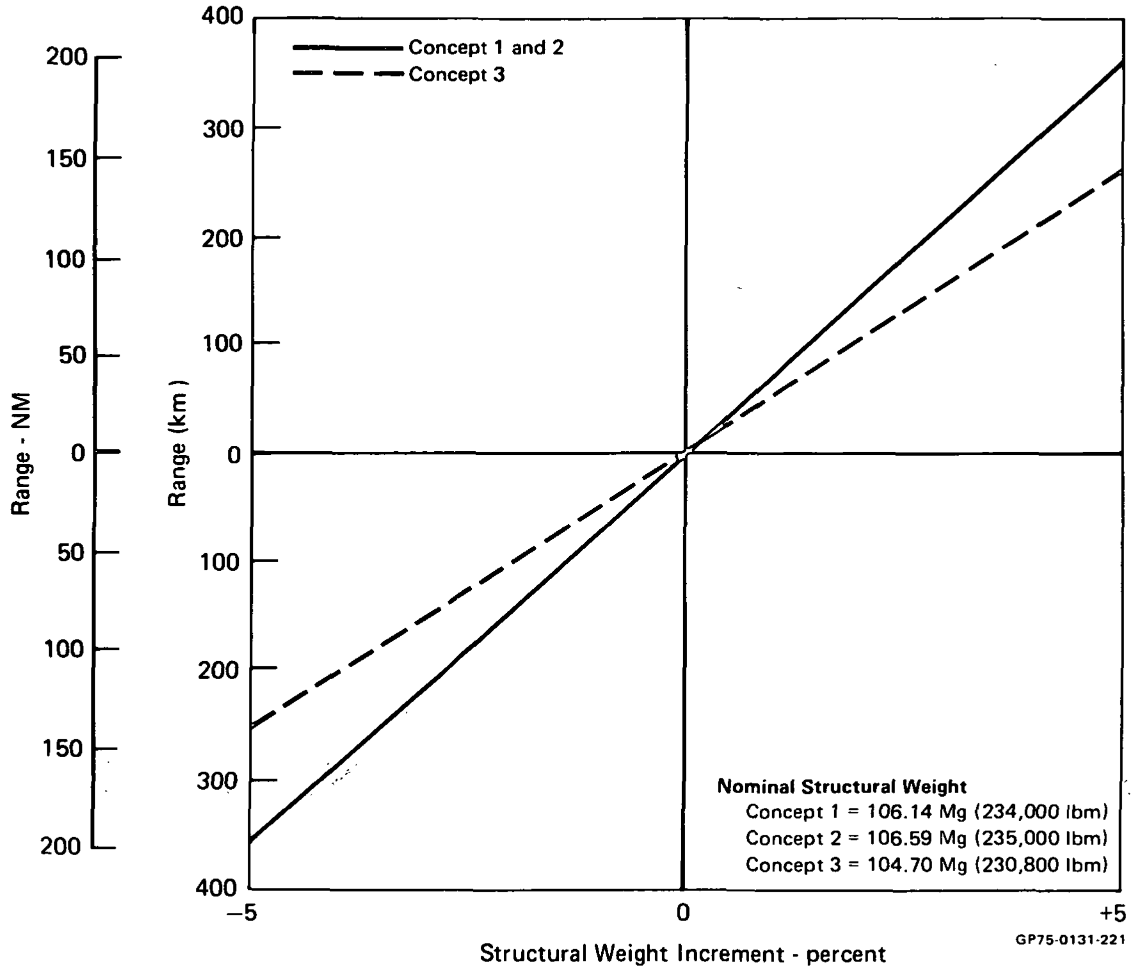


FIGURE 63
EFFECT OF STRUCTURAL WEIGHT ON RANGE

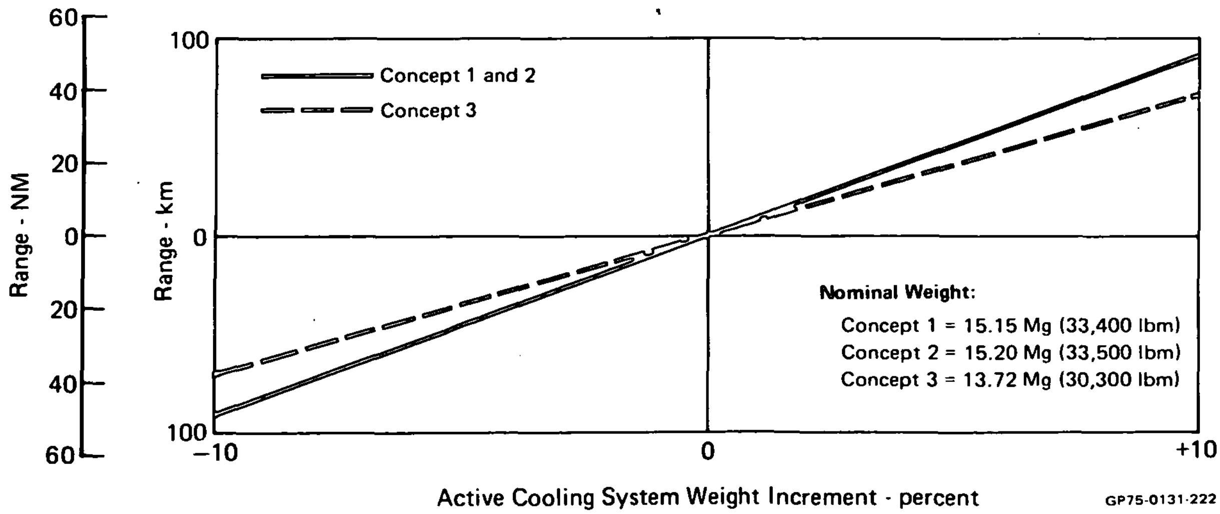


FIGURE 64
EFFECT OF COOLING SYSTEM WEIGHT ON RANGE

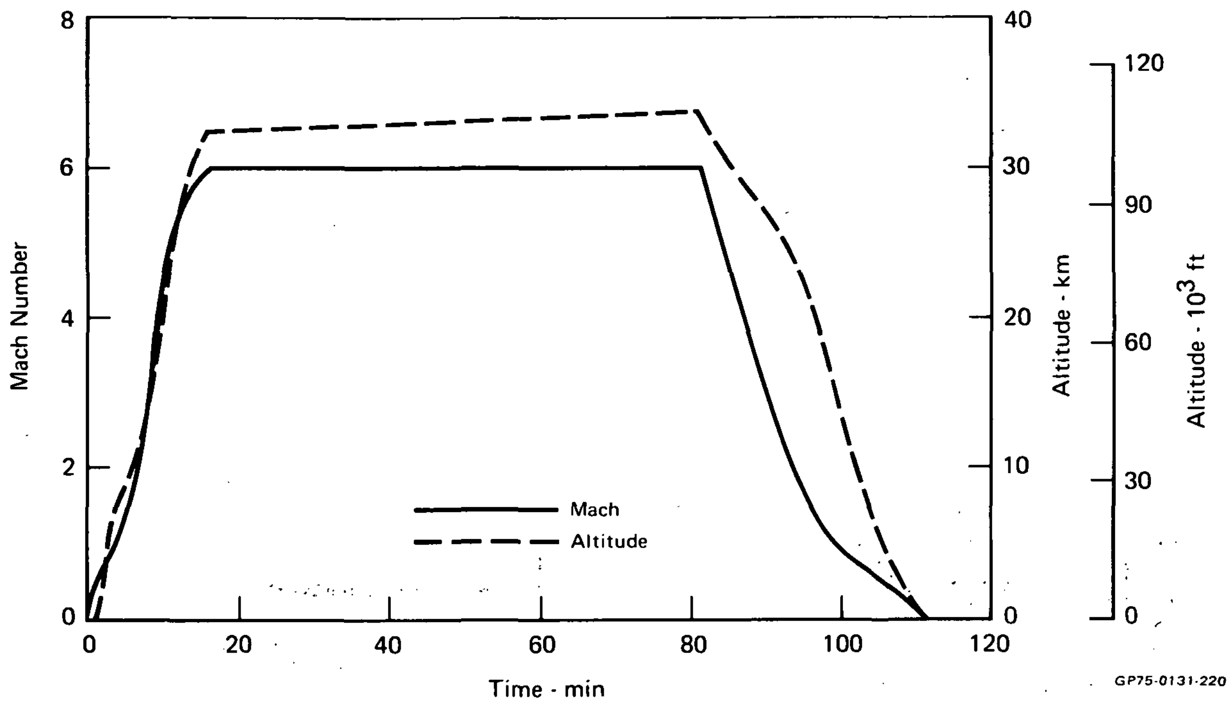


FIGURE 65
MACH NUMBER AND ALTITUDE vs TIME, CONCEPT 3

8. COMPARISON AND EVALUATION OF AIRCRAFT CONCEPTS

The study results show that integral fuel tanks combined with an elliptical-blended wing-body (Concept 3) results in the lightest weight and longest range configuration. This section presents a review of the predominant factors which influenced this conclusion.

The major factors affecting range are weight, volumetric efficiency and aerodynamic characteristics. These factors interact differently depending on the fuselage shape and type of tank structure.

Many pertinent elements driving the interactions were investigated in this study. For instance, it focused on two important structural technologies which are of concern to hypersonic vehicle designers: (1) actively cooled structures and thermal protection systems and (2) cryogenic tankage structural design. The analysis was generously supplemented with detailed configuration and structural layout