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FINAL REPORT

STUDY OF ACTIVE COOLING FOR SUPERSONIC TRANSPORTS

by G.D. Brewer and R.E. Morris

February 1975

Prepared under Contract NAS 1-13226

for

Langley Research Center

National Aeronautics and Space Administration

by

Lockheed-California Comapny

Burbank, California

A Division of Lockheed Aircraft Company

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FOREWORD

This is the final report of a study of Active Cooling for Supersonic Transports, performed under contract NAS 1-13226 for NASA-Langley Research Center, Hampton, Virginia. The report presents documentation of the substance of the work performed during the six months period, June through October, 1974.

The study was performed within the Science and Technology Branch of the Lockheed-California Company at Burbank, California, under the direction of G. Daniel Brewer as study manager. Robert E. Morris was project engineer. Other principal investigators were:

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SUMMARY

This study was a preliminary evaluation to determine the potential benefits of using the fuel heat sink of hydrogen-fueled supersonic transports to cool large portions of the aircraft wing and fuselage by means of an intermediate fluid such as an ethylene glycol-water solution. Advantages that it was anticipated might accrue to an actively-cooled vehicle included the use of lower cost aluminum in place of titanium structure, reduced cabin heat loads, and more favorable environmental conditions for the aircraft systems.

The two vehicles selected for a comparison of cooled versus uncooled versions both carry a payload of 22,226 kg (49,000 lbs), equivalent to 234 passengers, for 7,778 km (4,200 n. mi.). One was designed to cruise at Mach 2.7 and the other at Mach 3.2. The technology level is that assumed to exist in the early 1980's, to provide an initial in-service date of the early 1990's.

The work reported herein was a preliminary evaluation of a concept which, if judged sufficiently promising, was to be followed by a more comprehensive, rigorous design study. The technical approach which was employed involved establishing the characteristics of uncooled versions of aircraft for each cruise speed. Cooled versions were then generated to provide a basis for gross evaluation of advantages and/or disadvantages of cooling. The LH_2-fueled M 2.7 supersonic transport design from the study performed by Lockheed for NASA-Ames Research Center (Reference 3) was used for the reference uncooled vehicle at that cruise speed. For the Mach 3.2 uncooled reference design, a very quick study was performed to establish an acceptable basis for a quick-look comparison between the cooled and uncooled versions.

The cooled aircraft designs were analyzed to determine their fuel heat sink capability, the extent and location of feasible cooled surfaces, and the coolant passage size and spacing. The basic structural approach which had previously been selected for the uncooled aircraft was found to be well adapted to the incorporation of the coolant passages. The use of coolant allowed replacement of the hot titanium passenger compartment structure (skin, stringers and frames) with cooled aluminum

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since it was strength critical at the cruise temperature. The wing box was critical at low speed (cold) flight conditions and the titanium spar and rib substructure was retained for mininum weight. The cover skins were replaced with cooled aluminum. These structural changes, together with the weight saved in the ECS system and the weight of the coolant system itself, were then the basis for establishing the weight and cost implications of the active cooled versions. The effects of change in vehicle drag due to the cooled structure, the change in specific fuel consumption due to the addition of external heat, coolant pumping horsepower requirements, and excess fuel flow required during deceleration were considered in evaluating performance, weight, and cost of the cooled aircraft.

The final results and comparison of the aircraft are tabulated below:

		Mach 2-7		Mach 3-2	
WEIGHT DATA		Uncooled	Cooled	Uncooled	Cooled
Gross Weight	kg.	163,783	163,615	198,493	194,567
Operating Empty wt.	kg.	99,279	96,166	127,223	124,000
Structural wt.	kg.	57,500	56,700	78,300	75,100
Cooling system wt.	kg.	-	1,273	-	2,152
ECS system wt.	kg.	3,574	2,907	4,658	2,952
ALUMINUM UTILIZATION					
(% of wing and fuselage structure)		18.7	48.4	14.2	45
FUEL HEAT SINK UTILIZED - %		-	61	-	100
COST DATA					
RDT & E	$bil	3.28	3.42	4.72	4.84
Production Price	$mil	47.04	45.50	59.09	55.33
DOC	¢/AS km.	.941	.944	1.025	.992
ROI - % (After taxes)		7.01	7.02	3.80	4.97

The results of this preliminary analysis of the feasibility of actively cooling LH_2-fueled supersonic transport aircraft at two cruise speeds are summarized as follows:

Mach 2.7 Aircraft:

- The increase in usage of lower cost aluminum from 18.7 to 48.4 percent of the wing and fuselage structure allowed a price decrease of 3.7 percent at approximately the same gross weight.
- The cause of the slight increase in DOC of the cooled version was the increase in maintenance cost of the coolant system. As described in Section 4.7, this was estimated to be equivalent to a 25 percent increase in system maintenance or a 6 percent increase in total maintenance. Should no maintenance costs result, the DOC would be 1.727¢/AS nm or 1.3 percent lower than the uncooled aircraft.
- Since the cooled aircraft used only 61 percent of the available heat sink, more area could be cooled. This would involve diminishing returns however, because such surfaces (tail, flaps, ailerons, crew compartment) are either remotely located or involve complex plumbing connections, resulting in sizeable increases in coolant system and fluid weight.

Mach 3.2 Aircraft:

- The increase of aluminum utilization from 14.2 to 45 percent of wing and fuselage structure, together with the reduction in gross weight, allowed a price decrease of 6.4 percent for the cooled version.
- The DOC of the cooled aircraft is 3 percent less than that of the uncooled, with the increased maintenance cost of the cooling system balanced by reduced maintenance costs for the other systems permitted by the lower environmental temperatures. Should no maintenance costs result, the DOC would be 1.816¢/AS nm or 4.2 percent lower than the uncooled aircraft.
- Since the Mach 3.2 aircraft used 100 percent of the heat sink capability, no further area can be cooled. In fact, a slight reduction in cooled wing surface area, relative to the Mach 2.7 was required to meet this limitation.

General Conclusions:

Within the limited scope and ground rules of this study, no significant economic advantage was found for active cooling in the Mach 2.7 transport and only a slight advantage for the Mach 3.2.

The use of an active cooling system in a commercial transport operating environment requires consideration beyond that possible in this study as to what impact the system might have on maintenance costs, flight safety and dispatch reliability.

While the advantages of cooling were found to be marginal at Mach 2.7 and 3.2, it is significant that the trend shows increasing weight and economic benefits at the higher Mach number as the allowable stress levels decrease with higher strucural temperatures. This suggests that because of the trend toward lower L/D and increasing specific fuel consumption with Mach number, higher speeds will provide increasing fuel heat sink to maintain the required surface temperature as the heating load increased. Thus the greatest potential for active cooling will be at hypersonic cruise speeds, in particular the Mach 6-8 regime where scramjet propulsion is attractive and expensive super alloys at reduced allowables must be used if no cooling is employed.

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TABLE OF NOMENCLATURE

AR	=	Aspect Ratio
AST	=	Advanced Supersonic Technology
ATA	=	Air Transport Association
α_{FRL}	=	Angle of Attack - Fuselage Reference Plane
α_{WPR}	=	Angle of Attack - Wing Reference Plane
BL	=	Buttock Line
C_D	=	Drag Coefficient
C_{D_F}	=	Friction Drag Coefficient
C_{D_L}	=	Induced Drag Coefficient
C_{D_K}	=	Wing Camber Drag Coefficient
$C_{D_{TRIM}}$	=	Trim Drag Coefficient
C_{D_W}	=	Zero Lift Wave Drag Coefficient
$C_{D_{WING}}$	=	Drag Coefficient - Wing
C_f	=	Skin Friction Coefficient
C_L	=	Lift Coefficient
C_{L_K}	=	Lift Coefficient for Minimum Drag
DOC	=	Direct Operating Cost
DBTF	=	Duct Burning Turbofan
Δ	=	Increment
$\delta_{TE,\ LE}$	=	Flap Deflection - Trailing Edge or Leading Edge
ECS	=	Environmental Control System
FAR	=	Federal Air Regulation
FN	=	Installed Net Thrust

TABLE OF NOMENCLATURE (Continued)

IOC	=	Initial Operational Capability
K	=	Induced Drag Parameter
KEAS	=	Knots Equivalent Air Speed
LH_2	=	Liquid Hydrogen
L/D	=	Lift/Drag Ratio
M	=	Mach Number
M_{DES}	=	Design Mach Number
N_Z	=	Load Factor, Z Axis
ROI	=	Return on Investment
SFC	=	Specific Fuel Consumption
S_W	=	Wing Area
t/c	=	Wing Thickness Ratio
T/W	=	Thrust-to-Weight Ratio
W/S	=	Wing Loading (Weight/Area)

1.0 INTRODUCTION

This is the final report of a study performed by the Lockheed-California Company for NASA-Langley Research Center. The NASA Request for Proposal RFP1-12-4302, "A Study of Active Cooling for Supersonic Transports," dated April 1, 1974, sought a preliminary evaluation of the potential benefits of actively cooling the skin of liquid hydrogen fueled supersonic transports. The following were considered to be the principle areas of potential improvement:

- Lower structural temperatures would allow the use of aluminum with boron/epoxy reinforcement in place of titanium with boron/polyimide. This could result in lower development, material and fabrication costs.
- The addition of external heat to the hydrogen fuel would increase its enthalpy which would allow a lower fuel flow rate to maintain the same thrust level or engine temperature limit.
- Cooled vehicle external surfaces could reduce the weight and complexity of the environmental control system. In addition, the environment for hydraulic lines and equipment, brake fluid, and other subsystems would be improved, thereby also leading to reduced costs.
- Lower structural weights, lower SFC, and smaller, lighter components could allow iterative reduction of the vehicle gross and inert weights and lead to further cost savings.

The objective of this study (Contract NAS 1-13226) then was to provide a first-order comparison of weight, cost and performance of uncooled versus actively cooled airframes for two liquid hydrogen-fueled advanced supersonic transports; one designed to cruise at Mach 2.7 and the other at Mach 3.2. Since this initial evaluation was intended merely to provide guidance for determining the course of future effort, the effort was deliberately cursory in nature, planned to explore the basic elements of the problem just to the depth necessary to provide quantitative answers to the questions:

- is it feasible to actively-cool aluminum-skinned M 2.7 or M 3.2 LH_2 fueled supersonic transport aircraft, and, if the answers were both affirmative;
- which design cruise speed offers the most advantage in terms of cost, weight, and specific energy consumption?

If the results were sufficiently encouraging, it was intended that a more rigorous analysis of supersonic transport designs for selected cruise speeds would be performed.

All computations in this analysis were performed in customary English units and then converted to SI units.

2.0 BACKGROUND

Studies of aircraft over the subject flight speed spectrum show potentially large performance gains for liquid-hydrogen-fueled aircraft versus Jet A-fueled aircraft. In addition, the use of a cryogenic fuel opens up new possibilities for aircraft design through the use of the large heat sink capacity of the fuel. Studies (References 1 and 2) have shown that active cooling of an aluminum airframe for a hydrogen-fueled Mach 6 transport is possible with significant weight and cost reductions over the hot, superalloy structure. Other unpublished calculations at NASA-Langley Research Center indicated that the weight and cost trades could also be favorable for even a Mach 2.7 transport. In addition, it was considered that this tradeoff would be enhanced by the beneficial effect of cooling upon subsystems requirements such as the environmental control system for passenger comfort, etc. The possible gains to be made were sufficiently promising that this study was authorized to investigate the potential of airframe cooling for advanced supersonic transports.

3.0 TECHNICAL GUIDELINES

An existing design for the Mach 2.7, hydrogen-fueled supersonic transport as described in Reference 3 (slightly modified as described in section 4.2) was used as the uncooled baseline for the evaluation of active cooling at the lower Mach number. For the higher Mach number, the design of a baseline, uncooled hydrogen-fueled transport to cruise at Mach 3.2 consistent with the guidelines outlined in Reference 3 was to be defined in sufficient depth to determine the impact of active cooling on the aircraft. The active cooled aircraft for both cruise speeds were to have the same mission capability, equivalent design allowables, and airframe design as the uncooled aircraft. The structural design criteria for the active cooled aircraft were to meet the same airworthiness standards as the uncooled structure.

The active cooling technology applied to the cooled airframes was to be drawn from the studies summarized in References 1 and 2. These studies indicate that the most attractive cooling system was an internal convective cooling system which uses a secondary fluid (water-glycol) circulated through panel passages to transfer the structure heat load to hydrogen heat exchangers. For the present study it was specified that the contractor consider this system to be off-the-shelf insofar as possible in order to minimize considerations of the airframe cooling system design in the contract; however, innovation on the part of the contractor was not discouraged.

The basic guidelines followed in the design of the aircraft are those of the NASA-Ames study (Reference 3) and are reported below for convenience:

- Fuel - liquid hydrogen, available at airports.
- Planform - NASA Arrow - wing
- IOC - 1990
- Use of advanced materials and technology postulated to be developed by 1981. (Data available from Lockheed AST studies; References 4 and 5).
- Certification - FAR Part 25 and SST White Book
- Noise - FAR Part 36
- Fuel Reserves - FAR Part 121.648
- Runway Length Determination - FAR Part 25 for 305.6°K (90°F) day and 304.8 m (1000 ft) airport altitude.
- Operability - compatible with Air Traffic Control Systems and general operating environment envisioned for 1990, including capability for Category III-A operations.
- Aircraft Service Life - 50,000 flight hours
- Sonic Boom - no boom at ground level over populated areas
- Stability - control configured aircraft
- Cost - production base is 300 aircraft. Use modified ATA formulas for DOC evaluation at passenger load factor = 0.55. Use 1973 dollars.
- Payload - 22,226 kg (49,000 pounds) (234 passengers)
- Range - 7,778 km (4,200 NM)

Further performance constraints placed on the aircraft consist of a maximum takeoff field length of 3,200 m (10,500 ft.) and a maximum landing approach speed Of 82.3 m/s, (160 KEAS.)

4.0 TECHNICAL APPROACH

The study completed by Lockheed-California Company for NASA-Ames Research Center (Reference 3) resulted in definition of a supersonic transport aircraft of advanced design, fueled with liquid hydrogen and designed to cruise at Mach 2.7. The airframe structure is "uncooled", i.e., it is not actively cooled, and is designed to be fabricated basically of titanium reinforced with boron/polyimide. The general characteristics of the airplane are described in Section 4.2. This airplane design was used as the basis for evaluating the potential benefits of an actively-cooled version of an equivalent Mach 2.7 supersonic transport. The actively-cooled aircraft has the same configuration and type of propulsion system as the vehicle from Reference 3. An analysis was made to determine the feasibility of using internal convective cooling to transfer a large part of the aerodynamic heat load to the liquid hydrogen fuel and thus lower the working temperature of the skin and primary structure to the degree that aluminum, suitably reinforced with composites, could be employed as the primary structural material. A convective cooling system using water-glycol as the intermediate coolant which circulates in passages throughout the structure and which ultimately transfers the heat to the liquid hydrogen fuel was used to reduce the temperature of the aluminum skin and structure to acceptable working limits.

In the present study the focus was on determining generally whether active cooling offers potential advantage to the supersonic transport aircraft, as contrasted with the problem of designing specific convective cooling systems for those aircraft. Accordingly, the contractor was directed to use the cooling system technology summarized in References 1 and 2. Conceptual design methods as outlined in following sections were used to establish basis for comparing "cooled" vs. "uncooled" versions of both Mach 2.7 and Mach 3.2 aircraft.

For the Mach 3.2 case, an uncooled version employing composite-reinforced titanium structure was generated first, followed by modification of that design to reflect use of the water-glycol active cooling system to permit use of composite-reinforced aluminum skin and structure.

For purposes of this preliminary analysis a simple modification of the arrow-wing planform used in the Mach 2.7 design was employed to represent the Mach 3.2 aircraft. It was recognized that increasing the leading edge sweep to avoid shock impingement at the cruise condition would lead to low speed lift and control problems. However, it was felt the purposes of the investigation could be served, even though the Mach 3.2 airplane design is not completely verified at all flight conditions. The relative advantages and disadvantages of the cooled vs. The uncooled versions of the configuration could be weighed and evaluated without significant discrepancy. As originally proposed however, in the event the conclusion of this exploratory investigation showed sufficient promise for active cooling, a more rigorous analysis and determination of the characteristics of the Mach 3.2 airplane configuration would be required.

4.1 TECHNOLOGY DESCRIPTION

The technology level of this study was defined as that existing in the early 1980's with an IOC date of 1990-1995. For a complete description of the propulsion, aerodynamic, structures, weights and cost estimation methods used in the generation of the Mach 2.7 uncooled baseline LH_2 AST, see Reference 3. This section describes the aerodynamics and propulsion information developed for the Mach 3.2 aircraft. Weight and cost information are given in Sections 4.5 and 4.6 respectively.

4.1.1 Aerodynamic Data

In general, the characteristics of the Mach 3.2 aircraft were based on the contract work done on the Jet A-fueled Mach 2.2 and 2.7 aircraft for NASA-Langley (Reference 4). The wing camber drag for the Mach 3.2 design has been assumed the same as the Mach 2.7. The following figures for the Mach 3.2 airplane are included and are self-explanatory:

Figure 1	Drag Due-to-Lift Characteristics
Figure 2	Wave Drag Characteristics of Wing
Figure 3	Estimated Trim Drag Increment
Figure 4	Low-speed Drag Polars, Take-off and Landing
Figure 5	Low speed Lift Characteristics - Out of Ground Effect
Figure 6	Low speed Lift Characteristics - In Ground Effect

The total wave drag is dependent on relative fuselage size and nacelle shape and is calculated internally in the Advanced System Synthesis Evaluation Technique (ASSET) computer program as is the vehicle friction drag. Figure 5 shows that for the same tailscrape angle the Mach 3.2 airplane loses approximately 20 percent of the lift coefficient compared to the Mach 2.7 design. This loss is the primary reason for the reduced wing loading and the larger wing of the Mach 3.2 aircraft described later.

4.1.2 Propulsion Data

The engine used in the Mach 3.2 aircraft is a duct-burning turbofan (DBTF) fitted with a variable geometry nozzle incorporating a retractable noise suppressor and a thrust reverser. Turbine nozzle and blade cooling is by means of a closed loop liquid metal-to-hydrogen heat exchanger. Consequently, no cooling bleed-air penalty is required as would be the case with a hydrocarbon-fueled engine. Lockheed generated the cycle optimization data and installed performance using the in-house version of the SYNTHA engine cycle program. The design point characteristics of the

