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

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16. Abstract

A detailed analytical study was made to investigate the effects of fuselage cross section (circular and elliptical) and the structural arrangement (integral and non-integral tanks) on the performance of an actively cooled hypersonic cruise vehicle. The vehicle was a 200 passenger, liquid hydrogen fueled Mach 6 transport designed to meet a range goal of 9.26 Mm (5000 NM). A variety of trade studies were conducted in the area of configuration arrangement, structural design, and active cooling design in order to maximize the performance of each of three point design aircraft, circular wing-body with non-integral tanks, circular wing-body with integral tanks and elliptical blended wing-body with integral tanks. Aircraft range and weight were used as the basis for comparison. The resultant design and performance characteristics showed that the blended body integral tank aircraft weighed the least and had the greatest range capability, however, producibility and maintainability factors favor non-integral tank concepts.

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FOREWORD

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.

Detailed results are given in the following reports:

NASA CR-132668	Aircraft Design Evaluation
NASA CR-132669	Active Cooling System Analysis
NASA CR-132670	Structural Analysis.

The primary contributor to the contents of this volume was C. J. Pirrello. Assistance was provided by A. H. Baker and J. E. Stone.

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF FIGURES
(Continued)

LIST OF FIGURES
(Continued)

LIST OF ABBREVIATIONS AND SYMBOLS

Symbol	Definition
Btu	British Thermal Unit
CG	Center of Gravity
D	Diameter, Drag Force
DW	Design Weight
da/dn	Crack Growth Rate
E_c	Young's Modulus, Compression
°F	Degrees Fahrenheit
F_{cy}	Yield Compressive Strength
F_{TU}	Ultimate Tensile Strength
F_{TY}	Yield Tensile Strength
ft	Feet
g	Grams, Acceleration due to gravity
GH_2	Gaseous Hydrogen
H_2	Hydrogen
hr	Hour
in	Inch
Isp	Specific Impulse
K	Kelvin
K_c, K_{Ic}	Critical Stress Intensity Factors
ΔK	Stress Intensity Factor
L	Length, Lift Force
lbf	Pounds Force
lbm	Pounds Mass
LH_2	Liquid Hydrogen
m	Meter
$\dot{m}$	Coolant flowrate
M	Mach number
N	Newton, Running Load
n	Load Factor
N_2	Nitrogen

LIST OF ABBREVIATIONS AND SYMBOLS (Continued)

Symbol	Definition
NM	Nautical Mile
O.W.E.	Operating Weight, Empty
Pa	Pascal
psf	Pounds force per square foot
psi	Pounds force per square inch
$\dot{Q}$	Integrated heating rate = $\int \dot{q}dA$
$\dot{q}$	Heating rate per unit area
S	Theoretical wing area
s, sec	Second
T	Temperature, Thrust
t	Thickness
$\bar{t}$	Equivalent weight thickness
TOGW	Takeoff Gross Weight
TPS	Thermal Protection System
W	Watt, Weight
ρ	Density
Prefixes	
c	Centi (10^{-2})
k	Kilo (10^{3})
m	Milli (10^{-3})
M	Mega (10^{6})
Δ	Difference
Subscripts	
CR	Critical
F	Fuel
max	Maximum
S	Structural
SLS	Sea Level Static
x	Aircraft Axis - Longitudinal
y	Aircraft Axis - Lateral
z	Aircraft Axis - Vertical

SECTION 1
SUMMARY

A detailed analytical study was made to investigate the effects of fuselage cross section (circular and elliptical) and structural arrangement (integral and non-integral tanks) on the performance of an actively cooled hypersonic cruise vehicle. The vehicle was a 200 passenger, liquid hydrogen fueled Mach 6 transport designed to meet a range goal of 9.26 Mm (5000 NM).

Three specific point design aircraft, illustrated in Figure 1, were developed. The aerodynamic configurations were derived from the NASA HT-4 tailless delta aircraft configuration described in Reference (1).

A variety of trade studies were conducted in the area of configuration arrangement, structural design, and active cooling system design in order to maximize the performance of each of the three point design aircraft. Aircraft range and weight were used as the bases for comparison and the assumption was made that adequate liquid hydrogen was available to cool the aerodynamic surfaces of the aircraft.

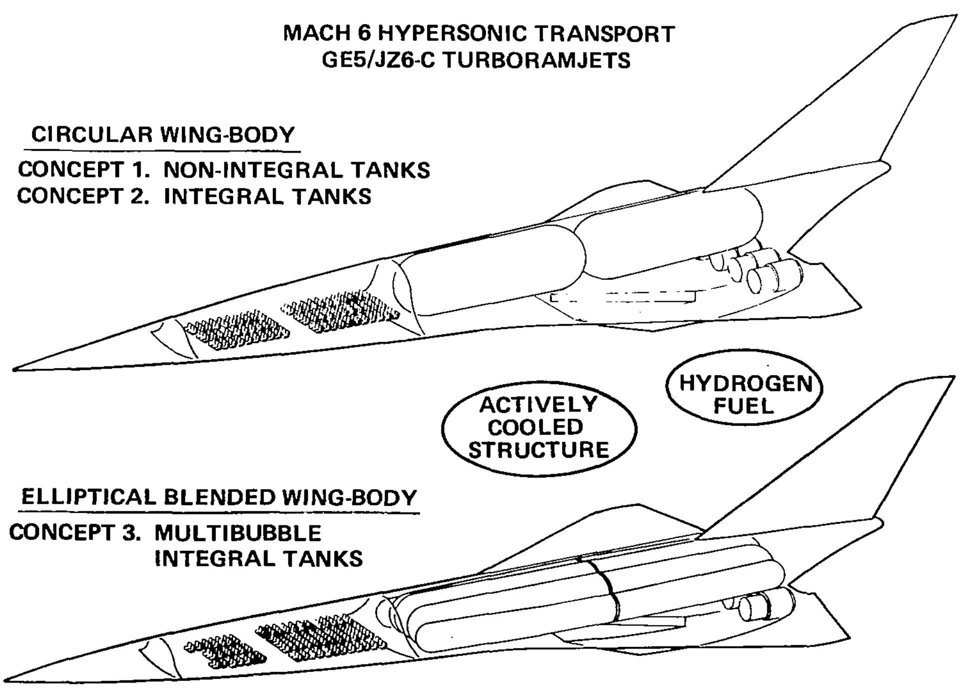
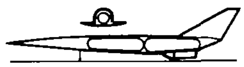
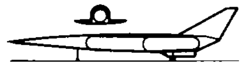


FIGURE 1
FUSELAGE/TANK STRUCTURE STUDY AIRCRAFT CONCEPTS

The resultant design and performance characteristics are summarized in Figure 2. Concept 3, the blended body, integral tank aircraft, weighed the least and had the greatest range capability (over 0.47 Mm (250 NM) more than the others). This superior performance is a result of the better aerodynamic characteristics and higher volumetric efficiency.

	CONCEPT 1 CIRCULAR  (360.5 FT) 109.9 m	CONCEPT 2 CIRCULAR  (360.5 FT) 109.9 m	CONCEPT 3 BLENDED BODY  (328.5 FT) 100.1 m
TANKAGE	NON – INTEGRAL	INTEGRAL	INTEGRAL
TANK WALL	MONOCOQUE	ISOGRID	ISOGRID
L/D	4.6	4.6	4.8
VOLUMETRIC EFFICIENCY	0.67	0.71	0.88
O.W.E. Mg (LBM)	190.2 (419,200)	190.6 (420,300)	187.2 (412,800)
TOGW Mg (LBM)	299.0 (659,200)	299.5 (660,300)	296.1 (652,800)
RANGE Mm (NM)	8.69 (4,690)	8.73 (4,715)	9.20 (4,968)

Note: L/D Basis - Nacelle on and Cool Wall
Vol eff = Tank vol/available volume in fuselage tank section

FIGURE 2
DESIGN AND PERFORMANCE CHARACTERISTICS

The relative producibility and serviceability of each of the three aircraft concepts when optimized for maximum range were assessed to provide an indication of cost trends. This analysis was not as detailed as the design studies, but the relative ranking of these factors is believed to be accurate. These factors are given below, with higher numbers indicating higher cost or greater maintenance requirements. Concept 1 is used as the baseline for comparison.

	Concept 1	Concept 2	Concept 3
Tank Wall	Monocoque	Isogrid Stiffened	Isogrid Stiffened
Producibility	1	3.5	3
Serviceability	1	1.2	1.3

Design approaches which would improve the producibility of the Concept 2 and 3 aircraft were briefly examined. All these approaches resulted in increased weight and therefore shorter range. The most attractive of these approaches was to use monocoque tank wall construction as in Concept 1. The effect of this approach, and the associated decrease in the range of each aircraft concept is presented below.

	Concept 1	Concept 2	Concept 3
Tank Wall	Monocoque	Monocoque	Monocoque
Producibility	1	1.6	1.8
Range Loss	0	452 km (244 NM)	117 km (63 NM)

SECTION 2

INTRODUCTION

Liquid hydrogen (LH_2) fuel has several potential advantages over conventional fuels when applied to hypersonic flight. A particularly attractive characteristic is its high specific impulse. Additionally, it offers a significant heat sink capacity for engine and inlet cooling, and also a potential cooling of the airframe itself.

However, a significant challenge for the aircraft designer is presented by liquid hydrogen's cryogenic storage temperature, 20.3 K (-423°F), and its extremely low density (approximately 1/12 that of conventional JP aircraft fuel). These factors, particularly the low density, result in a unique design problem. The large volumes required for fuel containment dictate a requirement for aircraft shapes which provide high volume per total aircraft surface area. Typically, blended-body and all-body shapes are attractive candidates. These shapes, combined with structural concepts in which fuel containment is integral with the basic structure generally lead to high volumetric utilization.

The specific major issues in this study were to evaluate the effects of fuselage cross section (circular and elliptical) and structural arrangement (integral and non-integral tanks) on the performance of a representative Mach 6 hydrogen fueled aircraft. The entire external surface of the aircraft was assumed to be actively cooled. In-depth studies were conducted on the design of the configuration and the active cooling system. Detailed strength analyses included an evaluation of the impact of fracture mechanics and fatigue on the design of the cryogenic tank structures. These analyses emphasized the development of minimum weight, long life structural concepts. They further provided a firm basis for calculating the weight of the detail elements of the fuselage/tank structure in lieu of using statistical weight estimates. Configuration studies focused on approaches that would maximize volume utilization, a requirement well recognized as a dominant issue.

The study was conducted with fuel weight and payload being fixed at 108.9 Mg (240,000 lbm) and 21.8 Mg (48,000 lbm), respectively. Aircraft size, weight, and range were dependent variables. Range was selected as a viable figure of merit.

The study logic, phasing and interactions are shown in Figure 3. The study was conducted in the sequence indicated, with the non-integral tank Concept 1 design being accomplished first, followed by analysis of the integral tank Concepts 2 and 3.

The design and performance characteristics of the baseline (or preliminary) configurations were developed and used as the basis for the aircraft trade-off and design refinement process.

General arrangement, structural design, active cooling system design and mission profile trade studies were accomplished for Concept 1 with fixed vehicle payload and a mission range goal of 9.26 Mm (5,000 NM). Using the results of these trade studies, the final sizing of the aircraft was accomplished and the fuel required to perform the mission was determined. This was set at 108.9 Mg (240,000 lbm).

For Concepts 2 and 3 the payload and the fuel weight (as determined from Concept 1) were held fixed and aircraft range was the variable. Sufficient hydrogen fuel was assumed to cool the aircraft aerodynamic surfaces. For all concepts, structural weights were determined as well as the take-off gross weight (TOGW). Prior to proceeding with the design refinement process for Concepts 2 and 3 a trade-off of candidate integral fuel tank thermal protection systems was accomplished. For all three aircraft concepts, refined design and performance characteristics were developed, and range sensitivity to various parameters was determined. Finally, the characteristics of the three concepts were compared and evaluated and conclusions were drawn.

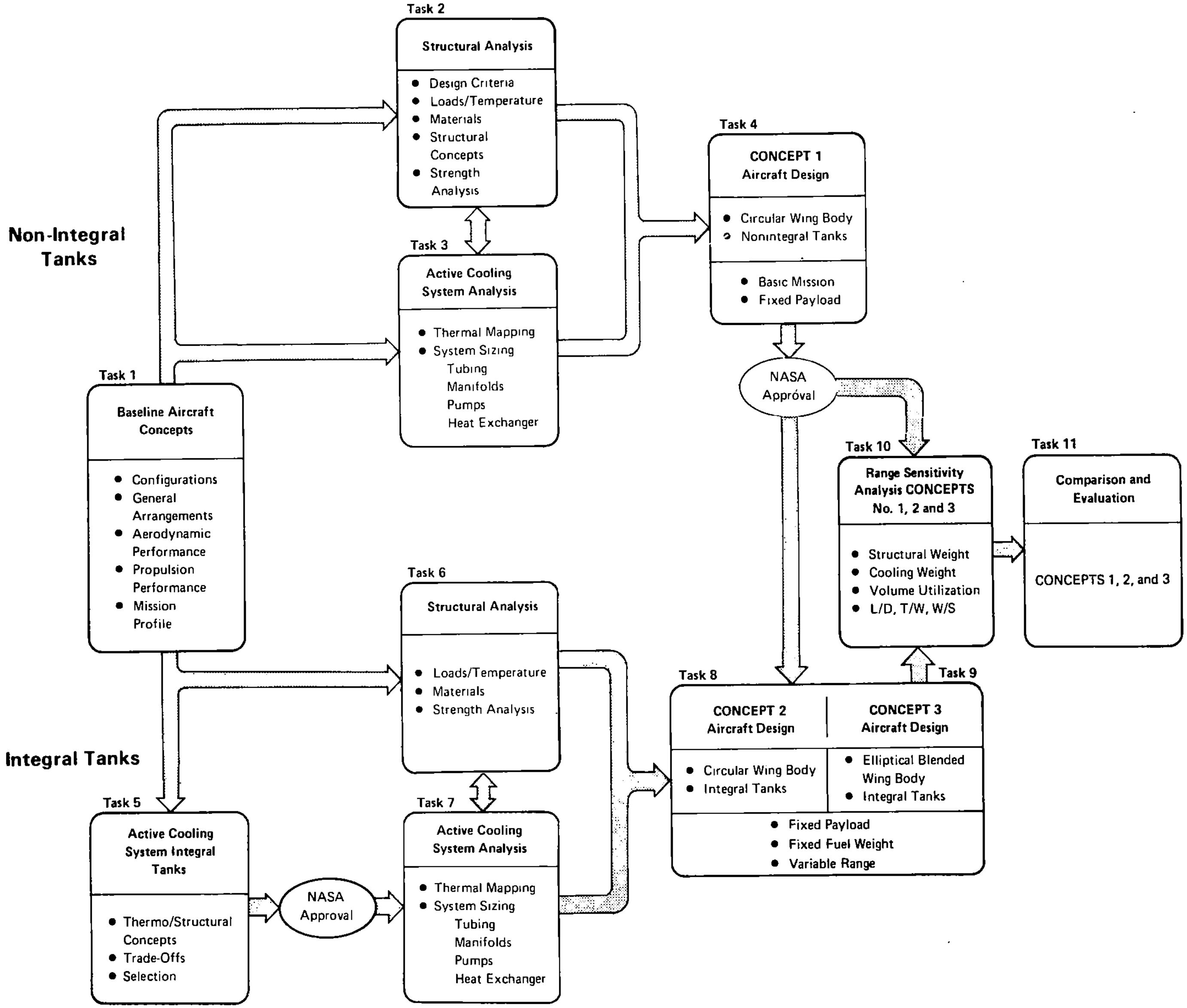


FIGURE 3
STUDY PLAN

SECTION 3

PERFORMANCE AND DESIGN REQUIREMENTS

Performance analyses for the three study aircraft concepts were accomplished using the flight profile, given in Figure 4, for the nominal design mission range of 9.26 Mm (5000 NM).

The ascent flight profile for each aircraft was constrained by sonic boom overpressure, dynamic pressure, inlet duct pressure and heating rate limits. Ascent along these limit lines was accomplished to reach the best (L/D) (I_{sp}) condition at start of Mach 6 cruise. From this point cruise was continued at the best (L/D) (I_{sp}) until start of descent. Descent was then accomplished at Max L/D.

Sufficient fuel reserves to allow loiter for 20 minutes at M = 0.8 and 12.2 km (40,000 ft) altitude were provided. Additional sea level reserve is sufficient for one go around (5 minutes at M = 0.4).

The mission performance requirements common to all three concepts, are:

1. Cruise Mach number = 6.
2. Payload = 21.8 Mg (48,000 lbm).
3. Fuel weight = 108.9 Mg (240,000 lbm) (determined from Concept 1 design synthesis) including allowance for boiloff during a preflight ground hold of one hour.

The propulsion systems were sized to meet the performance requirements for each aircraft concept. The "rubberized" engines were derived from the hydrogen burning GE5-JZ6 wrap around stoichiometric turboramjet. A new MCAIR approach to two dimensional, horizontal ramp, external compression air induction systems was used for all aircraft.

Design requirements, guidelines and assumptions common to all aircraft were:

1. All external surfaces cooled to a maximum structural temperature of 394K (250°F) except the nacelle. An unlimited fuel heat sink was assumed available to absorb the entire heat load.
2. Design Load Factors:

(a) Flight n_z = 2.5, -1.0 (limit)

(b) Taxi n_z = 2.0 (limit)

(c) Emergency landing n_z = 4.5, -2.0, n_x = 9.0, n_y = $\pm$1.5 (Ultimate)

3. Tank pressurization = 138 kPa (20 psi) gage limit.
4. Service Life = 10,000 hrs.
5. Fuel was LH_2 contained at its normal boiling point of 20.3 K (-423°F).
6. Space between structure and fuel tanks was purged to 3.45 kPa (0.5 psi) gage.
7. $(\text{Volume})^{2/3}/(\text{Planform Area})$ was to be approximately the same as the HT-4.
8. Structural material was to be primarily aluminum.
9. Airframe surface heating rates were based on sustained flight conditions. No allowance was made for flight maneuvers.
10. Designs were based on tank pressure stabilizing the structure.

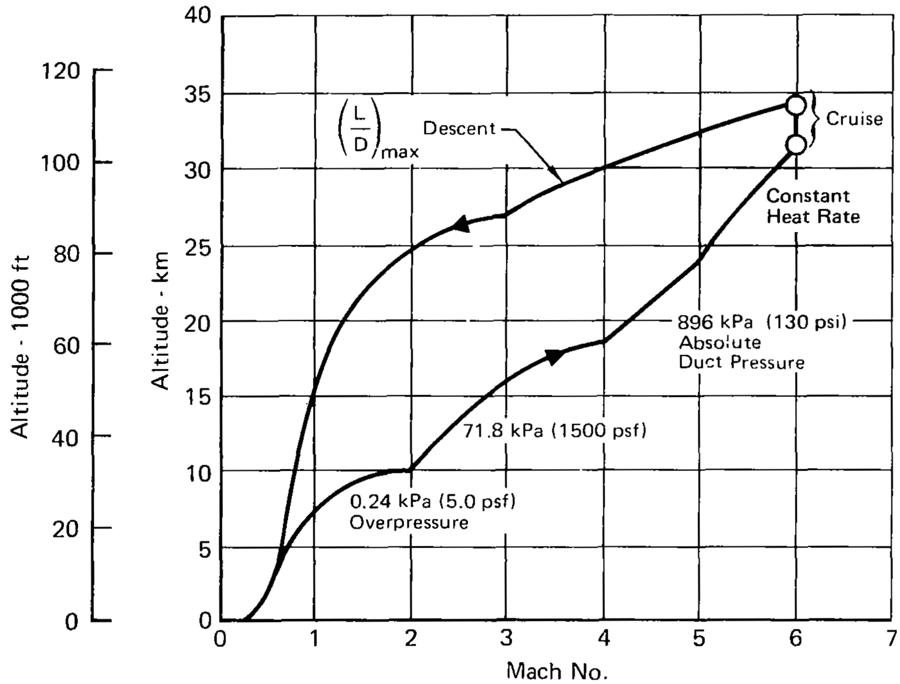


FIGURE 4
MISSION TRAJECTORY

SECTION 4

CIRCULAR BODY AIRCRAFT WITH NON-INTEGRAL TANKS (CONCEPT 1) SYNTHESIS

The final performance characteristics for Concept 1 were previously summarized in Figure 2. These characteristics were determined as a result of a series of trade studies and detailed structural and cooling systems analyses of the aircraft. The following sections summarize the refinement studies.