design studies. The thermal protection system analysis addressed thermal insulation, minimum heating rate trajectories and fuel boil-off weight penalties. Structural design addressed detail tank construction, support and material. Configuration design highlighted the effect of tank size, shape and method of support on the total system. Numerous tradeoffs supported the design selections. Consequently, the selected designs can be confidently compared using parameters which will enable the reader to gain insight into the technical reasons subordinate to the final result.

As an additional aid in comparison and evaluation, producibility and serviceability analyses were conducted on each of the study vehicles. The purpose of these studies was to gain an insight into relative costs, both production and operating. These were qualitative in nature and are not comparable in depth to the technological analyses.

8.1 COMPARISON OF AIRCRAFT CONCEPTS

A summary of the weights, volumes and aerodynamic characteristics of each concept is presented in this section so that a ready comparison can be made.

a. Weight - The difference in major section weights of each concept is shown in Figure 66. Operating Weight Empty (OWE) is the best parameter to use comparing total system weights because it does not include the fuel quantity which was held constant at 108.86 Mg (240,000 lbm).

	Concept 1	Concept 2	Concept 3
TOGW Mg (lbm)	299.03 (659,200)	299.50 (660,300)	296.10 (652,800)
O.W.E. Mg (lbm)	190.14 (419,200)	190.64 (420,300)	187.24 (412,800)
Structural Weight Mg (lbm)	106.14 (234,000)	106.59 ((235,000)	104.70 (230,800)
Fwd Fuselage Mg (lbm)	12.16 (26,800)	12.16 (26,800)	12.66 (27,900)
Center Fuselage Mg (lbm)	29.39 (64,800)	29.98 (66,100)	32.25 (71,100)
Tank Mg (lbm)	7.12 (15,700)	11.03 (24,300)	14.51 (32,000)
Actively Cooled Panels, Insulation and Supports Mg (lbm)	22.27 (49,100)	18.96 ((41,800)	17.74 (39,100)
Aft Fuselage Mg (lbm)	1.72 (3,800)	1.72 (3,800)	1.91 (4,200)
Remaining Structures Mg (lbm)	62.87 (138,600)	62.72 (138,300)	57.88 (127,600)
Active Cooling System Mg (lbm)	15.15 (33,400)	15.20 (33,500)	13.72 (30,300)
Aircraft Density $\left(\frac{\text{O.W.E.}}{\text{Total Aircraft Volume}}\right)$ kg/m3 (lbm/ft^3)	44.24 (2.76)	46.00 (2.87)	53.37 (3.33)
Structural Weight Fraction $\left(\frac{\text{Structural Weight}}{\text{TOGW}}\right)$	0.355	0.356	0.354