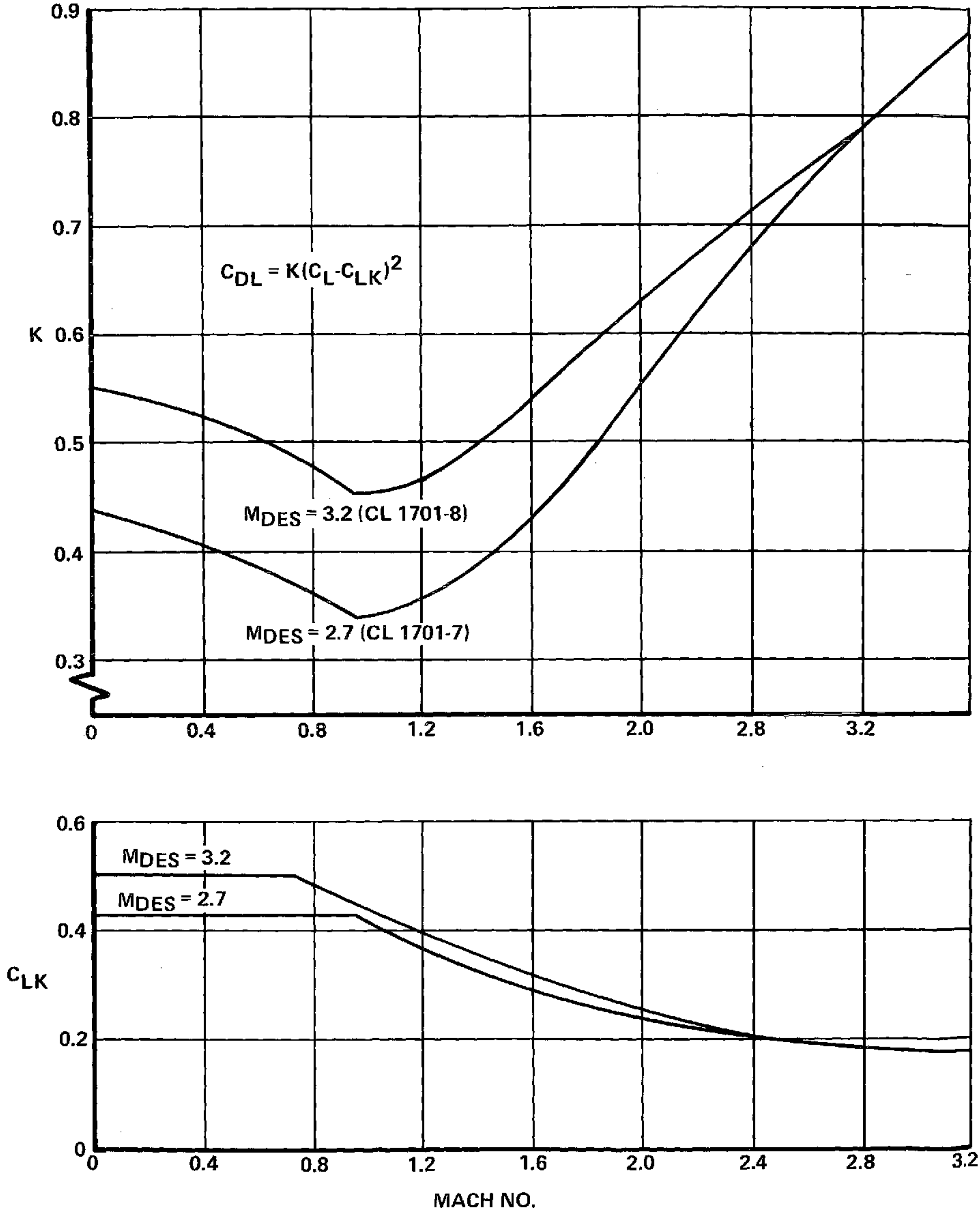

Figure 1. Drag Due-to-Lift Characteristics

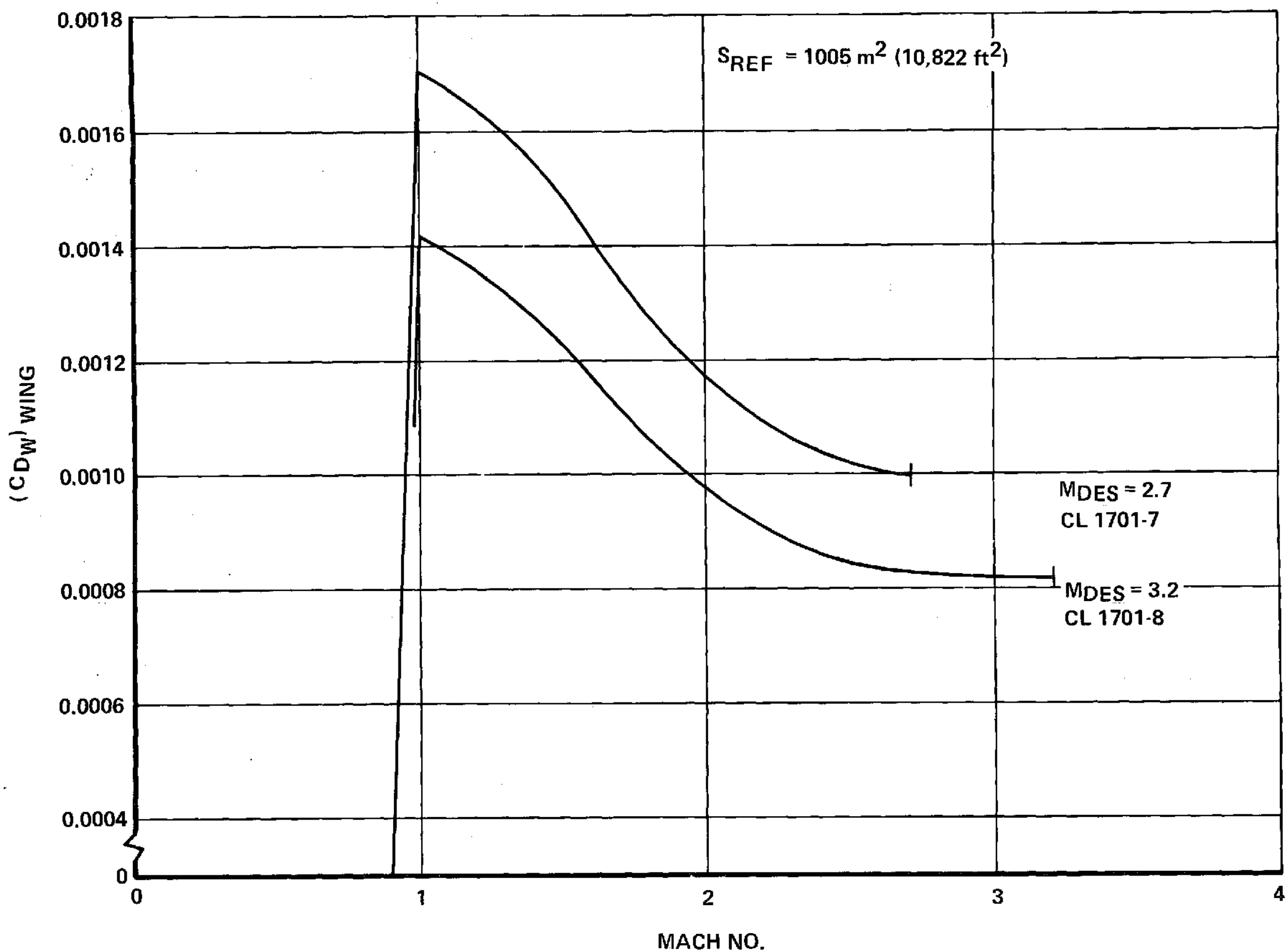

Figure 2. Wave Drag Characteristics of Wing

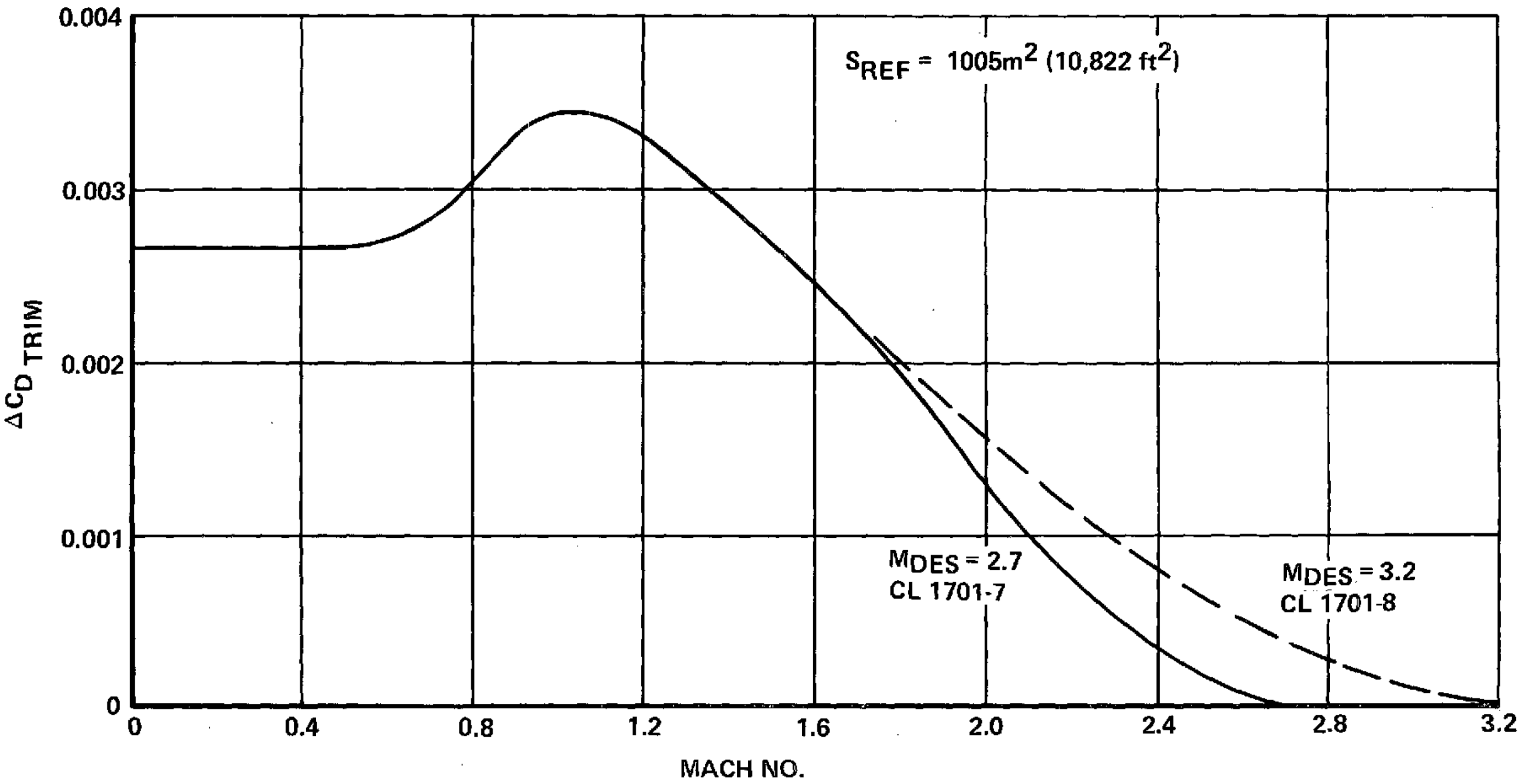

Figure 3. Estimated Trim Drag Increment

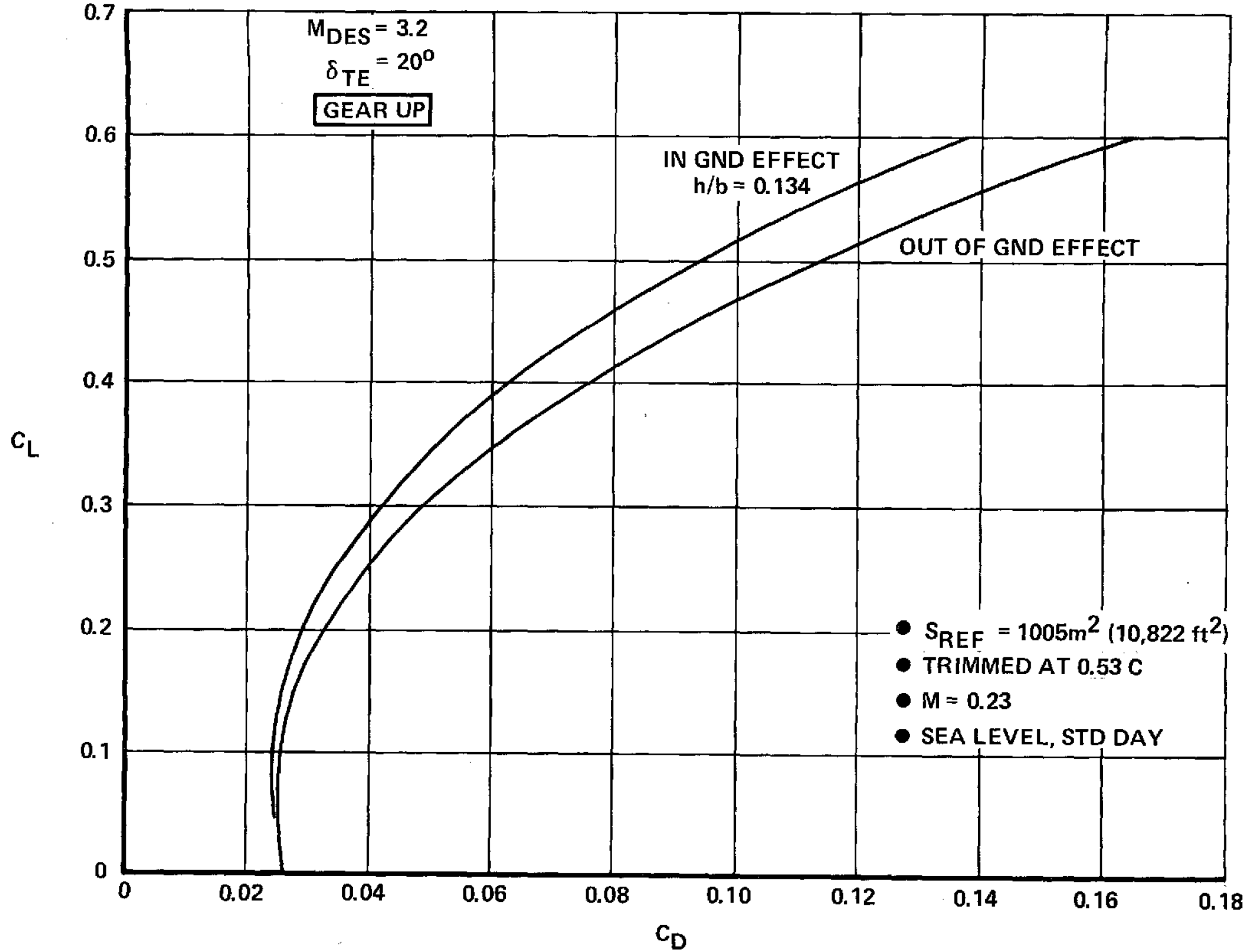

Figure 4. Low Speed Drag Polars, Takeoff and Landing

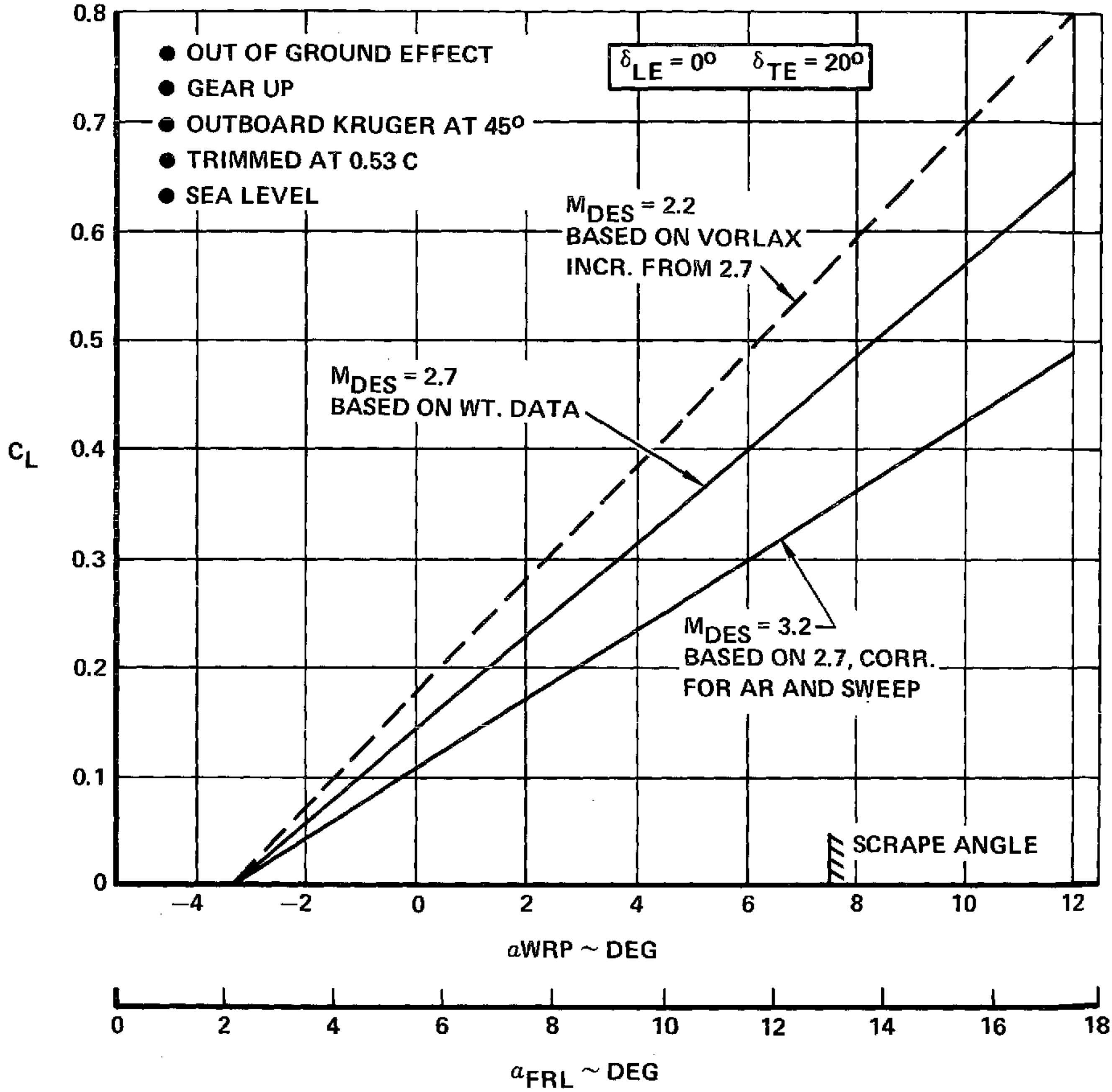

Figure 5. Low Speed Lift Characteristics - Out of Ground Effect

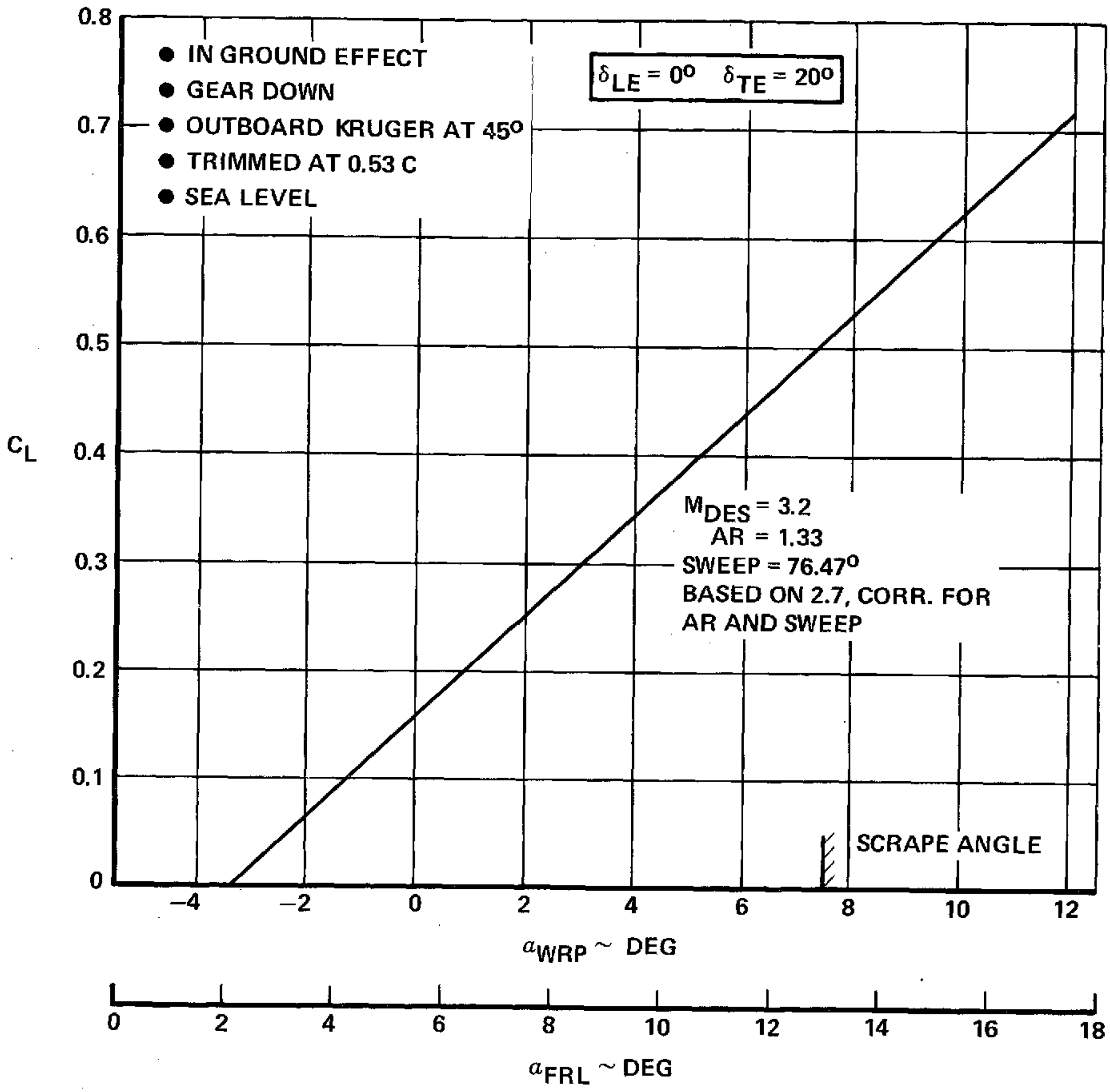

Figure 6. Low Speed Lift Characteristics - In Ground Effect

baseline-size engine are listed in Table 1. The installed performance is shown in Figures 7 thru 12. Installation losses include the effect of inlet recovery and drag, compressor bleed, nozzle losses and horsepower extraction.

TABLE 1. M3.2 LIQUID HYDROGEN DUCT BURNING TURBOFAN BASELINE CYCLE CHARACTERISTICS (SLS, UNINSTALLED)

Engine designation	LH2 TF -2
Engine type	DB TF
Design cruise Mach	3.2
Max thrust	38,100 daN (85800lb)
Specific fuel consumption	0.505 kg/hr daN (0.495 lb/hr/lb)
Corrected airflow	465 kg/Sec (1025 lb/Sec)
Bypass ratio	5.2
Fan pressure ratio	3.0
Fan adabatic efficiency	0.866
Compressor Pressure Ratio	6.0
Compressor adabatic efficiency	0.876
Overall pressure ratio	18.0
Nozzle velocity coefficient (duct)	0.981
Nozzle velocity coefficient (primary)	0 981
Max turbine inlet temperature	$1922^{\circ}K$ ($3460^{\circ}R$)
Max duct burning temperature	$1422^{\circ}K$ ($2560^{\circ}R$)
Fuel heating Value	119430 kJ/kg (51590 Btu/lb)
Peak fan polytropic efficiency	0.9
Peak compressor polytropic efficiency	0.915
HP turbine adabatic efficiency	0.92
LP turbine adabatic efficiency	0.91
Primary burner efficiency	1.0
Duct burner efficiency	0.962
Primary burner pressure loss ratio	0.060
Duct burner pressure loss ratic	0.047
Primary nozzle pressure loss ratio	0.005
Thrust to engine wt ratio	7.3daN/Kg (7.4 lb/lb)

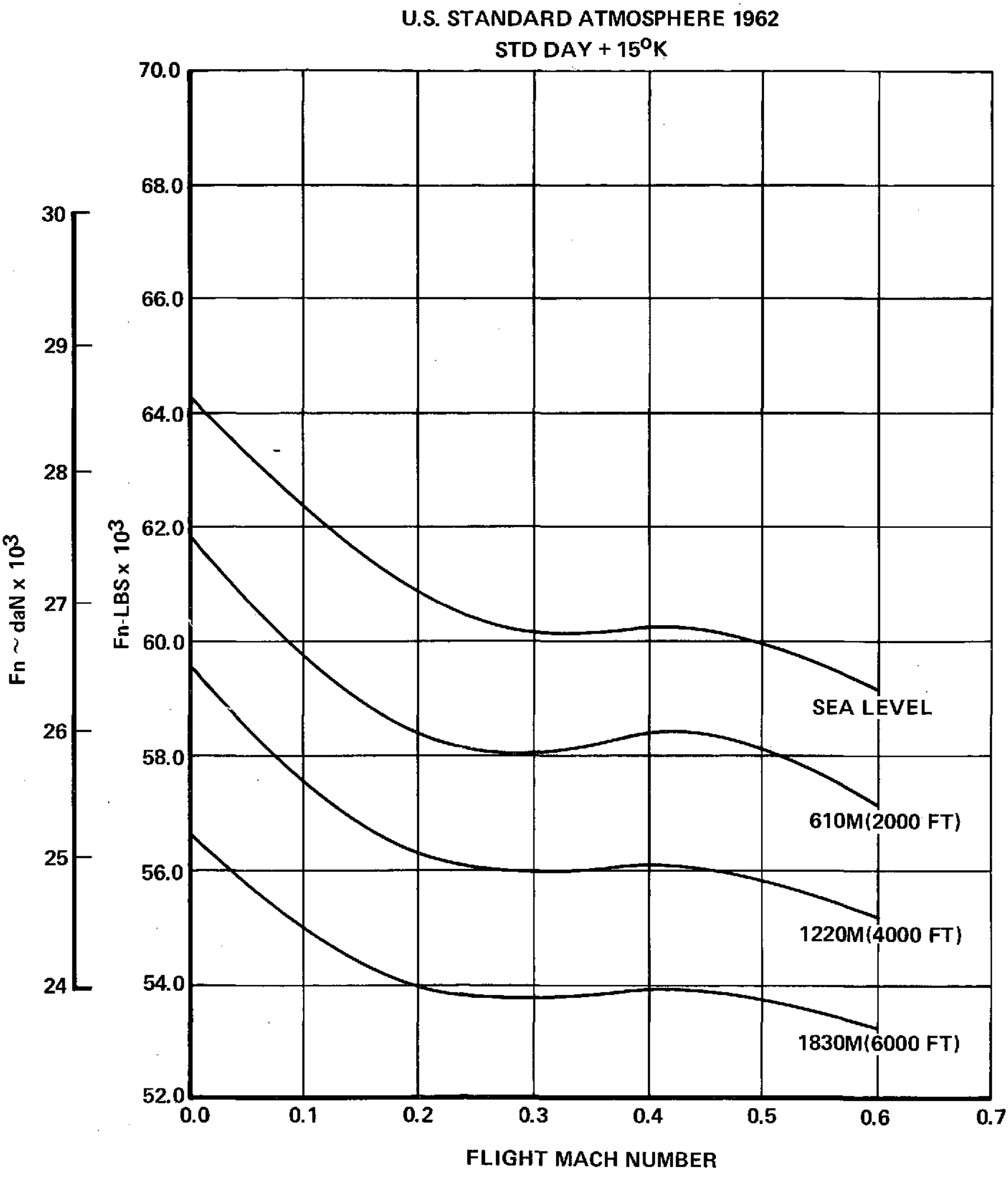

Figure 7. Installed Flight Performance - Noise Limited Takeoff Power

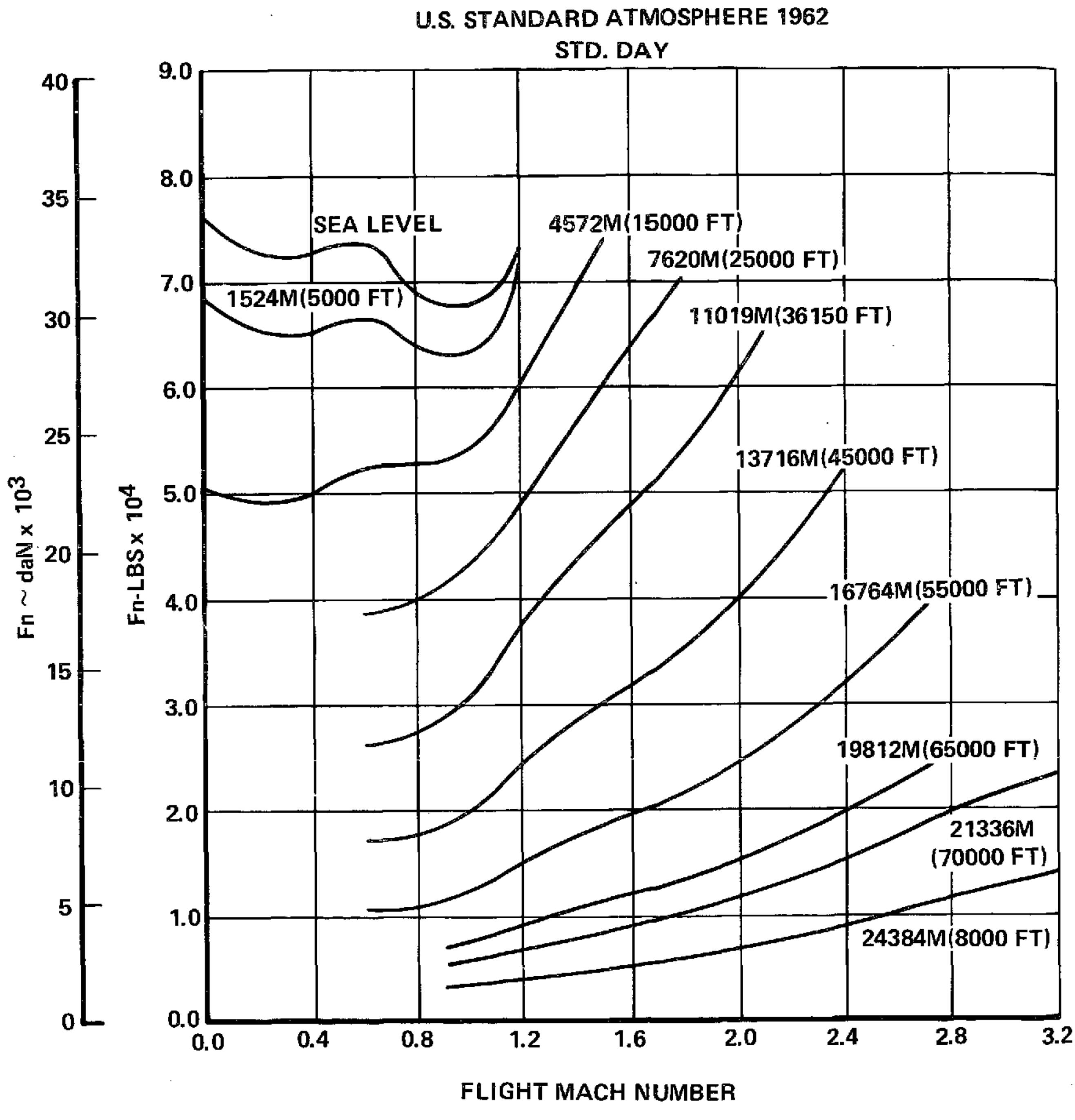

Figure 8. Installed Flight Performance - Augmented Max Climb

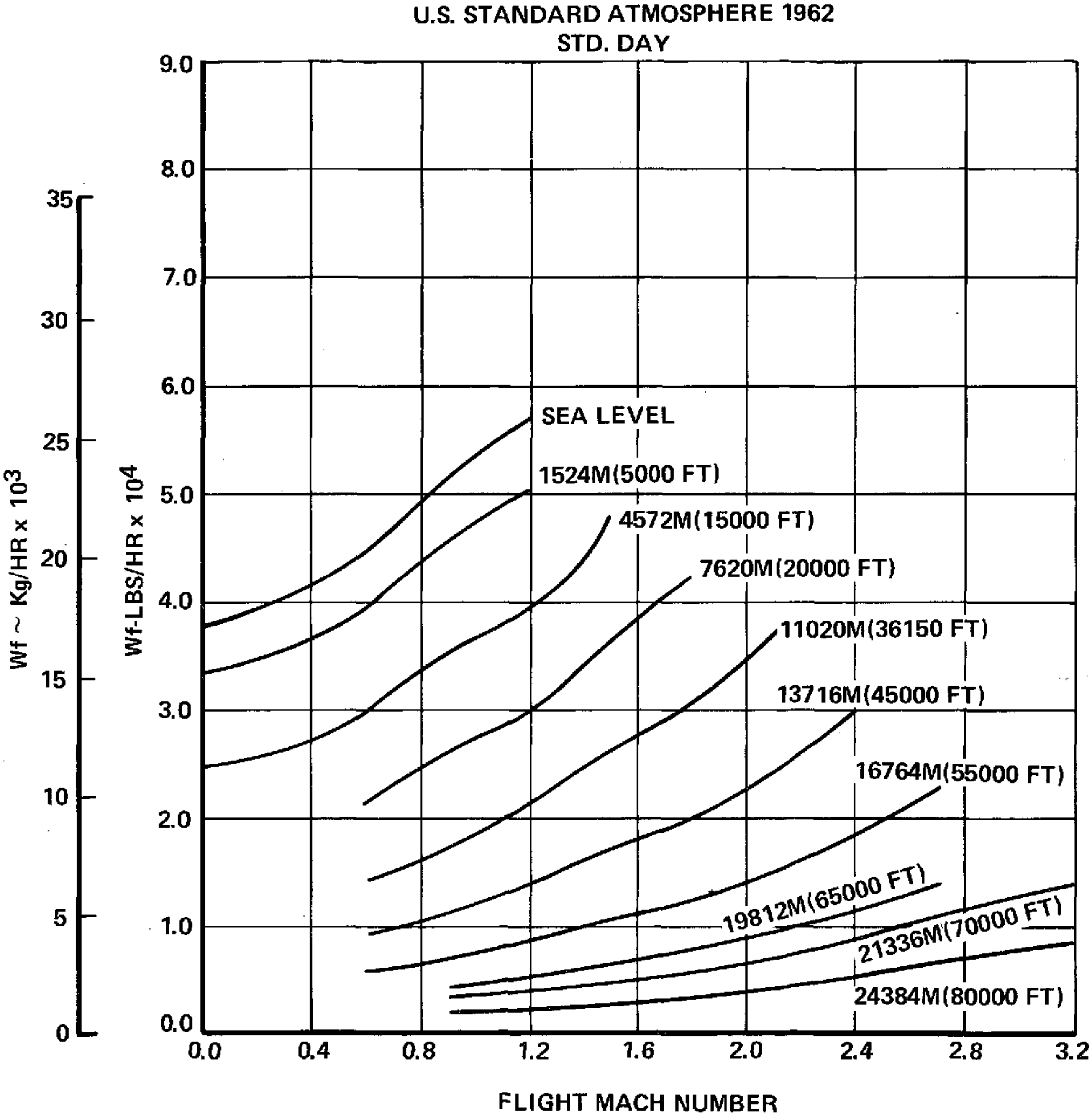

Figure 9. Installed Flight Performance- Augmented Max Climb

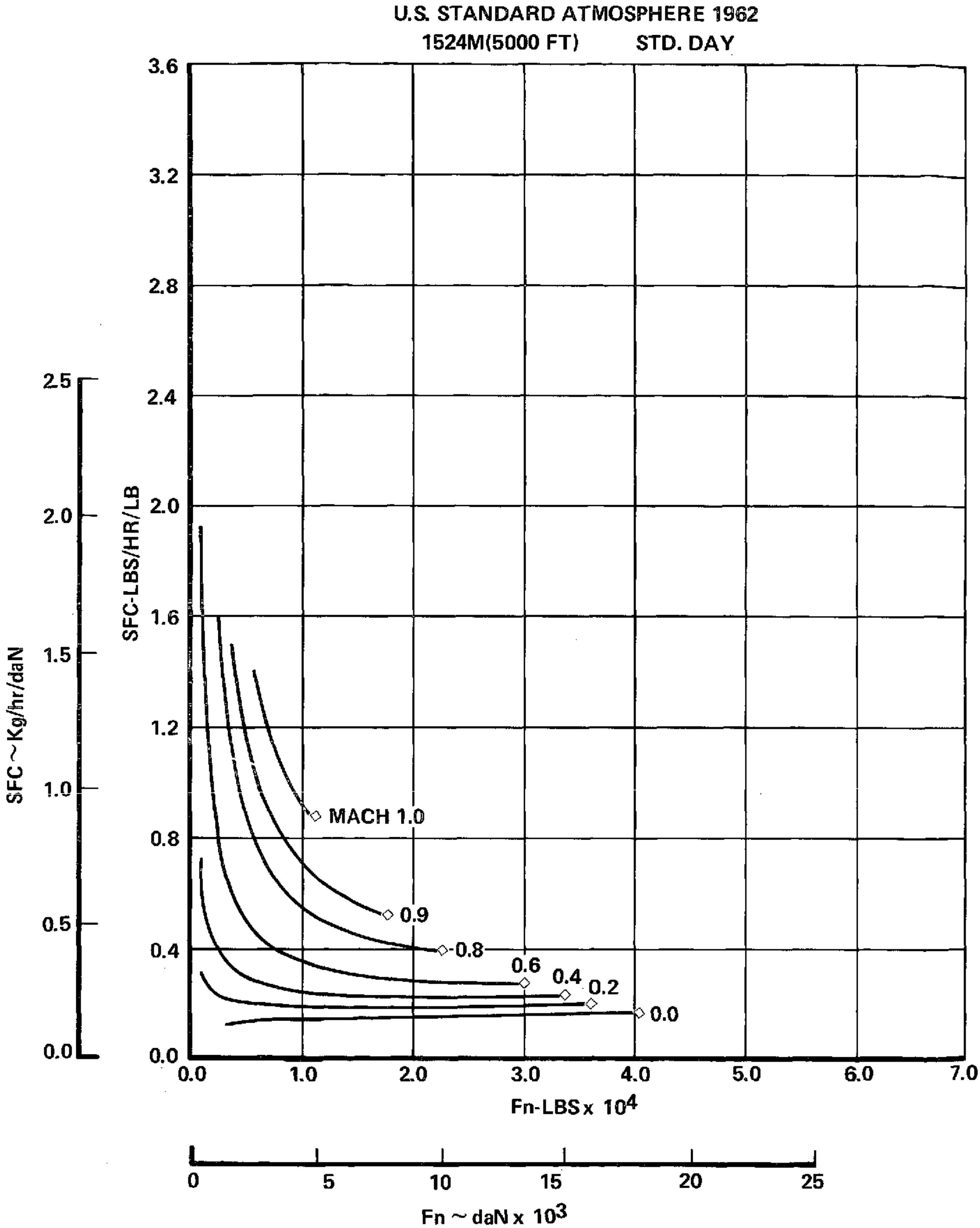

Figure 10. Installed Flight Performance - Non-Augmented Part Power

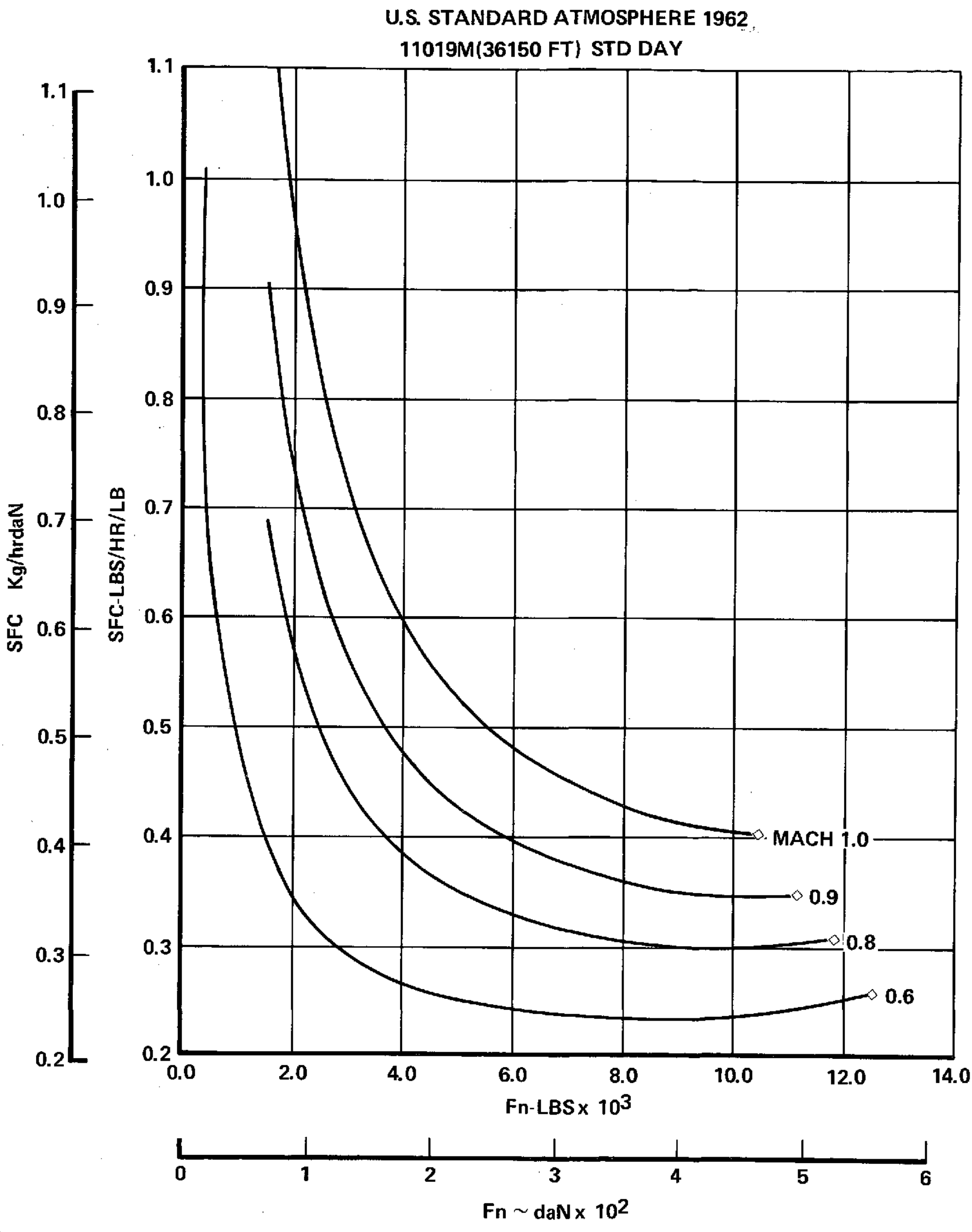

Figure 11. Installed Flight Performance - Non-Augmented Part Power

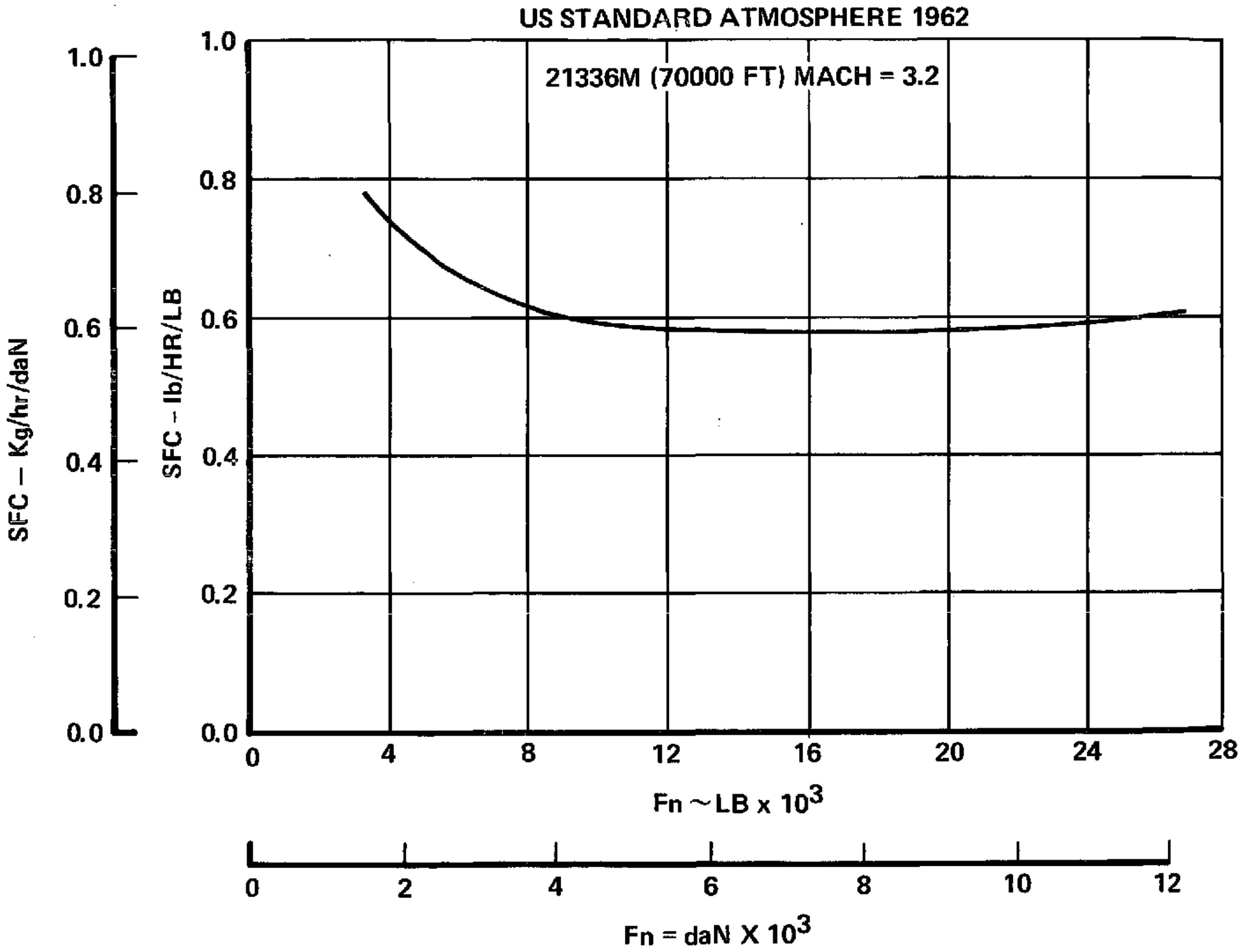

Figure 12. Installed Flight Performance - Augmented Part Power

4.2 UNCOOLED MACH 2.7 LH_2 TRANSPORT

The general characteristics of the airplane are listed in Table 2. Figures 13, 14, 15 and 16 are drawings showing its general arrangement, inboard profile and basic structural arrangement.

Detailed ASSET computer printouts of this design giving weight, cost, mission, and aerodynamic information are included in Appendix A. This aircraft is a refinement of the one reported on in Reference 3. It has a lower gross weight (164,000 kg) compared to the 167,000 kg of Reference 3). The essential difference is due to a modification of the airport noise prediction calculation technique and the increase of the landing approach speed from 79.3 to 82.3 m/s (154 to 160 KEAS). The wing reference area of this aircraft is 579m^2 (6232 ft.^{2}).

The interior arrangement is shown in Figure 14. It illustrates the passenger seating arrangement and the location of the liquid hydrogen fuel tanks. The large portion of fuselage volume devoted to LH_2 stowage is readily apparent. All LH_2 fuel is stowed in two large fuselage tanks arranged with one forward and one aft of the passenger compartment. Balance and c.g. management are facilitated by the location of fuel both forward and aft of the aircraft c.g. Use of fuselage stowage for fuel also provides an efficient ratio of tank volume to tank surface area and minimizes the fuel plumbing and tank insulation required. In addition, the integral tank structure also serves as the fuselage primary structure. Both the forward and aft fuel tank sections are divided into two separate tanks by means of a vertical divider. This divider is not a pressure bulkhead since provision is made for pressure equalization between the two compartments of each tank. It simply serves to provide fuel to each engine from a separate compartment.

With the payload in close proximity to the aircraft c.g., minimum c.g., movement results when the passenger and/or cargo load is varied. Passengers are seated six abreast on both levels of a double deck arrangement. This not only provides spacious accommodations but also minimizes the length of the payload section.