Material Selection

A study was conducted to select the materials to be used for primary and secondary structure. Primary structure was limited to the selection of an aluminum alloy. The material selection was based primarily on Concept 1 requirements. The selection was maintained for all concepts so that material properties would not be a variable in the final comparison and evaluation. That assumption was reviewed as the design of Concepts 2 and 3 progressed. No special design requirements were found which would make the original selection invalid.

Two considerations were paramount in selection of the aluminum alloys to be used. Those were the 394K (250°F) maximum temperature to which the moldline structure was to be cooled and the 20.3K (-423°F) cryogenic temperature at which the fuel tanks would operate. An additional consideration was the decision to assemble the tanks by welding. This provided the best vapor seal against leakage of the hydrogen fuel and minimized the weight of the joints required in assembling the large tanks.

A comparison was made of material properties for several aluminum alloys showing promise for primary structure. A summary of that comparison can be found in Figure 5. Based on this comparison, 2014-T6 and 7075-T6 alloys were eliminated because of their susceptibility to corrosion and stress corrosion cracking. The 7475-T761 material was not competitive from an elevated temperature strength standpoint and had the additional disadvantage of being available from only a single source with resulting high cost. Low strength properties at both room and elevated temperatures eliminated the 6061-T6 material and the T6 temper of the 2219 alloy.

Material	T = 300K (80°F)			T = 394K (250°F) (10,000 Hrs)					Advantages	Disadvantages
	Fatigue (20,000 Cycles)	K_C	da/dn ($\Delta K = 30$)	F_{TU}/ρ	F_{TY}/ρ	F_{CY}/ρ	$E_C^{0.333}/\rho$	$(E_C^{0.325} F_{CY}^{0.325})/\rho$		
2014-T6	1.00	0.67	NA	0.92	0.84	0.99	0.99	1.00		Susceptible to corrosion, exfoliation, and stress corrosion cracking
2024-T81	1.00	0.47	0.34	1.00	1.00	1.00	1.00	1.00	Good corrosion resistance, good elevated temperature mechanical properties	Low fracture toughness at room temperature. No elevated temperature fracture toughness data
2219-T6	0.64	0.92	NA	0.75	0.58	0.63	0.98	0.86	Stable for long time exposure at elevated temperature	Low initial strength, i.e., at low temperatures. No Elevated temperature K_C data
2219-T87	1.00	0.98	1.00	0.79	0.77	0.82	0.98	0.95	High fracture toughness, stable for long time exposure to elevated temperature. good corrosion resistance weldable, property data readily available at elevated temperature	Low initial strength, i.e., at low temperatures. No elevated temperature K_C data
6061-T6	0.69	1.00	0.41	0.59	0.57	0.61	1.00	0.85	High fracture toughness. Excellent corrosion resistance	Low strength. No elevated temperature K_C data
7075-T6	0.91	0.60	0.76	0.84	0.82	0.80	0.98	0.95		Susceptible to corrosion. Exfoliation, and stress corrosion cracking. Low fracture toughness. Temperature limited. No elevated temperature K_C data.
7475-T761	0.78	0.89	1.00	0.59	0.76	0.85	0.97	0.88	High fracture toughness	Sole source, premium price, temperature limited. No elevated temperature K_C data

Note: Index rating ratio property to highest value in column. Highest number is best rating

NA indicates data not available

FIGURE 5
SUMMARY MATERIAL EVALUATION - ELEVATED TEMPERATURE

Only 2024-T81 and 2219-T87 aluminum were finally considered as the basic construction material for the actively cooled panels. The failure modes considered to be most significant were: (1) stability, with (E_C) compressive modulus of elasticity being the figure of merit, and (2) fracture mechanics, with crack growth rate (da/dn) and fracture toughness coefficient (K_C) being the figures of merit. The 2219-T87 alloy was shown to be competitive in stability parameters and to have a definite superiority over 2024-T81 in fracture mechanics. 2219-T87 was therefore selected as the construction material for those structural elements operating at elevated temperature.

The choice of materials for cryogenic tank construction was limited by the aforementioned decision to utilize all-welded construction. A comparison of the three candidate materials is presented in Figure 6. From this comparison it became obvious that the crack growth characteristics (da/dn) of 6061-T6 would eliminate it from further consideration. The 2014-T6 material was also eliminated because of its inherent susceptibility to corrosion and stress corrosion cracking. Therefore, 2219-T87 aluminum alloy was chosen as the most acceptable material for tank fabrication.

	T = 394K (250°F)			T = 20.3K (−423°F)					
Material	Fatigue (20,000 Cycles)	K_C	da/dn (ΔK = 30)	F_{TU}/ρ	E_C/ρ	K_{IC}/ρ	Fatigue (20,000 Cycles)	Advantages	Disadvantages
2014-T6	1.0	0.67	NA	1.0	0.98	0.80	1.0		Susceptible to corrosion, exfoliation, and stress corrosion cracking
2219-T87	0.64	0.98	1.0	0.99	1.0	1.0	0.94	High fracture toughness, stable for long time exposure to elevated temperature. Good corrosion resistance, weldable, property data readily available at elevated temperature	Low room temperature strength
6061-T6	0.69	1.0	0.41	0.67	0.99	NA	0.83	High fracture toughness. Excellent corrosion resistance	Low strength. No cryogenic temperature K_C data

Note: Index rating-ratio of property to highest value in column. Highest number is best rating

NA indicates data not available

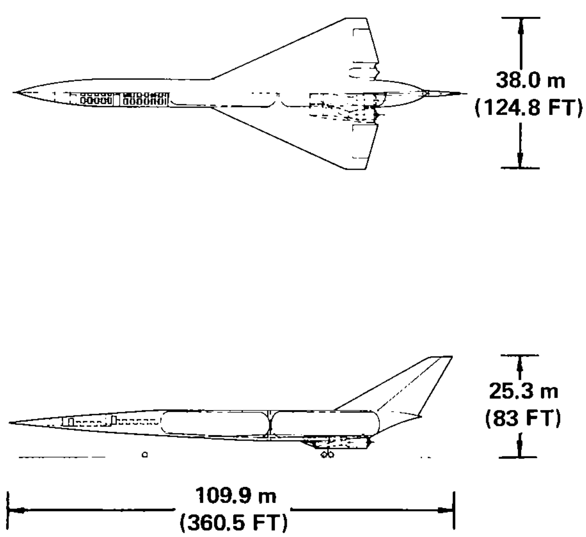
FIGURE 6
SUMMARY MATERIAL EVALUATION - CRYOGENIC TEMPERATURE

Annealed titanium alloy 6Al-4V was used in some limited secondary structure applications, such as fuselage links and fittings, where concentrated loads occur and lighter structure would result. The 6Al-4V alloy was selected because of its favorable combination of tensile strength, high fatigue allowables, and good fracture toughness, compared to several commonly used titanium alloys.

Trade Studies

The baseline (preliminary) Concept 1 aircraft characteristics are given in Figure 7. This baseline met all of the program performance requirements and was based on estimated weights of the fuselage/tank structure and the active cooling system. The fuel weight to perform the mission was determined as 108.9 Mg (240,000 lbm) which was then fixed for the Concept 2 and 3 studies.

Trade studies were conducted to determine the effect of considerations such as passenger compartment location, tank size, construction concepts, and thermal protection/active cooling system designs. Detail discussions of these trade studies are found in References (2), (3) and (4). In order to evaluate



PRELIMINARY CHARACTERISTICS

O.W.E. = 190 Mg (419,234 LBM)

WT_{FUEL} = 108.9 Mg (240,000 LBM)

NON-INTEGRAL FUEL TANKS

PAYLOAD = 21.8 Mg (48,000 LBM)
(200 PASSENGERS)

ENGINE (4) GE5/JZ6-C
T_{SLS} = 400 kN (90,000 LBF)
UNINSTALLED PER ENGINE

HOT NACELLE STRUCTURE

SPEED M = 6.0 (MAXIMUM)

SURFACE ACTIVELY COOLED
TO 394 K (250°F) MAX

RANGE GOAL = 9.26 Mm (5000 NM)

FIGURE 7
CONCEPT 1 BASELINE AIRCRAFT

trade study results in terms of aircraft range, the sensitivity curve shown in Figure 8 was developed for the aircraft. The results of these studies for the Concept 1 aircraft are presented below.

a. Payload/Fuel Location - The purpose of this trade study was to evaluate the effect on range of four different passenger compartment locations: a) at a forward position; b) at the aircraft center of gravity (CG) with an upper tier of seats; c) at the CG with a lower tier; and d) at the CG with a short compartment arranged vertically in several tiers of seats. The results are presented in Figure 9. The significant effect of the passenger compartment location on usable fuel volume is evident. The increased range resulting from the better volumetric efficiency is the primary reason for selecting the forward location as the most promising of the four locations examined. In addition, the forward location results in an ideal ground handling arrangement for boarding and deboarding passengers and for aircraft servicing. The distinct physical separation of the pressurized passenger compartment from the fuel tank compartment also provides the best fabrication scheme of the four arrangements examined.

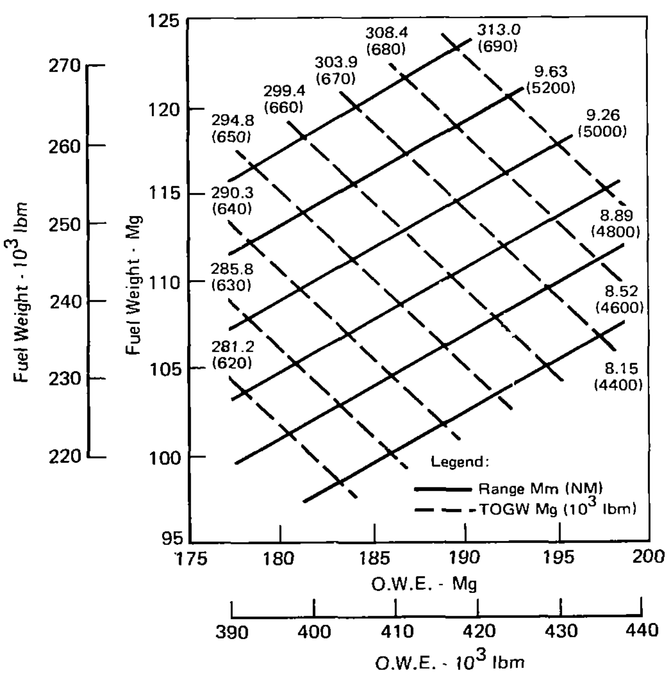
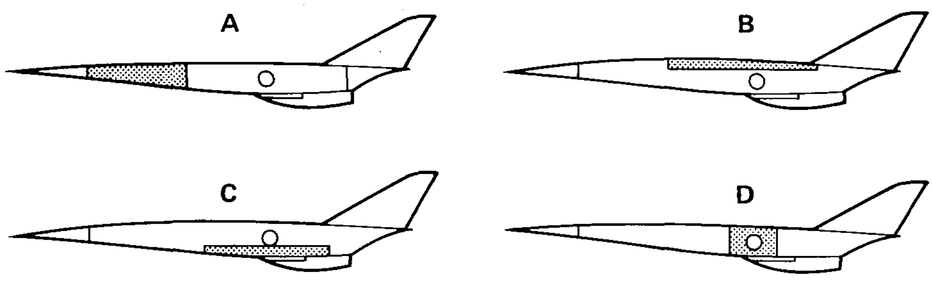


FIGURE 8
RANGE SENSITIVITY, CONCEPTS 1 AND 2



BASELINE: CONCEPT NO. 1 AIRCRAFT

PAYLOAD LOCATION	AVAILABLE FUEL VOLUME / TOTAL FUSELAGE VOLUME	ΔRANGE Mm (NM)
A	0.478	0
B	0.444	−0.74 (−400)
C	0.381	−2.04 (−1100)
D	0.388	−1.85 (−1000)

FIGURE 9
FORWARD PASSENGER COMPARTMENT SELECTED

b. Tank Length and Dome Shape - The objective of this study was to determine the combination of number of tanks and dome shape that maximized aircraft range. Combinations of two, three and four tanks with elliptical, torispherical and hemispherical domes were evaluated. The results are summarized in Figures 10 and 11. On the basis of this comparison a two tank configuration with elliptically domed tank ends was selected for the Concept 1 aircraft.

c. Actively Cooled Fuselage Covering - Actively cooled panels of both honeycomb and beaded construction were compared for the fuselage covering. The critical failure mode for these panels was compressive buckling. Coolant tube spacing was selected to limit skin temperature gradients to 56K (100°F) maximum such that thermal stresses had a negligible effect on panel general stability. Weights were determined for the panels considering not only the load carrying structure but also splices, manifolds, shear clips, adhesives, fasteners, residual coolant and a pumping power penalty. The study results are summarized in Figure 12. The honeycomb panel construction is lighter, within the Concept 1 load range, and provides for a greater range potential.

Number of Tanks	ΔRange km (NM)
2	0 (0)
3	−204 (−110)
4	−520 (−281)

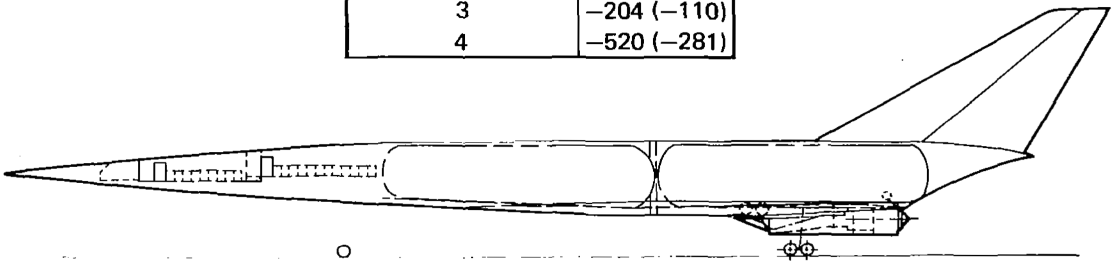


FIGURE 10
RESULTS-FUEL TANK LENGTH STUDY: (ELLIPTICAL DOMES)

Dome Shape	ΔRange km (NM)
Elliptical	0 (0)
Torispherical	−128 (−69)
Hemispherical	−233 (−126)

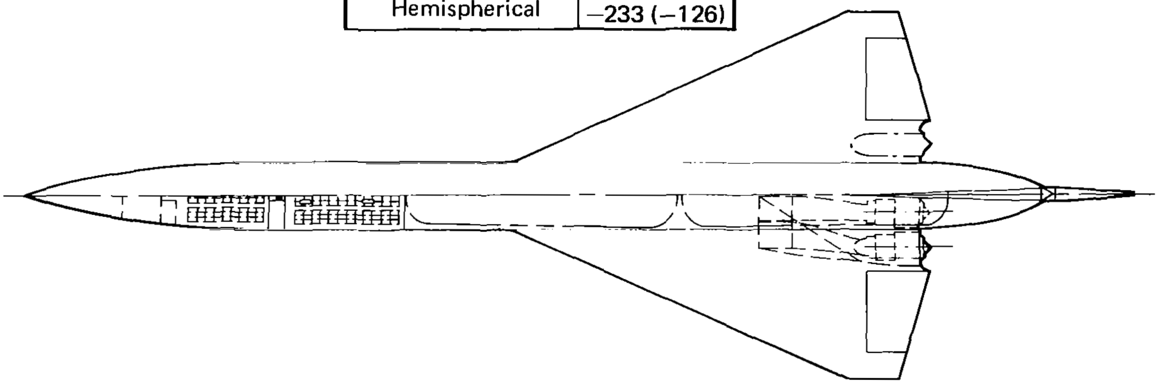
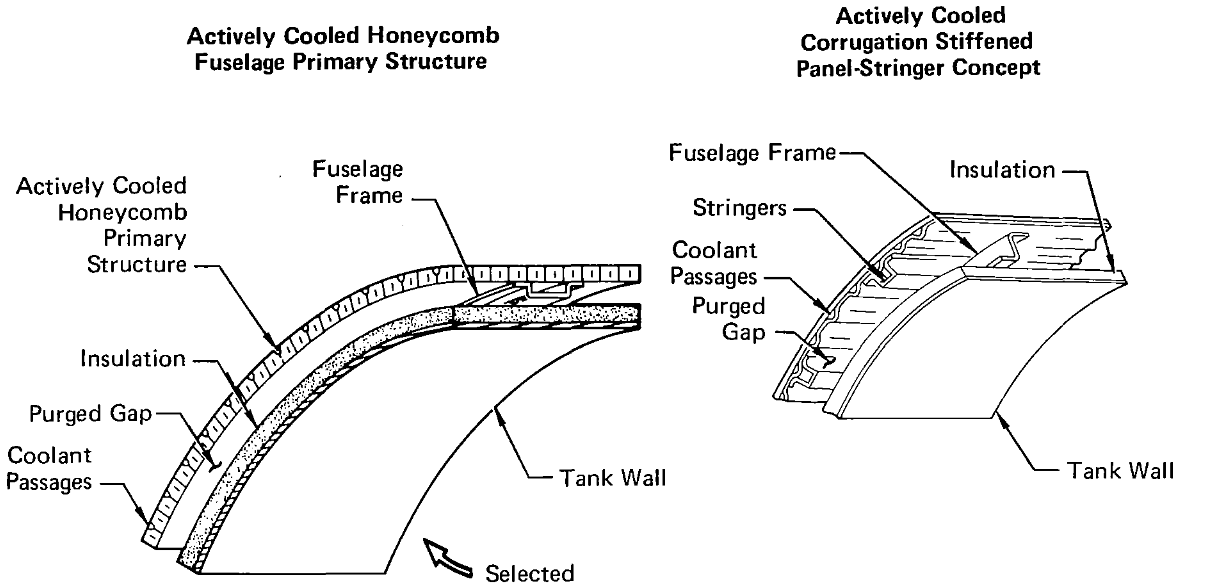
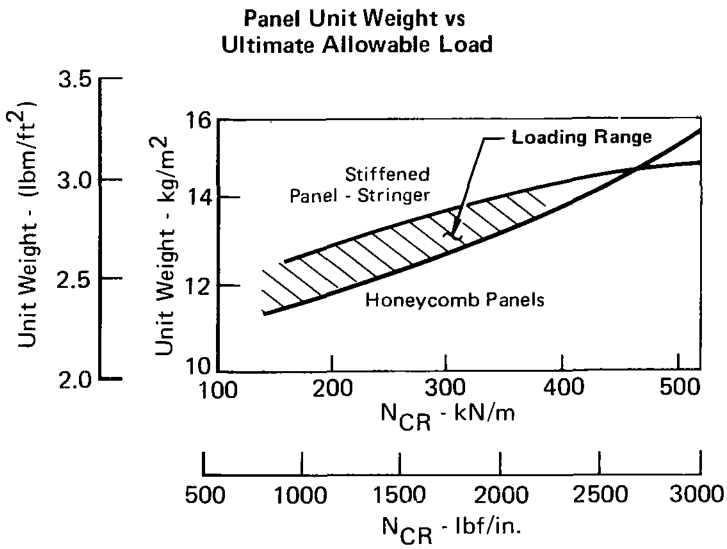


FIGURE 11
RESULTS-FUEL TANK DOME SHAPE STUDY: (2 TANKS)

These advantages, combined with the other desirable features listed in Figure 12, led to the selection of honeycomb panel construction for the fuselage covering.