GP75-0131-208

FIGURE 66
WEIGHT COMPARISON

As shown, the OWE of the integral tank arrangement (Concept 2) is only slightly greater than that of the non-integral tank of similar dee cross section (Concept 1). However, there is a 1.75% decrease in OWE for the integral tank with the elliptical cross section (Concept 3) compared to Concept 2. The effect of OWE on range can be assessed using the range sensitivity curves Figures 60 and 61. The reduced OWE of Concept 3 from Concept 2 results in a 205.6 km (111 NM) increase in range as taken from the sensitivity curves whereas the small difference between Concepts 1 and 2 has a negligible effect. It is noteworthy that three systems which differ so widely in structural concept have nearly the same structural weight fraction. For the Concepts 1 and 2 this is because of the compensating effect of active cooling system weight on the tank weight. The fundamental weight difference between the Concepts 1 and 2 exists in area of the center fuselage section as revealed in Figure 66. Under this center fuselage category, the smaller Concept 1 tank weight is practically compensated for by the increase in the Concept 2 fuselage cover structural weight which consists of actively cooled panels, insulation, and supports. (Note that Concepts 1 and 2 tank weights are only 7.6% and 10% of the structural weight respectively.) The Concept 3 tank structural weight is, however, considerably higher than either Concepts 1 or 2 (13.8% of the Concept 3 structural weight). But since the aircraft is smaller (for example, Concept 3 is 10% smaller in wing area as compared to Concepts 1 and 2), the non-tankage structural weights as shown in Figure 66 are all relatively less than either Concepts 1 or 2. Thus, the Concept 3 structural weight fraction is nearly equal to Concepts 1 or 2 where the increase of tank weight is offset by the smaller aircraft size resulting in less overall aircraft weight.

The integral bubble tank of Concept 3 weighs almost 24% more than Concept 2. It not only carries primary fuselage loads but acts as the wing carry through member. The greater tank weight is partly compensated by a lower active cooled panel weight for the center fuselage. These panels are non-structural versus the semi-structural panels of Concept 2. The lower panel weight is also due to the smaller surface area covered by panels on Concept 3.

b. Volume - The volumetric effiency is the most revealing parameter in evaluating the differences in the concepts. A summary of concept volumes is presented in Figure 67.

The summary indicates that the integral tanks use the fuselage volume more efficiently than non-integral tanks. The center fuselage volume needed to contain 108.86 Mg (240,000 lbm) of fuel, is 6.2% less for Concept 2 and 24% less for Concept 3 than Concept 1 respectively.

The greater volumetric efficiency of the integral tank results in a smaller sized vehicle. The wetted areas are indicative of the differences. Concept 2 fuselage is slightly smaller than Concept 1 whereas Concept 3 is dramatically smaller. The total wetted area of Concept 3 is 16% less than Concept 1.

c. Aerodynamic Characteristics Comparison - The aerodynamic characteristics which have the most influence on performance are compared in Figure 68. Of these, the lift to drag ratio (L/D) is the most influencing parameter on range, and is highest for Concept 3. The best correlating parameter for hypersonic L/D is the volume parameter $V^{2/3}/Sp$. The lowest value of the volume parameter indicating the best hypersonic cruise performance.