Cargo is stowed at the forward end of the lower deck so that the cutout for container installation/removal results in cutting only the relatively lightly loaded spar caps at the wing apex. Some of the electrical/electronic equipment is carried in the domed cavities in the pressure bulkheads at each end of the cabin in both decks to provide both good accessibility and a controlled environment. The space

TABLE 2. MACH 2-7 UNCOOLED LH_2 SUPERSONIC TRANSPORT

Payload	kg	(lb)	22,226	(49,000)
Range	km	(n.mi.)	7,778	(4,200)
Cruise Speed	Mach		2.7	
Takeoff Gross Weight	kg	(lb)	163,783	(361,074)
Operating Empty Weight	kg	(lb)	99,379	(218,869)
Fuel Weight, Mission	kg	(lb)	35,800	(78,995)
Total	kg	(lb)	42,278	(93,205)
Fuel Volume	m^3	(ft^3)	625	(22,086)
Wing Area	m^2	(ft^2)	57.9	(6,232)
Wing Loading (W/S) Takeoff	kg/m^2	(lb/ft^2)	283	(57.9)
Landing	kg/m^2	(lb/ft^2)	221	(45.3)
Span	m	(ft)	30.6	(100.6)
Overall Length	m	(ft)	99	(324.7)
Lift/Drag (cruise)			6.85	
Specific Fuel Consumption (cruise)	$\frac{kg}{hr}/daN$	(lb/hr/lb)	.562	(.553)
Thrust/Weight (SLS)	$\frac{N}{kg}$	(m)(lb/lb)	5.35	(.546)
Thrust Per Engine	N	(lb)	219,000	(49,286)
Weight Fractions	Percent			
Fuel			25.81	
Payload			13.57	
Structure			32.48	
Propulsion			16.62	
Equipment and Operating Items			11.52	
Energy Utilization	kJ/seat km	(BTU/Seat.n.mi)	5,190	(4,147)
DOC	¢/AS km	(¢/ASn.mi.)	.941	(1.744)
Price	$\$ \times 10^6$		47.04	

NOTE -
1. DIMS IN SI(ENGLISH) UNITS
2. LINEAR DIMS IN METERS (INCHES)
ANGLES IN RADIANS(DEGREES)
3. DATA RELATIVE TO ZRP
4. EXPOSED AREA

	WING (3)	HORIZ. TAIL (3)	F.S. VERT. TAIL (3)	WING VERT TAIL (EACH) (3)
AREA - M²(FT²)	639.173 (6880.0)	34.690 (373.4)	14.344 (154.4) (4)	13.773 (148.25)
ASPECT RATIO	1.607	1.707	0.517	0.435
TAPER RATIO	0.135	0.225	0.230	0.135
SPAN - M(IN)	32.052 (1261.89)	7.671 (302.0)	2.718 (107.0)	2.6035 (102.5)
ROOT CHORD - M(IN)	39.697 (1562.88)	7.331 (231.0)	8.585 (338.0)	9.236 (364.0)
TIP CHORD - M(IN)	5.023 (197.77)	1.664 (65.5)	1.976 (77.8)	1.262 (49.7)
MAC - M(IN)	27.2364 (1072.30)			
L.E. SWEEP RADIANS (DEGREES)	1.292 (74°) 1.236 (70.84°) 1.047 (60.0°)	1.058 (60.64°)	1.190 (68.2°)	1.281 (73.42°)

ADVANCED DESIGN
CL1701-7
GENERAL ARRANGEMENT -
LH₂ FUELED AST - POINT DESIGN
CL1701-7-1

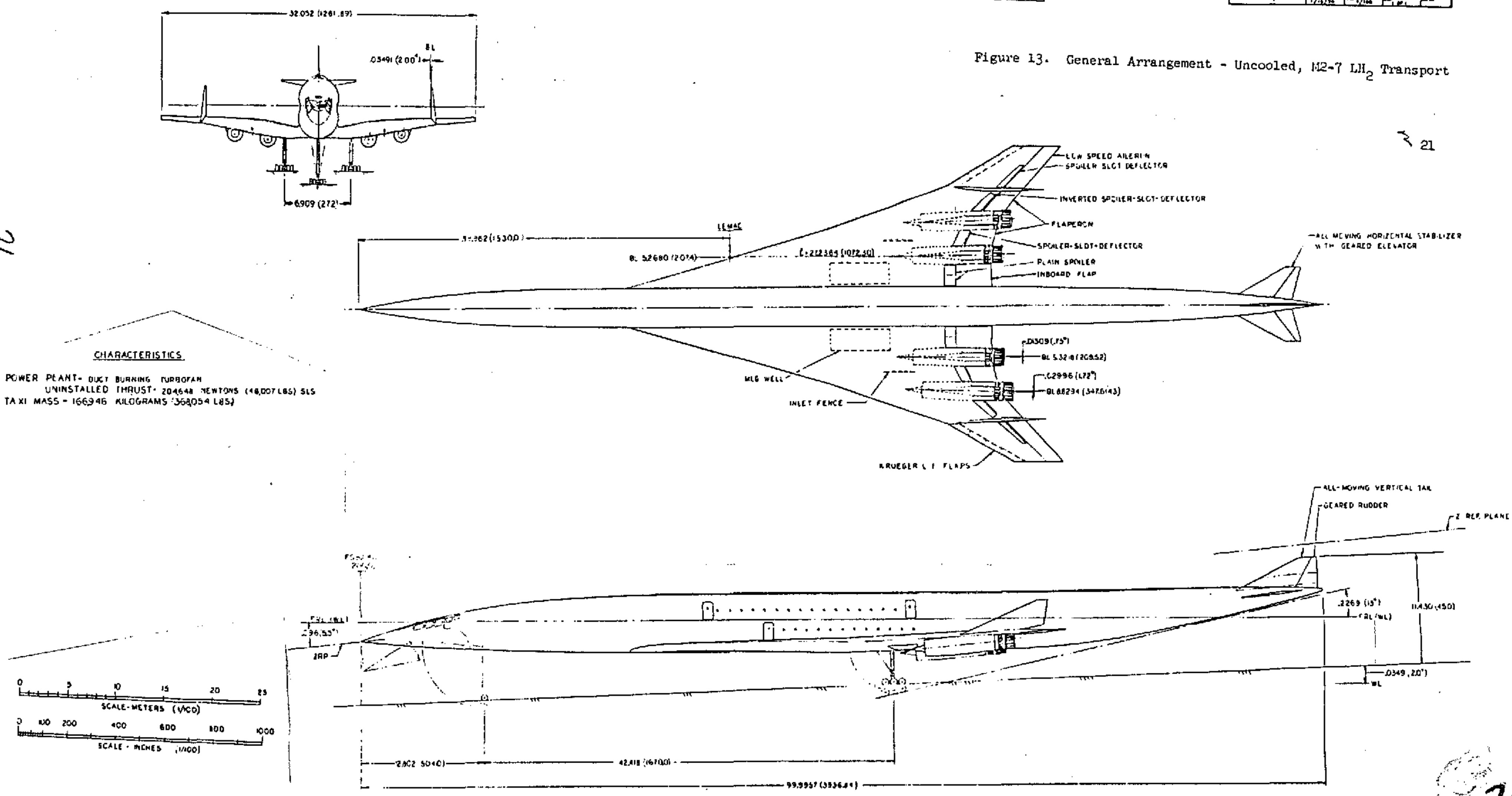

Figure 13. General Arrangement - Uncooled, M2-7 LH_2 Transport

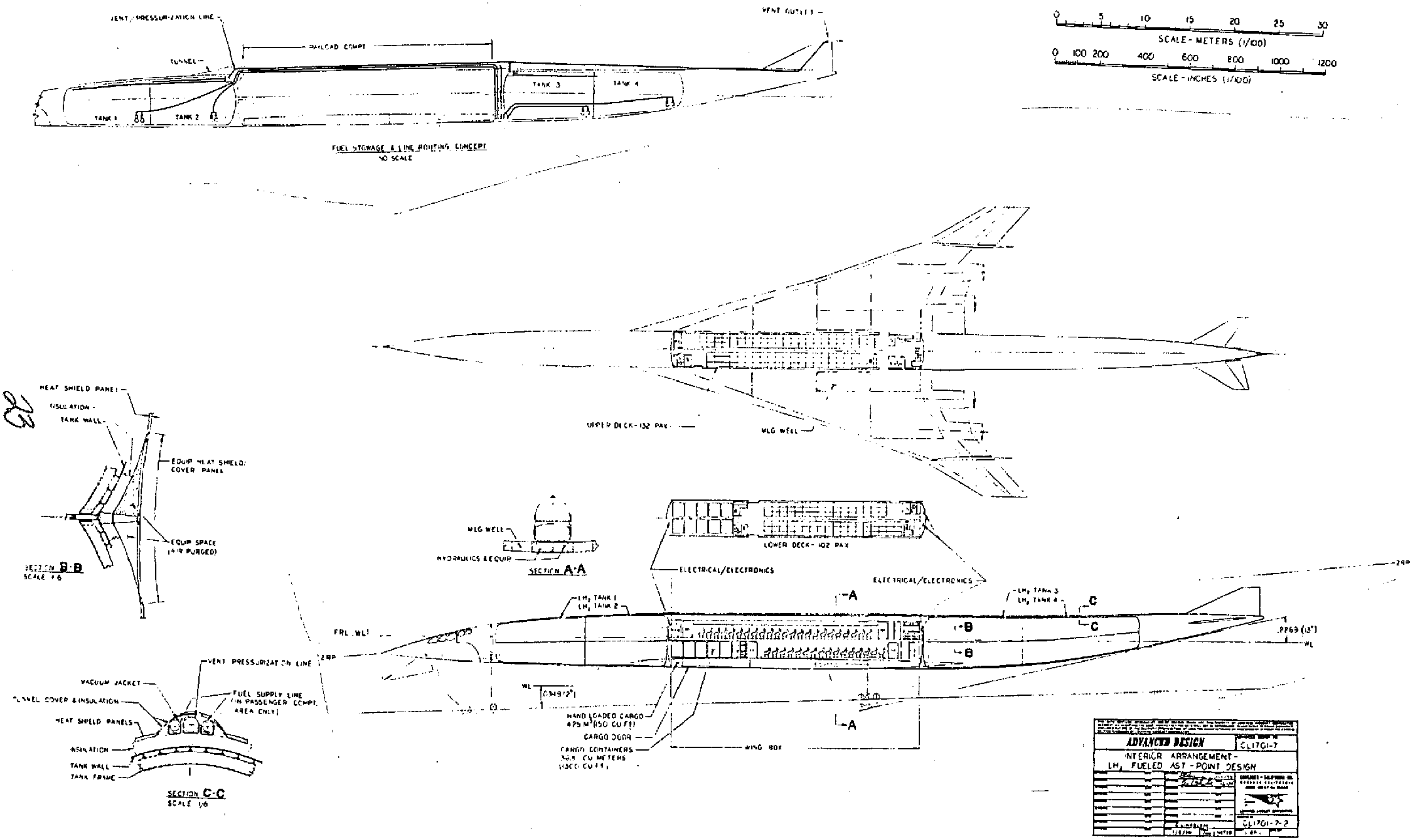

Figure 14. Interior Arrangement - Uncooled, M2-7 LH_2 Transport

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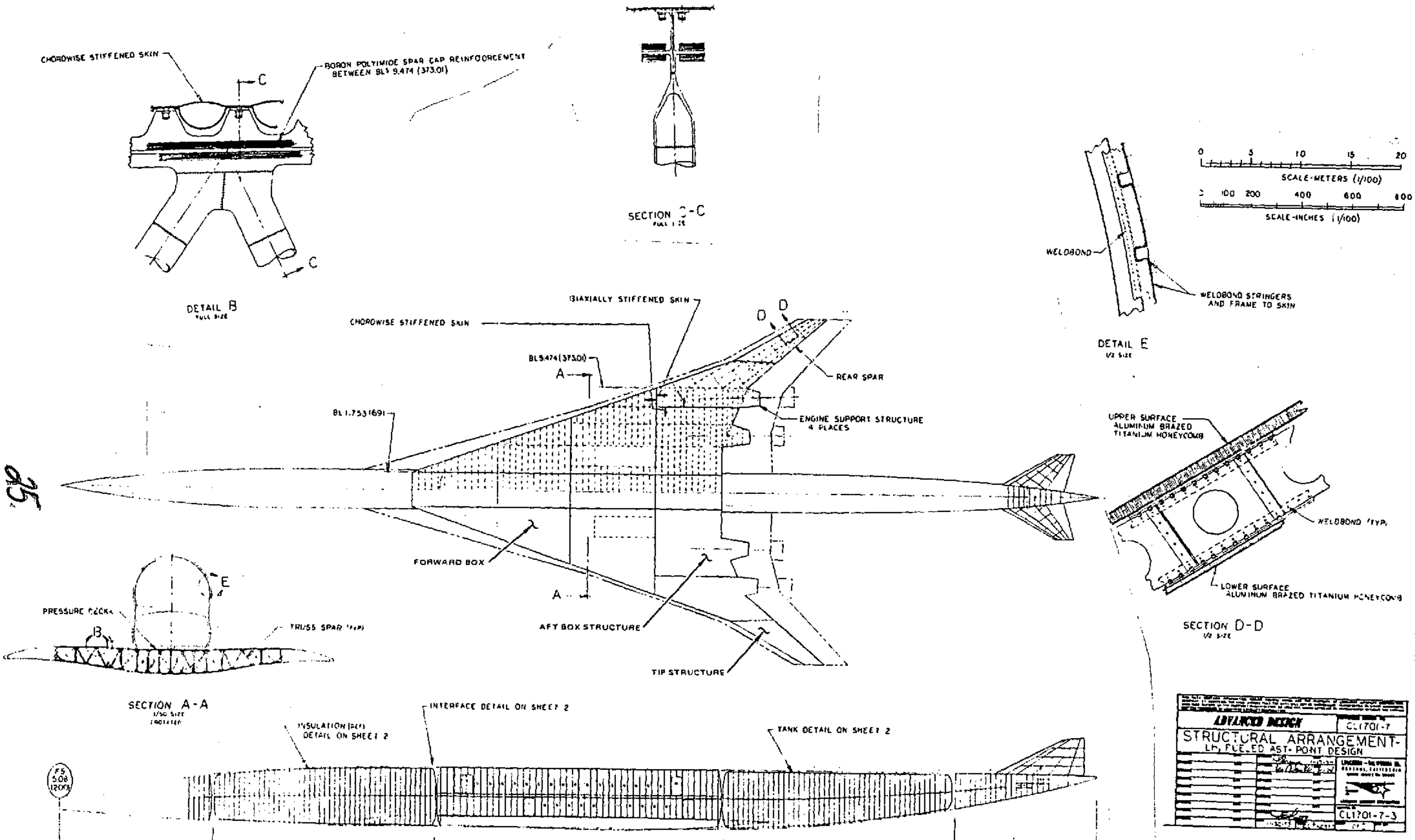

Figure 15. Structural Arrangement - Sheet 1

1. ALL MATERIAL TITANIUM ALLOY UNLESS SPECIFIED OTHERWISE

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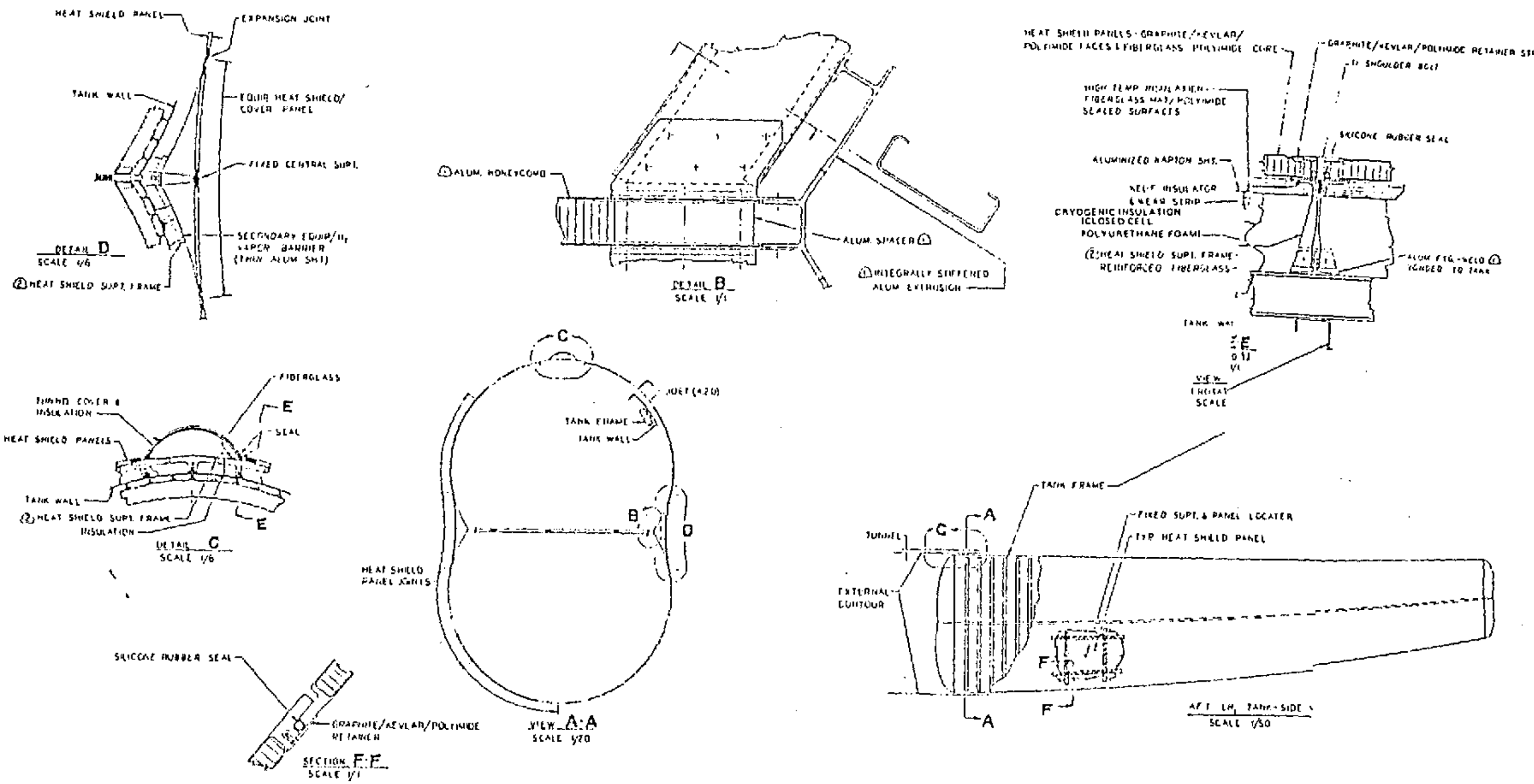
HEAT SHIELD PANEL
EXPANSION JOINT
TANK WALL
FIXED CENTRAL SUPT.
DETAIL D
SCALE 1/6
② HEAT SHIELD SUPT. FRAME
① ALUM. HONEYCOMB
ALUM. SPACER ①
① INTEGRALLY STIFFENED ALUM. EXTRUSION
DETAIL B
SCALE 1/1
ALUMINIZED KAPTON SHT.
CRYOGENIC INSULATION (CLOSED CELL POLYURETHANE FOAM)
SILICONE RUBBER SEAL
FIBERGLASS
SEAL
HEAT SHIELD PANELS
TANK WALL
② HEAT SHIELD SUPT. FRAME
INSULATION
DETAIL C
SCALE 1/6
TANK FRAME
TANK WALL
HEAT SHIELD PANEL JOINTS
VIEW A-A
SCALE 1/20
SILICONE RUBBER SEAL
GRAPHITE/KEVLAR/POLYIMIDE RETAINER
SECTION F-F
SCALE 1/1
TANK FRAME
FIXED SUPT. & PANEL LOCATER
TYP HEAT SHIELD PANEL
TUNNEL
EXTERNAL CONTOUR
AFT LH2 TANK - SIDE
SCALE 1/50

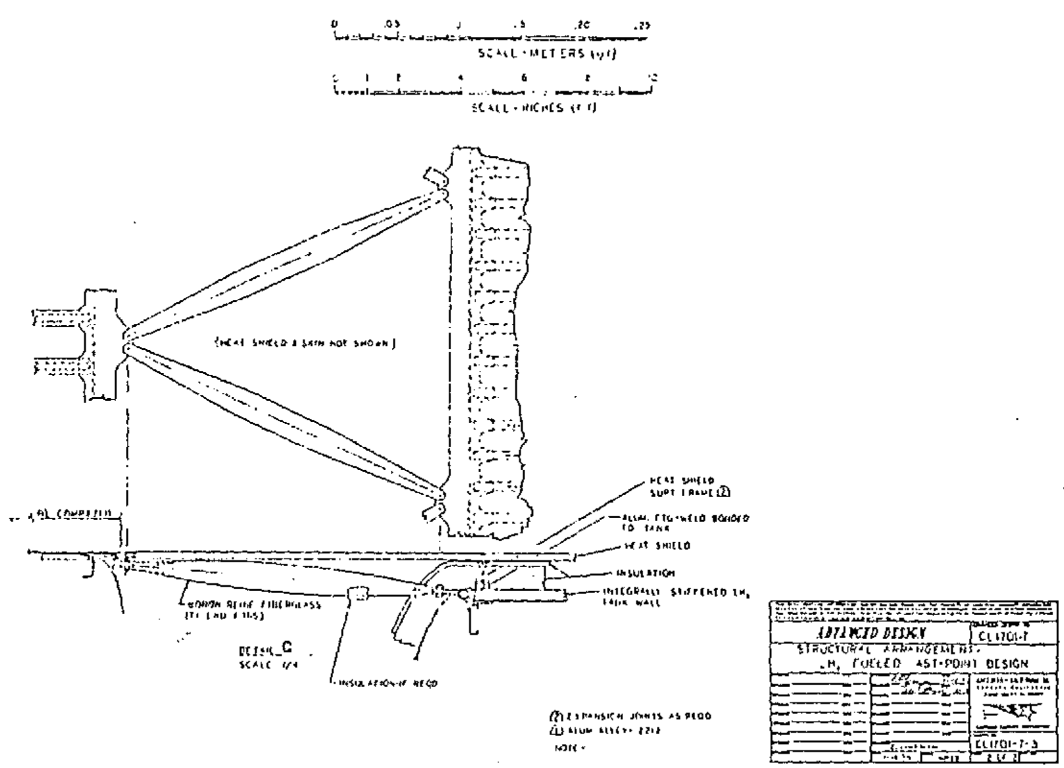

Figure 16. Structural Arrangement - Sheet 2

27a

below the floor and between the MLG wells is used for aircraft equipment and service centers.

Throughout the length of the payload section, fuel supply and vent lines are contained in a dorsal fairing above the fuselage so that any fuel vapors accidentally released will tend to rise away from the aircraft. Pressure bulkheads domed in opposite directions are shown in Figure 16 at the fuel tank/cabin interface joints. A truss type interstage structure provides the connection.

Flight control and high lift devices are shown in Figure 13. Pitch control is obtained from an all-moving horizontal stabilizer with a geared elevator while yaw control is provided by a fuselage-mounted all-moving vertical tail with a geared rudder. A fixed vertical fin is located on each side of the wing. The outer wing includes ailerons for roll control at low speeds and Krueger leading edge flaps for use at subsonic and transonic speeds. Plain spoilers next to the fuselage are used for deceleration on the ground. The Fowler inboard trailing edge flaps increase lift at low speeds while flaperons function, dependent on speed, as either high lift or roll control devices.

Wing-mounted main landing gears retract forward into the wing just outboard of the fuselage. Four duct burning turbofan engines are mounted in underwing pods having axisymmetric inlets and thrust reversers near the wing trailing edge.

The structural approach for the wing of the uncooled airplane is shown in Figure 15 and identified by the three major areas which include the forward box, aft box and tip structure.

Forward and Aft Box Structure: A chordwise stiffened arrangement is used for the forward and aft box structure which comprises the major portion of the basic wing. This arrangement is essentially a multispar structure with widely spaced ribs. The submerged spar caps of titanium alloy (Ti 6Al-4V annealed) are space approximately 20 inches on-center and are used to transmit the wing bending loads. These caps being submerged result in reduced temperatures, which in turn results in increased allowable stresses and also permits uncoupling of the spanwise and chordwise stiffness for vehicle flutter suppression.

Selective reinforcement of the basic metal structure is considered as the appropriate level of composite application for the near-term (1981) design.

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Composite reinforced spar cap details (Figure 15) show the application of unidirectional reinforcing with boron polyimide. Both truss-type and circular-arc corrugated webs are used as appropriate for access and manufacturing requirements.

The surface panel concepts for the forward and aft box in this arrangement have stiffening elements oriented in the chordwise direction. Structurally efficient circular-arc beaded-skin designs are used (Figure 15). These efficient circular-arc sections of sheet metal construction (Ti 6Al-4V annealed) provide effective designs when properly oriented in the airstream to provide acceptable aerodynamic performance as demonstrated on the NASA-Lockheed YF-12 airplane. The panel elements are weldbonded for improved fatigue life. The shallow protrusions provided smooth displacements under thermally induced strains and operational loads.

The stiffness-critical wing tip structure utilized monocoque construction (Figure 15) with biaxially stiffened panels which support the principal load in both the span and chord direction. The substructure is essentially a multispar design with full and partial ribs to provide support for the leading and trailing edge control surfaces and actuating system.

The monocoque construction has smooth-skinned aluminum brazed titanium honeycomb sandwich panel (Figure 16) that results in minimum aerodynamic drag. Thermal stresses are absorbed with minimal relief but criticality, defined by flutter suppression requirements, produces a minimum weight structural design for the tip structure.

Fuselage Structure:

The weather vision nose, payload and empennage sections of the CL1701 airplane are a conventional semimonocoque shell construction of titanium alloy material (Ti 6Al-4V annealed) with extensive use of weldbonding. The flight station enclosure tapers down from the constant cross-section of the forward tank and payload section which is formed by the intersection of two cylinders with a radius of 1.966 meters (77.4 inches). Structural continuity between the integral tank sections and the nose, payload, and empennage sections is provided by a truss arrangement, see Figure 16. Suitable longitudinal local reinforcements are used in truss member attachment areas to distribute the concentrated loads encountered.

The nose, payload and empennage structural arrangement is a uniaxial stiffened structure of skin and stringer with supporting frames. Weld bonding is utilized to improve the fatigue life of the structure. The skin and closed-hat stringers are supported by sheet metal frames that are spaced at approximately 0.508 meters (20-inch) intervals and aligned with the spars of the wing structure. Typical construction details of the frame and stringers are presented in Figure 15. A floor is provided at the intersection of the cylinders as well as above the wing box structure. Fore and aft intercoastals are provided over the wing box to support the lower cabin floor. Transverse beams which are attached to each frame are provided to support the upper cabin floor. The pressure boundary is provided by the upper surface of the wing box and pressure bulkhead at each end. The main frames that distribute concentrated wing and gear loads into the fuselage structure are built-up from titamium forgings or extrusions. The fuselage aft of the hydrogen tankage contains structural provisions for mounting the fin and horizontal stabilizer. A skin-stringer-frame construction similar to that provided in the pressurized area of the fuselage is used. The main rings that distribute the fin loads into the fuselage are titanium forgings

Empenage Structure:

The empennage structure utilizes sandwich construction with a multispar substructure. The empennage structural concepts and arrangements are dictated by the high sonic environment to which it is subjected, as well as engine exhaust temperatures.

Fuel Tanks:

The integral tanks are of welded construction and are integrally fabricated from 2219 aluminum alloy. The skin is stiffened with the stiffeners on the inside of the tank and with the outside surface of the tank smooth. This outside surface is .117 m (4.6 in) below contour, and the space between is occupied by insulation. The thermal protection system consists of two different types of insulations (see Figure 16 for details). Generally, the cryogenic insulation is a closed cell foam type material which is bonded to the smooth tank surface. The high temperature insulation is a fiberglas mat faced with a thin layer of polyimide resin. Heat shield panels of sandwich construction made up of fiberglas filler faced with graphite polyimide comprise the aircraft external surface. The heat shield panels are supported by low conductance fiberglas standoffs which are fastened to the tank surface. The integrally stiffened tank skin carries fuselage bending and shear loads as well as tank internal pressure loads.

4.3 UNCOOLED MACH 3.2 LH_2 TRANSPORT

The general characteristics of the airplane are listed in Table 3. The general arrangement is shown in Figure 17. The inboard profile and structural arrangement are considered to be similar to the Mach 2.7 version shown in Section 4.2. ASSET computer printout sheets giving weight, cost, mission and aerodynamic information of this design are presented in Appendix A.

The essential difference between the Mach 2.7 and 3.2 aircraft is in the increased wing sweep (reduced AR) for the higher speed design and the propulsion system inlet and engine. Other changes consist of the use of less aluminum, reduced material allowables and increased thermal protection weights for the hydrogen tankage. A further discussion of the comparison between the Mach 2.7 and 3.2 uncooled versions is given in Section 5.0.

TABLE 3. MACH 3.2 UNCOOLED LH_2 SUPERSONIC TRANSPORT

Payload	kg	(lb)	22,226	(49,000)
Range	km	(n.mi.)	7,778	(4,200)
Cruise Speed	Mach		3.2	
Takeoff Gross Weight	kg	(lb)	198,493	(437,594)
Operating Empty Weight	kg	(lb)	127,223	(280,474)
Fuel Weight, Block	kg	(lb)	39,497	(86,965)
Total	kg	(lb)	49,043	(108,120)
Fuel Volume	m^3	(ft^3)	725	(25,620)
Wing Area	m^2	(ft^2)	893	(9,613)
Wing Loading (W/S) Takeoff	kg/m^2	(lb/ft^2)	222	(45.5)
Landing	kg/m^2	(lb/ft^2)	178	(36.4)
Span	m	(ft)	34.4	(113)
Overall Length	m	(ft)	104.5	(343)
Lift/Drag (cruise)			7.72	-
Specific Fuel Consumption (cruise)	$\frac{kg/hr}{daN}$	$\frac{lb/hr}{lb}$	.608	(.597)
Thrust/Weight (SLS)	$\frac{N}{kg}$	(lb/lb)	5.2	(.531)
Thrust Per Engine	N	(lb)	258,639	(58,145)
Weight Fractions	Percent			
Fuel			24.71	
Payload			11.20	
Structure			36.18	
Propulsion			17.53	
Equipment and Operating Items			10.38	
Energy Utilization	kJ/seat km	(BTU/seat nm)	5,730	(4,565)
DOC	¢/ASkm	(¢/AS nm)	1.025	(1.895)
Price	$\$X10^6$		59.09	-

CHARACTERISTICS

POWER PLANT- DUCT BURNING TURBOFAN
UNINSTALLED THRUST- 249,410 NEWTONS (56,000 LBS) SLS

	WING 1	HORIZ TAIL 2	FUS VERT TAIL 2	WING VERT TAIL (EACH) 2
AREA-M²(FT²)	841.31 (9771)	33.5 (317.4)	18.9 (203.2)	16.3 (175.75)
ASPECT RATIO	1.3358	1.707	0.517	0.455
TAPER RATIO	0.135	0.225	0.130	0.136
SPAN-M(IN)	33.92 (1335.6)	9.75 (383.87)	3.11 (122.45)	2.84 (111.85)
ROOT CHORD-M(IN)	56.35 (2214.5)	9.31 (366.39)	9.87 (388.77)	10.11 (397.93)
TIP CHORD-M(IN)	6.38 (251.35)	2.09 (82.44)	2.27 (89.40)	1.37 (54.17)
MAC-M(IN)	34.93 (1375.2)	6.46 (254.35)	6.87 (270.35)	6.05 (238.54)
L.E. SWEEP RADIANS(DEGREES)	1.33 (76.47°) 1.28 (73.5°) 1.14 (65.05°)	1.14 (65.43°)	1.25 (71.8°)	1.33 (76.2°)

1 EXPOSED AREA
2 DATA RELATING TO ZRP
1. LINEAR DIMS IN METERS (INCHES) ANGLES IN RADIANS (DEGREES)
2. DIMS IN SI (ENGLISH) UNITS
NOTE -

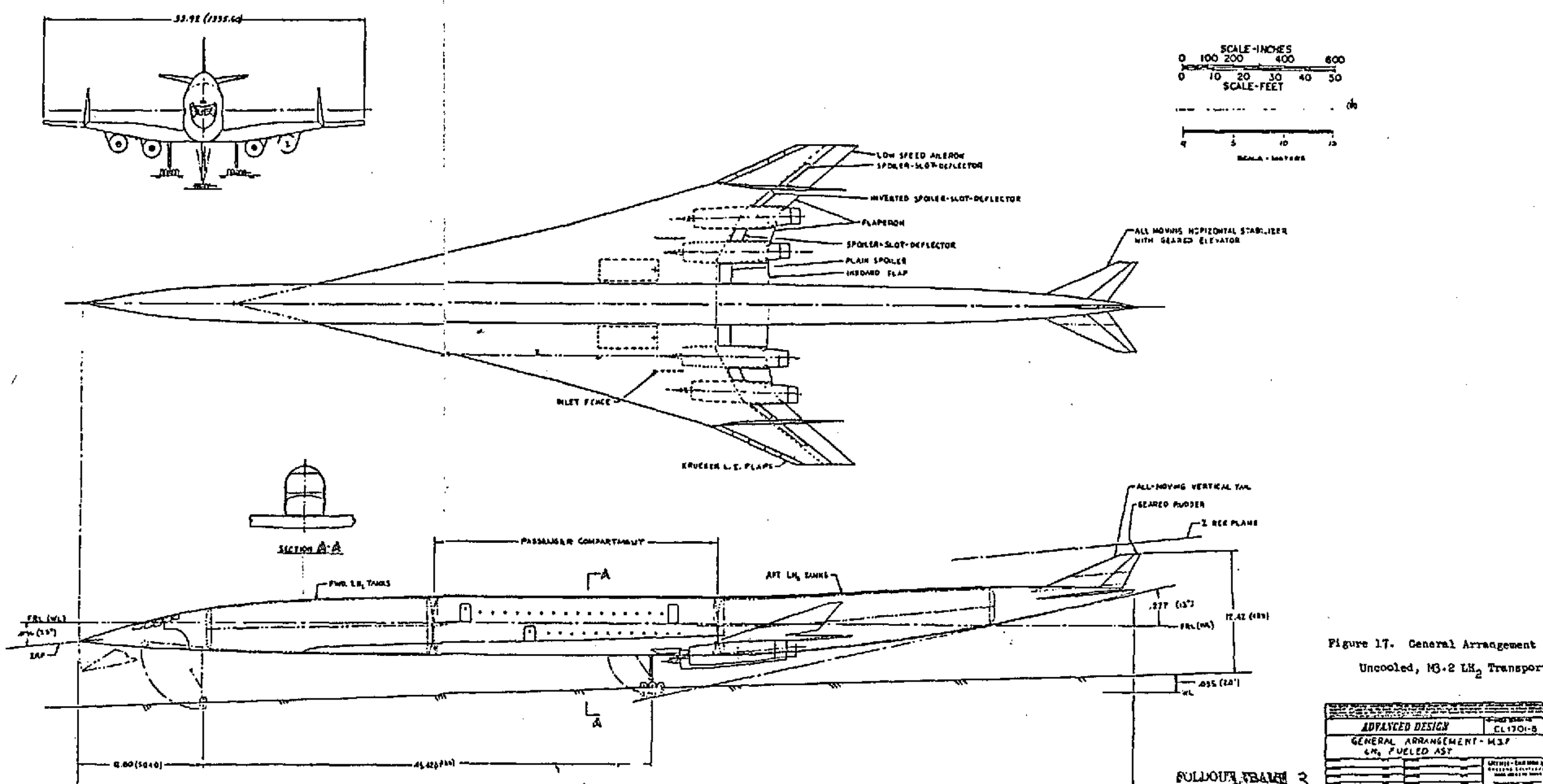

Figure 17. General Arrangement - Uncooled, M3.2 LH_2 Transport

FOLDOUT FRAME 3

FOLDOUT FRAME

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4.4 ANALYSIS OF COOLED STRUCTURE

4.4.1 Background

Cooling the wing and fuselage structure of the LH_2 AST aircraft requires sufficient removal of the heat loads due to aerodynamic heating to maintain maximum surface temperatures at or below 367°K (660°R). As discussed in Reference 6, the thermal analysis of an aircraft subject to aerodynamic heating is divided into four steps:

1. Determination of the nonviscous flow field about the aircraft. This step requires knowledge of the flight profile and the design atmosphere which along with the vehicle configuration, provide the basis for calculating the ambient air properties at the outer edge of the boundary layer.