Honeycomb Panels Selected

- 74.1 km (40 NM) Range Saving
- Lighter
- Fail-Safe Fluid Containment
- Minimize Fastener Installation
- More Efficient Structural Utilization with Frame Caps

FIGURE 12
COMPARISON BETWEEN HONEYCOMB AND STIFFENED STRUCTURAL CONCEPTS

d. Trajectory Shaping - A study related to the ascent trajectory provided visibility to the sensitivity of this class of aircraft to the maximum heating rate encountered in the flight profile. As indicated in Figure 13, the originally assumed ascent was based solely on sonic boom overpressure, dynamic pressure, and inlet duct pressure limitations. The design of the active cooling system to this profile required the absorption of a transient heat load

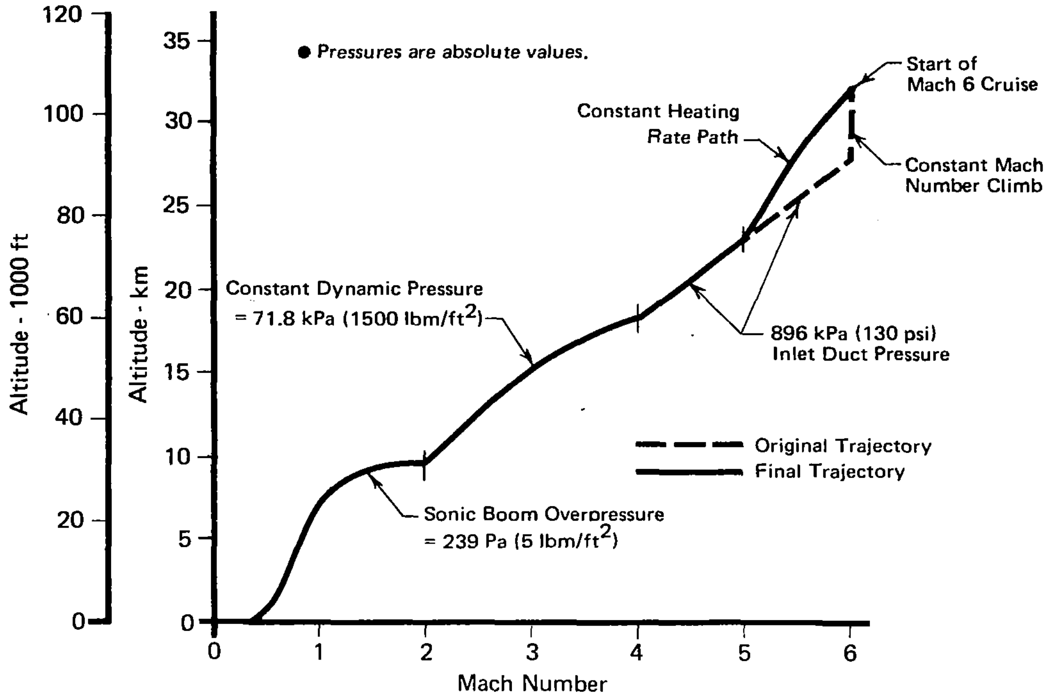


FIGURE 13
FLIGHT TRAJECTORY

that was 40% greater than the sustained heat load experienced during cruise as indicated in Figure 14. This initial approach resulted in a large, heavy active cooling system which was significantly overdesigned for most of the mission, including all of cruise.

A modification to the ascent trajectory, following a constant heating path from Mach 5 to the Mach 6 start of cruise at the best $(L/D)(I_{sp})$ condition, was then investigated. This approach resulted in a net range gain of 289 km (156 NM) attributable to a reduction in cooling system weight. This ascent trajectory modification was incorporated in the final trajectory.

e. Nacelle Cooling - The feasibility of cooling the nacelle module surfaces to the same temperature, 394 K (250°F), as the rest of the airframe was studied. It was found that although the surface area involved represented only 9.4% of the total airframe wetted surface area, cooling the nacelle surfaces added 23.8% to the total heat load to be absorbed.

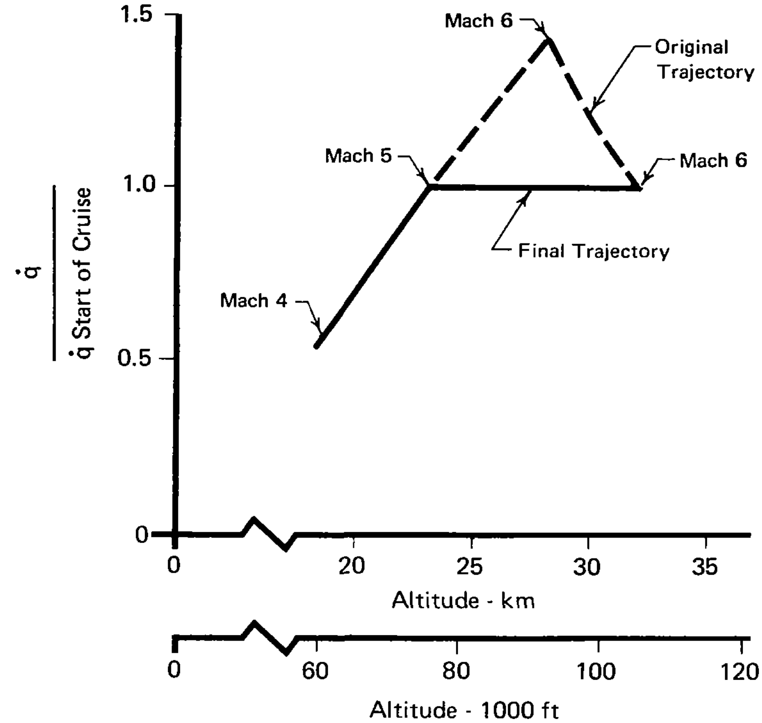


FIGURE 14
EFFECT OF ASCENT TRAJECTORY ON HEATING RATES

There are numerous reasons why the heating rates on the nacelle surface were much higher than the overall average surface heating rates.

- o The nacelle is located on the lower surface.
- o Most nacelle surface locations are near boundary layer origins which results in high heat transfer rates due to the short characteristic lengths involved.
- o Flow in the boundary layer diverter region is subsonic and therefore local adiabatic wall temperatures approach total temperature.
- o The external inlet ramps are positioned at high deflection angles.
- o Heating to the panels in some regions includes conduction from the internal duct walls in addition to external aerodynamic heating.

All of these factors combine to impose extreme cooling requirements for the nacelle surfaces.

A comparison with a "hot" structure nacelle design was made. Superalloy materials compatible with the resultant higher temperature environment were used. While the nacelle structural weight increased, a net aircraft system weight decrease of 2.39 Mg (5266 lbm) resulted, corresponding to a 137 km

(74 NM) gain in range. In addition, cooling the nacelle surface results in many practical design problems. Routing coolant lines across the fuselage/nacelle interface and designing cooling lines into the external inlet duct ramps and sidewalls would be complex and probably would result in volumetric penalties. Taking all of these factors into account, the "hot" structure approach was selected.

f. Hydrogen Tankage Thermal Protection - Thermal protection for the Concept 1 non-integral tankage consists of cooled surface panels, a nitrogen purged gap, and foam insulation applied to the external tank surfaces. A study was conducted to find the combination of insulation weight and fuel boiloff weight that maximized aircraft range. This study was based on the mission requirements which include a one hour ground hold preceding the flight as well as a 20 minute loiter at 12.2 km (40,000 ft) prior to descent.

Parametric data on weight effects alone indicate that the minimum combined weight of insulation and fuel boiloff occurs with an insulation thickness near 2.54 cm (1.0 inch) as shown in Figure 15. However, it was determined that achieving more usable fuel at the expense of additional insulation weight was beneficial from a range standpoint. Maximum range was determined to be achieved with an insulation thickness of 4.27 cm (1.68 in.) as shown in Figure 16.

Structural Analysis

The structural arrangement for Concept 1 is shown in Figure 17. In order to initially determine the vehicle size and performance capability, an "initial estimated weight" was determined. This weight was based on current MCAIR estimation techniques modified for use with actively cooled structure. Using these techniques, factors were applied to the forward fuselage pressurized passenger compartment to account for the non-circularity of the fuselage cross section shape. Weights for all structural components except those located in the fuselage/tank (center fuselage) region of the aircraft, as determined with

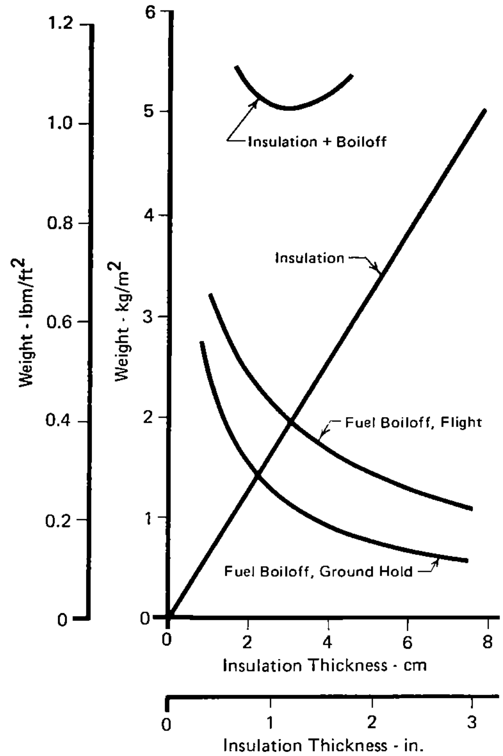


FIGURE 15
NON-INTEGRAL TANKAGE TPS CHARACTERISTICS

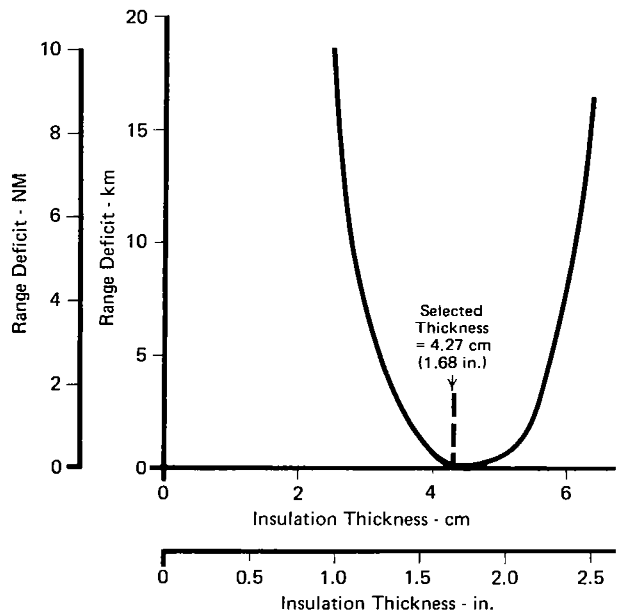


FIGURE 16
RANGE PENALTY ASSOCIATED WITH CHANGES IN
NON-INTEGRAL TANKAGE INSULATION THICKNESS

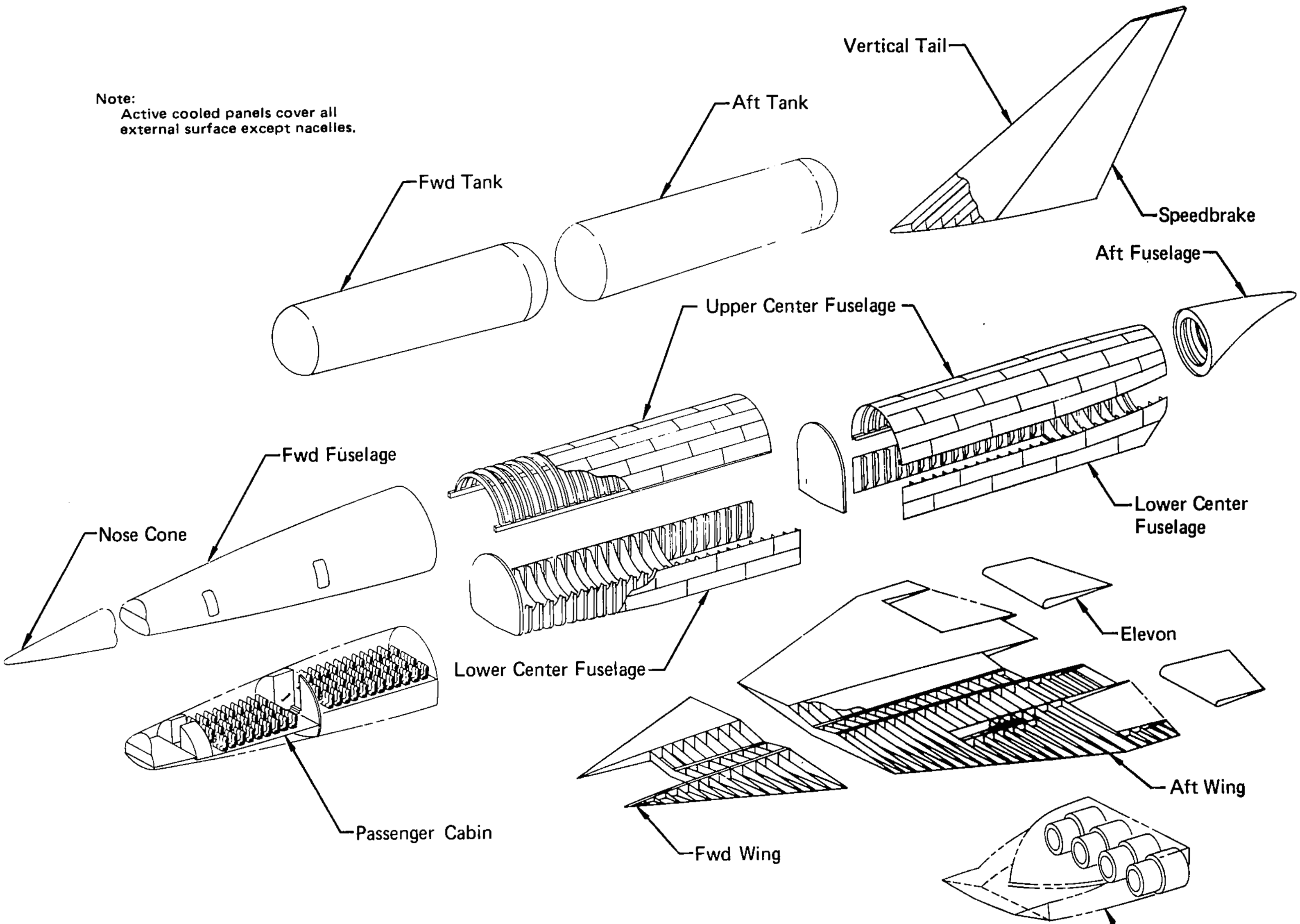


FIGURE 17
CONCEPT 1 STRUCTURE ASSEMBLY BREAKDOWN

these techniques, then remained fixed. Detailed structural analyses of representative structural components located in the fuselage/tank region were then conducted to refine the center fuselage structure and fuel tank weights. The resulting refined weights were then used to determine the final aircraft range and permit performance comparisons with the other aircraft concepts.

The weight refinement process started with the definition of a finite element computer model of the fuselage tank area which was utilized to determine internal load distributions resulting from overall aircraft loads. Although only the fuselage tank area was modelled, stiffness and load characteristics of adjacent structural regions were considered in the model. Because of the inherent stiffness characteristics of the monocoque construction of the forward and aft fuselage sections, it was logically assumed that load introduction from these elements would take the classic plane strain distribution. Input load vectors were therefore calculated on that basis. Wing load inputs took the form of cap loads and web shears in each of the spars. A simplified quick analysis indicated that the chordwise bending stiffness of the wing did not contribute significantly to the overall fuselage bending stiffness and was therefore neglected. Design loads used in the Concept 1 analysis are presented in Figures 18 and 19. The internal loads were then employed in strength analysis to determine member sizing and to assess the effects of fatigue and fracture mechanics requirements. This sizing analysis process was automated using the MCAIR Computer Aided Structural Design (CASD) program.

a. Finite Element Computer Model - The structural model used for analysis of the Concept 1 center fuselage is illustrated in Figure 20. With this model, 1055 joint degrees of freedom were used. A resizing routine was used with three iterations to obtain margins of safety near the desired zero value.

b. Fuselage Structure - Detailed stress analysis of the actively cooled panels, frames, bulkheads and longerons which form the fuselage/tank area primary structure resulted in a calculated weight of 16.41 Mg (36,185 lbm) compared with the initial estimated weight of 18.16 Mg (40,037 lbm). In the area of the monocoque structural shell, the actively cooled panels were heavier than originally estimated. However, the frame and bulkhead weights were lighter than estimated, largely as a result of the use of the panels as part of the frame caps. Accommodating the N_2 purge pressure requirement

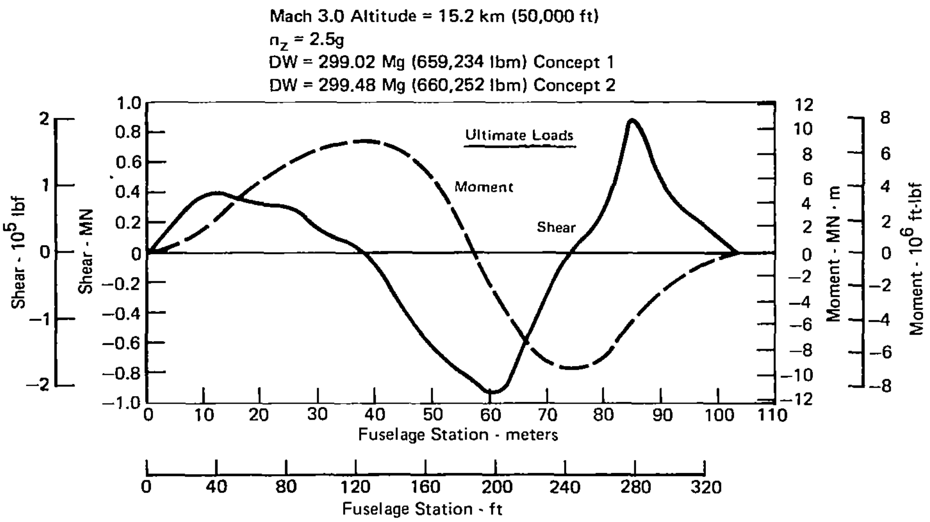


FIGURE 18
CONCEPT 1 AND 2
Net Aircraft Shear and Moment
2.5g Flight Condition

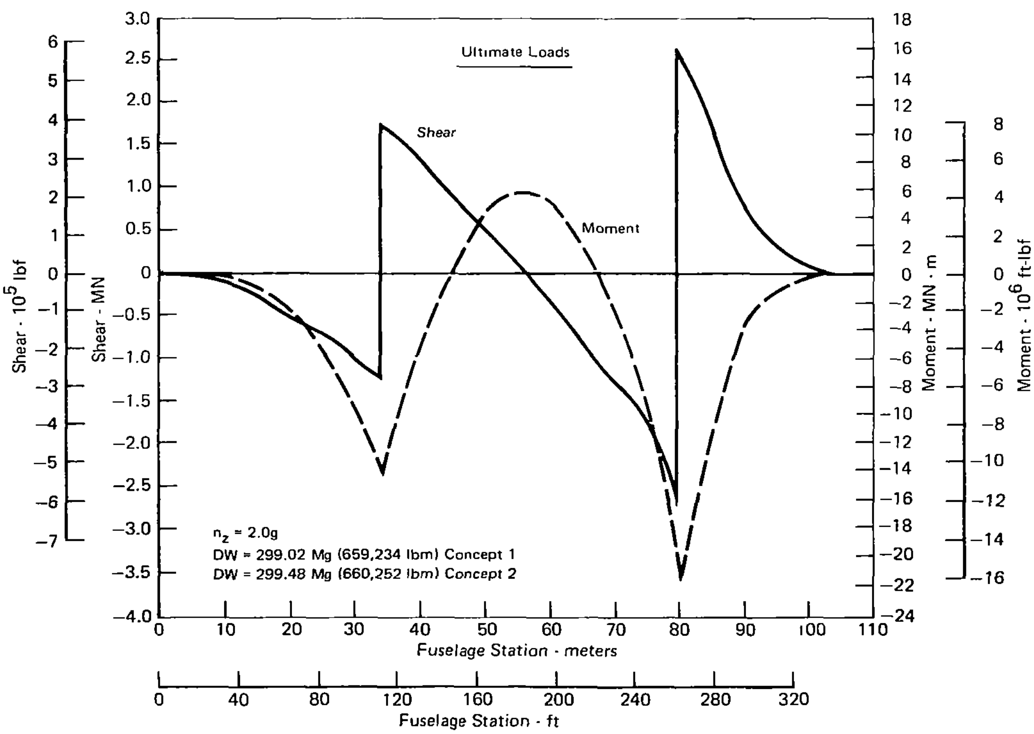


FIGURE 19
CONCEPT 1 AND 2
Net Aircraft Shear and Moment
2g Taxi Condition

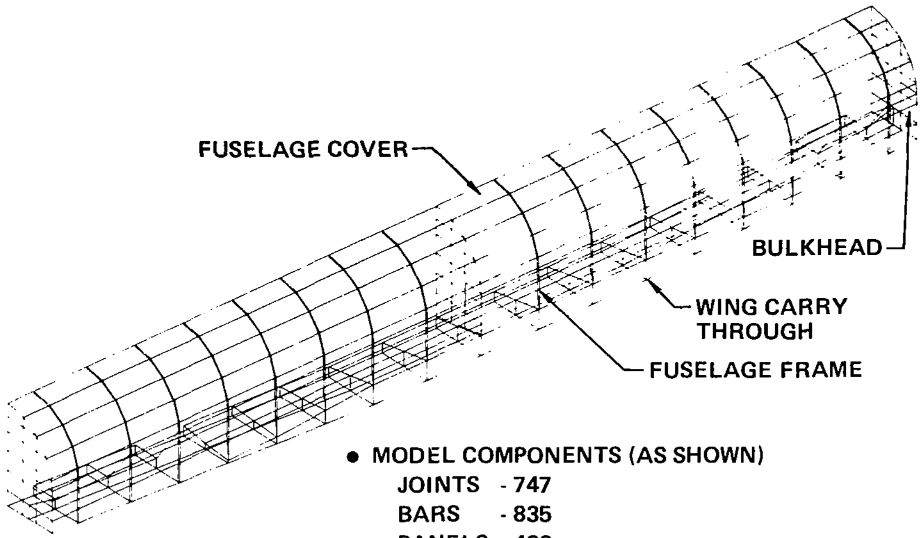


FIGURE 20
CONCEPT 1 FINITE ELEMENT COMPUTER MODEL
Fuselage/Tank Area

resulted in only a modest weight effect, specifically 44 kg (97 lbm), which affects range by only 2.53 km (1.4 NM).