Concept 3 is also the most efficient during other phases of the flight profile. Its low value of zero lift drag ($C_{D_o}S$) is indicative of the best

Figure of Merit	Concept 1 m^3 (ft^3)	Concept 2 m^3 (ft^3)	Concept 3 m^3 (ft^3)
Passenger Volume	333.9 (11,800)	339.6 (12,000)	288.7 (10,200)
Passenger + Baggage and Cargo Volume	489.6 (15,500)	481.1 (17,000)	455.6 (15,900)
Forward Fuselage	682.0 (24,100)	670.7 (23,700)	622.6 (22,000)
Center Fuselage Volume	2422.5 (85,600)	2272.5 (80,300)	1842.3 (65,100)
Aft Fuselage Volume	107.5 (3,800)	107.5 (3,800)	107.5 (3,800)
Nacelle Volume	467.0 (16,500)	467.0 (16,500)	489.6 (17,300)
Total Fuselage Volume	3212.1 (113,500)	3050.7 (107,800)	2572.5 (90,900)
Payload Efficiency $\left(\frac{\text{Passenger + Cargo Volume}}{\text{Total Fuselage Volume}}\right)$ x 10^2	15.2%	15.8%	17.7%
Volumetric Efficiency $\left(\frac{\text{Fuel Volume}}{\text{Center Fuselage Volume}}\right)$ x 10^2	67%	71%	88%

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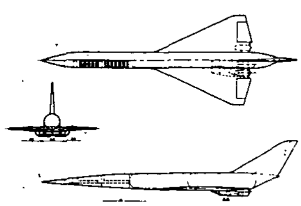
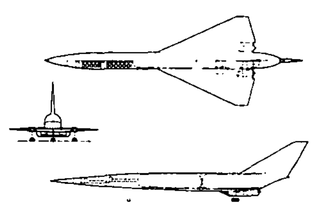
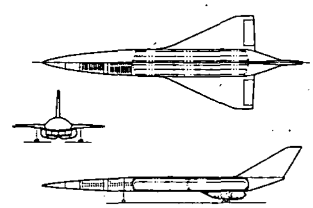
FIGURE 67
VOLUME COMPARISON

Pertinent Aerodynamic Parameters	Concept 1  Discrete Wing Body	Concept 2  Discrete Wing Body	Concept 3  Blended Wing Body
Range - Mm (NM)	8.69 (4,690)	8.73 (4,715)	9.20 (4,968)
Fineness Ratio, ℓ/d	13.45	14.0	13.1
$V^{2/3} \div S_p$	0.178	0.176	0.163
b^2/S_{wet}	0.357	0.363	0.387
L/D	4.6	4.6	4.8
$C_{D_0}S$ m^2 (ft^2)	9.82 (105.73)	9.55 (102.85)	9.24 (99.41)
km/$kg_{fuel_{cruise}}$ (NM/lbm) x 10^3	115.9 (28.4)	116.4 (28.5)	119.6 (29.3)

GP75-0131-205

FIGURE 68
AERODYNAMIC CHARACTERISTICS SUMMARY

acceleration efficiency whereas the highest value for the ratio of span squared to reference wetted area (b^2/S_{wet}) indicates it has the best subsonic cruise and loiter performance.

These superior aerodynamic characteristics of Concept 3 can be attributed to the excellent volumetric efficiency. Increased volume utilization permits a smaller vehicle size which results in lower friction drag. Friction drag is a significant factor on actively cooled aircraft because of the external wall cooling and the greater exterior surface roughness which results in relatively higher friction drag.

Concept 3 also benefits from wing-body blending. Experimental data indicates that blending reduces adverse interference effects often exhibited by low-wing designs. Also, the low side profile and flattened lateral shape reduce the destabilizing forebody inputs thereby reducing the vertical fin size. The fineness ratio of Concept 3 is somewhat smaller, however, the significance of this parameter in establishing wave drag levels is not clearly defined for blended shapes.

8.2 EVALUATION OF AIRCRAFT CONCEPTS

Aircraft concepts are normally evaluated in terms of several parameters. Among these are payload and range which are used to assess an aircraft's relative ability to accomplish specific missions. Other factors are operating weight empty and takeoff gross weight, considered as indicators of initial investment and operational costs. Aircraft costs are also affected by considerations such as development and testing, producibility and serviceability.

This study specifically addressed those factors which affected range, which was identified as the primary figure of merit. Payload and fuel weight were held constant so as not to affect the study results. Producibility and serviceability factors were developed to enhance the economic evaluation of the aircraft concepts. However, due to lack of depth in the development of these factors, additional study will be required to determine their real significance.