2. Selection of an appropriate expression for the rate of thermal energy transferred to the skin from the hot gases in the boundary layer (i.e., determination of the aerodynamic heat transfer coefficient).

3. Establishment of structural component thermophysical properties.

4. Selection of a mathematical model describing the heat flow paths within the structure.

Reference 6 applied these steps to the thermal analysis of a supersonic Jet A-fueled aircraft cruising at Mach 2.7. Since the aircraft design is similar to the LH_2 AST, the technical approach used in determining heat loads for the Jet A-fueled aircraft is applicable to the LH_2 AST. Details of the steps used in the development of aerodynamic heating coefficients and recovery temperatures are discussed in Appendix B.

Results of the analysis for the Jet-A aircraft are shown in Figure 18, a plot of the surface isotherms for Mach 2.7 cruise at 19,800 m (65,000 ft) altitude.

4.4.2 Thermal Analysis

The external heat transfer coefficients used for the determination of cooling loads are based on the results obtained with the above referenced Jet A-fueled aircraft. This is a larger aircraft than the LH_2 AST but has the same wing sweep

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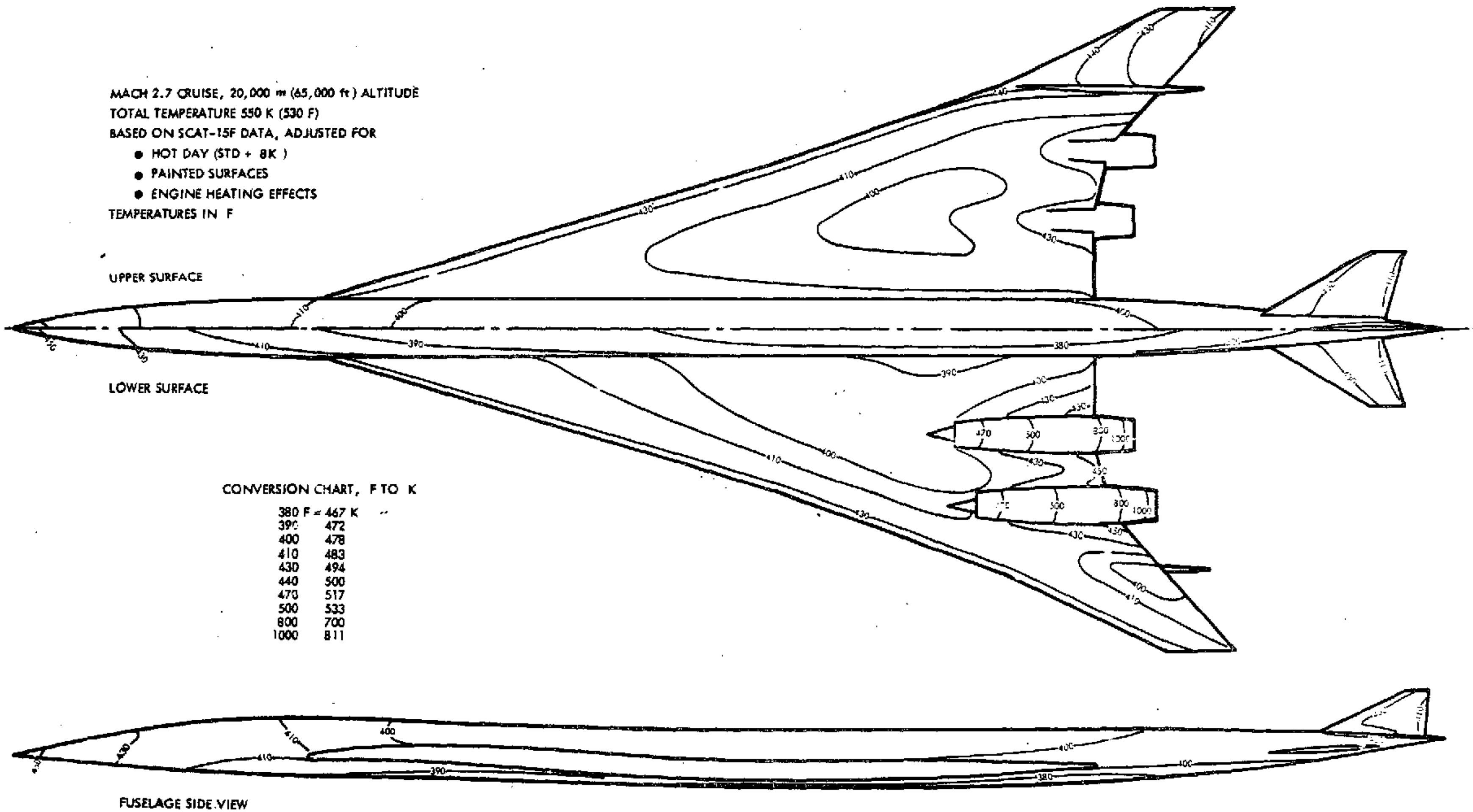

Figure 18. Surface Isotherms - Mach 2.7 Cruise (Jet A-Fueled)

angle. The cruise Mach number for both aircraft is 2.7 with cruise altitude of 20,720 m (68,000 ft) for the LH_2 AST and 19,850 m (65,000 ft) for the Jet A-fueled AST. The external heat transfer coefficients for the LH_2 AST wing are considered derivable from the Jet A-fueled AST on the basis that the airfoil shape is similar. Figure 19 shows the distribution of heat transfer coefficient values for both upper and lower wing surfaces for the hydrocarbon fueled AST at the 2.7 Mach number cruise. Similar locations were found for the LH_2 AST wing by proportioning the wing span and chord length. The heat transfer coefficient at any point, or more explicitly the Stanton number, is a function of the skin friction coefficient, which is dependent on the local Reynolds number. On the assumption that in turbulent flow the skin friction coefficient varies as the 0.2 power of the Reynolds number, the heat transfer coefficients for the LH_2 AST wing were modified from the Jet A-fueled AST wing data by the ratio of the distance from the leading edge raised to the 0.2 power. This was done to obtain heat transfer coefficients for both the fuselage and the upper and lower surfaces for the Mach 2.7 cruise case.

Cooling of the wing and fuselage surfaces results in higher skin friction coefficients. By the method of Reference 7, the average ratio of cooled to uncooled skin friction coefficients was determined and this factor was applied to the heat transfer coefficients previously obtained. The result of this analysis is discussed in Section 4.5

For the Mach 3.2 case, no previous thermal analysis accounting for local conditions was available. Since the Mach 3.2 aircraft cruises at 23,200 m (76,000 ft), it was found that for the fuselage surface the average heat transfer coefficient was less than that for the Mach 2.7 aircraft as scaled on the basis of the local Reynold's number raised to the 0.2 power. It was assumed that the integrated average values of heat transfer coefficients determined for the Mach 2.7 case could be similarly modified for the upper and lower wing surfaces.

The average wing loading during cruise is higher for the Jet A-fueled AST than the LH_2 AST. The higher angle of attack required for the former is expected to result in a higher ratio of integrated external heat transfer coefficients for the lower surface compared to the upper surface. The average integrated value for both surfaces is expected to be unchanged. The division of heat load to be absorbed by the coolant between upper and lower surfaces for the LH_2 AST was modified slightly to reflect this difference in wing loading.

MACH 2.7 CRUISE 19,850 M (65,000 FT.) ALTITUDE
HEAT TRANSFER COEFFICIENTS
(BTU/HR-FT^2°F)

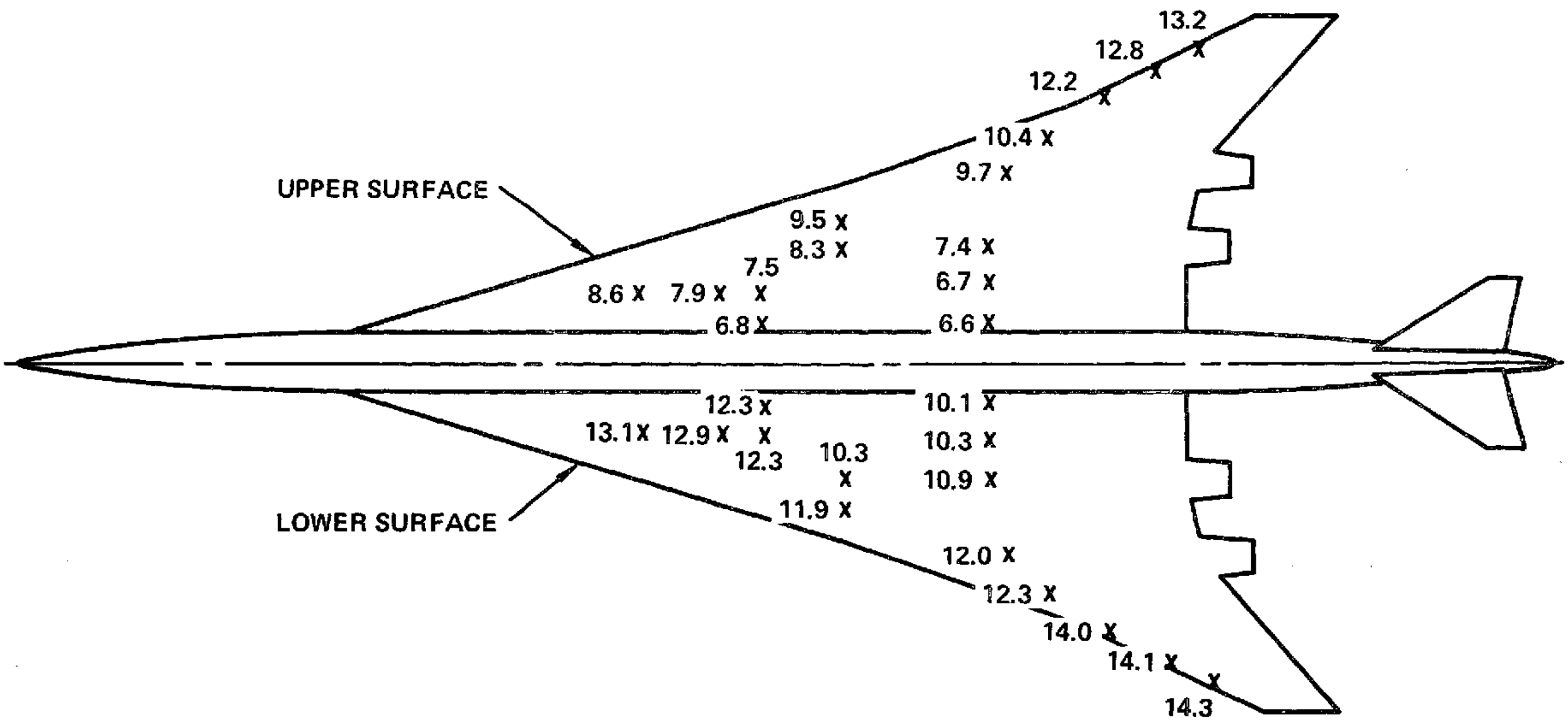

Figure 19. Distribution of External Heat Transfer Coefficients for Jet A-Fueled AST at M = 2.7

The final values of heat loads to be removed for both the Mach 2.7 and 3.2 LH_2 AST are given in a subsequent section on analytical results.

4.4.3 Panel Analysis

The analysis of skin temperatures depends upon the structural configuration, coolant temperature, coolant flow rate, coolant passage size and spacing of the passages as well as the external heat transfer coefficient. Assuming no internal heat transfer other than to the coolant, the following equation (from Reference 8) applies to the fin effect at any point along the passage:

$$\frac{t_{m_2} - t_x}{t_{m_2} - t_{m_1}} = \frac{\cosh A_2 (l_2 - x)}{(A_2/A_1) \sinh A_2 l_2 \cdot \coth A_1 l_1 + \cosh A_2 l_2} \tag{1}$$

where

l_2 = length of fin to the boundary condition where $dt/dx = 0$

l_1 = passage half-width

x - any point along the fin

t_x = temperature of any point along the fin

t_{m_2} = temperature of fin without fin effect

t_{m_1} = temperature of passage surface without fin effect, and

A = a function account for heat transport and dimensional properties, defined as:

$$\sqrt{\frac{h_1 + h_2}{K \delta}}$$

where h_1 and h_2 are external and internal convection heat transfer coefficients, respectively.

K = thermoconductivity of fin

δ = thickness of fin

The functions, A_1 and A_2, apply to the passage and fin sections, respectively. Differentiation equation (1) results in the following expression:

$$\frac{dt_x}{dx} = \frac{(A_2) \left[(t_{m_2} - t_{m_1}) \cdot \sinh A_2 (l_2 - x)\right]}{(A_2/A_1) \sinh A_2 l_2 \cdot \coth A_1 l_1 + \cosh A_2 l_2} \tag{2}$$

The heat flow rate from the fin at any point along the passage, q_{FIN}, is defined as

$$q_{FIN} = K \delta \, dy \left(\frac{dt_x}{dx}\right)_{x = 0} \tag{3}$$

where dy is the incremental passage length.

The heat flow the coolant is thus given by the following equation:

$$\frac{W}{2} \, cp.dt_y = K \delta \, dy \left(\frac{dt_x}{dx}\right)_{x = 0} + U (t_r - ty) \, l_1 \, dy \tag{4}$$

where

W = passage flow

c_p = specific heat of coolant

t_y = temperature of coolant at point y along passage

U = overall heat transfer coefficient

t_r = recovery temperature

By sustituting from equation (2) the equivalent expression for $\left(\frac{dt_x}{cx}\right)_{x=0}$, equation (4) may be rewritten as follows:

$$\frac{W}{2} \, cp.dt_y = \left[\left(P - \frac{Ph_1}{h_1 + h_2} + Ul_1\right) t_r - \left(\frac{Ph_1}{h_1 + h_2} + Ul_1\right) t_y\right] dy \tag{5}$$

where

$$P = \frac{K\delta\, A_2 \sinh A_2\, l_2}{(A_2/A_1) \sinh A_2\, l_2 \cdot \coth A_1\, l_1 + \cosh A_2\, l_2}$$

Equation (5) is easily integrated by the separation of variables so that the temperature rise of the coolant in the passage may be determined as follows:

$$t_{y_2} - t_{y_1} = (t_r - t_{y_1}) \left\{ 1 - e^{-\left[\frac{\left(\frac{K\delta A_1\, A_2 \sinh A_2\, l_2}{A_2 \sinh A_2\, l_2 \cdot \coth A_1\, l_1 + \cosh A_2\, l_2}\right)h_2 + h_1 h_2 l_1}{\frac{W}{2} c_p\, (h_1 + h_2)}\right] y} \right\} \tag{6}$$

where

t_{y_2} = temperature of coolant at end of passage length y

t_{y_1} = temperature of coolant at start of passage

h_1 = external heat transfer coefficient

h_2 = internal heat transfer coefficient

Equation (6) is limited in application because of the change in coolant thermophysical and heat transport properties with temperature. As a result the total heat load to be absorbed by the coolant must be numerically integrated by selecting small increments of "y" and averaging the values of all terms which are temperature dependent.

The most significant factor to be determined is h_2, the internal heat transfer coefficient. Reference 9 defines for heating and cooling viscous liquids flowing in non-isothermal streamline motion inside tubes the following recommended equation for determination of the Nusselt number, $h_a D/k$:

$$\frac{h_a D}{k}\left(\frac{\mu}{\mu_s}\right)^{-0.14} = 1.86 \left[\left(\frac{DG}{\mu}\right) \left(\frac{c_p \mu}{k}\right) \left(\frac{D}{L}\right)\right]^{1/3} \tag{7}$$

where

h_a = average heat transfer coefficient

D = hydraulic diameter

k = thermoconductivity of liquid

μ/μ_s = ratio of liquid viscosity at the average bulk temperature to its viscosity at the average temperature of the inside surface of the tube

$\frac{DG}{\mu}$ = Reynolds number

$\frac{c_p \mu}{k}$ = Prandtl number

L = length of passage

The above equation is applicable for Reynolds number less than 2100. Equation (7) is not usable for defining the heat transfer coefficient at various points along the passage. For any length L the equation integrates the local values and averages the results as follows:

$$h_a = \frac{\int_o^L h_L dL}{L}$$

where

h_L = local heat transfer coefficient

Substituting the above value of h_a in Equation (7) gives the following expression for h_L:

$$\int_o^L h_L \, dL = 1.86 \frac{k}{D}\left(\frac{\mu}{\mu_s}\right)^{0.14} \left[\left(\frac{DG}{\mu}\right) \left(\frac{c_p\mu}{k}\right) (D)\right]^{1/3} L^{2/3} \qquad (8)$$

Taking the derivative of both sides of equation (8) gives

$$h_L \, dL = 1.86 \frac{k}{D} \left(\frac{\mu}{\mu_s}\right)^{0.14} \left[\frac{D^2 G \, c_p \mu}{k}\right]^{1/3} \cdot \frac{2}{3} L^{-1/3} \, dL \tag{9}$$

or

$$h_L = \frac{2}{3} h_a$$

Equation (9) states that the local heat transfer coefficient at any point, L, is essentially 2/3 of the average value from zero to L. In the analysis of heat load absorbed by the coolant, the internal heat transfer coefficient was calculated from equation (9) at the midpoint of each increment of passage length and assumed to be the average for that increment for the laminar flow case.

When the coolant flow is fully turbulent, the heat transfer coefficient is defined by the following equation (Ref. 8):

$$\frac{h}{c_p G} = 0.027 \left(\frac{DG}{\mu}\right)^{-0.2} \left(\frac{c_p \mu}{k}\right)^{-2/3} \left(\frac{\mu}{\mu_s}\right)^{0.14} \tag{10}$$

where G = flow per unit area

Equation (1) applies at Reynolds number of 10,000 or higher. It is seen that the heat transfer coefficient is now independent of passage length. Reynolds number of 2100 to 10,000 covers the transition region. In this region the range of heat transfer coefficients is not defined but is assumed to increase from a minimum value at Re = 2100 to the maximum turbulent value at Re = 10,000. For the purpose of this analysis, a parabolic curve fit was assumed.

Other coolant properties such as c_p, k, density, and μ were evaluated at the average liquid bulk temperature over the particular passage interval. For μ_s, the average passage skin temperature was used. A computer program was written to evaluate the variation of skin and coolant temperatures along the passage length.

A fuselage panel was selected for the application of this calculation procedure for the estimation of cooling loads because an average external heat transfer coefficient could be easily determined and the passage lengths are uniform. The spacing of the passages was dependent upon the structural requirements. The cooling load was determined for the tube passages with the 80 mm (3.15 in) maximum separation distance. Since the temperature variation of the panel skin is an important design consideration, the passage spacing was held to this value as being fairly representative.

Results of a typical calculation are depicted in Figures 20 and 21 for a tube radius of 2.54 mm (0.1 in) and a passage length of 6.096 m (20 ft). It is seen that turbulent coolant flow was not fully established, remaining in the transitional Reynolds number region at the end of 6.096 m. The coolant flow and inlet temperature required to maintain the maximum skin temperature at 367°K (660°R) was found to be 90.72 kg (200 lb) per hour starting at 283°K (510°R). All coolant properties were based upon a mixture of 60 percent ethylene glycol/water. Calculations were made at intervals of one foot length.

To arrive at the selection of passage size, five tube radii were investigated. In each case the coolant inlet temperature was varied to determine its effect on coolant flow requirement. The smallest passage size with a reasonable pressure drop had a 2.54 mm (0.1 in) radius tube, using coolant inlet temperature of 283°K (510°R).

The passage sizes studied with their effects on flow rates and pressure drops at various inlet temperatures are tabulated as follows:

TUBE RADIUS		TEMP COOLANT IN		W		ΔP	
mm	(in.)	°K	(°R)	kg/hr	(lb/hr)	kPa	(psi)
7.12	(0.28)	256	(460)	204	(450)	27.5	(3.99)
		283	(510)	397	(875)	28.9	(4.19)
		311	(560)	272	(600)	9.9	(1.77)
		339	(510)	454	(1000)	26.6	(3.86)
5.08	(0.20)	256	(460)	193	(425)	98.8	(14.32)
		283	(510)	272	(600)	71.6	(10.4)
		311	(560)	204	(450)	35.8	(5.19)
		339	(610)	431	(950)	120.3	(17.45)

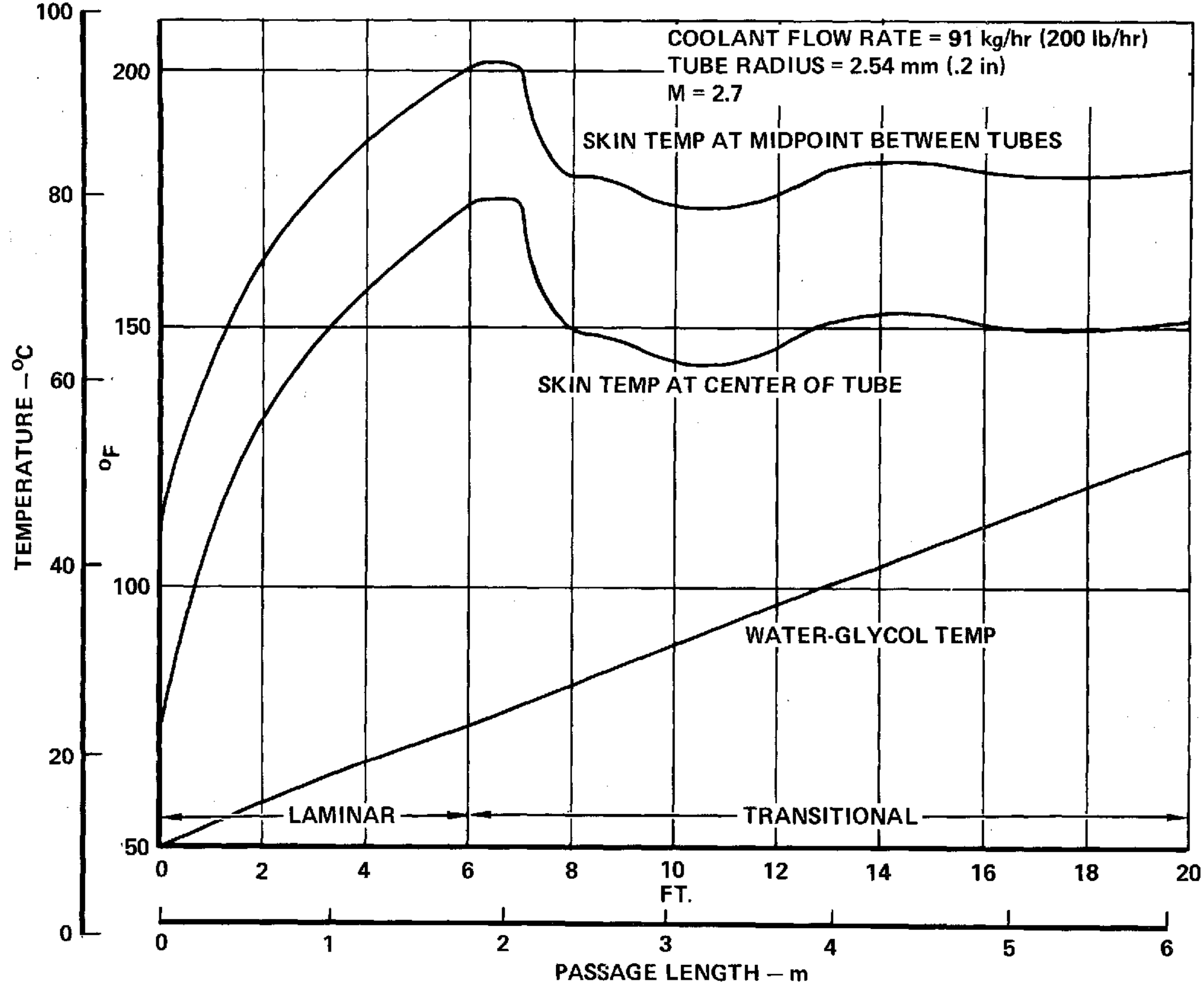

Figure 20. Variation of Skin and Coolant Temperatures Along Passage Length

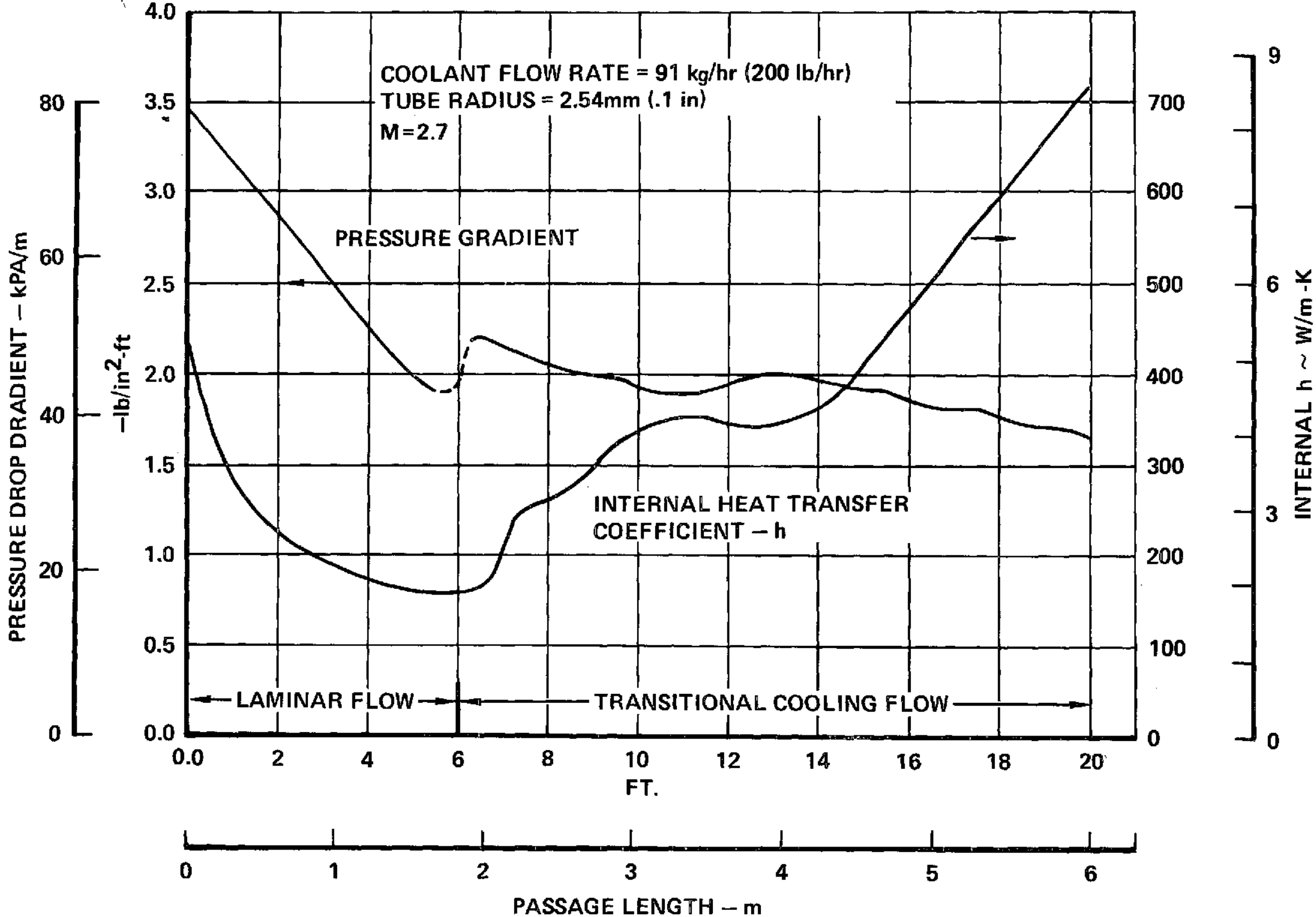

Figure 21. Variation of Pressure Gradient and Internal Heat Transfer Coefficient Along Passage Length

TUBE RADIUS		TEMP COOLANT IN		W		ΔP	
mm	(in.)	°K	(°R)	kg/hr	(lb/hr)	kPa	(psi)
3.81	(0.15)	256	(460)	200	(440)	329	(47.63)
		283	(510)	188	(415)	148	(21.46)
		311	(560)	163	(360)	93.5	(13.56)
		339	(610)	431	(950)	482	(68.41)
2.54	(0.10)	256	(460)	200	(440)	1672	(242.5)
		283	(510)	114	(250)	373	(54.1)
		311	(560)	136	(300)	460	(66.6)
		339	(610)	363	(800)	2390	(346)
		283	(510)	91	(200)	292	(42.4)
1.77	(0.05)	256	(460)	204	(450)	27,700	(4024)
		283	(510)	79	(175)	5,260	(764)
		311	(560)	114	(250)	8,830	(1281)
		339	(610)	363	(800)	64,200	(9316)

The actual maximum metal temperatures are 368°K (662°R) for the Mach 2.7 and 371°K (667°R) for the Mach 3.2 aircraft. These values were conservatively chosen to allow for the effects of overspeed and maneuver. A determination of the exact maximum temperature that would allow an aircraft life of 50,000 hours was felt to be beyond the scope of this preliminary analysis since it would involve the cumulative effect of time and temperature based on the probability of overspeed, frequency of maneuver and would require a transient thermal analysis considering local conditions at the point of maximum panel temperature of the location in question.

4.4.4 Final Results

The cooled areas of the wing and passenger compartment are shown in Figure 22. The rationale for selection of these areas is discussed in the following paragraphs.