The effects of the fatigue and fracture mechanics requirements were also found to be small as noted:

- o Weight included for fatigue = 0
- o Weight included for fracture mechanics = 268 kg (590 lbm), equivalent to 14.8 km (9 NM) range.

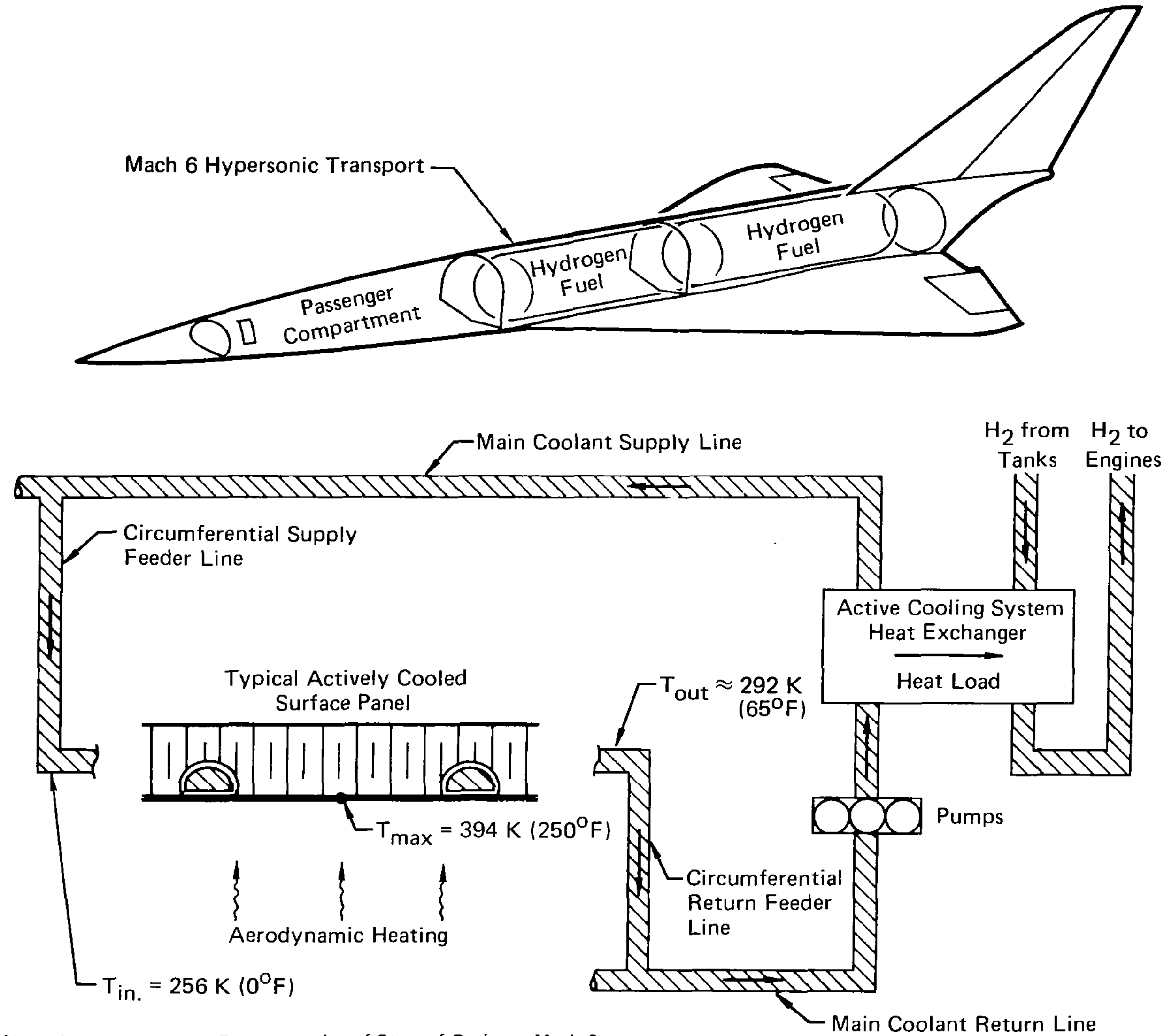
c. Fuel Tanks - A plain skin, monocoque construction was selected for use on the Concept 1 non-integral tanks. The skin thickness was established by burst pressure requirements. The pressure stabilized tanks showed adequate margins of safety for the crash and flight bending conditions. Use of stiffening on the tank walls was not required once the monocoque thickness was established by pressure requirements. The unique combination of tank dimensions and pressure were completely responsible for the selection of monocoque tank walls from a weight standpoint. Analysis of the two non-integral tanks led to a calculated weight of 7.12 Mg (15,699 lbm) as compared to the initial estimated weight of 7.09 Mg (15,635 lbm).

Cooling System Analysis

A detailed study of the aircraft's active cooling system was conducted to refine the system weight and provide a complete system definition for comparison with the other study aircraft. As indicated in Figure 21, the active cooling system uses coolant passing through the structural surface panels to absorb the aerodynamic heating input to the airframe. A distribution system routes coolant to and from the panels, passing through a heat exchanger where the heat load is transferred to the hydrogen fuel. The panels are held to a maximum surface temperature of 394 K (250°F). The coolant inlet temperature to the panels was assumed to be 256 K (0°F). Coolant tube spacing was chosen to permit a 56 K (100°F) outer skin thermal gradient and minimize panel structural weight. As a result, the average coolant outlet temperature was approximately 292 K (65°F). Sufficient hydrogen fuel heat sink capacity for airframe cooling was assumed, as part of the initial study ground rules. It was assumed that the LH_2 from the tanks was supplied to the system heat exchanger at 20.3 K (-423°F). Subsequent analyses, based on a fuel temperature rise of 235 K (423°F), revealed that engine fuel flowrates during cruise would, in fact, not provide sufficient heat capacity to absorb the total airframe heat load. These flowrates are only slightly more than half of that required. This subject is discussed relative to each concept in subsequent sections of this report.

Figure 22 indicates how coolant feeder lines, branching out of the main lines, service adjacent surface panels. Coolant is dispersed into manifolds at the panel ends to distribute flow evenly through coolant tubes. A distribution system routing, indicated in Figure 23, was derived based on 6.1 m (20 ft) long panels with the fuselage panels positioned in a staggered arrangement, which reduces feeder line size.

Aircraft subsystems (environmental control, hydraulic, and electrical systems) were defined only in sufficient detail to establish their cooling requirements. These systems were integrated with the airframe's coolant distribution system, as shown in Figure 24, so that these heat loads are also absorbed by the hydrogen fuel.



Note: Temperatures are Representative of Start of Cruise at Mach 6

FIGURE 21
AIRFRAME SURFACE ACTIVE COOLING SYSTEM

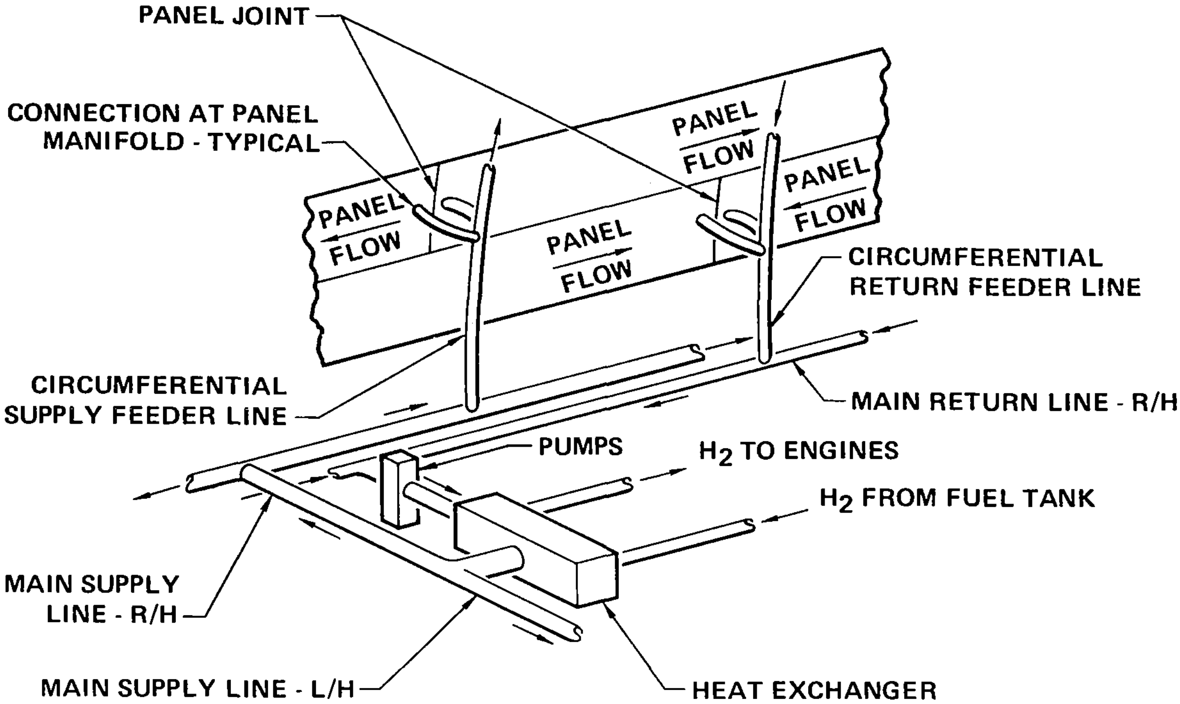
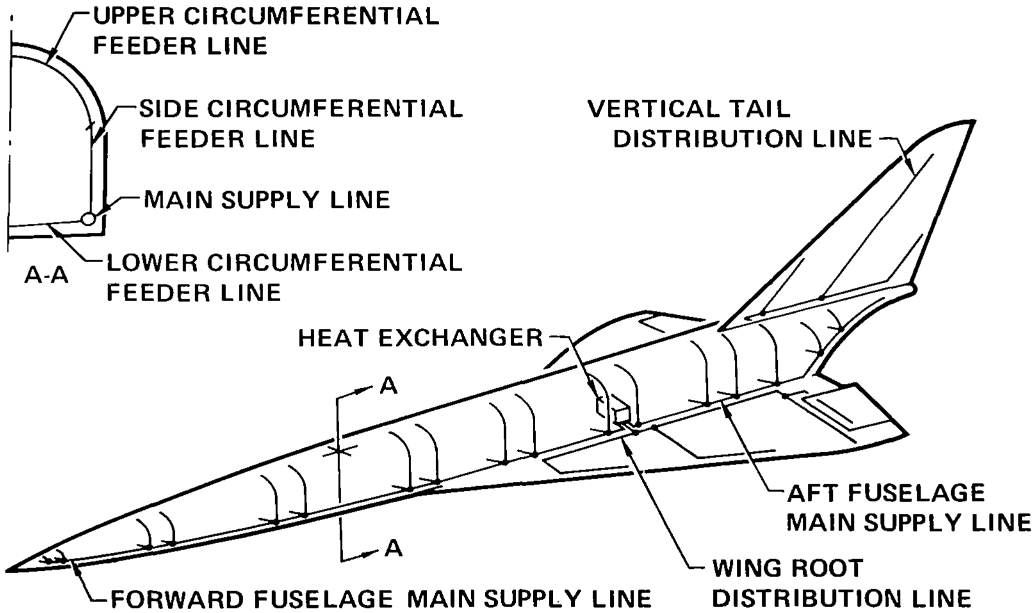


FIGURE 22
TYPICAL COOLANT DISTRIBUTION ROUTING
AT MAJOR COMPONENT LOCATION



- Supply Lines Only are Shown, Return Lines Spaced Similarly
- Lines Represent Those on One Side of Aircraft Only, Exception Vertical Tail Lines

FIGURE 23
SIMPLIFIED DISTRIBUTION SYSTEM SCHEMATIC,
CONCEPTS 1 AND 2

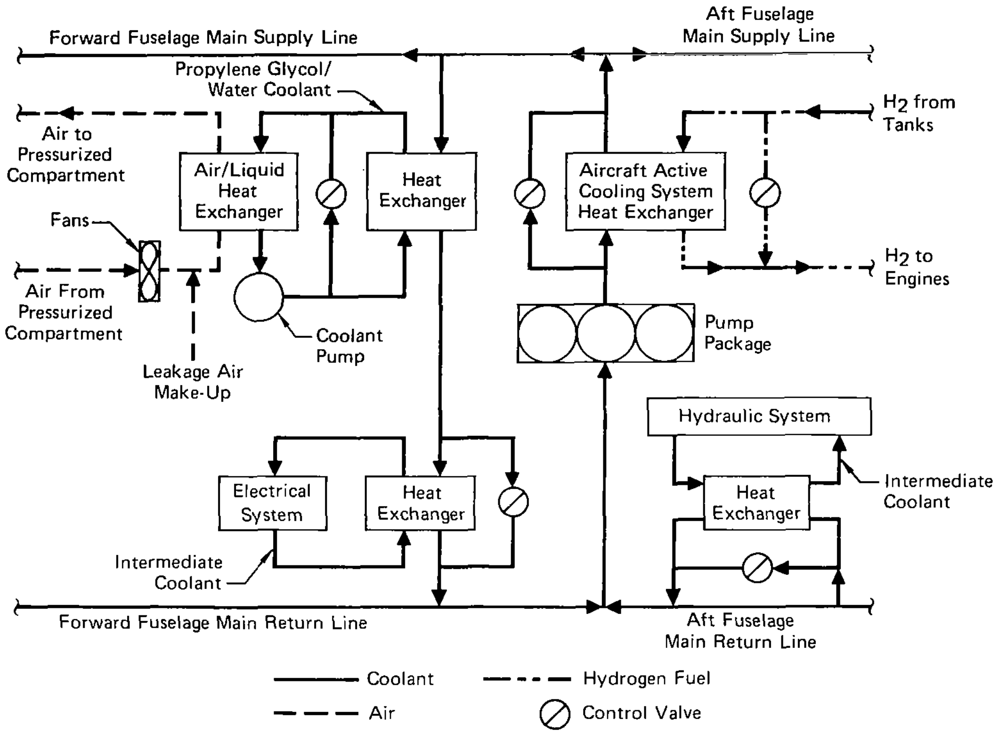
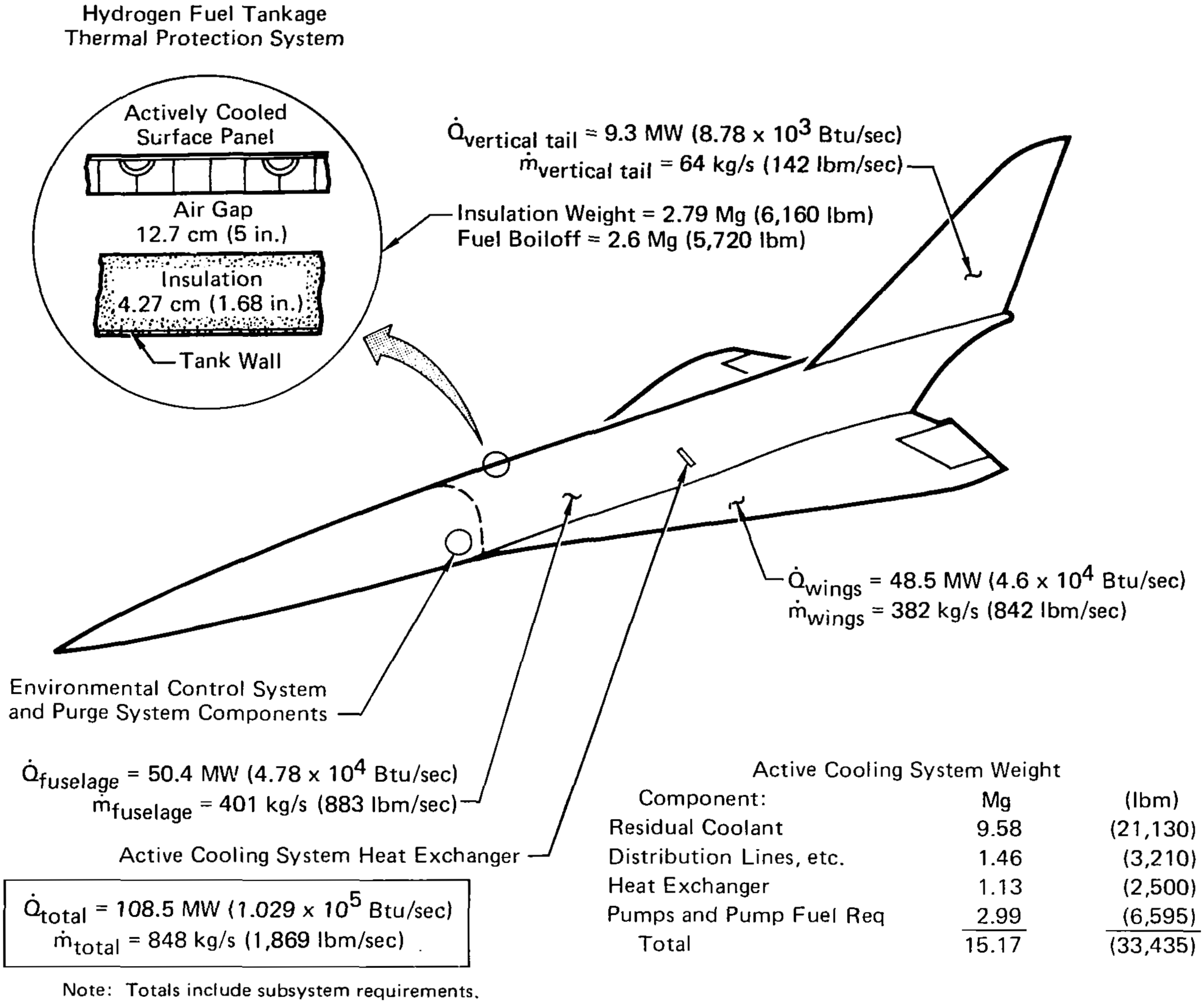


FIGURE 24
INTEGRATION OF SUBSYSTEMS WITH ACTIVE COOLING SYSTEM

Surface heating rates were established at the cooling system design point selected from the ascent trajectory trade study. These heating rates varied from a maximum of 199 kW/m^2 (17.5 Btu/sec ft^2) on the forward fuselage to a minimum of 12 kW/m^2 (1.1 Btu/sec ft^2) on the aft fuselage upper surfaces. The average heating rate over the entire cooled surface area was 29.4 kW/m^2 (2.59 Btu/sec ft^2). Parametric data generated with a computerized thermal model of a cooled panel were used to establish flowrate requirements. This information, combined with the distribution system definition, enabled the active cooling system weights to be determined.