A comparison of the major evaluation factors is presented in Figure 69.

	CONCEPT			
Factor	1	2	3	3a*
Range Mm (NM)	8.69 (4690)	8.73 (4715)	9.20 (4968)	9.10 (4905)
Volumetric Efficiency %	67	71	88	88
Producibility	1	3.5	3.0	1.88
Serviceability	1	1.2	1.3	-

*Alternate tank study using plain skin monocoque tanks.

FIGURE 69
EVALUATION SUMMARY

Range - This factor is the figure of merit for the study. The integral tank, blended body, Concept 3 aircraft is definitely superior. The major contributors to its superiority are its greater aerodynamic performance, smaller size, and lower weight.

Volumetric Efficiency - The study used constant fuel volume for all three study aircraft. A major factor, then, in aircraft size and resulting weight is the efficiency with which that fuel can be packaged. Figure 69, above, which measures volumetric efficiency as the ratio of fuel volume to center fuselage volume shows the multi-bubble tank/elliptical fuselage combination is definitely the most efficient.

Producibility - Contribution of the producibility factor to the cost of owning and operating the study aircraft is unknown. These factors are merely estimated ratios of the production cost of each aircraft to that of Concept 1. Although it was unexpected the integrally machined stiffening of the Concept 2 and 3 tanks and provisions for thermal strain relief drove these factors to 3.5 and 3, respectively, for Concepts 2 and 3. Concept 1, with its monocoque tanks, proved to be the least expensive production aircraft by a wide margin.

Serviceability - These factors are a measure of the relative difficulty of performing inspection, maintenance, repair and service tasks. The improved volumetric efficiency and thermal strain relief provisions proved to be the undoing of the integral tank aircraft when these factors were estimated. Limited access made them the least desirable. The effect of these factors on operating the aircraft, however, was not assessed.

9. CONCLUSIONS AND RECOMMENDED AREAS FOR FUTURE INVESTIGATION

Several conclusions were drawn from the design integration studies. Two of these conclusions are considered by MCAIR to have a significant effect on future design of hypersonic cruise vehicles. These form the basis for recommended future studies.

9.1 CONCLUSIONS

o The structural arrangement (Integral Vs Non-Integral Tanks) had a negligible effect on structural weight. This is exemplified by the fact the structural fraction of the total airplane (structural weight/TOGW) is essentially constant for the three concepts.

o The greater volumetric efficiency of integral tanks helps compensate for increased tank weight by reducing the wetted area of the using concept and, consequently, reducing active cooling system weight. The tank concept which makes maximum use of cross sectional area will provide the smallest integrated configuration. The smaller the size the more favorable the performance for a given fuel volume. The bubble tank of Concept 3 had excellent cross sectional utilization and consequently superior range performance.

o Integral cryogenic tanks with external insulation require extensive means of compensating for thermal expansion while at the same time reacting structural loads. The result is an increase in complexity from a producibility aspect (addition links, fittings, welding) leading to higher cost than a tank which does not react primary fuselage loads.

o The nacelle module required a disproportionate amount of weight and complexity to provide active cooling protection, even with unlimited heat sink capacity. The interface problem between the nacelle and fuselage is the driving problem which must be addressed in more detail, whether designed with hot structure or heat protected to use lower temperature materials.

9.2 RECOMMENDED AREAS FOR FUTURE INVESTIGATION

Non-Integral Tank Design - The cost advantage of monocoque non-integral tanks is extremely attractive. When combined with the inherent aerodynamic performance superiority of the blended wing-body configuration it could be superior to any of the three concepts studied. It is recommended that a study be conducted of a blended wing-body concept with non-integral tanks under the same ground rules and criteria used for this study supplemented with an in-depth economic analysis.

Nacelle Hot Structure Design - A study of the interface between the engine nacelle module and the fuselage is also suggested. As discussed in Section 9.1 of Reference (1), numerous thermo/structural considerations are involved which require definition before a practical design can be derived.

10. REFERENCES

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