As described in Section 4.2, the basic fuel tank concept involves the use of an integral or primary load carrying tank structure covered with both low (422°K max) and high temperature insulation. The insulation is protected with composite heat shield panels which must be removable to allow for inspection and repair of the insulation and tank. As a consequence of this basic design requirement for removability of the heat shields, cooling of the tank areas was considered to be impractical. A previous study (Reference 4) also examined the non-integral tank concept in which the tank is a non-load carrying pressure vessel located within the conventional fuselage structure. In this concept (non-integral) the use of cooled structure

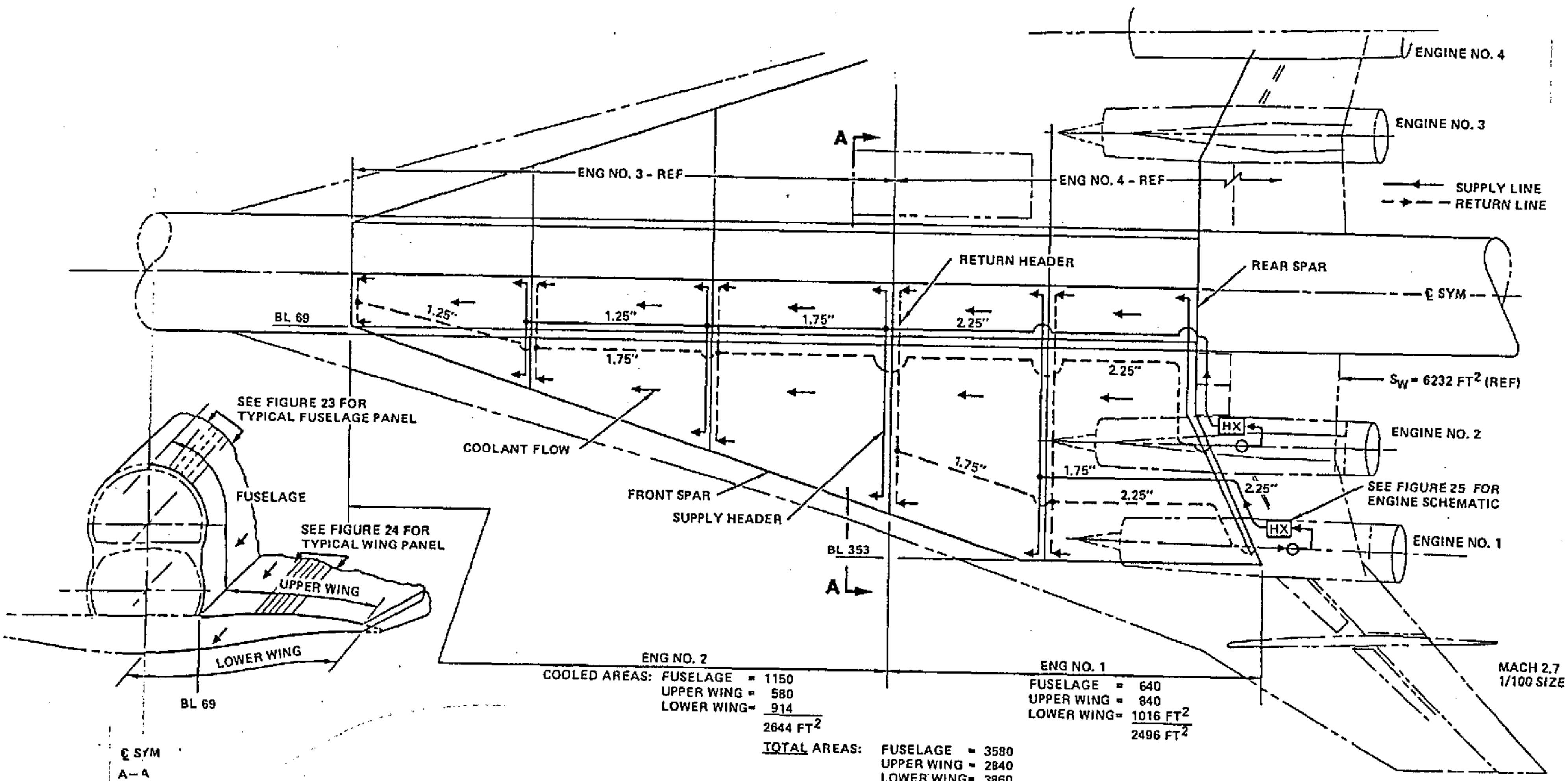

Figure 22. Cooled Panel Arrangement

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is feasible and would allow reduction of the insulation weight while maintaining a constant inflight boil-off of 2.7 percent for the Mach 2.7 aircraft. Using data from the previous study, a weight comparison of the uncooled integral tank and the cooled non-integral concept was made and is tabulated below:

	kg	(lb)
Total uncooled non-integral system weight including tank, insulation, supports and fuselage structure.	16,615	(36,630)
Total uncooled integral system weight including tank, insulation, tank supports and heat shield	14,210	(31,330)
Weight penalty for non-integral tankage	2,405	(5,300)

If the uncooled titanium fuselage of the non-integral concept is replaced with cooled aluminum structure, and insulation is removed to maintain the boil-off constant at 2.7 percent:

	kg	(lb)
Fuselage weight saved	295	(650)
Insulation weight saved	1,424	(3,140)
Penalty for cooling distribution system and fluid	858	(1,450)
Net weight reduction due to cooling	1,061	(2,340)

Total weight of cooled non-integral tankage:

= 15,554 - 1061 (36,630 - 2340) = 15,554 (34,290)

The final comparison shows a net weight penalty of 1344 kg (2960 lb) (15,554 - 14,210 kg) for the cooled non-integral concept compared to the uncooled integral and for this reason the choice was to not attempt cooling of the tank areas and to retain the uncooled integral tank concept.

Remote areas of the aircraft such as the crew compartment and movable surfaces were not considered for active cooling because of the complex plumbing connections and long line runs involved.

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The actual arrangement of the system is also shown in Figure 22. The areas cooled by the fuel used by each engine have been selected to equalize the heat load. Line sizes are indicated. Fuselage and wing panel details are shown in Figures 23 and 24 which also show alternate methods of connecting the individual passages to the headers. The three concepts shown consist of two in which the individual passages are each connected to the headers by either a flexible hose or tube and one in which each four foot wide panel has integral manifolds weld-bonded to the skin and connected in turn to the headers. This reduces the number of individual connections required. A weight comparison of these concepts is included in Section 4.5.2.

Figure 25 is an overall schematic of the coolant/H_2 system for one engine system.

For the fuselage an average heat transfer coefficient was applied for the heat load determination. For the wings, both upper and lower surfaces were divided into regions. An average heat transfer coefficient was calculated for each region as previously described. The total cooling load for the fuselage is based on the single panel with a 6.096 m (20 ft) long passage. The total cooling load for the upper and lower wing surfaces is obtained by summing up the results for the individual panels which have varying passage lengths.

Air conditioning requirements were based upon the use of bleed air from engine compressors, to maintain a cabin altitude of 1,828 m (6,000 ft) during cruise. The air is cooled by a ram air heat exchanger with final cooling accomplished by a separate glycol-to-air heat exchanger. The required air conditioning air flow is 132 kg (290 lb) per minute, which provides 20 CFM per passenger (and crew) of 23.9°C (75°F) air which is comparable to todays wide-body practice. Assuming that the fuselage surface will be cooled down to an average of 79.6°C (174°F), the ram air must be cooled down to about -11°C (12°F) in order to maintain a cabin temperature of 23.9°C (75°F) in cruise. The -11°C air is introduced into the cabin side wall by means of tubing as shown in Figure 26. By this means the sidewall temperature is maintained below 21.1°C (70°F) and the amount of sidewall insulation can be minimized.

The results of the thermal analysis made for the jet fueled AST wing showed that the average heat transfer coefficient for the lower surface was about 39 percent higher than that for the upper surface. This was modified for the LH_2 AST because of its lower angle of attack during cruise. It was estimated that the difference in

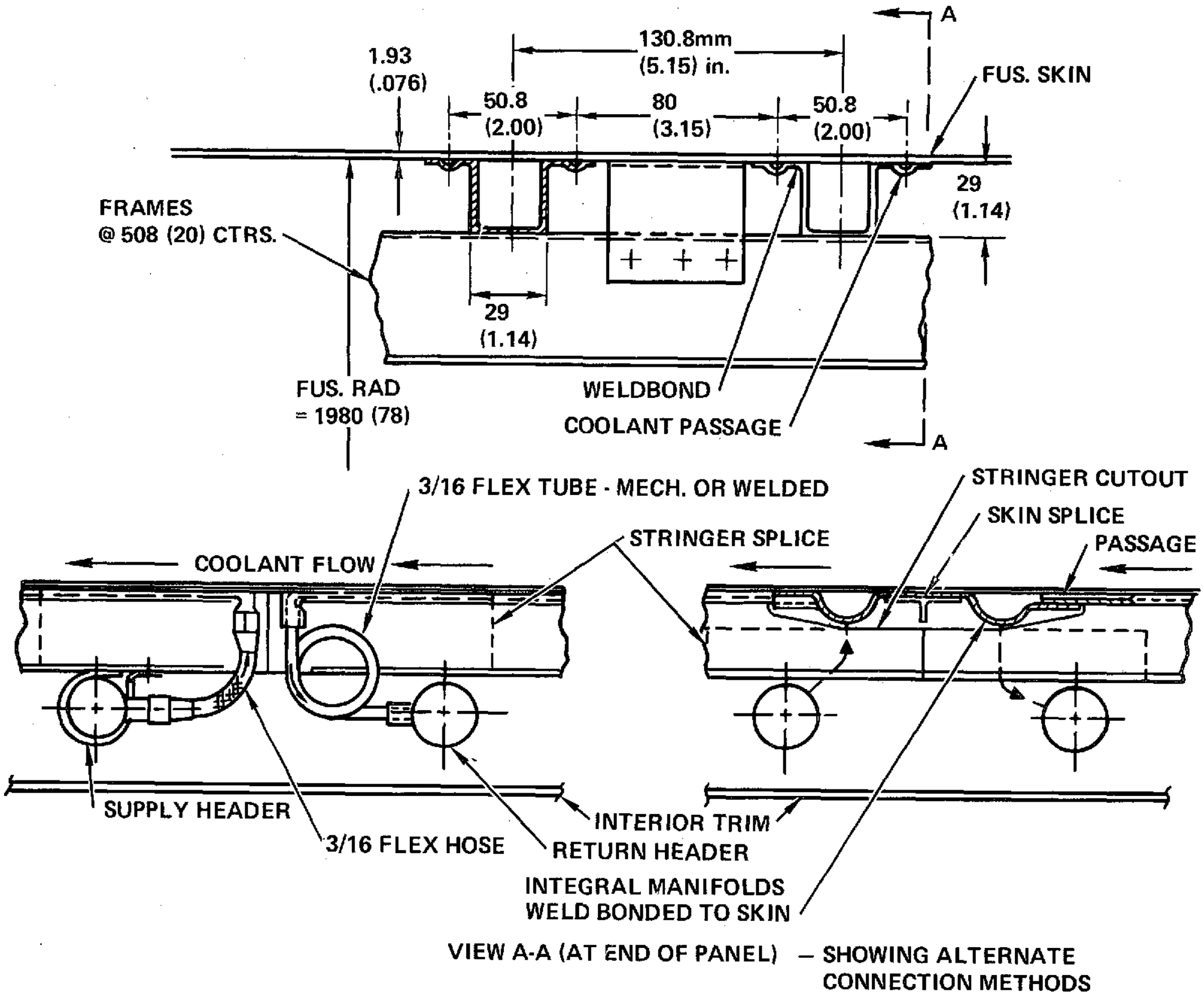

Figure 23. Fuselage Panel Detail

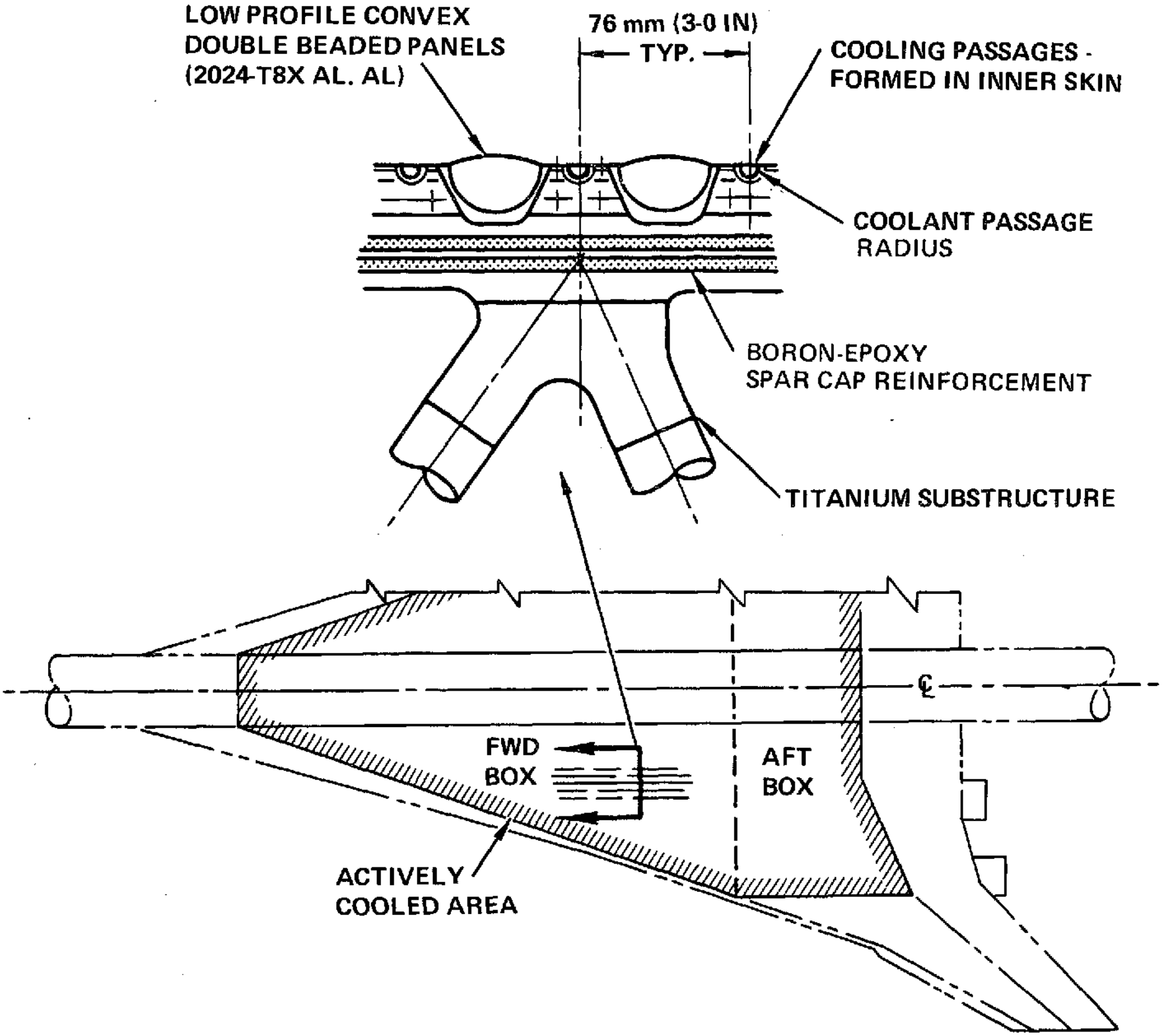

Figure 24. Wing Panel Detail

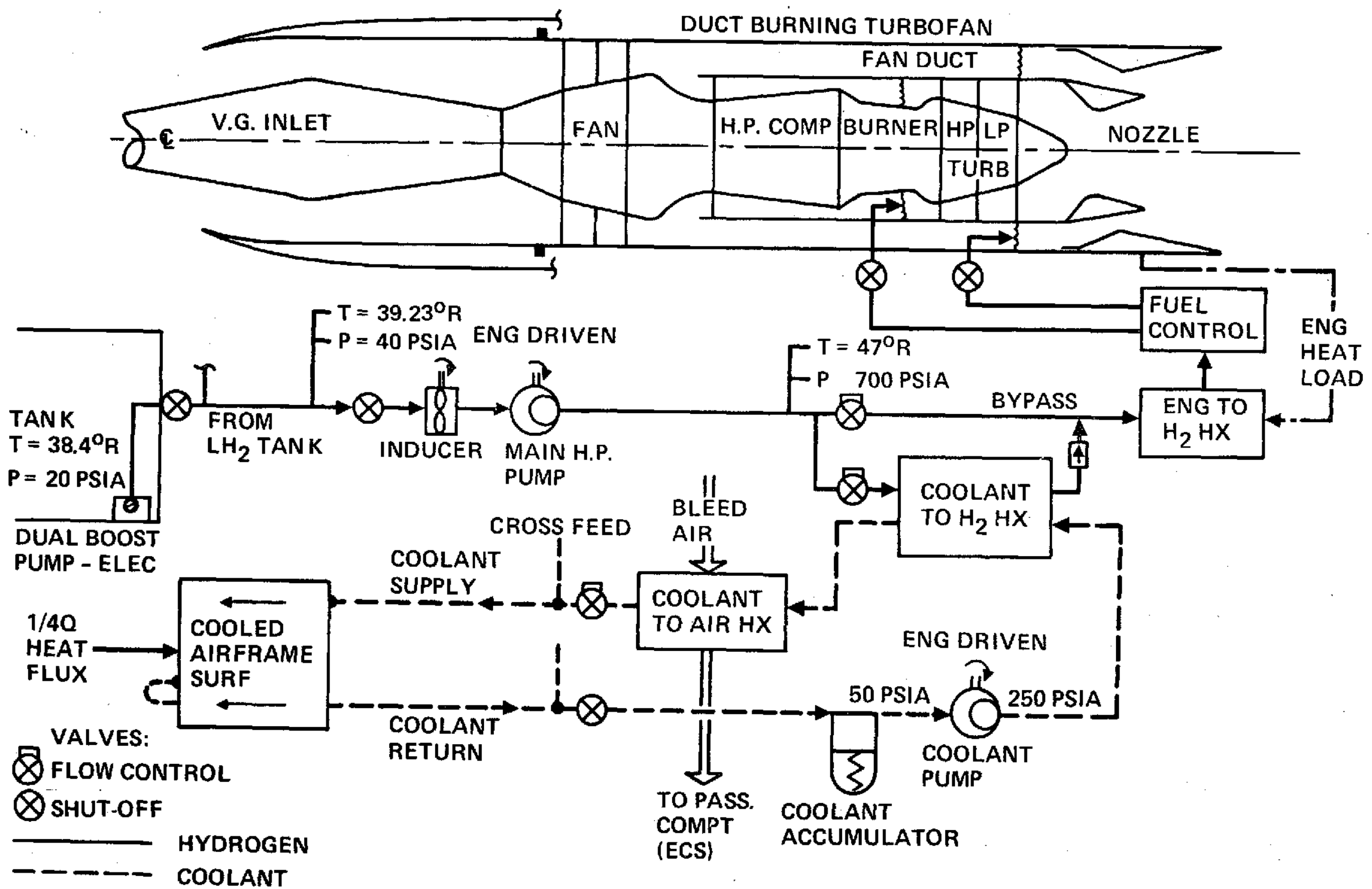

Figure 25. Coolant/H_2 Schematic

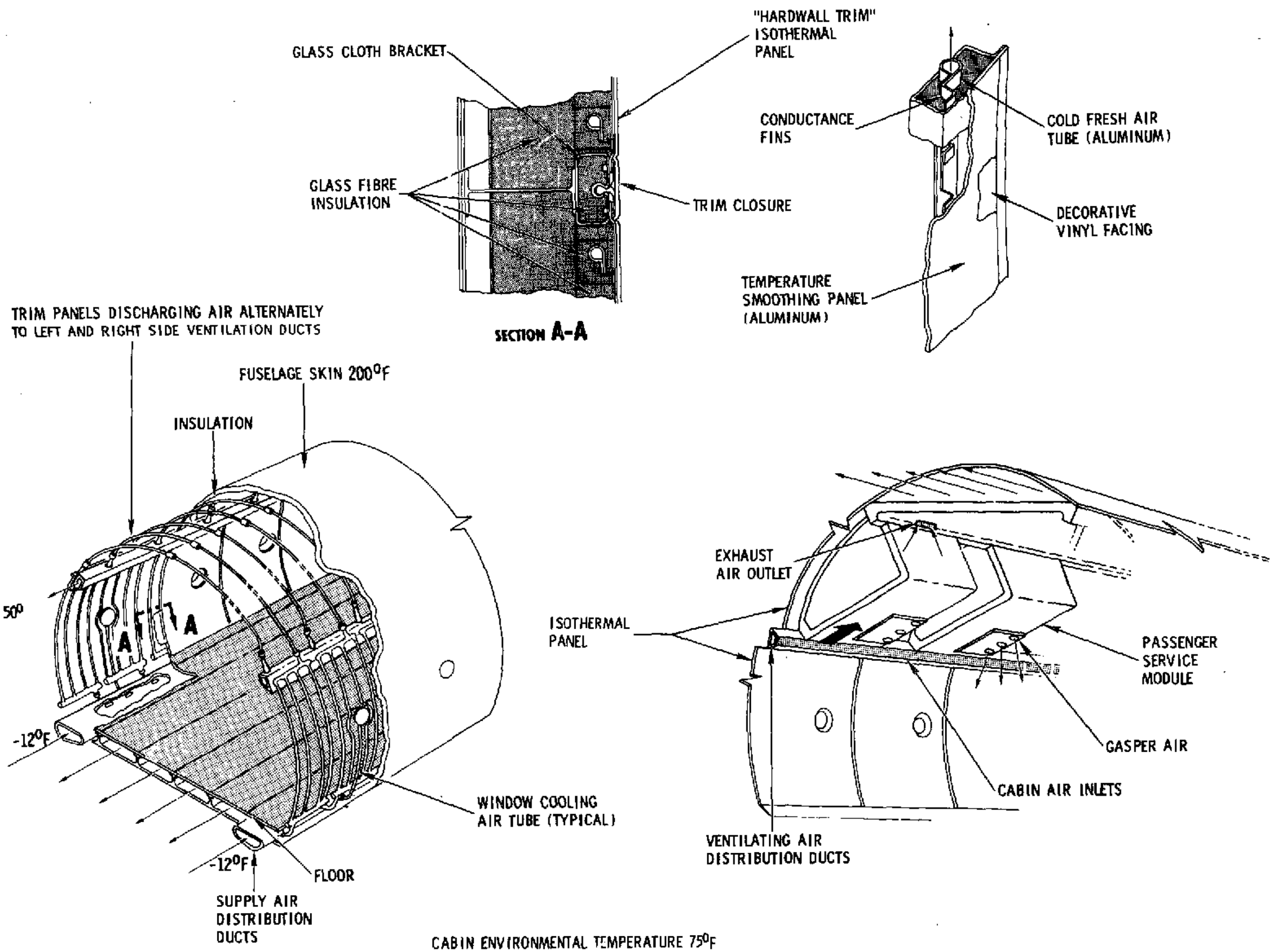

Figure 26. Cabin Air System

lift coefficient required would result in an 8 percent ratio decrease or a 36 percent higher coefficient for the lower surface than for the upper surface. For each wing panel the average of both upper and lower heat transfer coefficients was used in the calculations, and an average heat load determined for each panel. After obtaining the total cooling load for upper and lower wing areas, the ratio was applied to obtain separate loads for the upper and lower wing surfaces. These loads were further adjusted to account for the difference in wing upper and lower areas on the basis of the calculated unit heat load for each surface.

The above calculation procedure was used for the Mach 2.7 aircraft. For the Mach 3.2 aircraft, the Mach 2.7 cooling loads were modified by the ratios of external heat transfer coefficients, based on an average Reynolds number and by the ratios of temperature differences between the adiabatic wall temperature and the average surface temperature. Table 4 summarizes data for both the Mach 2.7 and 3.2 cooled aircraft.

As explained in notes B and E of Table 4, the Mach 3.2 aircraft used 100 percent of the hydrogen heat sink while cooling about 87 percent of the wing area available for cooling. In order to increase this heat sink capability the use of a hydrogen expansion turbine in place of the engine to drive the coolant pump was investigated. The main hydrogen pump and possibly other units could also be driven during cruise flight but this would require an alternate power source during lower speed flight.

The turbine was located at approximately the mid-temperature point of the hydrogen/coolant heat exchanger. Due to the high specific heat of hydrogen gas the pressure and temperature ratios across the turbine required to drive the coolant pump are very low. For example, to drive the 44.3 KW (59.3 HP) coolant pump (1/4 of the total) the pressure ratio is 0.92 and the temperature drop at 90 percent turbine efficiency is $3.3^{o}K$ ($5.9^{o}F$). This would provide an increase of only 1.1 percent in the heat sink assuming no line or turbine heat leak, consequently the concept was rejected.

It is recognized that other means, such as a secondary cooling loop, are possible that could reject heat to the hydrogen at a higher temperature but were considered beyond the scope of the technology described in Reference 2 on which this study was based.

TABLE 4. COOLED AIRCRAFT DATA

			MACH 2.7		MACH 3.2	
BASELINE AIRCRAFT (Ref.):						
Gross Weight	kg	(lbs)	163,783	(361,075)[A]	198,433	(437,594)
Wing Area	m^2	($ft.^2$)	579	(6,232)	893	(9,613)
Cruise Alt.	m	(ft)	20,726	(68,000)	23,165	(76,000)
Cruise L/D	-		6.85	6.85	7.72	7.72
Cruise SFC	$\frac{kg}{hr}$/da N	$\frac{lb}{hr}$/lb	.563	(0.553)	.608	(.597)
Cruise Fuel Flow	kg/hr	(lb/hr)	11,500	(25,300)	13,320	(29,400)
COOLED AREAS						
Fuselage	m^2	($ft.^2$)	333	(3,580)	333	(3,580)
Upper Wing	m^2	($ft.^2$)	264	(2,840)	344	(3,700)
Lower Wing	m^2	($ft.^2$)	359	(3,860)	464	(5,000)
Total			956	(10,280)	1,141	(12,280)
COOLING HEAT LOADS						
Fuselage	kW	(Btu/hr (10^6))	2,340	(8.00)	4,130	(14.10)
Upper Wing	kW	(Btu/hr (10^6))	2,230	(7.60)	4,760	(16.26)
Lower Wing	kW	(Btu/hr (10^6))	4,100	(14.00)	8,590	(29.30)
Envir. Control System	kW	(Btu/hr (10^6))	304	1.04)	422	(1.44)
Total			8,974	(30.64)	17,902	(61.10)
PASSAGE RADIUS						
Fuselage	mm	(in.)	2.54	(0.10)[C]	3.18	(0.125)[C]
Upper Wing	mm	(in.)	2.54	(0.10)	3.18	(0.125)
Lower Wing	mm	(in.)	3.05	(0.12)	3.55	(0.14)
COOLANT (60/40%)						
Coolant Temp. In	°K	(°R)	284	(510)	284	(510)
Coolant Temp. Out	°K	(°R)	327	(587)	332	(597)[B]
Total Coolant Flow	kg/hr	lb/hr	229,000	(505,000)	406,000	(897,000)
PRESSURE DROP (MAX.)						
Supply Manifold	kPa	($lbs/in.^2$)	296	(43)[C]	296	(43)[C]
Panel	kPa	($lbs/in.^2$)	372	(54)	372	(54)
Return Manifold	kPa	($lbs/in.^2$)	290	(42)	290	(42)
Heat Exchanger	kPa	($lbs/in.^2$)	420	(61)	420	(61)
Pump Pressure Rise	kPa	($lbs/in.^2$)	1378	(200)	1378	(200)
HEAT EXCHANGER						
H_2 Temp. In	°K	(°R)	26.2	(47)	26.2	(47)[D]
H_2 Temp. Out	°K	(°R)	200	(359.5)	324	(582)
Coolant Temp. In	°K	(°R)	327	(587)	332	(597)
Coolant Temp. Out	°K	(°R)	292	(507.2)	283	(508.2)
Min. T	°K	(°R)	392	(687.5)	264	(475)[E]
Max. T	°K	(°R)	512	(920.2)	513	(921.2)
Log Mean ΔT	°K	(°R)	183	(329)	384	(591)

NOTES:

A. These weights represent the uncooled aircraft before incorporation of the coolant system.

B. The cooled wing areas shown for the Mach 3.2 case represent about 86.5 percent of the area available for cooling. (100 percent was cooled at M 2.7). This limitation was caused by a lack of hydrogen heat sink. To alleviate this condition, the coolant out temperature was raised 10°F (to 137°) and the heat exchanger pinch point temperature was set at a minimum of 15°F. The maximum peak skin temperature (see Figure 20) is estimated to be 207°F at the transition point under this condition.

C. The passage size was chosen to limit the pressure drop to a maximum of 54 psig with the flow rate required by the panel heat load. The supply and return manifold pressure drops shown are for the most remote (forward) panels. See Section 4.5.2 for effect of pressure drop allocation on system weight.

D. This temperature includes the estimated rise in temperature across both the tank boost pump and the main engine pump.

E. This minimum pinch point temperature difference dictated the maximum area that could be cooled on the Mach 3.2 aircraft.

4.5 WEIGHTS

The parametric weight equations are the same as used previously in the NASA-Ames AST Concept Study - Hydrogen Fueled Configuration (Reference 3), except for the following items which are described in this section:

- Wing and Passenger Compartment Structural Weights
- Materials Distribution
- Mach 3.2 (New)
- Environmental Control System
- Cooling System

4.5.1 Structural Weights

This section describes the modifications and weight changes resulting from the incorporation of the cooling system in the uncooled design described in Section 4.2.

Wing: A chordwise stiffened wing design, as adopted for the uncooled airplane (Figure 15), is employed for the wing box structure from the fuselage side (BL 69) to the outboard engine pylon (BL 353). (See Figure 22.) This design was selected for structural efficiency (Reference 6), and was well suited for integrating the cooling system design with the structure with minimum changes. The stiffness-critical outer wing structure remains titanium honeycomb construction.

Strength and manufacturing considerations dictate the use of titanium alloy (Ti-6Al-4V annealed) for the wing substructure (spars, ribs) to achieve a minimum weight design. The submerged spar caps, which transmit the wing bending moments, are titanium alloy reinforced with unidirectional boron-epoxy composites.

Aluminum alloy (2024-T81) surface panels of a low profile, double-beaded skin design are used extensively. These efficient circular-arc sections of sheet metal construction have coolant passages formed integrally with the inner beaded skin (Figure 24), and are joined to the outer skin by weld bonding. The shallow protrusions provide smooth displacements under thermally induced strains and operational loads and offer significantly improved fatigue life. The uncooled design requires sheet thicknesses slightly greater than minimum gage in the aft box (Table 5). However, the buckling efficiency of the minimum gage aluminum panels provides an 8 percent weight saving in panel weight over the uncooled titanium alloy design. For the cooled design, the net weight saving in the wing box structure is approximately 2.6 percent as shown in Table 5.

TABLE 5. WING BOX DESIGN (MACH 2.7)

ITEM		UNITS	UNCOOLED		ACTIVELY COOLED		REMARKS
1. Material			Titanium Alloy - TI-6Al-4V annealed surface and substructure w/composite reinf.		Aluminum Alloy - 2024T81 surface; TI-6Al-4V annealed w/comp reinf. substr.		Actively cooled panels; min. wt substructure design - titanium alloy
2. Design Temperature		K (F)	Room Temp		Room Temp		Critical Condition at R.T.
3. Forward Box:	Upper-Outer	mm (in)	0.380	(0.015)	0.610	(0.024)	Minimum gage design for both titanium and aluminum is approximately the same weight (S_{FB} = 2607 ft^2)
	Upper-Inner	mm (in)	0.254	(0.010)	0.406	(0.016)	
	$\bar{t}_u$	mm (in)	0.736	(0.029)	1.14	(0.045)	
	Lower-Outer	mm (in)	0.508	(0.020)	0.813	(0.032)	
	Lower-Inner	mm (in)	0.254	(0.010)	0.406	(0.016)	
	$\bar{t}_\ell$	mm (in)	0.863	(0.034)	1.35	(0.053)	
	Box weight	kg (lb)	4,798	(10,577)	4,723	(10,413)	ΔW = 75 kg (164 lb)
4. Aft Box:	Upper-Outer	mm (in)	0.380	(0.015)	0.610	(0.024)	Minimum gage design for aluminum; inner skins for uncooled min gage (see fwd box)
	Upper-Inner	mm (in)	0.330	(0.013)	0.406	(0.016)	
	$\bar{t}_u$	mm (in)	0.838	(0.033)	1.14	(0.045)	
	Lower-Outer	mm (in)	0.508	(0.020)	0.813	(0.032)	
	Lower-Inner	mm (in)	0.345	(0.014)	0.405	(0.015)	
	$\bar{t}_\ell$	mm (in)	1.04	(0.040)	1.35.	(0.053)	
	Box weight	kg (lb)	3,835	(8,455)	3,628	(7,998)	ΔW = 207 kg (457 lb)
5. Tip	Box weight	kg (lb)	2,284	(5,036)	2,284	(5,036)	No cooling of stiffness critical tip structure
6. Wing Box	Total weight	kg (lb)	10,917	(24,068)	10,636	(23,447)	Cooled structure is 2.6% lighter than uncooled. Surface panel weight savings is 282 kg (621 lb)

Passenger Compartment: The passenger compartment structure is of aluminum alloy (2024T81) construction, cooled to a nominal $367^{o}K$ ($660^{o}R$) and is critical at the Mach 2.7 cruise condition. To provide a structure that will have a service life of 50,000 flight hours, appropriate multiplying factors are applied to the design life for use in establishing allowable design stresses. For structure subjected to a spectrum loading, such as the compartment stiffeners, the allowable stress (~50,000 psi) is selected using a factor of 2 times the service life of 50,000 hours. For areas of the fuselage structure such as the passenger compartment skin and frames subjected to constant amplitude loading, the allowable stresses are selected for 200,000 design flight hours of service (50,000 x 4). A larger factor is applied to this constant amplitude loading because the scatter in fatigue test data is larger for this type of loading. The maximum operational design stress level applicable to the aluminum alloy fuselage skin in hoop tension is 14,000 psi. This reduced value is also selected for the fuselage skin since it is subjected to biaxial stresses due to operating pressure, external aerodynamic pressure, and thermal loads. For design, the latter accounts for approximately 15 percent of the allowable design stress. The skin thickness required to limit the gross area stress to 11,900 psi (.85 x 14,000) is 1.93 mm (0.076 in.). This results in a 10.5 percent increase in weight over the uncooled titanium skin which is 1.09 mm (.043 in.) for the passenger compartment skin, as shown in Table 6.

The stiffeners are sized to provide the section modulus so that the applied bending moments for a positive maneuver (n_z = 2.5) results in adequate margins of safety consistent with the failure modes for compression design (i.e. crippling, column) at the appropriate design temperature. The buckling efficiency of the aluminum skin permits increased stiffener spacing circumferentially as shown on Figure 23. The aluminum stiffener design, with the integral cooling passages, results in 25 percent weight saving over the uncooled titanium design. The stiffener weight saving more than compensates for the heavier skins required and results in a 6.3 percent saving in passenger compartment shell structure weight. Pertinent results are shown in Table 6.

The materials distributed for the cooled versus uncooled wing and fuselage structure is given in Table 7.

The major structure weights for the uncooled Mach 3.2 aircraft, with the exception of the hydrogen tanks, are increased 5 percent due to the strength degradation with increased temperatures over the uncooled Mach 2.7.