Figure 25 provides a summary of the results of thermodynamic analyses of the Concept 1 aircraft. Heat loads and coolant requirements for each major section of the aircraft are shown along with totals which include subsystem requirements. A cooling system weight breakdown is provided and the fuel tank thermal protection system characteristics are summarized. These results are based on the specified assumption of unlimited fuel heat sink capacity to cool the entire aircraft surface (exclusive of the nacelle). As



Active Cooling System Weight

Component:	Mg	(lbm)
Residual Coolant	9.58	(21,130)
Distribution Lines, etc.	1.46	(3,210)
Heat Exchanger	1.13	(2,500)
Pumps and Pump Fuel Req	2.99	(6,595)
Total	15.17	(33,435)

FIGURE 25
THERMODYNAMIC SUMMARY, CONCEPT 1

mentioned previously, actual engine fuel flowrates during cruise are inadequate to absorb the total heat load. For Concept 1, it was estimated that the heat sink afforded by the fuel flow during cruise is approximately 50% of that required to absorb the design heating rate. However, refinement of the airframe thermal protection system via localized heat shielding, etc., to provide a matching of the airframe heating rates and engine fuel flow heat capacities compatible with the engine efficiencies used was considered to be beyond the scope of this study. The major goal of this study was establishment of a baseline system against which fuselage/tankage trade studies could be made.

SECTION 5

INTEGRAL TANKAGE THERMAL PROTECTION/ACTIVE COOLING SYSTEM SELECTION

Prior to the design refinement of Concepts 2 and 3, a study was conducted to select a thermal protection/active cooling system arrangement for the integral tank aircraft concepts. Trade studies of eight candidate arrangements were conducted and the most promising arrangement was selected based on the fact that no light weight vapor barrier non-permeable to gaseous hydrogen exists.

The eight candidate conceptual arrangements considered are shown in Figure 26. Concepts a, b, and c were specified in the study definition. Concepts d through h were evaluated to insure that a variety of competitive arrangements were investigated. Concept c, due to its thermodynamic similarity to the non-integral tankage thermal protection system (TPS) arrangement, was selected as the baseline concept. Range differences were determined to reflect differences in TPS weight, usable fuel, and TPS volume. The range differences reflect configuration variations on the upper half of the fuselage only. This simplification avoided numerous complexities involved with considerations unique to the lower half of the fuselage as justified in Reference (3). Therefore, the range differences shown are approximately one half the actual magnitude involved. Each concept was also evaluated in terms of relative fabrication difficulties, inspectability/maintainability, cost, and development status. A summary of this information is presented in Figure 27. As noted on the figure, Concept c, the baseline, was also selected as the integral tankage TPS concept.

It can be noted in Figure 27 that most of the candidate arrangements are reasonably competitive on a range basis or purely on a unit weight basis. Therefore, other considerations were prime drivers in the final configuration selection process. Concepts a and d, which require thick layers of insulation due to H_2 gas permeation, penalize the usable fuel volume to the extent that the resultant range losses are significant. Concepts b and f require diffusion bonded structure to insure against hydrogen leakage. Such structures would be expensive and difficult to inspect. Concept g was considered difficult to fabricate and inspect. Concept h, although analytically attractive, would require the development of an acceptable multilayer, evacuated insulation material. Finally, Concept e offers no significant advantages over Concept c and is less thermodynamically efficient. Hence, Concept c was selected for subsequent studies.

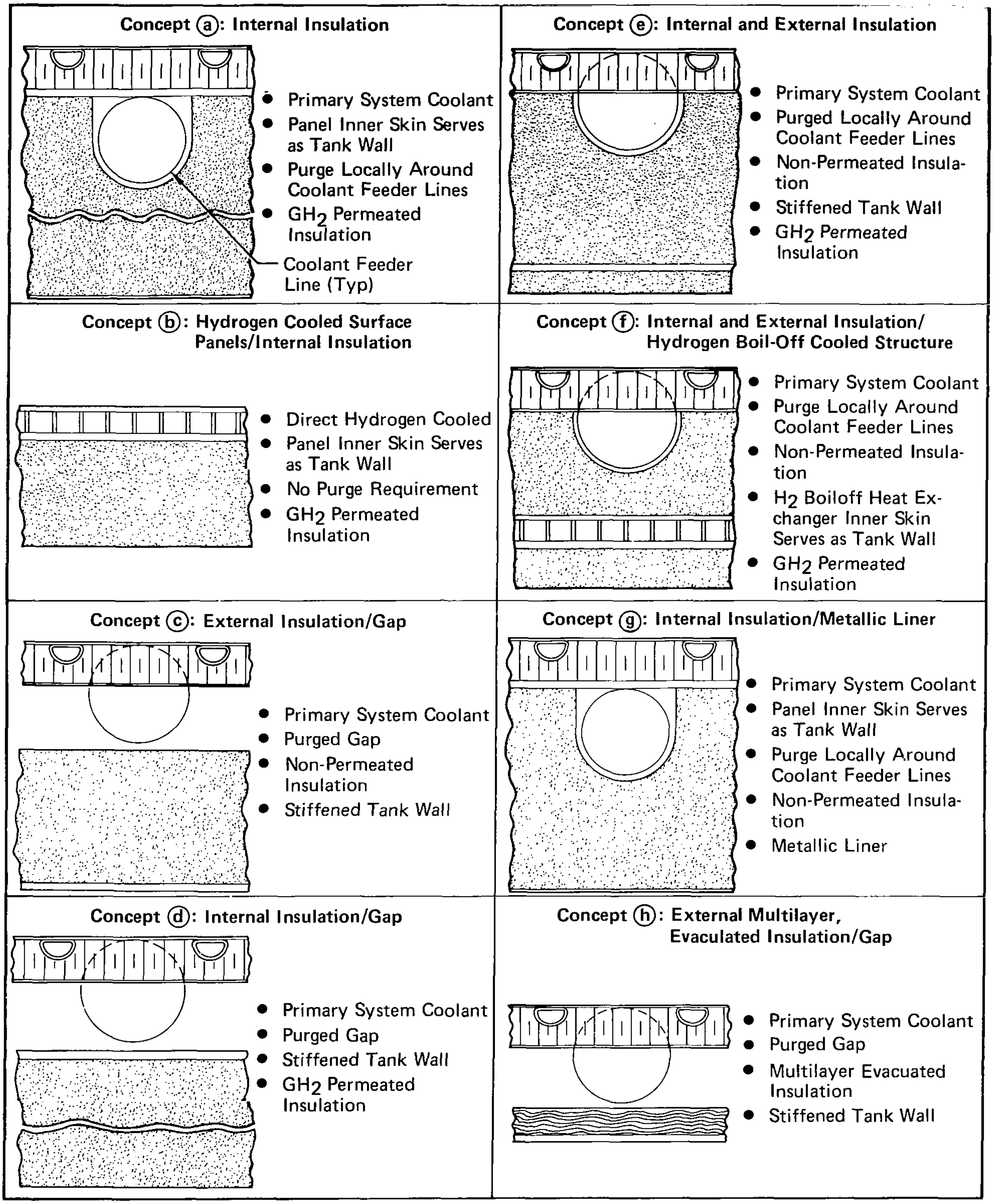


FIGURE 26
CANDIDATE INTEGRAL TANKAGE TPS CONCEPTS

CONCEPT	1/2 RANGE CHANGE FROM BASELINE CONCEPT c km (NM)	DIFFERENCES FROM SELECTED CONCEPT c					STRUCTURE + TPS UNIT FIXED WEIGHT kg/m^2 (lbm/ft^2)
		RANGE	FABRI-CATION	INSPECTABILITY/ MAINTAINABILITY	COST	DEVELOPMENT	
a STRUCTURAL COOLED PANEL, PERMEATED INSULATION	-352(190)	SIGNIFI-CANT LOSS	DIFFICULT	POOR	HIGH	------	23.7(4.86)
b STRUCTURAL H_2 COOLED PANEL, PERMEATED INSULATION *	-9(5) b1	----	DIFFICULT	POOR	HIGH	REQ'D	22.0(4.50)
	-74(40) b2	LOSS	DIFFICULT	POOR	HIGH	REQ'D	14.7(3.02)
c STRUCTURAL TANK WALL, NON-PERMEATED INSULATION, GAP		BASELINE AND SELECTED CONCEPT					19.4(3.98)
d STRUCTURAL TANK WALL, PERMEATED INSULATION, GAP	-174(94)	SIGNIFI-CANT LOSS	----	----	----	----	20.9(4.28)
e STRUCTURAL TANK WALL, NON-PERMEATED INSULATION	-44(24)	LOSS	----	----	----	----	20.5(4.19)
f STRUCTURAL BOILOFF H_2 COOLED TANK WALL, PERMEATED/NON-PERMEATED INSULATION	-98(53)	LOSS	DIFFICULT	POOR	HIGH	REQ'D	24.0(4.91)
g STRUCTURAL COOLED PANEL, NON-PERMEATED INSULATION, METALLIC LINER	-7(4)	----	DIFFICULT	POOR	HIGH	REQ'D	19.3(3.96)
h STRUCTURAL TANK WALL MULTILAYER, EVACUATED INSULATION, GAP	+39(21)	INCREASE	----	----	----	REQ'D	19.5(4.00)

* Concept b1 insulation thickness same as for Concept a. As a result, fuel boiloff is inadequate to cool surface structural panels. Concept b2 insulation thickness sized to provide adequate boiloff for structural cooling.

FIGURE 27
INTEGRAL TANK TPS SELECTION

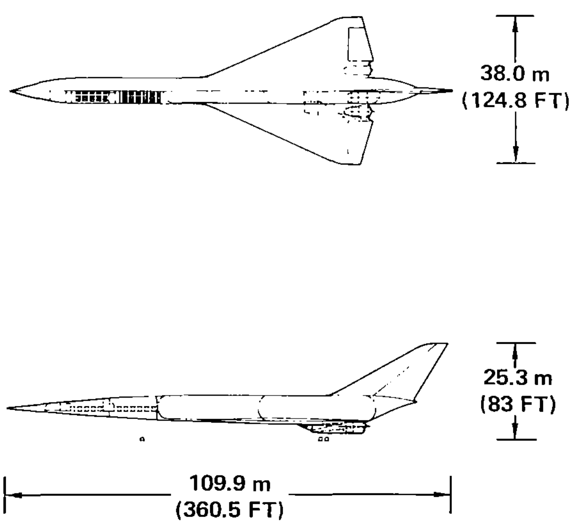
SECTION 6

CIRCULAR BODY AIRCRAFT WITH INTEGRAL TANKS (CONCEPT 2) SYNTHESIS

The Concept 2 aircraft is almost identical in external lines to Concept 1. It also has the same fuel weight, passenger payload and propulsion system. However, the structural arrangement of the fuselage/tank area is quite different, in that a single integral fuel tank replaced the non-integral tanks of Concept 1. Selection of the tank thermal protection and active cooling systems for Concept 2 was completed prior to refining the design of the airplane. The refinement studies were similar to those of Concept 1.

Trade Studies

The baseline (preliminary) Concept 2 aircraft characteristics are given in Figure 28. The airplane size was established by the payload and fuel weight requirements. Range was a "fall out" of those assumptions. Many of the trade study results discussed in Section 4 also apply to Concept 2. Passenger compartment location, elliptically domed tank ends, and a two compartment fuel tank are samples of items retained from Concept 1. The tank length



PRELIMINARY CHARACTERISTICS

O.W.E. = 189.7 Mg (418,106 LBM)

WT_{FUEL} = 108.9 Mg (240,000 LBM)
INTEGRAL FUEL TANKS

PAYLOAD = 21.8 Mg (48,000 LBM)
(200 PASSENGERS)

ENGINE (4) GE5/JZ6-C
T_{SLS} = 400 kN (90,000 LBF)
UNINSTALLED PER ENGINE

HOT NACELLE STRUCTURE

SPEED M = 6.0 (CRUISE)

SURFACE ACTIVELY COOLED
TO 394 K (250°F) MAX

FIGURE 28
CONCEPT 2 BASELINE AIRCRAFT

trade study showed that the two compartment tank arrangement was necessary for CG control and to limit the effect of crash condition pressure heads. The honeycomb construction actively cooled panels were also retained in order to provide minimum fuselage weight. The modified trajectory was utilized for all of the aircraft concepts, as well as the hot engine nacelle module configuration described in Section 4. The two major additional trade studies conducted for Concept 2 involved tank wall construction and the structural arrangement to be used for the fuselage covering in the tank area. Range sensitivity of the Concept 2 aircraft was found to be the same as Concept 1. Therefore the sensitivity curve of Figure 8 was used to evaluate these trade study results.

a. Tank Wall Construction - Non-stiffened tank walls were found to be a heavy approach for the Concept 2 integral tanks due to the higher bending loads. Integrally machined stiffeners were chosen over mechanically fastened stiffeners because of the problem associated with leakage of gaseous hydrogen through the fastener holes. The two most attractive external stiffener arrangements, a 0°-90° waffle pattern and Isogrid construction, are compared in Figure 29. Based on these results, the Isogrid construction was selected for Concepts 2 and 3 to realize the weight savings potential.

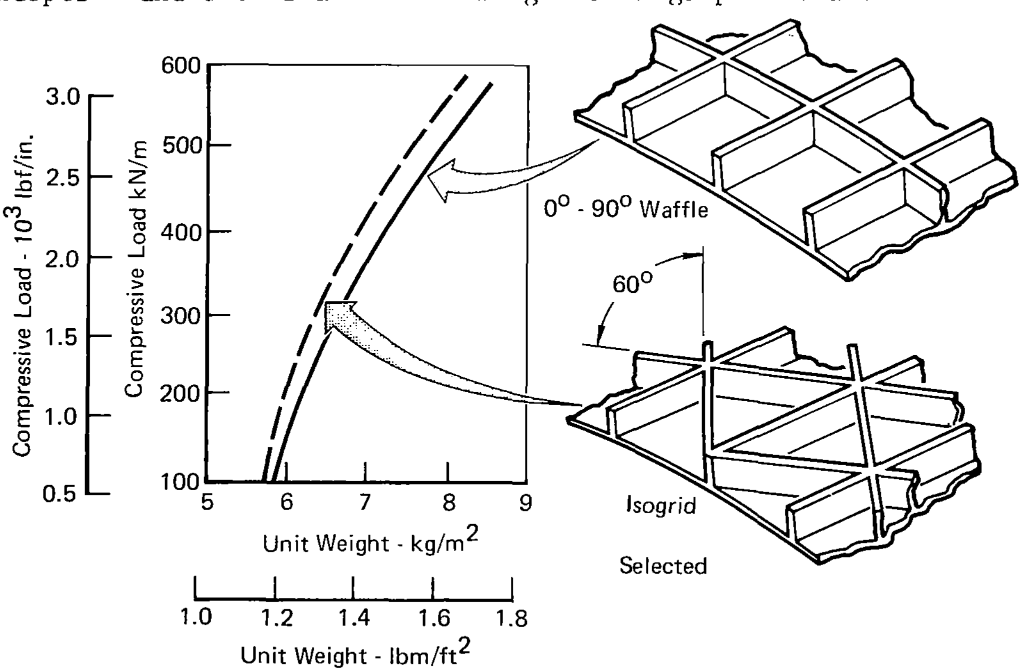


FIGURE 29
TANK WALL/PRIMARY STRUCTURE CONSTRUCTION
Externally Stiffened

Three other methods of tank wall construction were also considered for this study. The first was a 0-90° waffle pattern modified by the addition of discrete rings. These rings would also be used as supports for the actively cooled panels. Thermal gradients, however, from 20.3 K (-423°F) at the tank wall to 366 K (200°F) at the panel inner surface would create drastic thermal stress problems in a continuous ring. For this reason the rings were eliminated from further consideration.

Integral stiffening by means of $\pm 45°$ waffle patterns and plain monocoque skin construction were also considered but were not competitive from a weight standpoint. Many other approaches to integral stiffening are available. This study was limited to the concepts noted above because of their simplicity, limited height, and ready adaptability to application of cryogenic insulation.

b. Semi-structural vs Non-structural Fuselage Covering - Since the integral tank is the primary structural load path in the fuselage/tank area, the actively cooled panels initially were designed to be non-structural. However, the panels, as designed to the minimum height required to serve the cooling function, weigh approximately 90% as much as they would in a structural configuration. Therefore, a trade study was conducted to determine how much weight could be saved by using the panels as secondary bending structure. The non-structural panels were assumed to be supported individually from the integral tank and have slip joints around their periphery to allow relative motion between the panels. In the semi-structural arrangement all the panels were assumed to be interconnected and supported, on frames, from the upper wing surface. Major slip-joints were utilized at each end of the cover to allow thermally induced motion between the tank and cover. The semi-structural panels did not have to be increased in weight to carry the maximum compressive running loads. By using this arrangement, the integral tank bending loads are relieved sufficiently to permit a significant tank shell weight reduction. The results showed that 998 kg (2200 lbm) in tank weight could be saved, which corresponds to a range increase of 51 km (31 NM). The semi-structural fuselage cover arrangement selected for Concept 2 is illustrated in Figure 30.

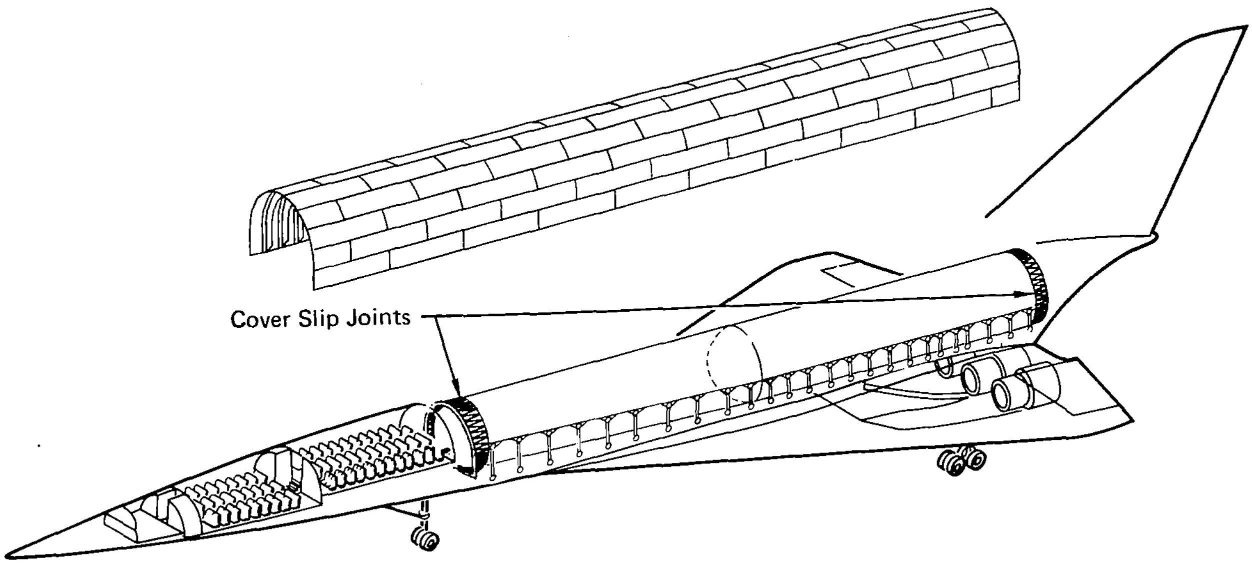


FIGURE 30
SEMI-STRUCTURAL FUSELAGE COVERING

Structural Analysis

The structural arrangement for Concept 2 is shown in Figure 31. The process of structural weight refinement described for Concept 1 was also followed for Concept 2. Design loads used in the Concept 2 analysis were the same as those used for Concept 1 as presented in Figures 18 and 19.

a. Finite Element Computer Model - The Concept 2 structural model is illustrated in Figure 32. 1618 joint degrees of freedom are used with this model. The fuselage and tank structures have been separated for this illustration, but are joined by the computer analysis program. As for Concept 1, a resizing routine with three iterations was employed to approach a zero margin of safety.

b. Fuselage Covering - The loads in the semi-structural covering are approximately 20% or less, of those for the primary structural covering on the Concept 1 aircraft. Detail stress analysis of the semi-structural actively cooled panels resulted in a calculated weight of 9.31 Mg (20,540 lbm) compared with the original estimate for the baseline of 8.90 Mg (19,625 lbm). No additional weight was required to satisfy either the fatigue or fracture mechanics requirement.