TABLE 6. PASSENGER COMPARTMENT SHELL DESIGN (MACH 2.7)

ITEM	UNITS	UNCOOLED		ACTIVELY COOLED		REMARKS
1. Material	-	Titanium Alloy TI-6Al-4V (annealed)		Aluminum Alloy 2024T81		Representative aluminum alloy for cooled design
2. Design Temperature	K (F)	422K	(300F)	366K	(200F)	Average stringer temp. at start of cruise
3. t_S, Skin Thickness	mm (in)	1.09	(0.043)	1.93	(0.076)	Minimum skin thickness required for cabin pressurization (80.67 kPa)
4. F_g, Allow gross area stress	kPa (psi)	172,369	(25,000)	96,527	(14,000)	Max circumferential (Hoop) stress. Assume 15% attrib. to thermal effects
5. A_{ST}, Stiffener Area	$mm^2 (in^2)$	151	(0.234)	225	(0.349)	Shell bending strength
6. S, Stiffener Spacing	mm (in)	112	(4.40)	131	(5.15)	
7. $\bar{t}_{ST}$, Equiv Thickness	mm (in)	1.35	(0.053)	1.72	(0.068)	$[A_{ST} \div S]$
8. $\bar{t}_{SHELL}$, Equiv Thickness	mm (in)	2.44	(0.096)	3.65	(0.144)	$[t_{SK} + \bar{t}_{ST}]$
9. I_{SHELL}, Moment-of-Inertia	$m^4 (in^4)$	0.152	(0.365×10^6)	0.228	(0.547×10^6)	$[121 \pi \times 10^6 \ \bar{t}_{SHELL}]$
10. C, Distance to Extreme Fiber	m (in)	2.96	(116.4)	2.96	(116.4)	[R + 39.0]

TABLE 6. PASSENGER COMPARTMENT SHELL DESIGN (MACH 3.2) (Continued)

ITEM	UNITS	UNCOOLED	ACTIVELY COOLED	REMARKS
11. M, Bending Moment	Nm(in-lb)	25×10^6 (225×10^6)	25×10^6) (225×10^6)	Positive Maneuver, $n_z = 2.5$
12. f_{b_c}, Bending Stress	kPa (psi)	495,000 (71,800)	330,000 (47,900)	[$Mc \div I_{SHELL}$]
13. F_{cc}, Allowable Stress	kPa (psi)	514,000 (74,500)	346,000 (50,200)	Crippling stress at design temperature
14. Ult. Margin of Safety	-	0.04	0.05	[$(F_{cc} \div f_{b_c})-1$]
PASSENGER COMPARTMENT WEIGHT SUMMARY (S_{REF} = 439 m^2 (4722 ft^2): Non-optimum factor = 1.14				
15. Skin	kg (lb)	2,419 (5,333)	2,672 (5,891)	$W_{COOLED} = 1.105\ W_{UNCOOLED}$
16. Stiffeners	kg (lb)	2,981 (6,573)	2,391 (5,271)	$W_{COOLED} = 0.802\ W_{UNCOOLED}$
17. Total Shell	kg (lb)	5,400 (11,906)	5,063 (11,162)	$W_{COOLED} = 0.938\ W_{UNCOOLED}$

TABLE 7. MATERIALS DISTRIBUTION (PERCENT)

	UNCOOLED STRUCTURE MACH 2.7	UNCOOLED STRUCTURE MACH 3.2	ACTIVELY-COOLED STRUCTURE MACH 2.7 AND 3.2
WING:			
Aluminum	4.6	0	22.4
Titanium	85.6	91.4	68.0
Steel	2	2	2
Composites	6.2	5	6
Other	1.6	1.6	1.6
FUSELAGE:			
Aluminum	32.6	32.6	74
Titanium	51.4	51.4	10
Steel	1.8	1.8	1.8
Composites	2.5	2.5	2.5
Other	11.7	11.7	11.7

Table 8 shows the final weight saving based on the total cooled wing and fuselage. The saving is lower than shown above for the wing box and fuselage shell since it represents the total group weight and includes the uncooled wing control surfaces, outboard tips, flight compartment, tail cone, interior, and fuel tanks. The Mach 3.2 case shows increased saving because its initial uncooled weights were increased 5 percent as explained above, thus allowing a larger saving when cooled aluminum structure is incorporated.

TABLE 8. WEIGHT SAVING FOR COOLED STRUCTURE

	MACH 2.7		MACH 3.2	
	UNCOOLED	COOLED	UNCOOLED	COOLED
Wing (Total)	0	-1.32%	0	-3.24%
Fuselage (Total)	0	-1.9%	0	-3.42%

4.5.2 Cooling System Weights

This section describes how the weights of the cooling system (and fluid) were determined. A general discussion is given below, followed by the actual weight break down.

Distribution System: A tradeoff study of the effect of the relative pressure drop between the panel and the distribution system on system weight was conducted for the Mach 2.7 system assuming that the total system pressure drop is 1380 kPa (200 psi), (see Table 4), with 420 kPa (61 psi) allowed for the heat exchanger. This leaves a total of 958 kPa (139 psi) to be allocated between the panel and the distribution system. The maximum metal temperature and consequently the heat flux was assumed to be unchanged in the panel. The results are presented in Figure 27 which shows that the design point panel pressure drop of 372 kPa (54 psi) is within 13.6 kg (30 lb) of the minimum total system weight at 40 psig. On this basis, a design point pressure drop of 372 kPa (54 psi) was used for both the Mach 2.7 and 3.2 aircraft. Using this pressure drop distribution, typical line sizes are tabulated below for the forward panels (Engines No. 2 and 3).

	MACH 2.7		MACH 3.2	
Supply and Return Dia.	mm	(in.)	mm	(in.)
Eng. to aft panel	57.2	(2.25)	69.8	(2.75)
Aft to mid panel	44.5	(1.75)	67.2	(2.25)
Mid to fwd panel	31.8	(1.25)	41.2	(1.62)
Headers (Typical)	28	(1.1)	31.8	(1.25)

The system maximum working pressure is 1722 kPa (250 lbs/in^2) and wall thickness was determined with a suitable factor of safety but in no case was it allowed to be less than 0.71 mm (0.028 in.) for practical installation and handling. Weight allowances for fittings, bellows and mounting were also estimated.

Three alternate methods of connecting the individual passages to the distribution system were shown in Figure 23. A weight comparison of these methods is tabulated below for the Mach 2.7 aircraft.

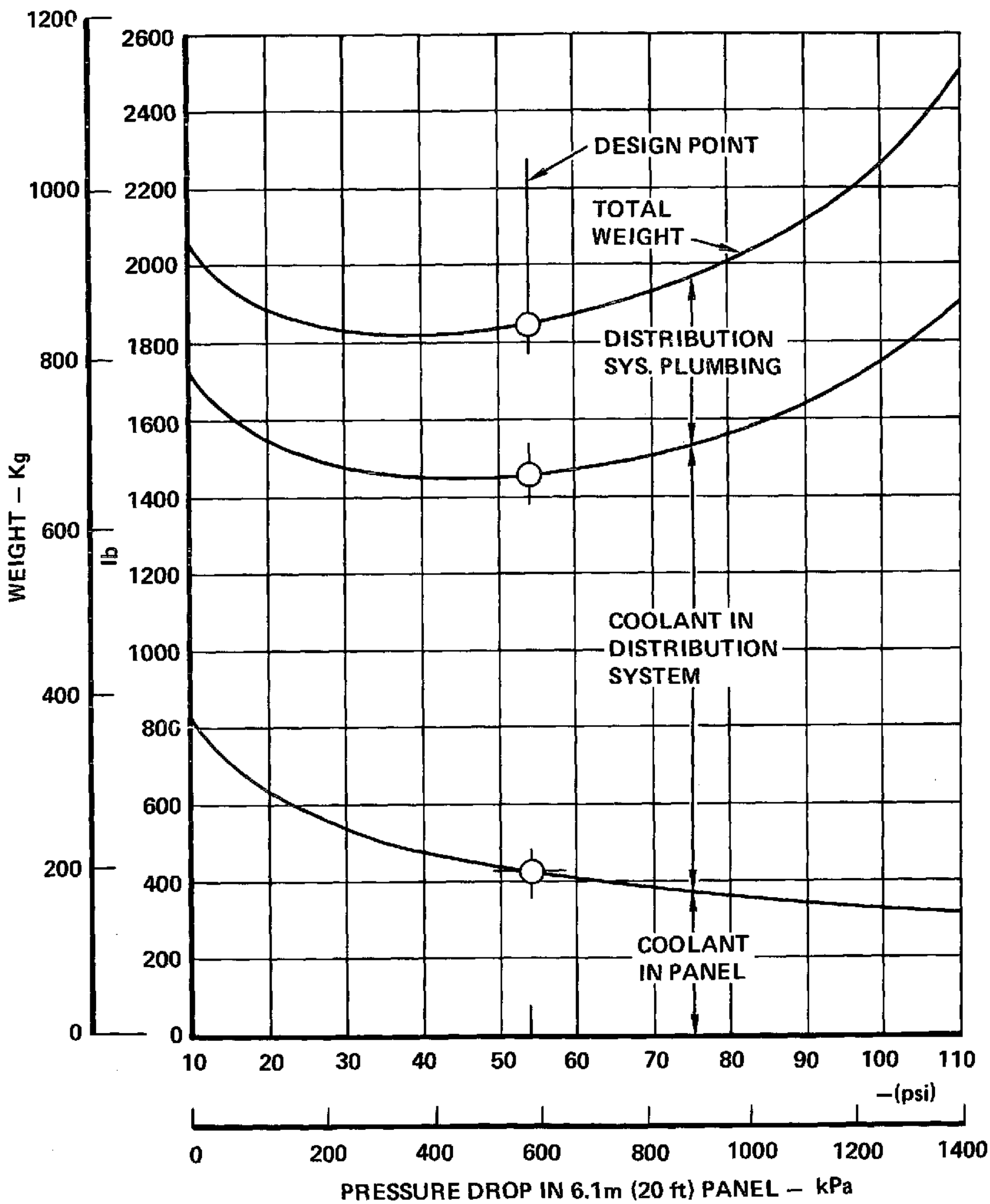

Figure 27. Effect of Pressure Drop Distribution on System Weight

	FLEX. HOSE		FLEX. (TUBE		INTEGRAL MANIFOLD	
Plumbing or manifold weight Kg (lb)	397	(765)	49	(108)	43.5	(96)
Fluid weight Kg (lb)	5	(11)	14	(31)	45.8	(109)
Total	352	(776)	63	(139)	89.4	(197)

The weight of the integral manifold system considered the weight saved by the stringer cutout and the reduction of individual connections, assuming 1.22 m (4 ft) wide panels. The flexible tube connection was chosen over the flexible hose because of weight and reliability advantages and was felt to be a less costly concept than the integral manifold approach. Furthermore, it was not susceptible to cracks parallel to the passages which would cause loss of the panel coolant as in the case of the integral manifold.

Pumps: The pumps are driven by a power takeoff unit (declutchable) from the engine gear box. The pumps are conventional, centrifugal type with an efficiency of 82 percent and a pressure rise of 1380 kPa (200 lb/in^2). This gives a power per pump (4 pumps) of 24.9 KW (33.4 HP) for the Mach 2.7 and 44.3 KW (59.3 HP) for the Mach 3.2 aircraft.

Reservoirs: Reservoirs were assumed to hold a system residual pressure of 345 kPa (50 lbs/in^2) and were sized by the change in total fluid volume caused by a fluid temperature excursion from 220 to 339°C (395 to 610°R).

Heat Exchangers: The coolant to hydrogen heat exchangers represent probably the greatest degree of uncertainty with regard to performance and weight. Funding limitations prevented the use of a computer program (similar to the panel analysis) that would be required to survey the many possibilities. The data of Reference 2 was reviewed but was not used as neither the coolant side heat transfer coefficient nor the heat exchanger weight could be confirmed. The difficulty encountered was in the correlation of available heat transfer data at the extremely low coolant film temperature involved. An estimate was made of the average coolant temperature using the log mean temperature difference with the following results:

			MACH 2.7		MACH 3.2	
Heat Load/Exchanger	kW	(Btu/Hr x 10^6)	2250	(7.66)	4475	(15.28)
Coolant in Temp.	°K	(°R)	327	(587)	332	(597)
Coolant out Temp.	°K	(°R)	282	(507.2)	282.5	(508.2)

			MACH 2.7		MACH 3.2	
Hydrogen in Temp	^oK	(^oR)	26.1	(47)	26.1	(47)
Hydrogen out Temp.	^oK	(^oR)	200	(359.5)	32.3	(580)
Log Mean Temp. ΔT	^oK	(^oR)	183	(329)	72.8	(131)
Heat Transfer Coeff:						
Coolant Side	W/mK	(Btu/hr ft^2 R^o)	3.51	(292)	5.7	(475)
Hydrogen Side	W/mK	(Btu/hr ft^2R^o)	9.6	(800)	9.6	800
Overall	W/mK	(Btu/hr ft^2R^o)	2.53	(214)	3.48	(298)
Heat Exchange Area	m^2	(ft^2)	10.12	(109)	36.3	(391)

The incrase in the coolant side coefficient for the Mach 3.2 case is due to the higher film temperature caused by the smaller log mean temperature difference.

The area calcaulated above was used as the basis for the heat exchanger core weight reported in Table 9.

4.5.3 Environmental Control System (ECS)

The cooling system weights listed in Table 9 are offset to some extent by the reduction in ECS weight. The cooled cabin wall allows a reduction in both equipment and insulation weight by limiting the heat load to essentially that of a Mach 2 aircraft. Further weight reduction is limited because of the basic requirement of providing a sufficient flow of cooled fresh air for ventilation as described in Section 4.4.4. A comparison of the uncooled and cooled aircraft ECS weights is given below. By comparison, the weight of the cooled aircraft systems are only about 30 percent heavier than the L-1011 on a per passenger basis:

	MACH 2.7		MACH 3.2	
Uncooled ECS Weight kg (lb)	3,575	(7,880)	4,658	(10,269)
Cooled ECS Weight kg (lb)	2,907	(6,408)	2,952	(6,508)
Weight Saving	668	(1,472)	1,706	(3,761)

The net effect of both the cooling and ECS system weights is a penalty of 607 kg (1338 lb) for the Mach 2.7 aircraft and 480 kg (1057 lb) for the Mach 3.2.

The slightly higher weight 45.4 kg (100 lb) of the Mach 3.2 system is due to the larger heat exchangers (coolant to air) required at the higher engine bleed temperature at Mach 3.2.

TABLE 9. COOLING SYSTEM WEIGHT SUMMARY

	MACH 2.7		MACH 3.2	
	kg	(lb)	kg	(lb)
EQUIPMENT				
1. Distribution system (including Headers)	201	(444)	268	(591)
Outbd Systems #1 and #4	47	(104)	59	(130)
Inbd Systems #2 and #3	105	(232)	134	(296)
Flex tubes and bosses (Header to Passages)	49	(108)	75	(165)
2. Pump Instl.	40	(88)	54	(118)
Pumps (4)	27	(60)	39	(85)
Power Takeoff (4)	9	(20)	11	(25)
Installation	4	(8)	4	(8)
3. Reservoir Instl.	26	(56)	37	(82)
Reservoir (4)	22	(48)	33	(74)
Installation	4	(8)	4	(8)
4. Heat Exchanger Instl.	107	(236)	232	(512)
Core Wt.	37	(80)	129	(284)
Headers	65	(144)	93	(206)
Installation	5	(12)	10	(22)
5. Controls, Valves, Sensors, Etc.	118	(260)	145	(320)
Sub-Total (Equipment)	492	(1,084)	736	1,623
FLUID				
1. Distribution System	448	(988)	806	(1,777)
Outbd System #1 and #4	141	(310)	248	(547)
Inbd Systems #2 and #3	307	(678)	558	(1,230)
2. Coolant in Panels	173	(380)	260	(574)
Fuselage	56	(123)	60	(132)
Upper Wing	40	(87)	74	(164)
Lower Wing	77	(170)	126	(278)
3. Pumps (4)	9	(20)	16	(35)
4. Reservoirs (4)	28	(62)	45	(99)
5. Heat Exchangers (4)	62	(136)	218	(480)
Sub-Total (Fluid)	720	(1,586)	1,345	(2,965)
TOTAL SYSTEM WEIGHT				
Equipment	492	(1,084)	736	(1,623)
Fluid	720	(1,586)	1,345	(2,965)
Contingency	63	(140)	105	(230)
Total Weight	1,275	(2,810)	2,186	(4,818)

The above system weights, while calculated for the uncooled aircraft, are scaled in proportion to the total cooled area when the cooled aircraft is resized.

Since relatively cool cabin exhaust air is used to cool the cargo compartment, some of the equipment, and the landing gear bays, no change in operating environment or weight was assumed from the incorporation of the cooling system.

The structural and system weights, together with the cost relations described in Section 4.7 form the basis for inputs to the ASSET vehicle synthesis program for determination of the cost and performance of the cooled vehicles.

4.5.4 Variations in Fuel Consumption Caused by Cooling

The effect on the basic vehicle caused by incorporation of the cooling system was examined with regard to the following areas:

- Skin friction increase in cooled areas
- SFC decrease due to fuel enthalpy increase
- Additional fuel required for descent cooling at end of cruise
- SFC penalty for coolant pump horsepower extraction

Typical calculations for the Mach 2.7 aircraft are discussed below:

Skin Friction: Table 10 shows the increase in skin friction in the cooled areas. These values were determined in the aerodynamic heating analysis program described in Appendix B. Integration of these values results in an overall increase of 9.82 percent in the cooled areas shown in Figure 22. Consideration of the total vehicle wetted area reduces this to an equivalent of 3.5 percent overall. Applying this value to the friction drag coefficient gives a decrease of 1.48 percent in L/D during cruise. This is equivalent to an increase of 374 kg (825 lb) of fuel required for cruise.

TABLE 10. SKIN FRICTION INCREASE IN COOLED AREAS

WING B L	$\Delta C_f/C_f$ UNCOOLED (%)
80 to 130 in.	9.20
130 to 180 in.	9.14
180 to 230 in.	9.11
230 to 280 in.	9.13
280 to 330 in.	9.22
330 to 390 in.	9.48
FUSELAGE	
F.S. 1610 to 2450 in.	10.9

SFC Decrease: The enthalpy added to the fuel by the coolant heat load amounts to 1190 Btu's/lb. The relative change in SFC is then:

$$\frac{\text{SFC uncooled}}{\text{SFC cooled}} = \frac{51590 + 1190}{51590} = 1.023 \text{ or } 2.3\%$$

where

51,590 B/lb = Fuel Heating value

This is equivalent to a fuel saving of 580 kg (1280 lb) during cruise.

Descent Cooling: The additional fuel required to maintain cooling at the end of cruise is estimated as 204 kg (450 pounds). This assumes that fuel in excess of that required by the engine must be expended down to Mach 1.95 at which time the skin temperature is 367°K (660°F).

Pump horsepower extraction: The fuel penalty for driving the coolant pump during cruise is estimated as 1.135 lb/HP-eng.

Therefore, since the pump HP/eng is 33.4:

$$\Delta W \text{ Fuel} = 1.135 \times 33.4 \times 4 \text{ eng} = 69 \text{ kg } (152 \text{ lb})$$

The final results are summarized below:

	Wt. Fuel kg	Wt. Fuel (lb)
• Fuel increase due to skin friction	+374	(+825)
• Fuel decrease due to SFC	-580	(-1280)
• Fuel increase due to descent cooling	+204	(+450)
• Fuel increase due to coolant pump	+69	(+152)
Net Change	+67	(+147)

Since the quantity of fuel involved is so small compared to the total fuel load (0.16 percent) the cooled vehicle was not charged with this penalty.

4.6 COST FACTORS

The costs for the actively cooled supersonic transport were determined in a manner described in Reference 3. The adjustments that were made to the basic input data are described below.

4.6.1 Structure and System

The additive cost for the structure to accommodate the active cooling system is accounted for in the weight increase and the added complexity. The cost from the added weight is simply the additional cost from the weight increase in the structure of the wing, fuselage and the addition of the plumbing, heat exchangers, pumps, reservoirs, and controls. The complexity of the system was taken into account through an increase in the labor hours for fabrication and assembly of the cooled panel structure and the added cost for the installation of the equipment and controls. The percentage increase in the labor hours for the structural fabrication and assembly over that of an uncooled panel are:

	% Increase-Labor
Wing	25
Body	33

The primary cause of this increase is the additional number of weldbonds that must be made (see Figures 23 and 24) and the need to proof pressure check each panel coolant passage after fabrication and before final assembly.

The cost for the non-structural elements of the system (pumps, heat exchangers, control, etc.) was based on the extrapolation of costs for systems such as environmental control system, hydraulics, and fuel system. The material dollar factor derived from these systems accounts for the purchase of the equipment and material and the labor hours accounts for the installation of this equipment. An example of these effects on production cost is given in Section 4.7.

4.6.2 Maintenance

The maintenance cost for the active cooling system was estimated by relating it to a similar system, in terms of function, and using that system's maintenance cost for the active cooling system. The active cooling system is a low pressure system (compared to aircraft hydraulic systems) and has components such as flow control valves and heat exchangers which are similar to an environmental control system, therefore, its maintenance requirements are assumed to be the same.

A breakdown of the maintenance cost for a DC-8 aircraft, as reported by Air Canada, is shown in Table 11. The system's maintenance cost is $35.58 out of the total of $159.73 or 22 percent. The DOC for the AST is calculated by a method that is more detailed than the ATA method and the system's maintenance cost may be isolated Isolating the systems maintenance cost for the AST shows a fairly good agreement with Air Canada experience for the DC-8 (26 percent for the AST; 22 percent for the DC-8). Using the air conditioning system maintenance cost as being representative of the active cooling system gives an increase of approximately 25 percent for system maintenance or a 6 percent increase in total maintenance.

Although the maintenance cost for the systems for the Mach 2.7 and the Mach 3.2 airplanes are increased by 25 percent to account for the active cooling system their total systems maintenance cost are considered equal. The active cooling system on the Mach 3.2 airplane will maintain an environment that is equivalent to the Mach 2.7 airplane as far as the systems are concerned. Since the environment is the same and the systems are identical the maintenance costs are assumed to be equal. The maintenance equations for the systems are adjusted to provide equal maintenance costs for the Mach 2.7 and the Mach 3.2 vehicle but the remainder of the maintenance costs are influenced by the characteristics of the two vehicles.

4.6.3 Reliability

Although not required in the scope of the study, an estimate was made of the overall reliability of the cooling system. Consdering that the system has not been defined at the component level such an analysis is highly speculative and involves an analogy to similar components in existing aircraft systems. The system was assumed to be non-redundant in that no components were duplicated. Such duplication would of course increase the overall system reliability but would involve a higher initial weight and cost and an increase in system maintenance. Suitable fault detection and isolation would be required to detect malfunctioning components and to abort supersonic flight to prevent a prolonged structural overtemperature condition.

The following tabulation is a first order reliability estimate using similar components and correcting for pressure and temperature effects where possible (see schematic Figure 25). Only primary failures were considered. The areas felt to present the highest uncertainty are the integrity of the skin panels and the hydrogen-to-coolant heat exchanger considering the high thermal stresses involved and the difficulty of inspection.

TABLE 11. REPORTED DC-8 MAINTENANCE COST (AIR CANADA)($/HR)

Average Flight Duration - 2 hours

(Corrected to 1973 American labor rate)

ATA System		Air Canada
*21 - Air Conditioning		$ 8.50
*22 - Auto Flight		.78
*23 - Communications		1.87
*24 - Electrical Power		3.41
25 - Equipment/Furnishings		15.63
*26 - Fire Protection		.34
*27 - Flight Controls		6.52
*28 - Fuel		2.33
*29 - Hydraulic Power		.83
*30 - Ice and Rain Protection		.46
*31 - Instruments		.31
32 - Landing Gear		12.77
*33 - Lights		.93
*34 - Navigation		5.72
*35 - Oxygen		.84
*36 - Pneumatic		1.72
*38 - Water/Waste		1.12
52 - Doors		.74
53 - Fuselage		3.08
54 - Nacelles/Pylons		2.29
55 - Stabilizers		.92
56 - Windows		.39
57 - Wings		2.67
	Total	$ 74.07
71-80 - Propulsion Items		66.59
Unassigned DMC (Airframe)		19.07
Grand Total (Excluding 71-80)		93.14
Grand Total (Including 71-80)		$159.73

*Systems = $35.58 (22 percent of total)

COMPOUND	NUMBER IN SYSTEM	FAILURE RATE (FAIL./HR x 10^{-6})	TOTAL FAILURES RATE/HR. x 10^{-6}
Air/coolant heat exchanger	4	30	120
H_2/coolant heat exchanger	4	160	640
Skin panels	10280 ft^2	0.04/ft^2	410
Panel passage connections	5350	0.1/connection	535
Distribution lines and connectors	All	100	100
Valves (H_2 and coolant)	20	20	400
Pump and drive	4	100	400
Sensors and circuits	All	200	200
Total system			2805

This is equivalent to 357 hours mean time between failures (MBTF) or 0.79 delays per 100 departures using an average flight time of 2.8 hours. This may be compared to a current target delay rate of 3.5 per 100 departures for all aircraft systems and equipment in a typical commercial aircraft with approximately the same flight time. The analysis did not consider the degradation in reliability of the engine fuel supply system where a flow control valve malfunction would cause the loss of an engine. The final consideration is that the addition of the cooling system could have a significant impact on both the aircraft dispatch reliability and total maintenance cost, and that the estimate of maintenance cost given above is reasonable.

4.6.4 Development Cost

The active cooling system is an added complexity which will affect the design, design support, testing, and tooling. The following percentage increases are estimated for the engineering development:

Design	-	15%
Testing	-	10%
Design Support	-	5%

The effect on the total design and test is determined by applying the percentage increase for each category to the percentage that category is of the total design effort.

Design	50% x 1.15 = 57.50%
Testing	20% x 1.10 = 22.00%
Design Support	30% x 1.05 = 31.50%
Total Design Engineering	111.00%

or an 11 percent increase for the total Design effort.

The incrase in tooling is considered as approximately the same increase as the design engineering and its cost was increased by 10 percent.

4.7 WEIGHT/COST TRENDS FOR COOLED VERSUS UNCOOLED AIRCRAFT

A major objective of the study was to find out if the substitution of lower cost, cooled aluminum structure in place of titanium could pay for the extra weight and complexity of the cooling system itself and hopefully even reduce the total weight and cost of the aircraft. The following example compares weight trends and production cost data for the wing and fuselage of the cooled and uncooled versions of the Mach 2.7 aircraft, assuming the aircraft gross weights are held constant.

WEIGHT AND MATERIAL DISTRIBUTION

	UNCOOLED AIRCRAFT		COOLED AIRCRAFT	
	kg	(lbs)	kg	(lbs)
WING:				
Aluminum: Uncooled	4,740	(2,171)	-	-
Cooled Skin	-	-	4,730	(10,426)
Titanium	18,330	(40,407)	14,300	(31,532)
Other Mat'l (Steel composites, etc.)	2,100	(4,627)	2,100	(4,627)
Total Wing	21,410	(47,205)	21,130	(46,584)
FUSELAGE:				
Aluminum: Uncooled	6,600	(14,445)*	7,915	(17,464)*
Cooled Skin	-	-	6,775	(14,934)
Titanium	10,400	(22,948)	1,930	(4,254)
Other Mat'l: (Steel, composites, etc.)	3,250	(7,144)	3,250	(7,144)
Total Fuselage	20,250	(44,646)	19,870	(43,796)

*Includes aluminum fuel tanks.

If we now apply the appropriate material and labor cost factors to the cooled and uncooled aircraft versions we can get a rough estimate of the potential structural cost savings. It should be emphasized that neither material cost nor labor learning curves have been applied to the following costs and they do not represent the true cumulative average production cost of the 300th airplane produced. (This was the production base used in the study in Reference 3):

STRUCTURAL COST COMPARISON

WING:

	MATL. COST $/LB.	LABOR HRS/LB.	RATE $/HR	TOTAL $/LB.	MATL. WT.LBS	TOTAL $
UNCOOLED:						
Uncooled Al[1]	12.72	4.80	16	89.52	2,171	194,345
TI[2]	52.35	8	16	180.35	40,407	7,287,402
					42,578	7,481,745
COOLED:						
Cooled Al[3]	12.72	6	16	108.72	10,426	1,133,515
TI[2]	52.35	8	16	180.35	31,532	5,686,796
					41,958	6,820,311

1 Non-primary structure

2 Primary sub-structure

3 Cooled skin

NET COST SAVING FOR WING: $7,481,745
6,820,311
\- 661,434

C - 2

FUSELAGE:

	MATL. COST $/LB.	LABOR HRS/LB.	RATE $/HR	TOTAL $/LB.	MATL. WT.LBS	TOTAL $
UNCOOLED:						
Uncooled AL.	12.72	6	16	108.72	14,554	1,582,310
TI	25.55	9	16	169.55	22,948	3,890,833
					37,502	5,473,143
COOLED:						
Uncooled AL[4]	12.72	6	16	108.82	17,464	1,898,686
Cooled AL[5]	12.72	8	16	140.72	14,934	2,101,512
TI	25.55	9	16	169.55	4,254	721,266
					36,652	4,721,464

4 Frame, floor beams, fuel tanks, etc.

5 Cooled skin and stringers

NET COST SAVING FOR FUSELAGE: $5,473,143
-4,721,464
751,679

THE TOTAL POTENTIAL STRUCTURAL COST SAVING IS THEN = $ 661,434
751,679
$1,413,113

Note that the higher material cost for titanium in the wing compared to the fuselage reflects the increased use of higher cost extrusions and forgings with attendant machining loses.

The above saving will be reduced by the cooling system cost and increased by the ECS system cost saving as follows:

	EQUIVALENT EQUIP. AND MATL. COST $/LB	LABOR HRS/LB	RATE $/HR	TOTAL $/LB	LBS. EQUIP.	COST $
Cool. System	80	3	16	128.00	1084	+139,000
ECS System	51.60	2.58	16	92.90	1472 (lbs saved)	-137,000
		NET ADDED SYSTEM COST			=	2,000

The final net saving is then $1,413,113 less $2,000 or $1,411,113. This comparison does not reflect the change in gross weight resulting from the incorporation of the cooling system and structural weight changes.

The next section will examine the cumulative effects of these cost savings including the effect of resizing, development cost increases and cooling system maintenance on both weight, price and operating cost.

5.0 COMPARISON OF COOLED AND UNCOOLED AIRCRAFT

In this section, two comparisons of final results are presented; the effect of cruise speed on the characteristics and cost of the uncooled aircraft, and the effect of active cooling versus no active cooling on aircraft designed for each of the subject cruise speeds. These aircraft have been resized to perform their respective missions and thus reflect gross weights and costs consistent with the limitations and ground rules of the study.

5.1 Comparison of Mach 2.7 and 3.2 Uncooled Aircraft

Tables 12 and 13 show that for the same mission the gross weight of the M 3.2 airplane is 21 percent higher than the M 2.7. This can be attributed mainly to the increased structural weight and the poorer low speed lift characteristics of the Mach 3.2 aircraft (see Section 4.1). The ground rule to limit landing approach speed to a maximum of 160 KEAS required that the M 3.2 airplane have a much larger wing (lower wing loading) than the Mach 2.7. This was offset to some extent by the lower wave drag of the larger winged M 3.2 airplane which showed a higher L/D than the M 2.7. This is apparent in the cruise efficiency [M (L/D)/SFC] of 41.4 for the Mach 3.2 aircraft compared to 33.4 for the Mach 2.7. This results in a reduced mission fuel fraction of 19.8 percent for the Mach 3.2 compared to 21.8 for the Mach 2.7.

The higher speed results in an increase in development cost of 43 percent for the Mach 3.2 airplane. Aircraft price is up 25 percent and direct operating cost of the Mach 3.2 is 8.7% higher than for the Mach 2.7.

The ROI's shown are purely arbitrary calculations based on speed, utilization, revenue, and costs without regard to the real world of airline scheduling, demand and operations.

TABLE 12. PERFORMANCE COMPARISON OF COOLED AND UNCOOLED AIRCRAFT (SI UNITS)

		MACH 2.7		MACH 3.2	
		UNCOOLED	COOLED	UNCOOLED	COOLED
GROSS	kg	163,783	163,615	198,493	194,567
FUEL WEIGHT	kg	42,278	42,222	49,043	48,337
PAYLOAD	kg	22,226	22,226	22,226	22,226
OPERATING EMPTY WT.	kg	99,279	99,166	127,223	124,003
EMPTY WT.	kg	94,760	94,649	122,491	119,294
COOLING SYSTEM WT.	kg	-	1,273	-	2,152
ECS SYSTEMS WT.	kg	3,577	2,907	4,658	2,952
WING AREA	m^2	579	579	893	876
THRUST/ENG.	N	219,224	219,002	258,629	253,514
APPROACH SPEED	m/s	82.3	82.3	82.3	82.3
CRUISE ALT.	m	20,726	20,726	23,165	23,165
CRUISE $^L/D$	-	6.85	6.85	7.72	7.68
CRUISE SFC	$\frac{kg}{hr}$/daN	.563	.563	.608	.609
RANGE	km	7,778	7,778	7,778	7,778
PASSENGERS	-	234	234	234	234
BLOCK FUEL	kg	35,832	35,799	39,447	38,871
ENERGY UTILIZATION	$\frac{kJ}{seat\ km}$	5,196	5,191	5,720	5,636

TABLE 12. PERFORMANCE COMPARISON OF COOLED AND UNCOOLED AIRCRAFT (Continued)

(CUSTOMARY UNITS)

		MACH 2.7		MACH 3.2	
		UNCOOLED	COOLED	UNCOOLED	COOLED
GROSS WEIGHT	lb.	361,074	360,704	437,594	428,939
FUEL WEIGHT	lb.	93,205	93,084	108,120	106,563
PAYLOAD	lb.	49,000	49,000	49,000	49,000
OPERATING EMPTY WT.	lb.	218,869	218,620	280,474	273,337
EMPTY WT.	lb.	208,907	208,662	270,041	262,993
COOLING SYSTEM WT.	lb.	--	2,806	--	4,745
ECS SYSTSMS WT.	lb.	7,880	6,408	10,269	6,508
WING AREA	$ft.^2$	6,232	6,238	9,613	9,431
THRUST/ENG.	lb.	49,286	49,236	58,145	56,995
APPROACH SPEED	Keas	160	160	160	160
CRUISE ALT.	ft.	68,000	68,000	76,000	76,000
CRUISE $^{L}/D$		6.85	6.85	7.72	7.68
CRUISE SFC	$\frac{lb}{hr}/lb$	.553	.553	.597	.598
RANGE	nm	4,200	4,200	4,200	4,200
PASSENGERS		234	234	234	234
BLOCK FUEL	lb.	78,995	78,921	86,965	85,695
ENERGY UTILIZATION	$\frac{Btu}{Seat\ nm}$	4,147	4,143	4,565	4,498

TABLE 13. COST COMPARISON OF COOLED AND UNCOOLED AIRCRAFT

		MACH 2.7		MACH 3.2	
		UNCOOLED	COOLED	UNCOOLED	COOLED
RDTE	BIL. $	3.28	3.42	4.72	4.84
AIRCRAFT PRICE	MIL. $	47.04	45.50	59.09	55.33
DOC	¢/Seat nm				
Crew		.097	.097	.085	.085
Fuel & Oil		.713	.712	.785	.773
Insurance		.133	.131	.149	.141
Depreciation		.428	.420	.480	.453
Maintenance		.373	.390	.396	.387
TOTAL DOC		1.744	1.750	1.895	1.839
ROI (After Taxes)	%	7.01	7.02	3.80	4.97

5.2 Comparison of Cooled and Uncooled Aircraft

Tables 12, 13 and 14 show the performance, cost and structural weight characteristics of the final, resized cooled aircraft compared to the uncooled baseline. Some general observations regarding the Mach 2.7 results are listed:

- The gross weight of the Mach 2.7 cooled aircraft stayed about the same as the uncooled while the price went down 3.7 percent and the DOC went up slightly.
- The gross weight remained essentially the same because the weight saved in the wing, fuselage, and ECS system of the cooled aircraft was approximately the same as the penalty for the cooling system.
- The total utilization of aluminum in the wing and fuselage increased from 18.7% in the uncooled to 48.4% in the cooled aircraft.
- The cost per pound of aircraft empty weight dropped from $225 for the uncooled version to $218 in the cooled aircraft due to the increased use of lower cost aluminum.

General trends of the Mach 3.2 aircraft results are as follows:

- The gross weight of the cooled version decreased about 2 percent compared to the uncooled while the DOC went down 3 percent. However, the price of the cooled aircraft decreased 6.4 percent, about twice that of the Mach 2.7 case.
- Compared to the Mach 2.7 case, more weight was saved in the wing, fuselage and ECS system of the cooled aircraft resulting in the 2 percent reduction of gross weight.
- The total utilization of aluminum in the wing and fuselage increased from 14.3 percent in the uncooled to 45 percent in the cooled aircraft.
- The average cost of a pound of empty weight dropped from $219 in the uncooled to $210 in the cooled version due to the increased use of aluminum.

Detailed ASSET computer printouts of all four designs giving weight, cost, mission, and aerodynamic information are included in Appendix A.

TABLE 14. COMPARISON OF COOLED AND UNCOOLED STRUCTURE

(SI UNITS)

STRUCTURE WEIGHT	lb.	MACH 2.7 UNCOOLED	MACH 2.7 COOLED	MACH 3.2 UNCOOLED	MACH 3.2 COOLED
WING:		(19,491)	(19,208)	(29,983)	(28,425)
ALUMINUM		897	4,302	0	6,367
TITANIUM		16,684	13,061	27,404	19,327
STEEL		390	384	600	399
COMP.		1,206	1,153	1,499	1,706
OTHER		312	308	480	455
FUSELAGE:		(19,879)	(19,484)	(23,287)	(22,155)
ALUMINUM		6,481	14,418	7,591	16,395
TITANIUM		10,218	1,948	11,970	2,215
STEEL		358	351	419	399
COMP.		497	487	582	554
OTHER		2,326	2,280	2,725	2,592

(CUSTOMARY UNITS)

STRUCTURAL WEIGHT	kg.	UNCOOLED	COOLED	UNCOOLED	COOLED
WING:		(42,970)	(42,345)	(66,099)	(62,665)
ALUMINUM		1,977	9,485	0	14,037
TITANIUM		36,782	28,794	60,414	42,612
STEEL		859	847	1,322	1,253
COMP.		2,664	2,541	3,305	3,760
OTHER		688	678	1,058	1,003
FUSELAGE:		(43,825)	(42,954)	(51,338)	(48,843)
ALUMINUM		14,287	31,786	16,736	36,144
TITANIUM		22,526	4,295	26,388	4,884
STEEL		789	773	924	879
COMP.		1,096	1,074	1,283	1,221
OTHER		5,128	5,026	6,007	5,715

6.0 STUDY CONCLUSIONS

Mach 2.7 Aircraft:

- The increase of lower cost aluminum usage from 18.7 to 48.4 percent of the wing and fuselage structure allowed a price decrease of 3.7 percent at approximately the same gross weight.
- The cause of the slight increase in DOC of the cooled version was the increase in maintenance cost of the coolant system. As described in Section 4.7, this was estimated to be equivalent to a 25 percent increase in system maintenance or a 6 percent increase in total maintenance. Should no maintenance costs result, the DOC would be 1.724¢/ASnm or 1.3 percent lower than the uncooled aircraft.
- Since the cooled aircraft used only 61 percent of the available heat sink, more area could be cooled. This would involve diminishing returns however, because such surfaces (tail, flaps, ailerons, crew compartment) are either remotely located or involve complex plumbing connections, resulting in sizeable increases in coolant system and fluid weight.

Mach 3.2 Aircraft:

- The increase of aluminum utilization from 14.2 to 45 percent of wing and fuselage structure, together with the reduction in gross weight allowed a price decrease of 6.4 percent for the cooled version.
- The DOC of the cooled aircraft is 3 percent less than that of the uncooled with the increased maintenance cost of the cooling system balanced by reduced maintenance costs for the other systems permitted by the lower environmental temperatures. Should no maintenance costs result, the DOC would be 1.816¢/ASnm or 4.2 percent lower than the uncooled aircraft.
- Since the Mach 3.2 aircraft used 100 percent of the heat sink capability, no further area can be cooled. In fact, a slight reduction in cooled wing surface area, relative to the Mach 2.7 was required to meet this limitation.

GENERAL

Within the limited scope and ground rules of this study, no significant economic advantage was found for active cooling in the Mach 2.7 transport and only a slight advantage for the Mach 3.2. While this conclusion is based on the addition of active cooling in an existing structural design concept (Reference 6), this design resulted from the consideration of many concepts and it is not felt that the incorporation of the small coolant passages would have dictated the choice of a different design.

The use of an active cooling system in a commercial transport operating environment requires consideration beyond that possible in this study as to what impact the system might have on maintenance costs, flight safety and dispatch reliability.

While the advantages of cooling were found to be marginal at Mach 2.7 and 3.2, it is significant that the trend shows increasing weight and economic benefits at the higher Mach number as the allowable stress levels decrease with higher structural temperatures. This suggests that because of the trend of lower $^{L}/D$ and increasing specific fuel consumption with Mach number, higher speeds will provide increasing fuel heat sink to maintain the required surface temperature as the heating load increases. Thus the greatest potential for active cooling will be at hypersonic cruise speeds, in particular the Mach 6-8 regime where scramjet propulsion is attractive and expensive superalloys at reduced allowables must be used if no cooling is employed.

REFERENCES

1. Helenbrook, R. G. and Anthony, F. M.: "Design of a Convective Cooling System for a Mach 6 Hypersonic Transport Airframe," NASA CR-1918, Dec 1971.

2. Anthony, F. M., Dukes, W. H., and Helenbrook, R. G.: "Data and Results from a Study of Internal Convective Cooling Systems for Hypersonic Aircraft," Bell Aerospace. NASA CR-132,432. June, 1974.

3. G. D. Brewer, "Final Report, Advanced Supersonic Technology Concept Study - Hydrogen Fueled Configuration." Lockheed-Calif. Co. NASA CR-114,718. Jan 1974.

4. Contract NAS 1-11940 "Studies of the Impact of Advanced Technologies Applied to Supersonic Transport Aircraft," NASA-Langley Research Center to Lockheed-Calif. Col, Sept 11, 1972.

5. Contract NAS 1-12288, "Study of Structural Design Concepts for an Arrow-Wing Supersonic Transport Configuration," NASA-Langley Research Center to Lockheed-Calif. Co., May 21, 1973.

6. I. F. Sakata and G. W. Davis, "Arrow-Wing Supersonic Transport Configuration Structural Concepts Evaluation" NASA CR-132575, Volume 1-3, 1975.

7. Hopkins, E. J., "Charts for Predicting Turbulent Skin Friction from the Van Driest Method (II)

8. McAdams, W. H. "Heat Transmission," McGraw-Hillbook Co., Inc., 1954.

9. "SAE Aerospace Applied Thermodynamics Manual," 1969.

APPENDIX A

COMPUTER PRINTOUT - ASSET PARAMETRIC ANALYSIS

CL-1701-61 and CL-1701-8

LH_2 - AST D-B TURBOFAN ENGINES

	Page
Mach 2.7 - Uncooled	A-1 thru A-9
Mach 2.7 - Cooled	A-10 thru A-17
Mach 3.2 - Uncooled	A-18 thru A-24
Mach 3.2 - Cooled	A-25 thru A-32

SCNR 3 ID NO. 201 A S S E T P A R A M E T R I C A N A L Y S I S OCTOBER 8 1974

AIRCRAFT MODEL --CL 1701-6
I.O.C. DATE --1990
DESIGN SPEED --SUPERSONIC

ENGINE I.D. -- 1000
SLS SCALE 1.0 = 81330
NUMBER OF ENGINES = 4.

WING QUARTER CHORD SWEEP = 68.63 DEG
WING TAPER RATIO = 0.0

1 W/S	57.9	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2 T/W	0.546	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3 AR	1.62	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4 T/C	3.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5 RADIUS N. MI	4200	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
6 GROSS WEIGHT	361074	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
7 FUEL WEIGHT	93205	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
8 OP. WT. EMPTY	218869	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
9 ZERO FUEL WT.	267869	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
10 THRUST/ENGINE	49286	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
11 ENGINE SCALE	0.606	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
12 WING AREA	6232.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
13 WING SPAN	166.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
14 H. TAIL AREA	458.3	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
15 V. TAIL AREA	268.6	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
16 BODY LENGTH	324.7	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
COST DATA																
17 RDTE - BIL.	3.276	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
18 FLYAWAY - MIL.	66.80	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
19 INVESTMENT-BIL.	6.985	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20 DOC - C/SM	1.744	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
21 IOC - C/SM	0.796	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22 ROI A.T. - O/O	7.03	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
CONSTRAINT OUTPUT																
23 TAKEOFF DST(1)	~~[illegible]~~	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
24 CLIMB GRAD(1)	~~[illegible]~~	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
25 TAKEOFF DST(2)	~~[illegible]~~	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
26 CLIMB GRAD(2)	~~[illegible]~~	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
27 CTOL LNDG D(1)	8668	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
28 AP SPEED-KT(1)	160.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
29 CTOL LNDG D(2)	7779	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
30 AP SPEED-KT(2)	161.4	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
31 CTOL LNDG D(3)	8890	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
32 AP SPEED-KT(3)	162.8	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0

FAR T.O. FLD. LENGTH = 6785
2ND SEG. CLIMB GRADIENT = .0731 (ENG. OUT)

A-2

ORIGINAL PAGE IS OF POOR QUALITY

MACH 2.7 - UNCOOLED

ORIGINAL PAGE IS OF POOR QUALITY

CL 1701-6   LH2-AST   D-B TURB[illegible]N ENGINES

M2.7 UNCOOLED

T/C	AR	W/S	T/W
3.00	1.62	57.9	0.546

W E I G H T   S T A T E M E N T

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
TAKE-OFF WEIGHT	(361075.)		
FUEL AVAILABLE	93205.	FUEL	25.81
ZERO FUEL WEIGHT	(267870.)		
PAYLOAD	49000.	PAYLOAD	13.57
OPERATING WEIGHT	(218870.)		
OPERATING ITEMS	5367.	OPERATING ITEMS	2.76
STANDARD ITEMS	4555.		
EMPTY WEIGHT	(208907.)		
WING	42970.		
TAIL	6070.		
BODY	43828.	STRUCTURE	32.48
LANDING GEAR	16540.		
SURFACE CONTROLS	4545.		
NACELLE AND ENGINE SECTION	2929.		
PROPULSION	(60005.)	PROPULSION	16.62
WEIGHT OF LIFT ENGINES	0.		
VECTOR CONTROL SYSTEM	0.		
ENGINES	26664.		
THRUST REVERSAL	0.		
AIR INDUCTION SYSTEM	10644.		
FUEL SYSTEM	21349.		
ENGINE CONTROLS + STARTER	1348.		
INSTRUMENTS	1090.		
HYDRAULICS	2744.		
ELECTRICAL	4528.		
AVIONICS	1900.	EQUIPMENT	8.76
FURNISHINGS AND EQUIPMENT	11500.		
ENVIRONMENTAL CONTROL SYSTEM	7880.		
AUXILIARY GEAR	1980.		
A.M.P.R.	(169786.)	TOTAL	(100.00)
EXCESS FUEL CAPACITY - BODY	-0.		
EXCESS FUEL CAPACITY - WING	0.		
EXCESS BODY LENGTH - FT	0.0		

STRUCTURE   ALUMINUM

A-3

WEIGHT MATRIX

ELEMENT/ / MATERIAL	AL	TIT.	STEEL	COMP.	OTHER	TOTAL
WING	1977.	36782.	859.	2664.	688.	42970.
TAIL	273.	5639.	61.	0.	97.	6070.
FUSEL	14287.	22526.	789.	1096.	5128.	43825.
L. G.	17.	4235.	6505.	0.	6183.	16940.
NACELLE	56.	435.	974.	0.	0.	1465.
AIR INDUCT	490.	9431.	106.	0.	617.	10644.
S. CTLS	1091.	205.	954.	68.	2227.	4545.
TOTALS	18190.	79253.	10249.	3828.	14940.	126460.

CONFIGURATION GEOMETRY

BASIC WING--

AREA(SQ.FT)	SPAN(FT)	TAPER RATIO	C/4 SWEEP	L.E. SWEEP	CR(FT)	MAC(FT)
6231.9	100.55	0.0	66.626	72.500	123.96	82.64

INBOARD WING--

AREA(SQ.FT)	EXP. AREA	L.E. SWEEP	REF L(FT)	SFLE(SQ.FT)	AVG T/C
6231.9	4789.5	72.50	72.45	0.0	3.00

OUTBOARD WING--

AREA(SQ.FT)	Y BRK(FT)	L.E. SWEEP	REF L(FT)	SFLE(SQ.FT)	AVG T/C
0.0	0.0	72.50	72.45	0.0	3.00

TOTAL WING--

AREA(SQ.FT)	EFF AR	AVG T/C	CR(FT)	CT(FT)	(B/2)/LW	P
6231.9	1.62	3.00	123.96	0.0	0.315	0.389

WING TANK--

CBAR1(FT)	CBAR2(FT)	FTL(FT)	FVWING(CU FT)	FVBOX(CU FT)
104.06	0.0	43.82	0.0	0.0

FUSELAGE--

LENGTH(FT)	S WET(SQ FT)	BWW(FT)	EQUIV D(FT)	SPI(SQ FT)
324.70	13327.9	12.40	16.44	212.25

BW(FT)	BH(FT)	SBW(SQ FT)	FVB(CU FT)
12.90	19.43	13327.86	22086.39

TAIL--

SHT(SQ.FT)	SHTX(SQ.FT)	HT REF L(FT)	SVT(SQ.FT)	SVTX(SQ.FT)	VT REF L(FT)
458.33	371.37	15.03	268.56	268.58	19.63

PROPULSION--

ENG L(FT)	ENG D(FT)	POD L(FT)	POD D(FT)	POD S WET	NO. PODS	INLET L(FT)
18.18	5.14	31.35	6.00	2365.28	4.	0.0

M I S S I O N   S U M M A R Y

CL 1701-6   LH2-AST   D-B TURBOFAN ENGINES

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG L/D RATIO	AVG SFC (FF/T)	MAX OVER PRES
TAKEOFF POWER 1	0.	0.0	361075.	451.	451.	0.	0.	10.0	10.0	0.	-1101.	0.	0.0	0.150	0.0
POWER 2	0.	0.300	360624.	676.	1127.	0.	0.	0.4	10.4	0.	1209.	0.	5.89	0.359	0.0
CLIMB	0.	0.300	359948.	908.	2035.	4.	4.	1.1	11.5	0.	1209.	0.	7.90	0.377	0.0
CRUISE	5000.	0.414	359040.	605.	2640.	0.	4.	4.0	15.5	0.	-1101.	0.	8.52	0.215	0.0
ACCEL	5000.	0.414	358435.	189.	2829.	3.	8.	0.6	16.1	0.	1101.	0.	9.53	0.233	0.0
CLIMB	5000.	0.539	358245.	4192.	7021.	99.	107.	13.1	29.2	0.	1101.	0.	9.70	0.324	0.0
CLIMB	34000.	0.989	354053.	12491.	19512.	315.	422.	17.0	46.2	0.	1206.	0.	6.25	0.557	0.0
CLIMB	63000.	2.700	341562.	322.	19834.	14.	436.	0.5	46.8	0.	1206.	0.	6.82	0.574	0.0
CRUISE	66000.	2.700	341240.	57819.	77653.	3564.	4000.	137.9	184.7	0.	-1201.	0.	6.85	0.553	0.0
DECEL	70000.	2.700	283421.	19.	77673.	27.	4027.	1.1	185.8	0.	1501.	0.	6.86	-0.222	0.0
DESCENT	70000.	2.337	283402.	208.	77880.	134.	4162.	11.9	197.8	0.	1501.	0.	7.97	-0.126	0.0
CRUISE	69000.	2.700	283194.	560.	78448.	36.	4200.	1.5	199.2	0.	-1201.	0.	6.83	0.557	0.0
CRUISE	5000.	0.414	282626.	547.	78995.	0.	4200.	5.0	204.2	0.	-1101.	0.	9.41	0.219	0.0
RESET	0.	0.0	282080.	0.	78995.	0.	4200.	0.0	204.2	0.	0.	0.	0.0	0.0	0.0
RESET	0.	0.0	282080.	0.	78995.	-4200.	0.	*****	0.0	0.	0.	0.	0.0	0.0	0.0
RESERVE	0.	0.0	282080.	5530.	84524.	0.	0.	0.0	0.0	0.	0.	0.	0.0	0.0	0.0
CLIMB	0.	0.200	276550.	562.	85086.	3.	3.	0.7	0.7	0.	1209.	0.	8.03	0.375	0.0
CLIMB	1500.	0.505	275988.	3123.	88209.	99.	101.	12.8	13.5	0.	1101.	0.	9.17	0.296	0.0
CRUISE	37000.	0.900	272865.	1503.	89712.	93.	195.	10.9	24.4	0.	-1201.	0.	9.69	0.296	0.0
DESCENT	38000.	0.900	271361.	131.	89844.	52.	246.	7.3	31.7	0.	1501.	0.	9.15	-0.168	0.0
CRUISE	37000.	0.900	271230.	216.	90060.	13.	260.	1.6	33.2	0.	-1101.	0.	9.69	0.296	0.0
CRUISE	15000.	0.503	271014.	3145.	93205.	0.	260.	30.0	63.2	0.	-1101.	0.	9.61	0.224	0.0

TOGRWT= 361074.6   FUEL A= 93205.1   FUEL R= 93205.0

PRODUCTION

	PRODUCTION YEARS 1	2	3	4	5	6	7	8	9	10	TOTAL
AIRFRAME	833.18	774.24	852.18	934.10	1013.58	938.82	886.18	846.17	814.25	787.90	8680.59
ENGINEERING											
HOURS	2997.	2581.	2731.	2903.	3068.	2780.	2579.	2427.	2306.	2207.	26580.
LABOR RATE	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	
OVERHEAD RATE	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	
TOTAL	52.05	44.84	47.45	50.42	53.29	48.30	44.80	42.16	40.06	38.33	461.70
TOOLING											
HOURS	3596.	3098.	3278.	3483.	3681.	3337.	3095.	2913.	2768.	2648.	31896.
LABOR RATE	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	
OVERHEAD RATE	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	
TOTAL	66.35	57.15	60.48	64.26	67.92	61.56	57.11	53.74	51.06	48.86	588.49
MANUFACTURING											
HOURS	29968.	25813.	27315.	29026.	30677.	27804.	25793.	24272.	23064.	22070.	265803.
LABOR RATE	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	474.69	408.88	432.67	459.78	485.93	440.42	408.57	384.47	365.33	349.58	4210.32
QUALITY CONTROL											
HOURS	5994.	5163.	5463.	5805.	6135.	5561.	5159.	4854.	4613.	4414.	53161.
LABOR RATE	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	101.95	87.82	92.93	98.75	104.36	94.59	87.75	82.57	78.46	75.08	904.26
MATERIAL											
RAW AND PURCH	41.70	55.72	70.47	84.87	98.92	96.72	95.06	93.75	92.65	91.72	821.59
PURCHASED EQUIP.	77.45	103.48	130.88	157.62	183.72	179.62	176.55	174.10	172.07	170.34	1525.82
TOTAL	119.15	159.20	201.35	242.50	282.64	276.34	271.61	267.85	264.72	262.06	2347.42
MISCELLANEOUS											
HOURS	1195.	1033.	1093.	1161.	1227.	1112.	1032.	971.	923.	883.	10632.
LABOR RATE	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	18.99	16.36	17.31	18.39	19.44	17.62	16.34	15.38	14.61	13.98	168.41
ENGINES	174.32	204.82	244.99	283.68	320.62	306.10	295.44	287.09	280.25	274.48	2671.79
AVIONICS	6.00	9.00	12.00	15.00	18.00	18.00	18.00	18.00	18.00	18.00	150.00
PROFIT	124.98	116.14	127.83	140.12	152.04	140.82	132.93	126.93	122.14	118.18	1302.09
INSUR.+TAXES	83.32	77.42	85.22	93.41	101.36	93.88	88.62	84.62	81.43	78.79	868.06
WARRANTY	41.66	38.71	42.61	46.71	50.68	46.94	44.31	42.31	40.71	39.39	434.03
TOTAL FLYAWAY	1263.46	1220.34	1364.82	1513.01	1656.27	1544.56	1465.48	1405.11	1356.78	1320.70	14110.50

ORIGINAL PAGE IS OF POOR QUALITY

A-8

COST SUMMARY

RDT AND E	TOTAL*
PROTOTYPE AIRCRAFT	627.79
DESIGN ENGINEERING	762.78
DEVELOPMENT TEST ARTICLES	283.38
FLIGHT TEST	86.20
ENGINE DEVELOPMENT CRUISE	664.41
ENGINE DEVELOPMENT LIFT	0.0
AVIONICS DEVELOPMENT	0.0
MAINTENANCE TRAINER DEVEL	0.0
OPERATOR TRAINER DEVELOP	0.0
DEVELOPMENT TOOLING	683.77
SPECIAL SUPPORT EQUIPMENT	12.56
DEVELOPMENT SPARES	99.22
TECHNICAL DATA	16.30
TOTAL RDTE	3276.41

INVESTMENT	TOTAL*	PER PROD A/C**
PRODUCTION AIRCRAFT	14110.50	47035.01
PRODUCTION ENGINEERING	0.0	0.0
MAINTENANCE TRAINERS	0.0	0.0
OPERATOR TRAINERS	0.0	0.0
PRODUCTION TOOLING	416.29	1387.65
SPECIAL SUPPORT EQUIPMENT	705.53	2351.75
PRODUCTION SPARES	2148.62	7162.08
TECHNICAL DATA	80.90	269.68
TOTAL INVESTMENT	17467.84	58226.13

DIRECT OPERATIONAL COST (DOC)	C/SM***	PERCENT
FLIGHT CREW	0.09697	5.56035
FUEL AND OIL	0.71263	40.86186
INSURANCE	0.13308	7.63079
DEPRECIATION	0.42819	24.55208
MAINTENANCE	0.37313	21.39497
TOTAL DOC	1.74400	100.000

INDIRECT OPERATIONAL COST (IOC)	C/SM***	PERCENT
SYSTEM	0.00313	0.39315
LOCAL	0.09163	11.50931
AIRCRAFT CONTROL	0.00513	0.64417
CABIN ATTENDANT	0.06979	8.76548
FOOD AND BEVERAGE	0.02412	3.02920
PASSENGER HANDLING	0.13656	17.15260
CARGO HANDLING	0.00849	1.06621
OTHER PASSENGER EXPENSE	0.33550	42.14024
OTHER CARGO EXPENSE	0.00278	0.34890
GENERAL + ADMINISTR.	0.11903	14.95072
TOTAL IOC	0.79615	100.000

MISC. DATA	
RANGE (ST. MILES)	4833.02
BLOCK SPEED (MPH)	1322.72
FARE ($)	248.72
FLEET SIZE	14.25
PRODUCTION BASIS	300.00
REV.PASSENG.(MIL.PER YR)	1.81
AVER. CARGO PER FLIGHT	2000.00
FLIGHT PER A/C PER YEAR	465.26

RETURN ON INVESTMENT (ROI)	
TOTAL REVENUE PER YEAR *	469.72
TOTAL EXPENSE PER YEAR *	403.29
TOTAL INVESTMENT * INCL. FACILITIES	985.25
ROI BEFORE TAXES	13.49
ROI AFTER TAXES	7.01

* - MILLIONS OF DOLLARS
** - 1000 OF DOLLARS PER PRODUCTION A/C
*** - CENTS PER SEAT MILE

RESEARCH DEVELOPMENT TEST AND EVALUATION (RDTE)

	DEVELOPMENT AND DESIGN	CONTRACTOR TEST AND EVALU	DEVELOPMENT AIRCRAFT	TOTAL RDT AND E
AIRFRAME	1275.25	321.38	428.56	2025.19
ENGINEERING				
HOURS	39187.	7233.	2134.	48554.
LABOR RATE	8.17	8.17	8.17	8.17
OVERHEAD RATE	9.20	9.20	9.20	9.20
TOTAL	680.68	125.63	37.07	643.38
TOOLING				
HOURS	25464.	1778.	3557.	34799.
LABOR RATE	6.09	6.09	6.09	6.09
OVERHEAD RATE	12.36	12.36	12.36	12.36
TOTAL	594.58	32.81	65.63	693.02
MANUFACTURING				
HOURS		7114.	14228.	21342.
LABOR RATE		5.12	5.12	5.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		112.68	225.37	338.05
QUALITY CONTROL				
HOURS		1423.	2846.	4266.
LABOR RATE		6.29	6.29	6.29
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		24.20	48.40	72.60
MATERIAL				
RAW AND PRCHSD		7.54	15.08	22.62
PURCHASED EQUIP		14.00	28.00	42.00
TOTAL		21.54	43.08	64.62
MISCELLANEOUS				
HOURS		285.	569.	654.
LABOR RATE		5.12	5.12	5.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		4.51	9.01	13.52
ENGINES	684.41		68.67	753.08
AVIONICS	0.0		2.00	2.00
PROFIT(AIRFRAME)	191.29	48.21	64.28	303.78
INSUR.+TAXES			42.86	42.86
WARRANTY			21.43	21.43
SUBTOTAL	2150.96	369.58	627.79	3148.34
OTHER ITEMS				128.07
TOTAL (RDTE)				3276.41

SUMMA[illegible] ID NO. 202   A S S E T   P A R A M E [illegible] R I C   A N A L Y S I S   FEBRUARY [illegible] 1975

AIRCRAFT MODEL --CL 1701-6
I.O.C. DATE --1990
DESIGN SPEED --SUPERSONIC

ENGINE I.D. -- 1000
SLS SCALE I.O = 81330
NUMBER OF ENGINES = 4.

WING QUARTER CHORD SWEEP = 68.63 DEG
WING TAPER RATIO = 0.0

1 W/S	57.8	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2 T/W	[illegible].546	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3 AR	1.62	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4 T/C	3.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5 RADIUS N. MI	4200	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
6 GROSS WEIGHT	360704	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
7 FUEL WEIGHT	93084	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
8 OP. WT. EMPTY	216620	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
9 ZERO FUEL WT.	267620	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
10 THRUST/ENGINE	49236	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
11 ENGINE SCALE	0.605	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
12 WING AREA	6238.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
13 WING SPAN	100.6	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
14 H. TAIL AREA	459.3	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
15 V. TAIL AREA	268.9	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
16 BODY LENGTH	324.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
COST DATA																
17 RDTE - BIL.	3.419	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
18 FLYAWAY - MIL.	65.55	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
19 INVESTMNT-BIL.	0.966	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20 DOC - C/SM	1.750	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
21 IOC - C/SM	0.797	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22 ROI A.T. - 0/0	7.02	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
CONSTRAINT OUTPUT																
23 TAKEOFF DST(1)	~~[illegible]~~	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
24 CLIMB GRAD(1)	~~[illegible]~~	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
25 TAKEOFF DST(2)	~~[illegible]~~	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
26 CLIMB GRAD(2)	~~[illegible]~~	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
27 CTOL LNDG D(1)	8655	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
28 AP SPEED-KT(1)	159.8	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
29 CTOL LNDG D(2)	8766	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
30 AP SPEED-KT(2)	161.2	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
31 CTOL LNDG D(3)	8876	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
32 AP SPEED-KT(3)	162.6	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0

FAR T.O. FLD LENGTH = 6785
2ND SEG. CLIMB GRADIENT = .073 (ENG. OUT)

MACH 2.7 - COOLED

CL 1701-6 LH2-AST D-B TURBOFAN ENGINES

T/C	AR	W/S	T/W
3.00	1.62	57.8	0.546

W E I G H T S T A T E M E N T

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
TAKE-OFF WEIGHT	(360704.)		
FUEL AVAILABLE	93084.	FUEL	25.81
ZERO FUEL WEIGHT	(267620.)		
PAYLOAD	49000.	PAYLOAD	13.58
OPERATING WEIGHT	(218620.)		
OPERATING ITEMS	5367.	OPERATING ITEMS	2.76
STANDARD ITEMS	4592.		
EMPTY WEIGHT	(208662.)		
WING	42345.		
TAIL	6081.		
BODY	42954.	STRUCTURE	32.10
LANDING GEAR	16926.		
SURFACE CONTROLS	4541.		
NACELLE AND ENGINE SECTION	2926.		
PROPULSION	(59936.)	PROPULSION	16.62
WEIGHT OF LIFT ENGINES	0.		
VECTOR CONTROL SYSTEM	0.		
ENGINES	26637.		
THRUST REVERSAL	0.		
AIR INDUCTION SYSTEM	10632.		
FUEL SYSTEM	21320.		
ENGINE CONTROLS + STARTER	1347.		
INSTRUMENTS	1090.		
HYDRAULICS	2741.		
ELECTRICAL	4528.		
AVIONICS	1900.	EQUIPMENT	9.14
FURNISHINGS AND EQUIPMENT	11500.		
ENVIRONMENTAL CONTROL SYSTEM	6408.		
AUXILIARY GEAR	1980.		
COOLING	2806.		
A.M.P.R.	(170014.)	TOTAL	(100.00)

EXCESS FUEL CAPACITY - BODY	-0.
EXCESS FUEL CAPACITY - WING	0.
EXCESS BODY LENGTH - FT	0.0

W E I G H T   M A T R I X

ELEMENT/ / MATERIAL	AL	TIT.	STEEL	COMP.	OTHER	TOTAL
WING	9485.	28794.	847.	2541.	678.	42345.
TAIL	274.	5649.	61.	0.	97.	6081.
FUSEL	31786.	4295.	773.	1074.	5026.	42954.
L. G.	17.	4232.	6500.	0.	6178.	16926.
NACELLE	56.	435.	973.	0.	0.	1463.
AIR INDUCT	489.	9420.	106.	0.	617.	10632.
S. CTLS	1090.	204.	954.	68.	2225.	4541.
TOTALS	43196.	53030.	10213.	3683.	14820.	124942.