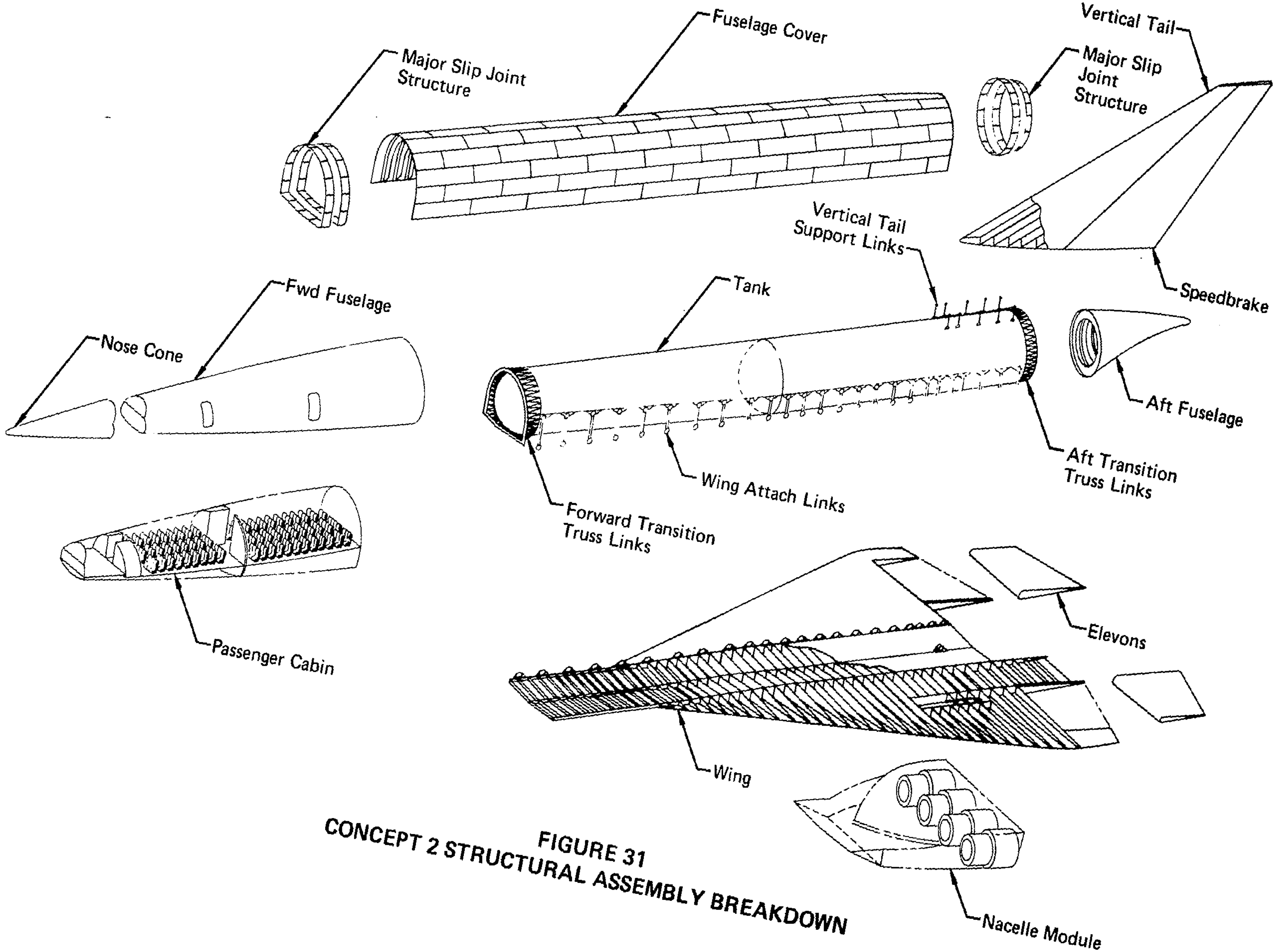
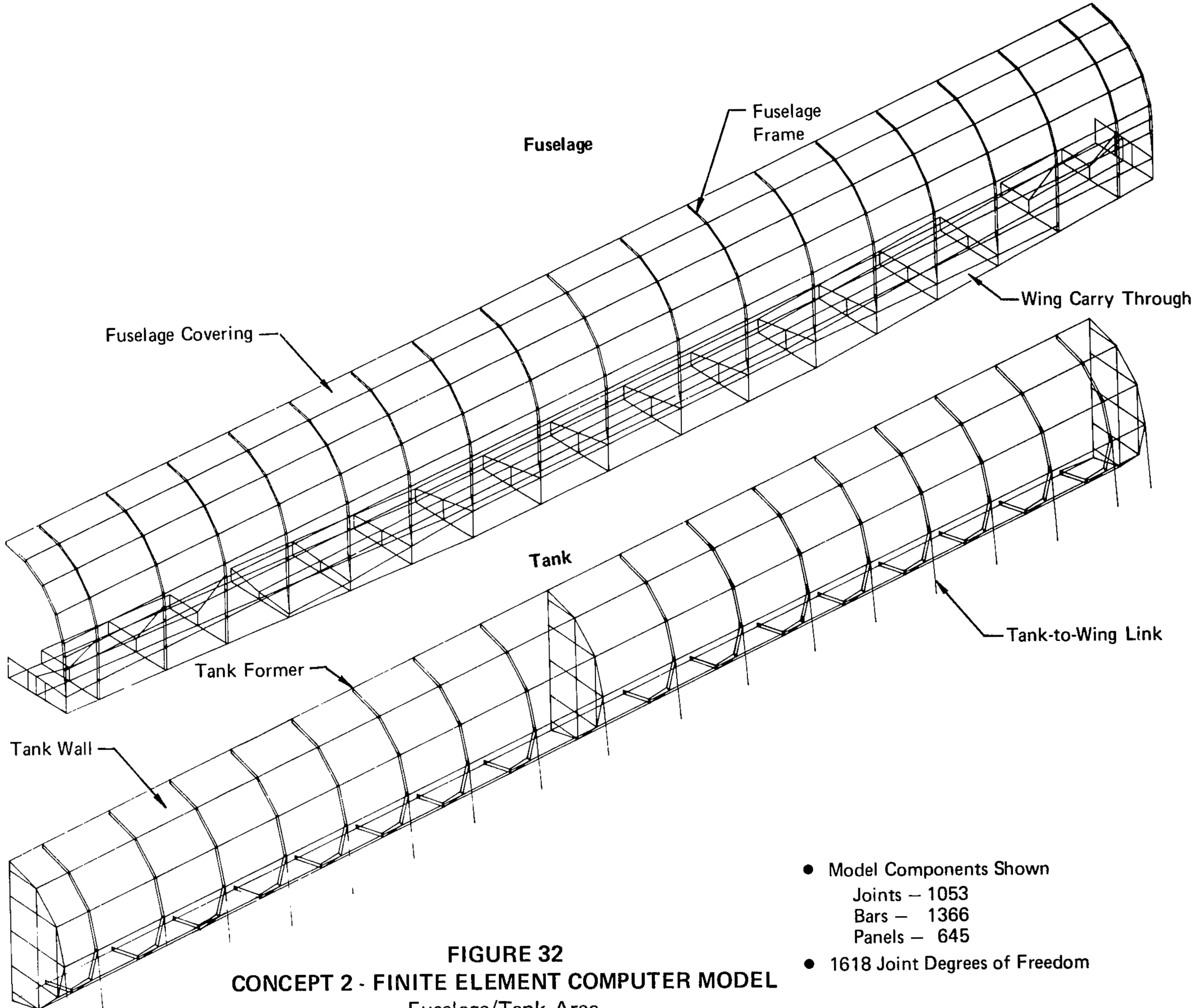


FIGURE 31
CONCEPT 2 STRUCTURAL ASSEMBLY BREAKDOWN



- Model Components Shown
  - Joints – 1053
  - Bars – 1366
  - Panels – 645
- 1618 Joint Degrees of Freedom

FIGURE 32
CONCEPT 2 - FINITE ELEMENT COMPUTER MODEL
Fuselage/Tank Area

Accommodating the N_2 purge requirement resulted in only a modest weight effect, as was the case with Concept 1. Specifically the weight effect was an increase of 44 kg (97 lbm) which transforms to a range increment of 2.53 km (1.4 NM)

c. Tank - In Concept 2 the structure was interconnected with link systems that accommodate the thermal strains while still maintaining reliable structural load paths. As an example the wing-to-tank connection is made with a series of links which have monoball bearings at each end to allow the links to swivel and yet allow one end of the link to move with respect to the other. Each link has full axial load capability. A series of nearly vertical links is used to attach each side of the fuel tank to the upper surface of the wing. The longitudinal location of the tank is fixed by a single aft link which attaches to the wing carry-through at the centerline. Side motion of the tank is prevented by a series of transverse links.

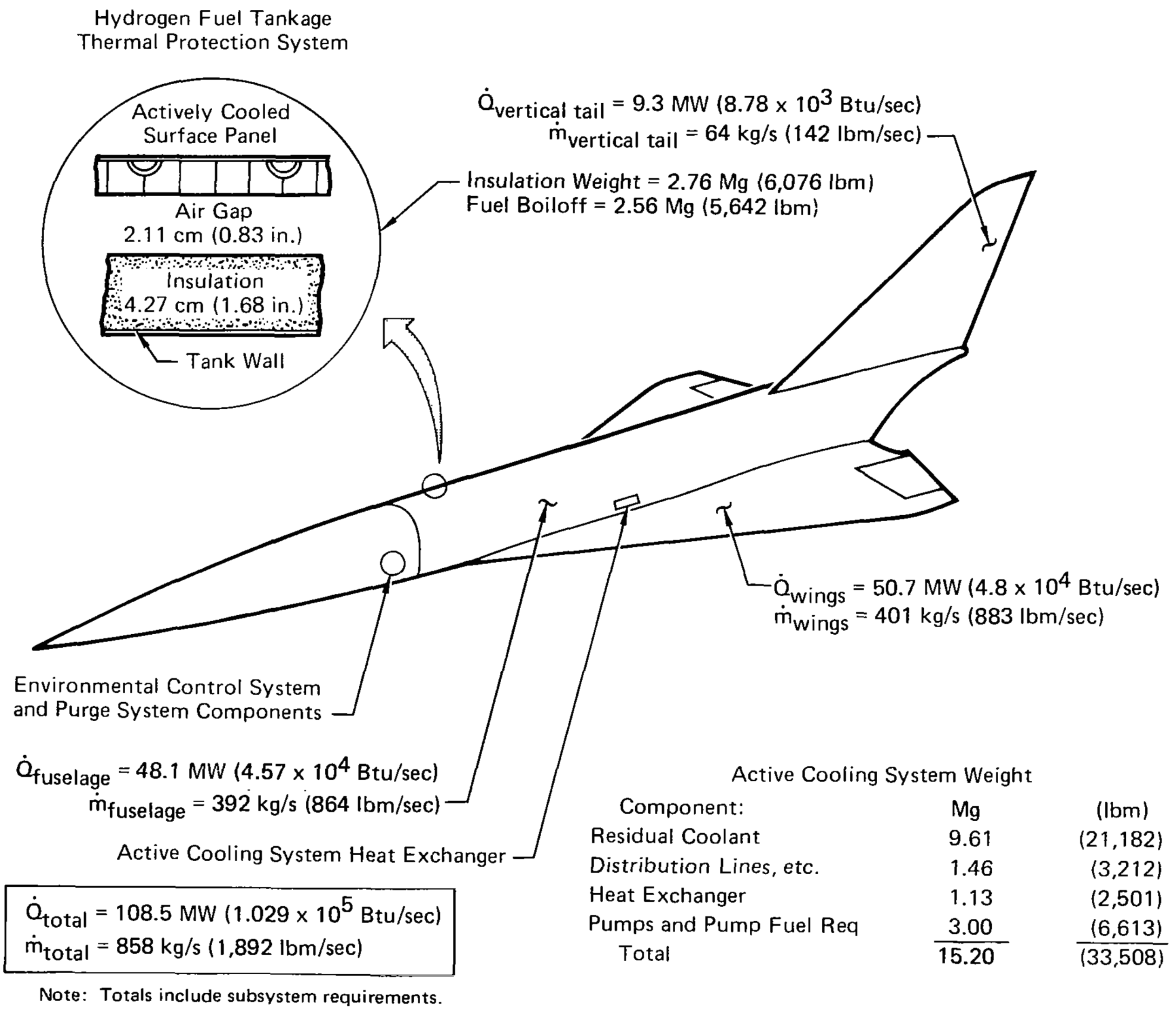
Thermal contraction of the tank is accommodated by these links, which travel in an arc and induce a bending stress of only about 3.45 MPa (500 psi) in the tank at the peak of the arc. Truss networks formed of the same type of links provide thermal strain relief at the splice joints where the forward and aft fuselage sections and the vertical tail attach to the tank.

Testing of the Isogrid stiffening concept has shown biaxially loaded structure stress concentration factors to be limited to a maximum of 1.5. This value was employed in the integral tank analysis.

Detail analysis of the Concept 2 integral tank resulted in a tank weight of 11.03 Mg (24,309 lbm) compared with the initial estimate of 8.76 Mg (19,303 lbm). This analysis also resulted in a final weight of the frame, bulkheads and longerons of 3.61 Mg (7,955 lbm) compared with the initial estimate of 5.51 Mg (12,150 lbm). Although these differences between the individual initial and final estimates are significant, the totals were in good agreement. Weight added specifically for fatigue considerations was minor, 20.4 kg (45 lbm), and none was added for fracture mechanics.

Cooling System Analysis

The Concept 2 cooling system was analyzed in a manner similar to that described for Concept 1. Since Concept 1 and 2 moldline contours and areas are nearly identical, the surface heating rates are similar and the cooling system requirements are essentially equal. The most significant difference between the cooling system designs involved a minor relocation of the cooling system heat exchanger. This relocation, necessitated by the lack of available space between the tank wall and the outer skin, did not significantly impact cooling system weight. A summary of the results is presented in Figure 33. As with Concept 1, the engine fuel flowrates during cruise are sufficient to cool about 50% of the airframe heat load.



Active Cooling System Weight

Component:	Mg	(lbm)
Residual Coolant	9.61	(21,182)
Distribution Lines, etc.	1.46	(3,212)
Heat Exchanger	1.13	(2,501)
Pumps and Pump Fuel Req	3.00	(6,613)
Total	15.20	(33,508)

FIGURE 33
THERMODYNAMIC SUMMARY, CONCEPT 2

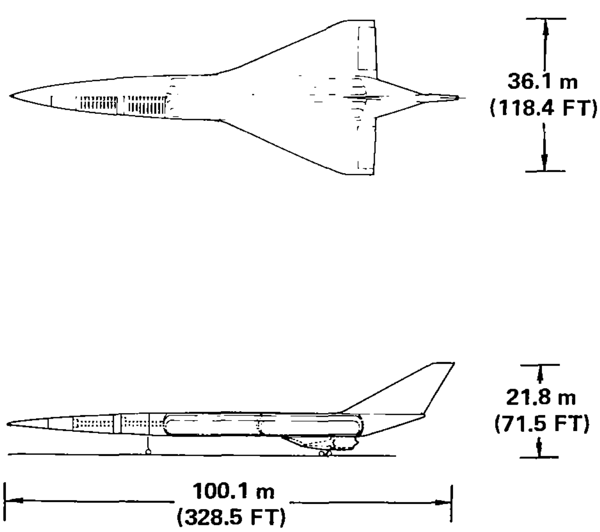
SECTION 7

ELLIPTICAL (BLENDED) BODY AIRCRAFT WITH INTEGRAL TANKAGE (CONCEPT 3) SYNTHESIS

The Concept 3 aircraft, with its blended wing body configuration, while similar in planform, is quite different in cross section from the previous study aircraft. Its elliptical fuselage cross section resulted in a different approach to configuration of the integral fuel tank. It did, however, have the common basis of fuel weight, passenger payload, and propulsion system that were used on Concepts 1 and 2. Tank thermal protection and the active cooling systems for integral tank aircraft (Concepts 2 and 3) were discussed previously. Refinement of the design into the final configuration was accomplished in the same manner as previously described for the Concept 1 and 2 aircraft.

Trade Studies

The baseline (preliminary) Concept 3 characteristics may be found in Figure 34. Payload and fuel weight requirements sized the aircraft, with the aircraft range as a "fall out". Passenger compartment location, elliptically



PRELIMINARY CHARACTERISTICS

O.W.E. = 187.4 Mg (413,204 LBM)

WT_{FUEL} = 108.9 Mg (240,000 LBM)
INTEGRAL FUEL TANKS

PAYLOAD = 21.8 Mg (48,000 LBM)
(200 PASSENGERS)

ENGINE (4) GE5/JZ6-C
T_{SLS} = 400 kN (90,000 LBF)
UNINSTALLED PER ENGINE

HOT NACELLE STRUCTURE

SPEED M = 6.0 (MAXIMUM)

SURFACE ACTIVELY COOLED
TO 394 K (250°F) MAX

FIGURE 34
CONCEPT 3 BASELINE AIRCRAFT

domed tank ends, and a two compartment fuel tank are samples of items retained from Concept 1. The tank length trade study showed that the two tank arrangement was necessary for CG control and to limit the effect of crash condition pressure heads. The multibubble tanks of Concept 3 retained Isogrid stiffening of the tank walls. The honeycomb construction actively cooled panels were retained but modified to be non-structural. A semi-structural configuration similar to Concept 2 was considered but found to require complex support structure and result in a range deficit. The major additional trade study conducted for Concept 3 involved tank cross section optimization.

This trade study involved tailoring the fuel tank within the elliptical fuselage to obtain maximum aircraft range. Assessment of the range effect utilized the sensitivity curve of Figure 35 developed specifically for Concept 3. This was accomplished by comparing structural containment efficiency (weight of fuel/weight of containment structure) and volumetric efficiencies for several combinations of bubble tank configurations as shown in Figure 36. The results of those comparisons indicated that the five bubble cross section is best in terms of structural containment efficiency. Aircraft range levels

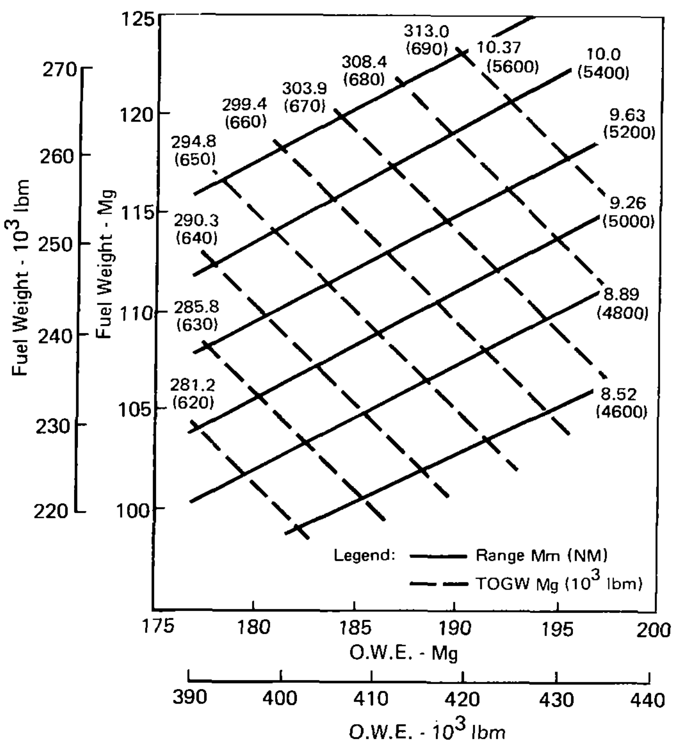


FIGURE 35
RANGE SENSITIVITY, CONCEPT 3

fall off for any bubble number above five, as indicated in Figure 36. The five bubble configuration was selected on the basis of being a fighter and more easily fabricated configuration, than the seven bubble configuration.

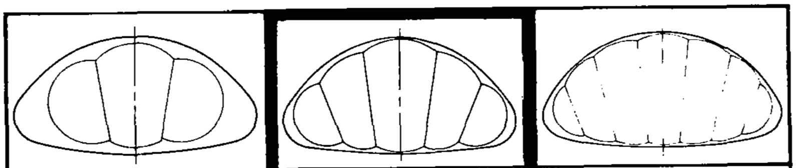


FIGURE OF MERIT	THREE BUBBLE	FIVE BUBBLE	SEVEN BUBBLE
CROSS-SECTIONAL AREA UTILIZATION	73%	90%	91%
WEIGHT EFFICIENCY WEIGHT FUEL / WEIGHT STRUCTURE	17.6	19.6	18.2
FABRICATION COST	LOW	MODERATE	HIGH

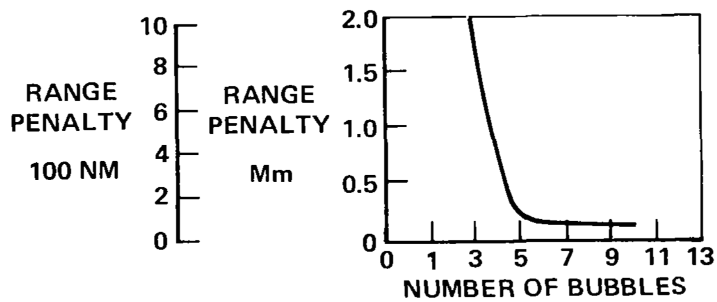


FIGURE 36
TANK CROSS SECTION SELECTION

Structural Analysis

The structural arrangement for Concept 3 is shown in Figure 37. The process of structural weight refinement previously described was also followed for Concept 3. Design loads, developed for the Concept 3 aircraft, are presented in Figures 38 and 39.

a. Finite Element Computer Model - The Concept 3 structural model is illustrated in Figure 40. 1016 joint degrees of freedom are used with this model. The resizing routine described previously was also employed for the Concept 3 analysis.

b. Tank - Detailed analysis of the Concept 3 tank was conducted in a manner similar to that described for Concept 2. However in this analysis the fuselage covering was considered to be completely non-structural. This analysis resulted in a tank weight of 14.54 Mg (32,047 lbm) compared with the

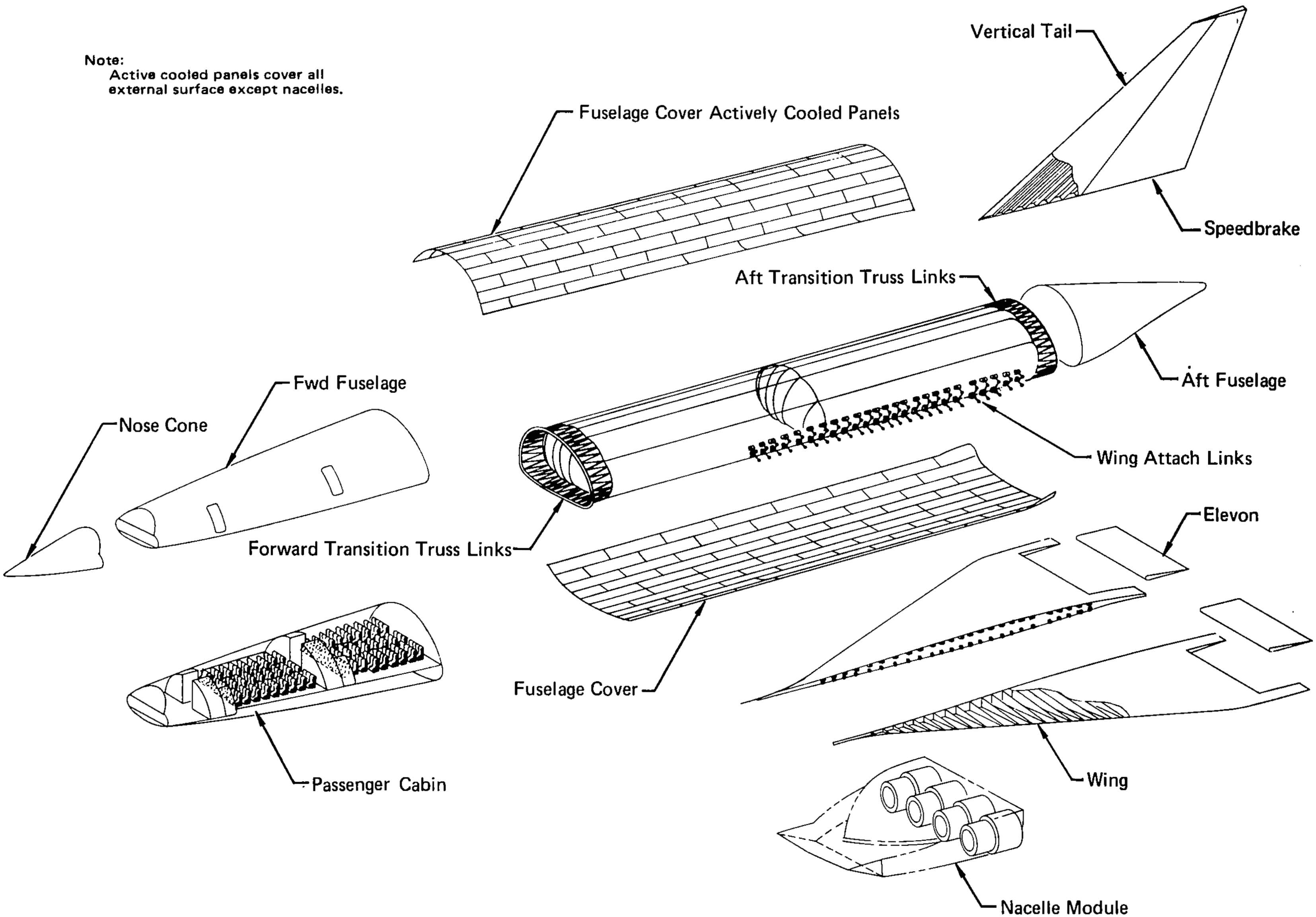


FIGURE 37
CONCEPT 3 STRUCTURAL ASSEMBLY BREAKDOWN

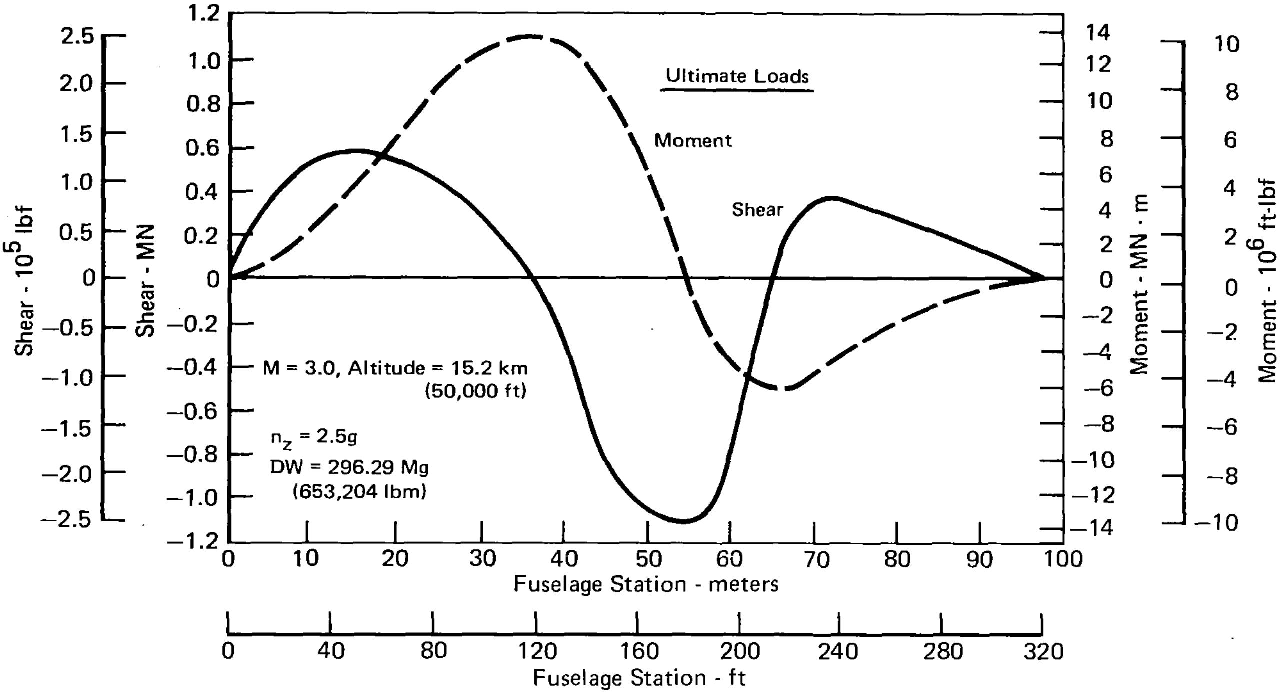


FIGURE 38
CONCEPT 3
Net Aircraft Shear and Moment
2.5g Flight Condition

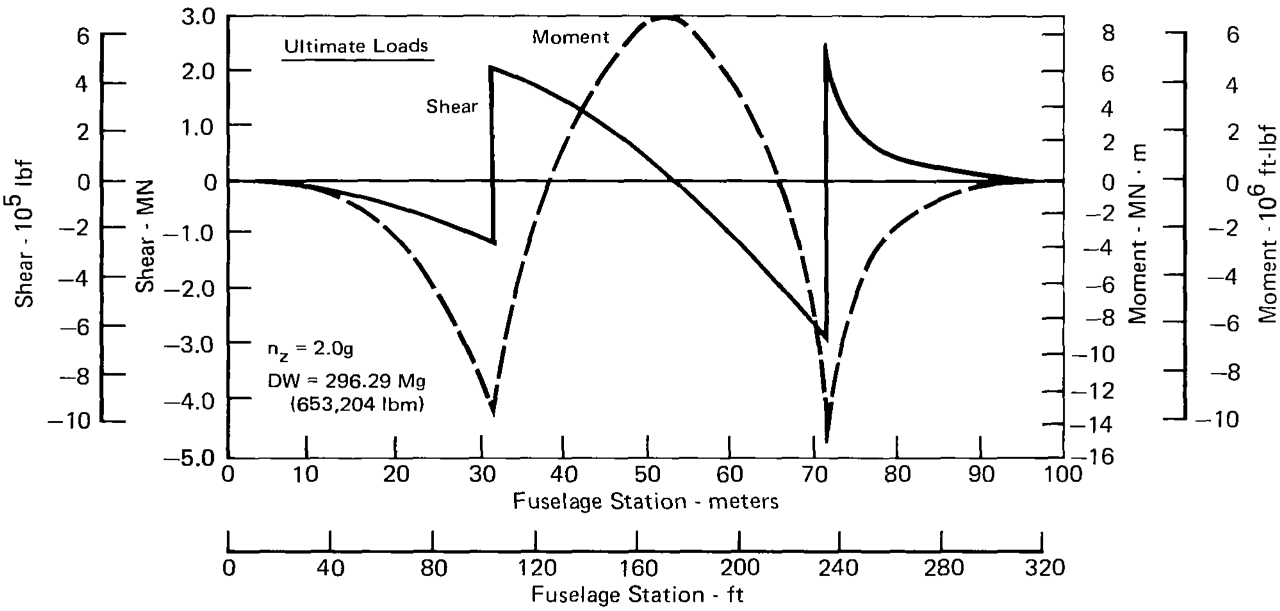


FIGURE 39
CONCEPT 3
Net Aircraft Shear and Moment
2g Taxi Condition

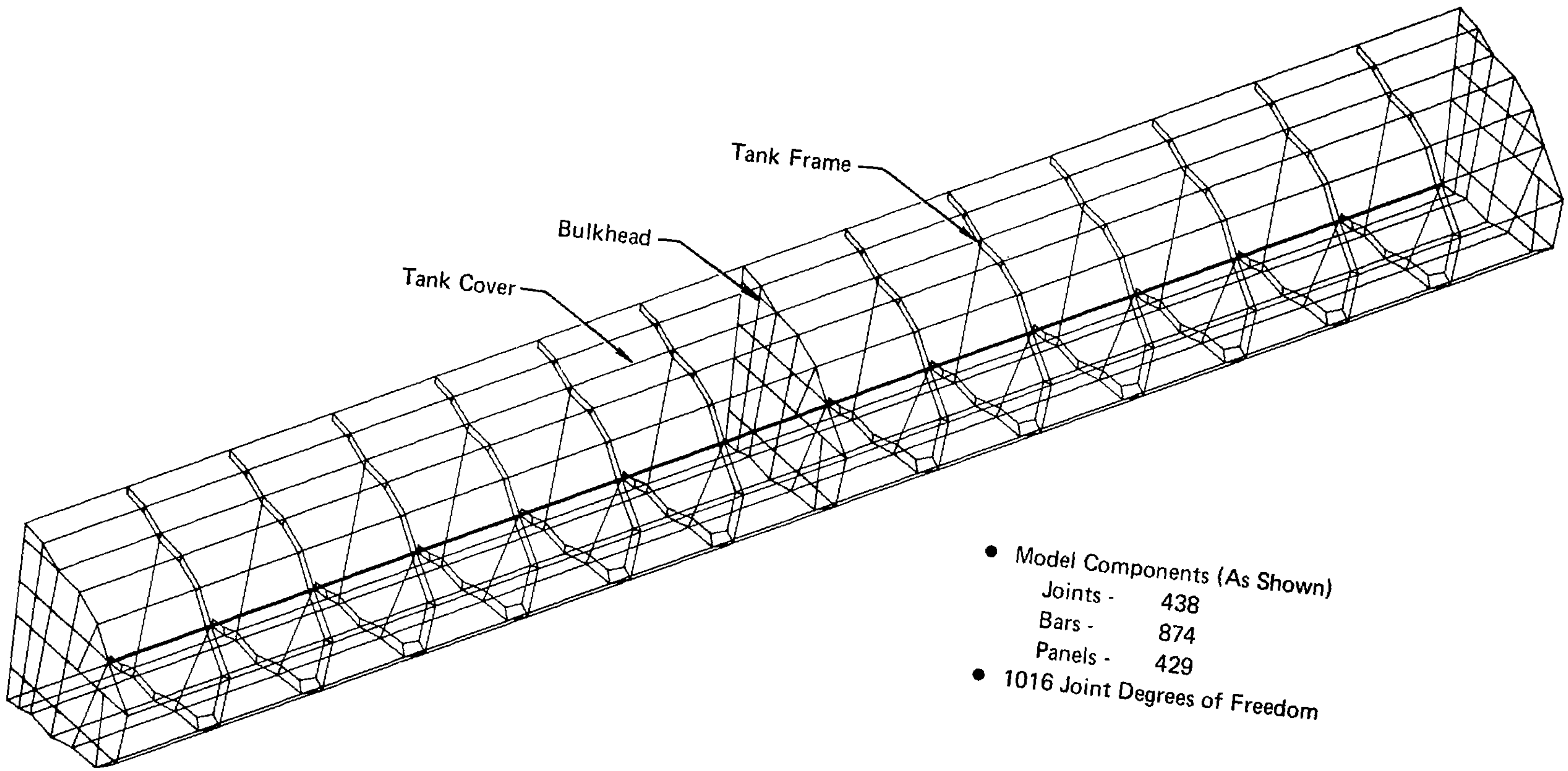
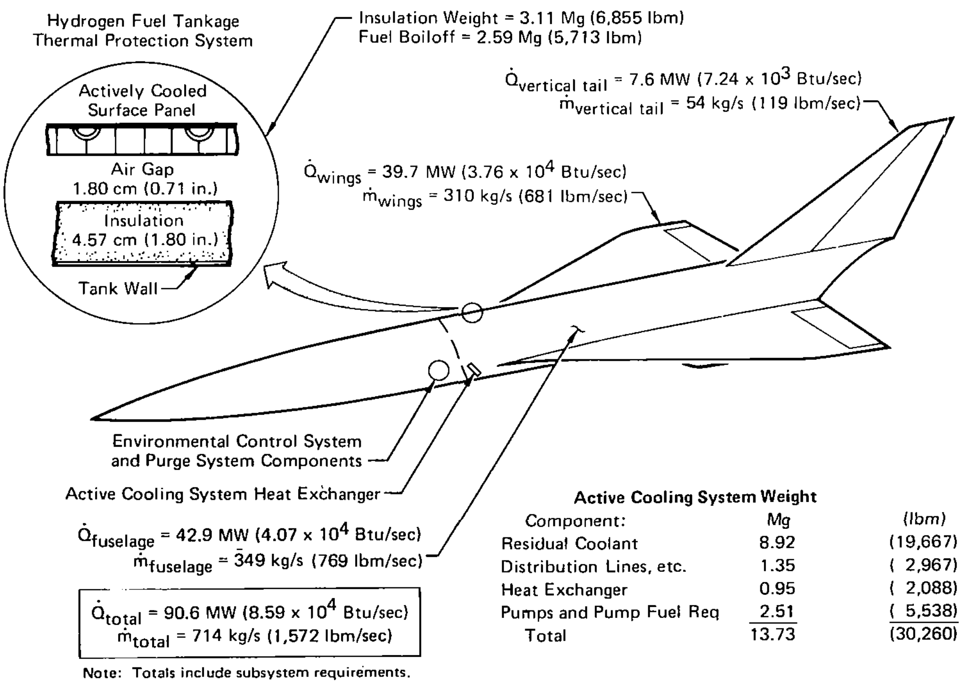


FIGURE 40
CONCEPT 3 FINITE ELEMENT COMPUTER MODEL
Fuselage/Tank Area

initial estimate of 10.57 Mg (23,292 lbm). A large amount of the weight increase in this tank structure is attributable to the rings which act as the wing carrythrough. A slight amount of weight, 213 kg (470 lbm) was added to the tank for fatigue considerations in the tank rings and no weight addition was required for fracture mechanics.

Cooling System Analysis

The procedures used to analyze the Concept 3 cooling system were similar to those described for Concept 1. However, Concept 3 is a smaller aircraft and its moldline contours are different. These differences resulted in lower airframe heat loads and coolant flowrate requirements. Due to the significantly different fuselage shaping and volume utilization, the primary cooling system components were relocated to an area forward of the tankage. This non-centralized location resulted in a weight penalty due to larger distribution lines. These results are summarized in Figure 41. Similar to the other air craft concepts, the engine fuel flowrates during cruise are insufficient for cooling the entire airframe. It was estimated that the fuel heat sink capacity for Concept 3 is also approximately 50% of that required.



Active Cooling System Weight

Component:	Mg	(lbm)
Residual Coolant	8.92	(19,667)
Distribution Lines, etc.	1.35	(2,967)
Heat Exchanger	0.95	(2,088)
Pumps and Pump Fuel Req	2.51	(5,538)
Total	13.73	(30,260)

FIGURE 41
THERMODYNAMIC SUMMARY, CONCEPT 3

SECTION 8

FINAL AIRCRAFT CHARACTERISTICS

The tradeoff studies and design refinement process resulted in final design and performance characteristics of the three concepts studied. These results are summarized in Figure 42. The lowest weight and best range is achieved with the blended body integral tank aircraft, Concept 3. Group weight statements for each aircraft are presented in Figure 43.

Analyses were conducted to determine the relative producibility and serviceability of the three study aircraft. Although these analyses were greatly simplified, their relative values are believed to be accurate. Each analysis concentrated on those areas where structural and arrangement differences were significant.

Producibility analyses, summarized in Figure 44, included both material and labor. Differing items such as the center fuselage cover, tanks and wing attachment were considered in some depth while common items, such as the forward fuselage, wing and tail, were included in a producibility factor which

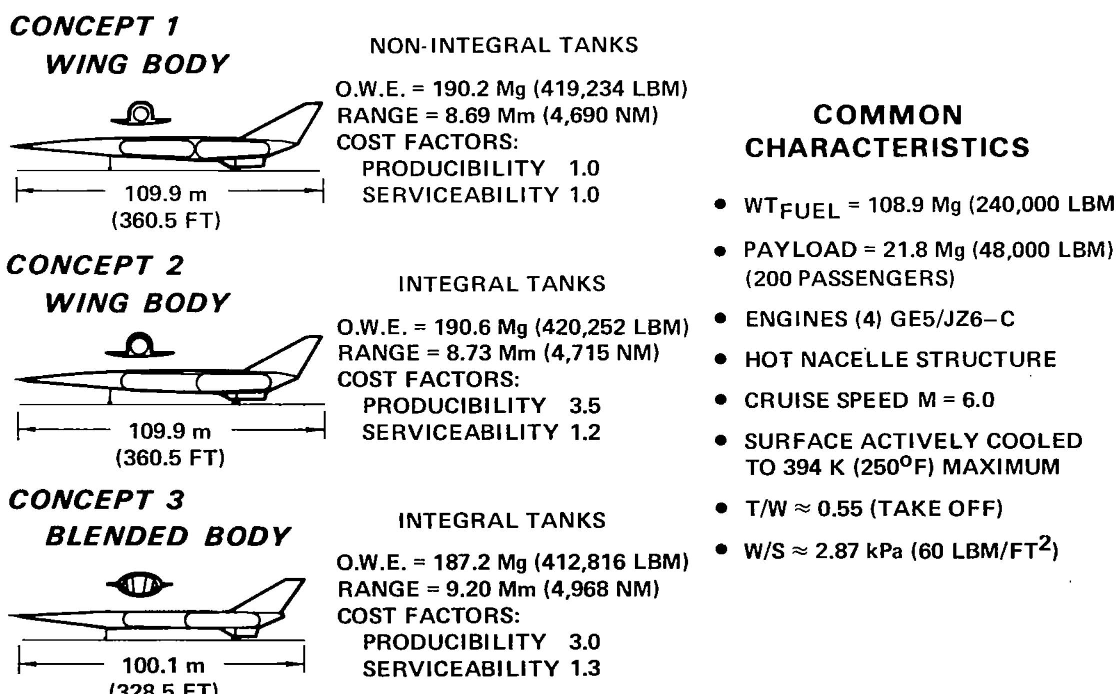


FIGURE 42
FINAL AIRCRAFT CHARACTERISTICS

	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)

FIGURE 43
FUSELAGE/TANK-GROUP WEIGHT STATEMENTS

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

FIGURE 44
RELATIVE COST RATIOS

was common to all three aircraft. Relative cost ratios are presented for the fuselage/tank area and the total aircraft and provide a measure of the relative fly-away cost for each concept. Since all three aircraft concepts carried the same fuel load and number of passengers, an indication of the relative operational costs may be attained by comparing aircraft range as well as the serviceability factors discussed below. These ratios show Concept 1 to have the lowest production cost. A similar study for Concepts 2 and 3, replacing integrally stiffened tank walls with plain skin monocoque walls, resulted in reduction of total vehicle cost factors from 3.5 to 1.6 for Concept 2 and from 3 to 1.8 for Concept 3.

Serviceability analysis also concentrated on those differences in aircraft arrangement which significantly affected maintenance actions. The net result of these analyses given in Figure 45 indicated Concept 1 to be the most easily maintained with Concepts 2 and 3 being more difficult by factors of 1.2 and 1.3 respectively.

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

FIGURE 45
RELATIVE SERVICEABILITY FACTORS

SECTION 9

OBSERVATIONS AND CONCLUSIONS

This report has presented the results from a structural design study of actively cooled, hydrogen fueled, hypersonic transport aircraft. In addition to the final design and performance description of the three aircraft concepts studied, detailed design trade studies and sensitivity studies were performed in the early phases. Simplified maintainability and producibility studies were also conducted. Thus, a broad basis is available for assessing the important design factors which must be considered for this class of aircraft.

The final aircraft characteristics, weights, and relative cost and serviceability factors were presented in Section 8. From Figures 42 and 43, it is seen that Concept 3, the blended body, integral tank aircraft has the lightest weight and the greatest range capability, (over 0.47 Mm (250 NM) more than the others). This superior performance is a result of the better aerodynamic characteristics and higher volumetric efficiency.

The structural weight differences between integral and non-integral tankage are small. This was examined in some detail in order to understand the factors which influence this finding.

Figure 46 shows a typical cross section through the fuselage/tank structure for each aircraft. In all cases the cooled wall is made of aluminum honeycomb panels with "dee" shaped cooling passages bonded to the outer skin. These panels absorb the aerodynamic heating. The tank wall, which serves the function of fuel containment and pressure vessel, is monocoque structure for the non-integral concept and isogrid stiffened structure for the integral concepts. The outer wall cooling function and tank wall pressure function are thus common to all concepts. However, for each concept, the fuselage bending function is performed differently and wing loads are carried differently.

In the non-integral Concept 1, the cooled outer wall serves the additional function of providing strength for the fuselage primary structure. This is accomplished by increasing the depth of the honeycomb panel and the thickness of the outer skin. However, these required increases are small, so the cooled outer wall panels are only slightly heavier than those of the integral concepts.

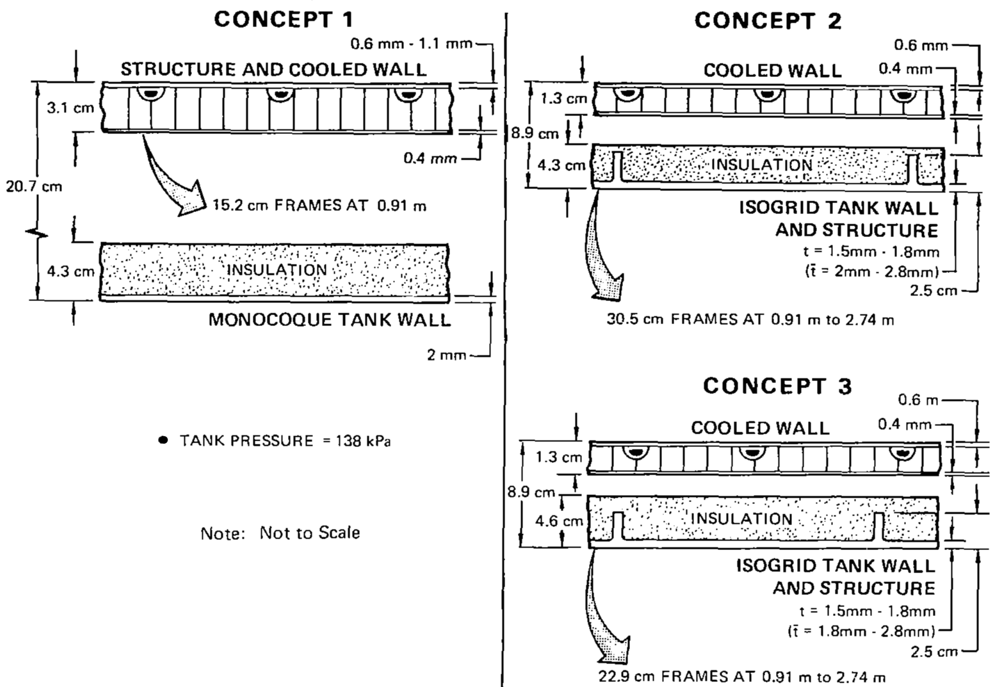


FIGURE 46(a)
STRUCTURAL ARRANGEMENTS
S.I. Units

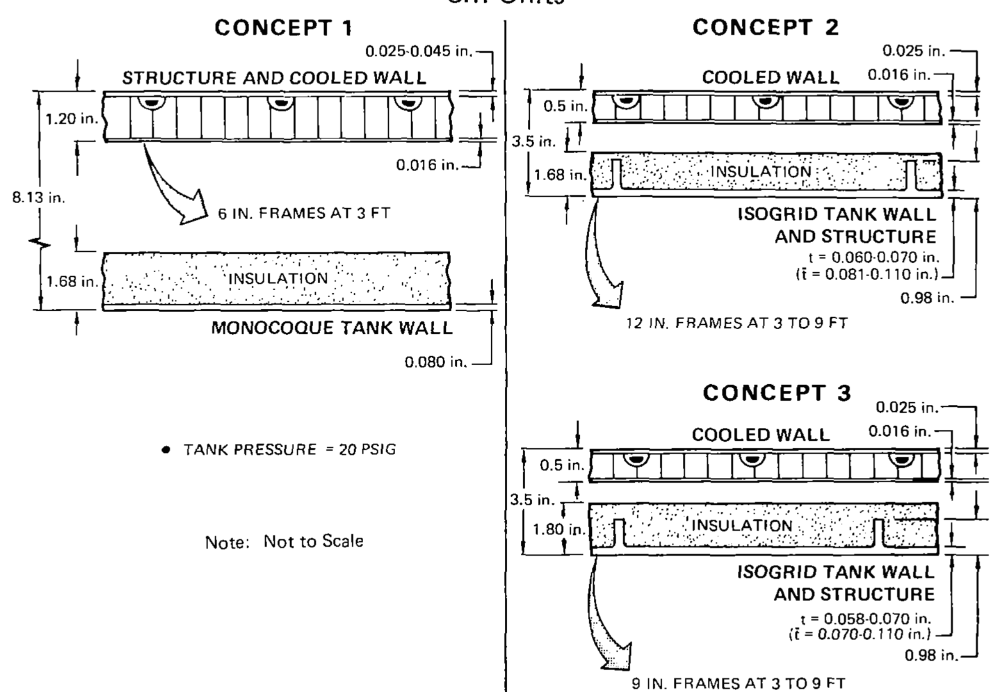


FIGURE 46(b)
STRUCTURAL ARRANGEMENTS
Customary Units

For the integral tank concepts the tank wall provides fuselage primary structure strength using a stiffened skin approach. Of the stiffened design candidates analyzed, an isogrid stiffening pattern produced the lightest weight. If the volume of the stiffening members is averaged over the surface, the equivalent weight thickness ($\bar{t}$ of Figure 46) is only slightly greater than that for the non-integral monocoque tank wall. Thus, the stiffened tank wall is only slightly heavier than the unstiffened non-integral tank wall.

The combination of outer panel weight plus tank weight results in approximately the same structural weight for each of the three aircraft concepts. This result appears to be strongly influenced by the fact that actively cooled structure is used.

While not proven by this study it appears that use of actively cooled structure favors the non-integral structural concepts. Consider the complete structural system of the non-integral Concept 1. It essentially has two structural elements. The outer element serves as a heat shield as well as primary structure. The inner element contains the fuel. The integral Concepts 2 and 3 also have two structural elements. The outer element serves as a heat shield and the inner element serves the dual functions of primary structure and fuel containment. Consider the situation if a non-actively cooled structure were employed, say a radiation heat shield/insulation system. Then the non-integral Concept 1 would require three structural elements: an outer heat shield/insulation, a primary structure and an inner fuel container. However, the integral Concepts 2 and 3 would still only require two structural elements. Thus it appears logical that the addition of a third structural element to the non-integral concepts would result in an incremental weight penalty which is not present for the actively cooled structure.

The small differences in structural weight are also influenced by the type of structure selected, the tank design pressures, the material properties of the tanks at cryogenic temperature and the very large size of the circular fuselage. Change to any or all of these elements could change the results. For example, the large size of the vehicle is a dominant factor in allowing the usage of monocoque structure for the non-integral Concept 1. If the tank diameter were cut in half, the wall thickness needed to sustain pressure would also be halved. The equivalent bending strength would, however, be reduced by a factor of eight. To regain the necessary bending strength, substantial stiffening would be required. Thus, tank pressure and size are extremely significant elements in driving the comparative results.

The principal aircraft performance results are plotted on Figure 47, to provide a quick visual comparison of the three aircraft concepts studied. Considering the effect of tank construction, it is seen that there is very little difference between the circular body non-integral (Concept 1) and circular body integral (Concept 2) aircraft. Each results in comparable range and TOGW. However, when considering the effect of body shape there is a clear advantage for the integral tank blended body (Concept 3) over the integral tank circular body (Concept 2). As stated previously, this is due to the improved aerodynamic characteristics and better volumetric utilization. To provide a basis of comparison between Concepts 2 and 3, it is seen that by off loading 4.54 Mg (10,000 lbm) of fuel, the Concept 3 range becomes the same as Concept 2, but TOGW is significantly reduced.

In summary, the results of this study are judged to have identified significant technical factors relating to integral and non-integral tank designs, in addition to providing performance effects of body shape.

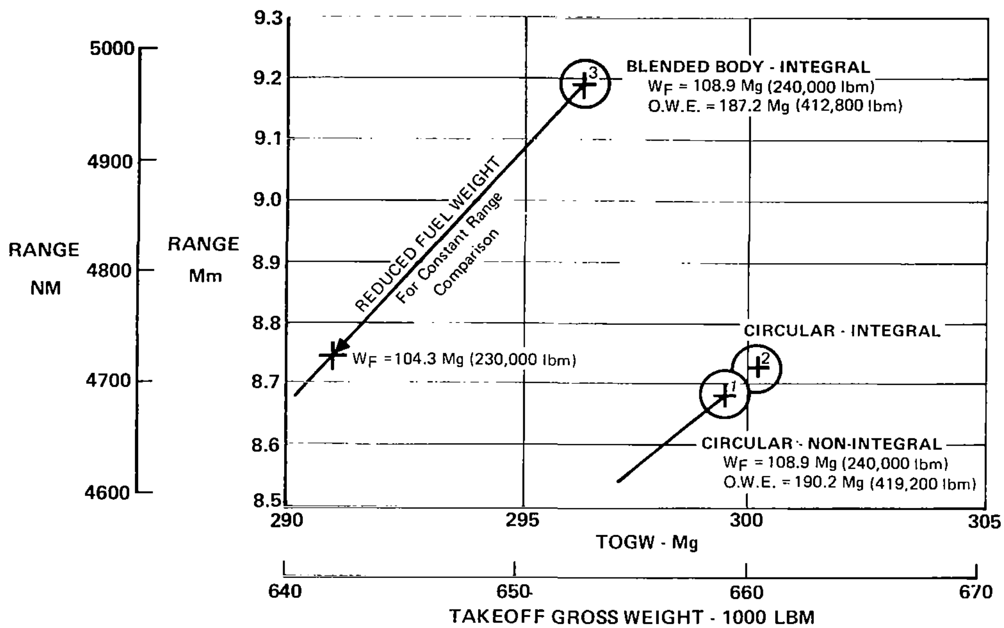


FIGURE 47
BEST PERFORMANCE WITH BLENDED BODY

The principal overall study conclusions are:

a. Integral vs Non-Integral Tanks

   1. There is very little weight difference for circular body concepts.

   2. Producibility and maintainability factors favor the non-integral tank concepts.

   3. Multi-bubble configurations require more welding and assembly time and thus are more costly to produce than circular tank configurations.

b. Circular vs Blended Body Shape

   1. Higher L/D and greater volumetric efficiency are achievable with the blended body.

   2. Better performance (i.e., greater range for the same weight) can be achieved with the blended body shape.

c. Volumetric Efficiency

   1. The efficient utilization of all available volume is a dominant factor in maximizing performance. This was shown in a number of trade studies as well as in the comparison of the circular and blended body shape. The studies of passenger compartment arrangement and location, along with studies of dome shape and number of tanks, all showed significant performance improvements when volumetric utilization was increased.

   2. Design compromises which increase fuel volume can improve aircraft range even though the compromise results in increased structural weight. The range sensitivity curves for Concept 1 and 2 show that range will be increased if for every additional unit of fuel weight less than 1.6 units of structural weight are added. Similar values for Concept 3 are 1.9 units of structural weight per unit of fuel weight. Thus the importance of achieving high fuel volumetric efficiency is again evident.

d. Structural Design

   1. Monocoque tank walls provide an attractive fabrication approach from a producibility standpoint. However, they may result in increased weight and decreased range, depending upon mission requirements. For Concept 2, use of a monocoque rather than

stiffened tank wall results in a 71% increase in tank weight and a range loss of about 463 km (250 NM). For Concept 3, the figures are 16% and 185 km (100 NM).

2. The monolithic non-integral tanks were designed to burst pressure requirements and not limited by fatigue or fracture mechanics considerations. However, the stiffened integral tanks were critical at the tank frames for fatigue design, although no weight was added for fracture mechanics. Thus for the design service life (10,000 hr) used in this study fatigue and fracture mechanics requirements generally had negligible performance effects. However, for a higher service life requirement these considerations would be more significant.

3. Vehicle size and tank design pressures had a strong influence on the results. The combination of these two factors allowed the non-integral tank to be of monocoque design and still be at minimum weight. If tank pressures and/or the vehicle size were reduced, the minimum weight design would probably have been a stiffened structure design.

4. Provisions for gaseous N_2 purging between the panels and the tankage can be met with minimal structural weight addition.

e. Thermal Design

1. The engine fuel flow rate demands during cruise are 52% to 62% of those that would be required if the entire airplane surface exclusive of the nacelle were cooled.

2. The use of internal insulation in the tanks was uncompetitive, because of GH_2 permeation of the insulation. Covering the insulation with a gaseous hydrogen vapor barrier would have distinct advantages. Thermal protection weight and volume requirements would be reduced and structural design problems would be minimized without the need for thermal expansion allowances.

3. Trajectory tailoring to minimize peak transient heating loads is important in minimizing the size and weight of the cooling system components and the coolant distribution lines.

4. As was found in the nacelle cooling study, designing the regions subjected to high heating rates as hot structure may reduce aircraft weight. It appears that this observation would also apply to remote areas, where active cooling would drive up the size and weight of the coolant distribution system.

f. Fly-away Cost

1. Producibility factors favor the simple monocoque non-integral tank approach.

g. Operational Cost

1. Maintainability factors favor the non-integral concept.
2. Comparison of the three concepts at equal range gives an indication of comparative fuel costs, since the payload was the same for all three concepts. This comparison favors the blended body shape over the circular body shape, since slightly less fuel is used for the same mission.

SECTION 10

CRITICAL AREAS REQUIRING ADDITIONAL RESEARCH AND TECHNOLOGY

Vapor Barriers

The accommodation of thermal strains and the requirement for integrally machined stiffening concepts for the tank walls are the major contributors to the high relative costs of integral tankage. The development of a viable vapor barrier against gaseous hydrogen leakage would allow the practical use of insulation on the inside of the tank wall. This in turn would reduce or eliminate the temperature differential between the tank wall and actively cooled structure, permitting use of lighter and simpler structural concepts. Large structural thermal deflections would not have to be accommodated and volumetric efficiency would be enhanced. It is possible that the honeycomb construction active cooling concept and load carrying tank wall could be combined in a manner to accomplish the load carrying function as well as fuel containment function.

Further, for the structural approaches presented in this report, vapor barriers would result in undegraded insulation properties and hence volumetric efficiency would be enhanced and weight lowered due to reduced insulation requirements.

Vapor barrier research and development therefore offers interesting design options.

Materials

There is currently considerable interest in the use of cryogenic hydrogen for aircraft systems, particularly as an energy conservation measure. Practical construction of tankage for this fuel will depend on extensive new knowledge of material properties specifically suited for this purpose. The structural designer needs material properties and design allowables for cryogenic tankage construction materials. Such tank structures probably will be weldments. Thus welding data, along with fatigue and fracture mechanics data, are needed. Development of light alloy weldable materials would enhance the ability to design reliable lightweight tankage for future hydrogen fueled aircraft.

Radiative Thermal Protection Systems

Studies which match airframe cooling requirements to available fuel heat sink capacity are needed to establish realistic system weights. The potential of combining insulation and active cooling in panel designs should be evaluated. Radiative systems will probably be required over part or all of the surfaces to reduce the total heat load absorbed by the cooling system and match the available heat sink capacity. Such systems would offer the potential of fail safe capability and also could even reduce the total system weight. It appears that the design and development of radiative protection systems will be needed for actively cooled aircraft.

Subcooled LH_2 Fuel

The effect of the tank size and design pressures on establishing the tank wall thickness for Concept 1 was dominant in the study. Use of sub-cooled (or slush) hydrogen would allow a lower fuel system tank pressurization and also reduce the amount of boil-off fuel lost during the mission (both ground operations and flight). Fuel system components for handling sub-cooled hydrogen do not represent major technological advancements. However, the potential benefits should be further investigated to determine the optimum degree of sub-cooling in order that design requirements can be established. Based on these requirements, system design and development investigations would be of value.

Cooling System Optimization

Investigations of several methods that could result in active cooling system weight reductions are warranted. Trade-off studies involving the following considerations are suggested:

a. Reduce coolant flowrate requirements by maximizing allowable surface temperatures and outer skin thermal gradients. - While structural weight penalties may result, the tradeoff with cooling system weight should be clearly established. For example, designing for a 422 K (300°F) maximum temperature rather than 394 K (250°F) would reduce flowrate requirements by nearly 40%. Increasing the allowable skin thermal gradient from 56 K (100°F) to 72 K (130°F) would reduce the number of coolant tubes required by about 10%.

b. Optimize coolant system design pressure. - A system design pressure of 1.03 MPa (150 psi) absolute was chosen for this study. A higher design pressure would permit larger pressure drops in the distribution lines, hence smaller lines and less residual coolant. Obviously, higher design pressures necessitate structural weight increases and increased pumping power. It is estimated that a design pressure between 1.38 MPa (200 psi) absolute and 1.72 MPa (250 psi) absolute, would reduce system weight by about 907 kg (2,000 lbm).

c. Establish the significance of centralizing the location of major cooling system components. As discussed in Section 7, the heat exchanger location for Concept 3 resulted in a significant weight penalty. The penalty involved in providing volume for components at a more favorable location should be assessed.

d. Refine feeder line sizing technique. - Each feeder line could be sized to match the local available pressure drop between the main supply and return lines. This study considered only a constant pressure drop per unit line length in establishing feeder line sizes. It is estimated that, by considering locally higher line pressure drops, a weight savings in the order of 454 kg (1000 lbm) is possible.

e. Consider alternate line routing schemes. - There are an infinite number of possible line routings. While no attempt was made during this study to find an optimum configuration, it seems logical that benefits could be derived.

11. REFERENCES

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