CL 1701-6 LH2-AST D-B TURBOFAN ENGINES

T/C	AR	W/S	T/W
3.00	1.62	57.8	0.546

CONFIGURATION GEOMETRY

BASIC WING--	AREA(SQ.FT)	SPAN(FT)	TAPER RATIO	C/4 SWEEP	L.E. SWEEP	CR(FT)	MAC(FT)
	6238.4	100.60	0.0	68.626	72.500	124.02	82.69

INBOARD WING--	AREA(SQ.FT)	EXP. AREA	L.E. SWEEP	REF L(FT)	SFLE(SQ.FT)	AVG T/C
	6238.4	4795.2	72.50	72.49	0.0	3.00

OUTBOARD WING--	AREA(SQ.FT)	Y BRK(FT)	L.E. SWEEP	REF L(FT)	SFLE(SQ.FT)	AVG T/C
	0.0	0.0	72.50	72.49	0.0	3.00

TOTAL WING--	AREA(SQ.FT)	EFF AR	AVG T/C	CR(FT)	CT(FT)	(B/2)/LW	P
	6238.4	1.62	3.00	124.02	0.0	0.315	0.389

WING TANK--	CBAR1(FT)	CBAR2(FT)	FTL(FT)	FVWING(CU FT)	FVBOX(CU FT)
	108.12	0.0	43.85	0.0	0.0

FUSELAGE--	LENGTH(FT)	S WET(SQ FT)	BWW(FT)	EQUIV D(FT)	SPI(SQ FT)
	324.55	13319.9	12.40	16.44	212.25

BW(FT)	BH(FT)	SBW(SQ FT)	FVB(CU FT)
12.90	19.43	13319.91	22057.71

TAIL--	SHT(SQ.FT)	SHTX(SQ.FT)	HT REF L(FT)	SVT(SQ.FT)	SVTX(SQ.FT)	VT REF L(FT)
	459.29	372.24	15.05	268.87	268.87	19.64

PROPULSION--	ENG L(FT)	ENG D(FT)	POD L(FT)	POD D(FT)	POD S WET	NO. PODS	INLET L(FT)
	18.18	5.14	31.34	6.00	2363.07	4.	0.0

MISSIO SUMMARY

CL 1701-6 LH2-AST D-B TURBOFAN ENGINES

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG L/D RATIO	AVG SFC (FF/T)	MAX OVER PRES
TAKEOFF POWER 1	0.	0.0	360704.	451.	451.	0.	0.	10.0	10.0	0.	-1101.	0.	0.0	0.150	0.0
POWER 2	0.	0.300	360254.	674.	1124.	0.	0.	0.4	10.4	0.	1209.	0.	5.90	0.359	0.0
CLIMB	0.	0.300	359580.	907.	2031.	4.	4.	1.1	11.5	0.	1209.	0.	7.91	0.377	0.0
CRUISE	5000.	0.414	358673.	603.	2635.	0.	4.	4.0	15.5	0.	-1101.	0.	8.53	0.215	0.0
ACCEL	5000.	0.414	358070.	189.	2823.	3.	8.	0.6	16.1	0.	1101.	0.	9.54	0.233	0.0
CLIMB	5000.	0.539	357881.	4188.	7011.	99.	107.	13.1	29.2	0.	1101.	0.	9.70	0.324	0.0
CLIMB	34000.	0.989	353693.	12498.	19509.	315.	422.	17.0	46.3	0.	1206.	0.	6.25	0.557	0.0
CLIMB	63000.	2.700	341196.	322.	19831.	14.	436.	0.5	46.8	0.	1206.	0.	6.82	0.574	0.0
CRUISE	66000.	2.700	340873.	57750.	77581.	3564.	4000.	137.9	184.7	0.	-1201.	0.	6.85	0.553	0.0
DECEL	70000.	2.700	283124.	19.	77600.	27.	4027.	1.1	185.8	0.	1501.	0.	6.87	-0.222	0.0
DESCENT	70000.	2.337	283104.	207.	77808.	134.	4162.	11.9	197.8	0.	1501.	0.	7.97	-0.126	0.0
CRUISE	69000.	2.700	282897.	568.	78375.	38.	4200.	1.5	199.2	0.	-1201.	0.	6.83	0.557	0.0
CRUISE	5000.	0.414	282329.	546.	78921.	0.	4200.	5.0	204.2	0.	-1101.	0.	9.42	0.219	0.0
RESET	0.	0.0	281783.	0.	78921.	0.	4200.	0.0	204.2	0.	0.	0.	0.0	0.0	0.0
RESET	0.	0.0	281783.	0.	78921.	-4200.	0.	*****	0.0	0.	0.	0.	0.0	0.0	0.0
RESERVE	0.	0.0	281783.	5524.	84445.	0.	0.	0.0	0.0	0.	0.	0.	0.0	0.0	0.0
CLIMB	0.	0.200	276259.	561.	85006.	3.	3.	0.7	0.7	0.	1209.	0.	8.04	0.375	0.0
CLIMB	1500.	0.505	275698.	3121.	88127.	99.	101.	12.8	13.5	0.	1101.	0.	9.17	0.296	0.0
CRUISE	37000.	0.900	272577.	1501.	89629.	94.	195.	10.9	24.4	0.	-1201.	0.	9.69	0.295	0.0
DESCENT	38000.	0.900	271075.	131.	89760.	52.	246.	7.3	31.7	0.	1501.	0.	9.15	-0.168	0.0
CRUISE	37000.	0.900	270944.	216.	89975.	13.	260.	1.6	33.2	0.	-1101.	0.	9.69	0.296	0.0
CRUISE	15000.	0.503	270729.	3140.	93115.	0.	260.	30.0	63.2	0.	-1101.	0.	9.62	0.224	0.0

TOGRWT= 360704.4 FUEL A= 93084.1 FUEL R= 93114.7

PRODUCTION

	PRODUCTION YEARS 1	2	3	4	5	6	7	8	9	10	TOTAL
AIRFRAME	800.64	743.54	818.03	896.38	972.38	900.46	849.83	811.35	780.65	755.31	8328.7
ENGINEERING											
HOURS	2886.	2486.	2631.	2796.	2955.	2678.	2484.	2338.	2221.	2126.	25600.
LABOR RATE	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	
OVERHEAD RATE	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	
TOTAL	50.13	43.16	45.70	48.56	51.32	46.51	43.15	40.61	38.58	36.92	444.66
TOOLING											
HOURS	3463.	2983.	3157.	3355.	3545.	3213.	2981.	2805.	2666.	2551.	30720.
LABOR RATE	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	
OVERHEAD RATE	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	
TOTAL	63.90	55.04	58.24	61.89	65.41	59.29	55.00	51.76	49.18	47.06	566.77
MANUFACTURING											
HOURS	28862.	24861.	26307.	27956.	29545.	26778.	24842.	23377.	22213.	21255.	255996.
LABOR RATE	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	457.18	393.80	416.71	442.82	468.00	424.17	393.49	370.29	351.85	336.68	4054.98
QUALITY CONTROL											
HOURS	5772.	4972.	5261.	5591.	5909.	5356.	4968.	4675.	4443.	4251.	51199.
LABOR RATE	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	98.19	84.58	89.50	95.10	100.51	91.10	84.51	79.53	75.57	72.31	870.90
MATERIAL											
RAW AND PURCH	39.60	52.91	66.93	80.60	93.94	91.85	90.28	89.03	87.99	87.10	780.23
PURCHASED EQUIP	73.55	98.27	124.29	149.69	174.47	170.53	167.66	165.34	163.41	161.76	1449.01
TOTAL	113.15	151.18	191.22	230.29	268.41	262.42	257.94	254.36	251.40	248.86	2229.24
MISCELLANEOUS											
HOURS	1154.	994.	1052.	1118.	1182.	1071.	994.	935.	889.	850.	10240.
LABOR RATE	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	18.29	15.75	16.67	17.71	18.72	16.97	15.74	14.81	14.07	13.47	162.20
ENGINES	174.22	204.70	244.84	283.50	320.42	305.91	295.26	286.91	280.08	274.31	2670.15
AVIONICS	6.00	9.00	12.00	15.00	18.00	18.00	18.00	18.00	18.00	18.00	150.00
PROFIT	126.13	111.53	122.70	134.46	145.86	135.07	127.47	121.70	117.10	113.30	1249.31
INSUR.+TAXES	80.08	74.35	81.80	89.64	97.24	90.05	84.98	81.14	78.06	75.53	832.87
WARRANTY	40.04	37.18	40.90	44.82	48.62	45.02	42.49	40.57	39.03	37.77	416.44
TOTAL FLYAWAY	1221.31	1180.29	1320.27	1463.79	1602.52	1494.51	1418.04	1359.67	1312.92	1278.01	13651.32

ORIGINAL PAGE IS OF POOR QUALITY

COST SUMMARY

RDT AND E	TOTAL*
PROTOTYPE AIRCRAFT	606.44
DESIGN ENGINEERING	878.18
DEVELOPMENT TEST ARTICLES	272.55
FLIGHT TEST	86.33
ENGINE DEVELOPMENT CRUISE	684.03
ENGINE DEVELOPMENT LIFT	0.0
AVIONICS DEVELOPMENT	0.0
MAINTENANCE TRAINER DEVEL	0.0
OPERATOR TRAINER DEVELOP	0.0
DEVELOPMENT TOOLING	766.05
SPECIAL SUPPORT EQUIPMENT	12.13
DEVELOPMENT SPARES	96.74
TECHNICAL DATA	17.01
TOTAL RDTE	3419.47

INVESTMENT	TOTAL*	PER PROD A/C**
PRODUCTION AIRCRAFT	13651.32	45504.40
PRODUCTION ENGINEERING	0.0	0.0
MAINTENANCE TRAINERS	0.0	0.0
OPERATOR TRAINERS	0.0	0.0
PRODUCTION TOOLING	414.66	1382.18
SPECIAL SUPPORT EQUIPMENT	682.57	2275.22
PRODUCTION SPARES	2095.36	6984.52
TECHNICAL DATA	84.22	280.73
TOTAL INVESTMENT	16928.11	56427.04

DIRECT OPERATIONAL COST (DOC)	C/SM***	PERCENT
FLIGHT CREW	0.09697	5.54090
FUEL AND OIL	0.71196	40.68027
INSURANCE	0.13060	7.46229
DEPRECIATION	0.42021	24.00995
MAINTENANCE	0.39040	22.30658
TOTAL DOC	1.75015	100.000

INDIRECT OPERATIONAL COST (IOC)	C/SM***	PERCENT
SYSTEM	0.00345	0.43230
LOCAL	0.09154	11.47874
AIRCRAFT CONTROL	0.00513	0.64312
CABIN ATTENDANT	0.06979	8.75130
FOOD AND BEVERAGE	0.02412	3.02430
PASSENGER HANDLING	0.13656	17.12459
CARGO HANDLING	0.00849	1.06447
OTHER PASSENGER EXPENSE	0.33550	42.07144
OTHER CARGO EXPENSE	0.00278	0.34833
GENERAL + ADMINISTR.	0.12011	15.06139
TOTAL IOC	0.79745	100.000

MISC. DATA	
RANGE (ST. MILES)	4833.02
BLOCK SPEED (MPH)	1322.70
FARE ($)	248.72
FLEET SIZE	14.25
PRODUCTION BASIS	300.00
REV.PASSENG.(MIL.PER YR)	1.81
AVER. CARGO PER FLIGHT	2000.00
FLIGHT PER A/C PER YEAR	985.25

RETURN ON INVESTMENT (ROI)	
TOTAL REVENUE PER YEAR *	469.72
TOTAL EXPENSE PER YEAR *	404.47
TOTAL INVESTMENT * INCL. FACILITIES	966.42
ROI BEFORE TAXES	13.50
ROI AFTER TAXES	7.02

* - MILLIONS OF DOLLARS
** - 1000 OF DOLLARS PER PRODUCTION A/C
*** - CENTS PER SEAT MILE

RESEARCH DEVELOPMENT TEST AND EVALUATION (RDTE)

	DEVELOPMENT AND DESIGN	CONTRACTOR TEST AND EVALU	DEVELOPMENT AIRCRAFT	TOTAL RDT AND E
AIRFRAME	1429.77	312.07	412.17	2154.01
ENGINEERING				
HOURS	43963.	7129.	2055.	53148.
LABOR RATE	8.17	8.17	8.17	8.17
OVERHEAD RATE	9.20	9.20	9.20	9.20
TOTAL	763.64	123.84	35.70	923.18
TOOLING				
HOURS	33010.	1713.	3426.	38148.
LABOR RATE	6.09	6.09	6.09	6.09
OVERHEAD RATE	12.36	12.36	12.36	12.36
TOTAL	666.13	31.60	63.20	760.94
MANUFACTURING				
HOURS		6851.	13703.	20554.
LABOR RATE		5.12	5.12	5.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		108.53	217.05	325.58
QUALITY CONTROL				
HOURS		1370.	2741.	4111.
LABOR RATE		6.29	6.29	6.29
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		23.31	46.62	69.93
MATERIAL				
RAW AND PRCHSD		7.16	14.32	21.48
PURCHASED EQUIP		13.30	26.59	39.89
TOTAL		20.46	40.91	61.37
MISCELLANEOUS				
HOURS		274.	548.	822.
LABOR RATE		5.12	5.12	5.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		4.34	8.68	13.02
ENGINES	684.03		68.63	752.65
AVIONICS	0.0		2.00	2.00
PROFIT(AIRFRAME)	214.47	46.81	61.83	323.10
INSUR.+TAXES			41.22	41.22
WARRANTY			20.61	20.61
SUBTOTAL	2328.27	358.88	606.44	3293.59
OTHER ITEMS				125.88
TOTAL (RDTE)				3419.47

AIRCRAFT MODEL --CL 1701-8
I.O.C. DATE --1990
DESIGN SPEED --SUPERSONIC

ENGINE I.D. -- 101000
SLS SCALE 1.0 = 85800
NUMBER OF ENGINES = 4.

WING QUARTER CHORD SWEEP = 72.22 DEG
WING TAPER RATIO = 0.0

1 W/S	45.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2 T/W	0.531	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3 AR	1.34	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4 T/C	3.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5 RADIUS N. MI	4200	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
6 GROSS WEIGHT	437593	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
7 FUEL WEIGHT	108119	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
8 OP. WT. EMPTY	260473	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
9 ZERO FUEL WT.	329473	0.	0	0	0	0	0	0	0	0	0	0	0	0	0	0
10 THRUST/ENGINE	58145	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
11 ENGINE SCALE	0.678	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
12 WING AREA	9613.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
13 WING SPAN	113.3	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
14 H. TAIL AREA	837.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
15 V. TAIL AREA	364.4	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
16 BODY LENGTH	343.4	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
COST DATA																
17 RDTE - BIL.	4.722	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
18 FLYAWAY - MIL.	65.69	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
19 INVESTMNT-BIL.	1.104	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20 DOC - C/SM	1.896	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
21 IOC - C/SM	0.810	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22 PGI A.T. - O/O	3.80	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
CONSTRAINT OUTPUT																
23 CTOL LNDG D(1)	1095	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
24 AP SPEED-KT(1)	160.2	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
25 CTOL LNDG D(2)	8177	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
26 AP SPEED-KT(2)	161.3	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
27 CTOL LNDG D(3)	8259	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
28 AP SPEED-KT(3)	162.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0

FAR.T.O. FLD LENGTH = 7280
2ND SEG. CLIMB GRADIENT = .069 (ENG. OUT)

MACH 3.2 - UNCOOLED

MACH 3.2 LH2 AST

T/C	AR	W/S	T/W
3.00	1.34	45.5	0.531

W E I G H T   S T A T E M E N T

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
TAKE-OFF WEIGHT	(437594.)		
FUEL AVAILABLE	108120.	FUEL	24.71
ZERO FUEL WEIGHT	(329474.)		
PAYLOAD	49000.	PAYLOAD	11.20
OPERATING WEIGHT	(280474.)		
OPERATING ITEMS	5390.	OPERATING ITEMS	2.38
STANDARD ITEMS	5043.		
EMPTY WEIGHT	(270041.)		
WING	66099.		
TAIL	10944.		
BODY	51338.	STRUCTURE	36.18
LANDING GEAR	20743.		
SURFACE CONTROLS	5623.		
NACELLE AND ENGINE SECTION	3659.		
PROPULSION	(76708.)	PROPULSION	17.53
WEIGHT OF LIFT ENGINES	0.		
VECTOR CONTROL SYSTEM	0.		
ENGINES	31716.		
THRUST REVERSAL	0.		
AIR INDUCTION SYSTEM	16044.		
FUEL SYSTEM	27539.		
ENGINE CONTROLS + STARTER	1410.		
INSTRUMENTS	1118.		
HYDRAULICS	3492.		
ELECTRICAL	4768.		
AVIONICS	1900.	EQUIPMENT	8.00
FURNISHINGS AND EQUIPMENT	11500.		
ENVIRONMENTAL CONTROL SYSTEM	10269.		
AUXILIARY GEAR	1980.		
A.M.P.R.	(223776.)	TOTAL	(100.00)
EXCESS FUEL CAPACITY - BODY	-0.		
EXCESS FUEL CAPACITY - WING	0.		
EXCESS BODY LENGTH - FT	0.0		

W E I G H   M A T R I X

ELEMENT/ / MATERIAL	AL	TIT.	STEEL	COMP.	OTHER	TOTAL
WING	0.	60414.	1322.	3305.	1058.	66099.
TAIL	0.	10562.	108.	0.	174.	10844.
FUSEL	16736.	26388.	924.	1283.	6007.	51338.
L. G.	0.	5207.	7965.	0.	7571.	20743.
NACELLE	0.	613.	1217.	0.	0.	1829.
AIR INDUCT	0.	14953.	160.	0.	931.	16044.
S. CTLS	0.	1603.	2755.	84.	1181.	5623.
TOTALS	16736.	119739.	14452.	4673.	16920.	172520.

M I S S I O N   S U M M A R Y

MACH 3.2 LH2 AST

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG L/D RATIO	AVG SFC (FF/T)	MAX OVER PRES
TAKEOFF															
POWER 1	0.	0.0	437594.	546.	546.	0.	0.	10.0	10.0	0.	-101101.	0.	0.0	0.150	0.0
POWER 2	0.	0.300	437047.	1033.	1580.	0.	0.	0.3	10.3	0.	101211.	0.	6.13	0.504	0.0
CLIMB	0.	0.300	436014.	1415.	2994.	4.	4.	0.9	11.2	0.	101211.	0.	8.20	0.526	0.0
CRUISE	5000.	0.414	434599.	752.	3746.	0.	4.	4.0	15.2	0.	-101101.	0.	8.77	0.228	0.0
ACCEL	5000.	0.414	433847.	376.	4123.	1.	5.	0.3	15.5	0.	101211.	0.	9.56	0.537	0.0
CLIMB	5000.	0.539	433471.	5908.	10031.	44.	49.	5.6	21.1	0.	101211.	0.	9.12	0.567	0.0
CLIMB	34000.	0.989	427563.	21253.	31283.	483.	531.	23.5	44.6	0.	101208.	0.	6.38	0.596	2.39
CLIMB	69500.	3.194	406310.	805.	32088.	38.	569.	1.2	45.8	0.	101208.	0.	7.61	0.606	1.36
CRUISE	74500.	3.200	405505.	53687.	85775.	3391.	3960.	110.1	155.9	0.	-101201.	0.	7.72	0.597	1.27
DECEL	77500.	3.200	351818.	29.	85804.	43.	4003.	1.5	157.4	0.	101501.	0.	7.68	-0.376	1.17
DESCENT	77500.	2.789	351789.	271.	86075.	185.	4189.	13.9	171.3	0.	101501.	0.	7.69	-0.149	1.95
CRUISE	77500.	3.200	351519.	169.	86243.	11.	4200.	0.4	171.7	0.	-101201.	0.	7.69	0.600	1.14
CRUISE	5000.	0.414	351350.	722.	86965.	0.	4200.	5.0	176.7	0.	-101101.	0.	9.46	0.234	0.0
RESET	0.	0.0	350628.	0.	86965.	0.	4200.	0.0	176.7	0.	0.	0.	0.0	0.0	0.0
RESET	0.	0.0	350628.	0.	86965.	-4200.	0.	*****	0.0	0.	0.	0.	0.0	0.0	0.0
RESERVE	0.	0.0	350628.	6088.	93052.	0.	0.	0.0	0.0	0.	0.	0.	0.0	0.0	0.0
CLIMB	0.	0.200	344541.	948.	94000.	2.	2.	0.6	0.6	0.	101211.	0.	8.06	0.524	0.0
CLIMB	1500.	0.505	343593.	4719.	98719.	33.	35.	4.4	5.0	0.	101211.	0.	8.52	0.565	0.0
CRUISE	37000.	0.900	338874.	4254.	102972.	145.	180.	16.9	21.8	0.	-101201.	0.	9.19	0.413	0.0
DESCENT	37000.	0.900	334621.	139.	103111.	49.	228.	6.9	28.7	0.	101501.	0.	8.53	-0.168	0.0
CRUISE	37000.	0.900	334482.	912.	104023.	31.	260.	3.6	32.4	0.	-101201.	0.	9.17	0.412	0.0
CRUISE	15000.	0.503	333570.	4188.	108210.	0.	260.	30.0	62.4	0.	-101101.	0.	9.62	0.243	0.0

TOGRWT= 437593.5   FUEL A=108119.6   FUEL R=108210.2

PRODUCTION

	PRODUCTION YEARS 1	2	3	4	5	6	7	8	9	10	TOTAL
AIRFRAME	1129.63	1047.09	1151.57	1261.51	1368.16	1266.74	1195.34	1141.08	1097.79	1062.05	11719.93
ENGINEERING											
HOURS	4074.	3510.	3714.	3946.	4171.	3780.	3507.	3300.	3136.	3001.	36139.
LABOR RATE	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	
OVERHEAD RATE	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	
TOTAL	70.77	60.96	64.51	68.55	72.45	65.66	60.91	57.32	54.47	52.12	627.73
TOOLING											
HOURS	4689.	4212.	4457.	4736.	5005.	4536.	4208.	3960.	3763.	3601.	43366.
LABOR RATE	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	
OVERHEAD RATE	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	
TOTAL	90.21	77.70	82.22	87.37	92.34	83.70	77.64	73.06	69.43	66.43	800.11
MANUFACTURING											
HOURS	40744.	35096.	37138.	39464.	41709.	37803.	35069.	33001.	31357.	30006.	361386.
LABOR RATE	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	645.39	555.92	588.26	625.12	660.67	598.80	555.49	522.73	496.70	475.29	5724.36
QUALITY CONTROL											
HOURS	8149.	7019.	7428.	7893.	8342.	7561.	7014.	6600.	6271.	6001.	72277.
LABOR RATE	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	139.61	119.40	126.34	134.26	141.89	128.61	119.30	112.27	106.68	102.08	1229.44
MATERIAL											
RAW AND PURCH	55.24	73.30	93.35	112.42	131.03	128.11	125.92	124.17	122.73	121.49	1088.27
PURCHASED EQUIP	102.59	137.07	173.36	208.79	243.35	237.92	233.85	230.61	227.92	225.62	2021.07
TOTAL	157.83	210.37	266.71	321.21	374.38	366.03	359.77	354.78	350.65	347.11	3109.33
MISCELLANEOUS											
HOURS	1630.	1404.	1486.	1579.	1668.	1512.	1403.	1320.	1254.	1200.	14455.
LABOR RATE	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	25.82	22.24	23.53	25.00	26.43	23.95	22.22	20.91	19.87	19.01	228.97
ENGINES	152.31	178.96	214.05	247.86	280.14	267.45	258.14	250.84	244.87	239.83	2334.45
AVIONICS	6.00	9.00	12.00	15.00	18.00	18.00	18.00	18.00	18.00	18.00	150.00
PROFIT	169.29	157.06	172.74	189.23	205.22	190.01	179.30	171.16	164.67	159.31	1757.99
INSUR.+TAXES	112.86	104.71	115.16	126.15	136.82	126.67	119.53	114.11	109.78	106.21	1171.99
WARRANTY	56.43	52.35	57.58	63.08	68.41	63.34	59.77	57.05	54.89	53.10	586.00
TOTAL FLYAWAY	1625.53	1549.17	1723.09	1902.82	2076.75	1932.21	1830.08	1752.24	1689.99	1643.85	17725.71

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COST SUMMARY

RDT AND E

	TOTAL*
PROTOTYPE AIRCRAFT	817.51
DESIGN ENGINEERING	1272.85
DEVELOPMENT TEST ARTICLES	384.29
FLIGHT TEST	149.84
ENGINE DEVELOPMENT CRUISE	949.72
ENGINE DEVELOPMENT LIFT	0.0
AVIONICS DEVELOPMENT	0.0
MAINTENANCE TRAINER DEVEL	0.0
OPERATOR TRAINER DEVELOP	0.0
DEVELOPMENT TOOLING	990.20
SPECIAL SUPPORT EQUIPMENT	16.35
DEVELOPMENT SPARES	117.77
TECHNICAL DATA	23.49
TOTAL RDTE	4722.07

INVESTMENT

	TOTAL*	PER PROD A/C**
PRODUCTION AIRCRAFT	17725.71	59085.73
PRODUCTION ENGINEERING	0.0	0.0
MAINTENANCE TRAINERS	0.0	0.0
OPERATOR TRAINERS	0.0	0.0
PRODUCTION TOOLING	645.94	2153.14
SPECIAL SUPPORT EQUIPMENT	886.29	2954.29
PRODUCTION SPARES	2503.32	8344.41
TECHNICAL DATA	108.81	362.69
TOTAL INVESTMENT	21870.07	72900.19

DIRECT OPERATIONAL COST (DOC)

	C/SM***	PERCENT
FLIGHT CREW	0.08477	4.47374
FUEL AND OIL	0.78447	41.39876
INSURANCE	0.14924	7.87591
DEPRECIATION	0.48018	25.34076
MAINTENANCE	0.39624	20.91087
TOTAL DOC	1.89490	100.000

INDIRECT OPERATIONAL COST (IOC)

	C/SM***	PERCENT
SYSTEM	0.00352	0.43430
LOCAL	0.11105	13.71035
AIRCRAFT CONTROL	0.00513	0.63318
CABIN ATTENDANT	0.06101	7.53230
FOOD AND BEVERAGE	0.02108	2.60304
PASSENGER HANDLING	0.13656	16.85991
CARGO HANDLING	0.00849	1.04801
OTHER PASSENGER EXPENSE	0.33550	41.42282
OTHER CARGO EXPENSE	0.00278	0.34296
GENERAL + ADMINISTR.	0.12484	15.41312
TOTAL IOC	0.80994	100.000

MISC. DATA

RANGE (ST. MILES)	4833.21
BLOCK SPEED (MPH)	1513.07
FARE ($)	248.73
FLEET SIZE	12.46
PRODUCTION BASIS	300.00
REV.PASSENG.(MIL.PER YR)	1.61
AVER. CARGO PER FLIGHT	2000.00
FLIGHT PER A/C PER YEAR	1127.00

RETURN ON INVESTMENT (ROI)

TOTAL REVENUE PER YEAR *	469.74
TOTAL EXPENSE PER YEAR *	429.45
TOTAL INVESTMENT * INCL. FACILITIES	1104.15
ROI BEFORE TAXES	7.30
ROI AFTER TAXES	3.80

* - MILLIONS OF DOLLARS
** - 1000 OF DOLLARS PER PRODUCTION A/C
*** - CENTS PER SEAT MILE

RESEARCH DEVELOPMENT TEST AND EVALUATION (RDTE)

	DEVELOPMENT AND DESIGN	CONTRACTOR TEST AND EVALU	DEVELOPMENT AIRCRAFT	TOTAL RDT AND E
AIRFRAME	1967.87	464.50	581.16	3013.54
ENGINEERING				
HOURS	63721.	11464.	2902.	78086.
LABOR RATE	8.17	8.17	8.17	8.17
OVERHEAD RATE	9.20	9.20	9.20	9.20
TOTAL	1106.82	199.12	50.40	1356.35
TOOLING				
HOURS	42668.	2418.	4836.	49922.
LABOR RATE	6.09	6.09	6.09	6.09
OVERHEAD RATE	12.36	12.36	12.36	12.36
TOTAL	861.05	44.61	89.22	994.88
MANUFACTURING				
HOURS		9672.	19344.	29016.
LABOR RATE		5.12	5.12	5.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		153.20	306.41	459.61
QUALITY CONTROL				
HOURS		1934.	3869.	5803.
LABOR RATE		6.29	6.29	6.29
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		32.90	65.81	98.71
MATERIAL				
RAW AND PRCHSD		9.99	19.97	29.96
PURCHASED EQUIP		18.55	37.09	55.64
TOTAL		28.53	57.06	85.60
MISCELLANEOUS				
HOURS		387.	774.	1161.
LABOR RATE		5.12	5.12	5.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		6.13	12.26	18.38
ENGINES	949.72		60.00	1009.72
AVIONICS	0.0		2.00	2.00
PROFIT(AIRFRAME)	295.18	69.68	87.17	452.03
INSUR.+TAXES			58.12	58.12
WARRANTY			29.06	29.06
SUBTOTAL	3212.77	534.18	817.51	4564.46
OTHER ITEMS				157.62
TOTAL (RDTE)				4722.07

SUMMARY ? NO. 2 A S S E T P A R A M E T R I C A N A L Y S I S FEBRUARY 1? 1975

AIRCRAFT MODEL --CL 1701-8
I.O.C. DATE --1990
DESIGN SPEED --SUPERSONIC

ENGINE I.D. -- 101000
SLS SCALE 1.0 = 85800
NUMBER OF ENGINES = 4.

WING QUARTER CHORD SWEEP = 72.22 DEG
WING TAPER RATIO = 0.0

1 W/S	45.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2 T/W	0.531	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3 AR	1.34	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4 T/C	3.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5 RADIUS N. MI	4200	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
6 GROSS WEIGHT	428939	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
7 FUEL WEIGHT	106562	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
8 OP. WT. EMPTY	273376	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
9 ZERO FUEL WT.	322376	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
10 THRUST/ENGINE	56995	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
11 ENGINE SCALE	0.664	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
12 WING AREA	9431.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
13 WING SPAN	112.2	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
14 H. TAIL AREA	817.7	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
15 V. TAIL AREA	357.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
16 BODY LENGTH	341.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
COST DATA																
17 RDTE - BIL.	4.844	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
18 FLYAWAY - MIL.	80.81	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
19 INVESTMNT-BIL.	1.042	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20 DOC - C/SM	1.839	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
21 IOC - C/SM	0.806	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22 ROI A.T. - O/O	4.97	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
CONSTRAINT OUTPUT																
23 CTOL LNDG D(1)	8083	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
24 AP SPEED-KT(1)	160.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
25 CTOL LNDG D(2)	8166	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
26 AP SPEED-KT(2)	161.2	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
27 CTOL LNDG D(3)	8256	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
28 AP SPEED-KT(3)	162.3	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0

FAR T.O. FLD. LENGTH = 7270'
2ND SEG. CLIMB GRADIENT = .063 (ENG-OUT)

MACH 3.2 — COOLED

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MACH 3.2 LH2 AST

T/C	AR	W/S	T/W
3.00	1.34	45.5	0.531

W E I G H T   S T A T E M E N T

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
TAKE-OFF WEIGHT	(428939.)		
FUEL AVAILABLE	106563.	FUEL	24.84
ZERO FUEL WEIGHT	(322377.)		
PAYLOAD	49000.	PAYLOAD	11.42
OPERATING WEIGHT	(273377.)		
OPERATING ITEMS	5387.	OPERATING ITEMS	2.42
STANDARD ITEMS	4996.		
EMPTY WEIGHT	(262993.)		
WING	62665.		
TAIL	10604.		
BODY	48843.	STRUCTURE	35.35
LANDING GEAR	20415.		
SURFACE CONTROLS	5528.		
NACELLE AND ENGINE SECTION	3586.		
PROPULSION	(75437.)	PROPULSION	17.59
WEIGHT OF LIFT ENGINES	0.		
VECTOR CONTROL SYSTEM	0.		
ENGINES	31088.		
THRUST REVERSAL	0.		
AIR INDUCTION SYSTEM	15689.		
FUEL SYSTEM	27258.		
ENGINE CONTROLS + STARTER	1402.		
INSTRUMENTS	1114.		
HYDRAULICS	3423.		
ELECTRICAL	4745.		
AVIONICS	1900.	EQUIPMENT	8.37
FURNISHINGS AND EQUIPMENT	11500.		
ENVIRONMENTAL CONTROL SYSTEM	6508.		
AUXILIARY GEAR	1980.		
COOLING	4745.		
A.M.P.R.	(218605.)	TOTAL	(100.00)
EXCESS FUEL CAPACITY - BODY	-0.		
EXCESS FUEL CAPACITY - WING	0.		
EXCESS BODY LENGTH - FT	0.0		

W E I G H T   M A T R I X

ELEMENT/ / MATERIAL	AL	TIT.	STEEL	COMP.	OTHER	TOTAL
WING	14037.	42612.	1253.	3760.	1003.	62665.
TAIL	0.	10329.	106.	0.	170.	10604.
FUSEL	36144.	4884.	879.	1221.	5715.	48843.
L. G.	0.	5124.	7839.	0.	7451.	20415.
NACELLE	0.	601.	1192.	0.	0.	1793.
AIR INDUCT	0.	14622.	157.	0.	910.	15689.
S. CTLS	0.	1575.	2709.	83.	1161.	5528.
TOTALS	50180.	79747.	14136.	5064.	16409.	165536.

MACH 3.2 LH2 AST

T/C	AR	W/S	T/W
3.00	1.34	45.5	0.531

C O N F I G U R A T I O N   G E O M E T R Y

BASIC WING--	AREA(SQ.FT)	SPAN(FT)	TAPER RATIO	C/4 SWEEP	L.E. SWEEP	CR(FT)	MAC(FT)
	9431.4	112.24	0.0	72.218	75.500	168.05	112.04

INBOARD WING--	AREA(SQ.FT)	EXP. AREA	L.E. SWEEP	REF L(FT)	SFLE(SQ.FT)	AVG T/C
	9431.4	7215.5	75.50	98.00	0.0	3.00

OUTBOARD WING--	AREA(SQ.FT)	Y BRK(FT)	L.E. SWEEP	REF L(FT)	SFLE(SQ.FT)	AVG T/C
	0.0	0.0	75.50	98.00	0.0	3.00

TOTAL WING--	AREA(SQ.FT)	EFF AR	AVG T/C	CR(FT)	CT(FT)	(B/2)/LW	P
	9431.4	1.34	3.00	168.05	0.0	0.259	0.387

WING TANK--	CBAR1(FT)	CBAR2(FT)	FTL(FT)	FVWING(CU FT)	FVBOX(CU FT)
	148.74	0.0	49.67	0.0	0.0

FUSELAGE--	LENGTH(FT)	S WET(SQ FT)	BWW(FT)	EQUIV D(FT)	SPI(SQ FT)
	341.49	14230.5	14.07	16.44	212.25

BW(FT)	BH(FT)	SBW(SQ FT)	FVB(CU FT)
12.90	19.43	14230.54	25251.38

TAIL--	SHT(SQ.FT)	SHTX(SQ.FT)	HT REF L(FT)	SVT(SQ.FT)	SVTX(SQ.FT)	VT REF L(FT)
	817.73	652.07	19.99	357.55	357.55	22.67

PROPULSION--	ENG L(FT)	ENG D(FT)	POD L(FT)	POD D(FT)	POD S WET	NO. PODS	INLET L(FT)
	20.44	5.09	42.67	7.88	4226.84	4.	0.0

MACH 3.2 LH2 AST

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG L/D RATIO	AVG SFC (FF/T)	MAX OVER PRES
TAKEOFF															
POWER 1	0.	0.0	428939.	536.	536.	0.	0.	10.0	10.0	0.	-101101.	0.	0.0	0.150	0.0
POWER 2	0.	0.300	428404.	1013.	1548.	0.	0.	0.3	10.3	0.	101211.	0.	6.13	0.504	0.0
CLIMB	0.	0.300	427391.	1387.	2935.	4.	4.	0.9	11.2	0.	101211.	0.	8.20	0.526	0.0
CRUISE	5000.	0.414	426004.	738.	3673.	0.	4.	4.0	15.2	0.	-101101.	0.	8.76	0.228	0.0
ACCEL	5000.	0.414	425266.	369.	4042.	1.	5.	0.3	15.5	0.	101211.	0.	9.54	0.537	0.0
CLIMB	5000.	0.539	424897.	5805.	9847.	44.	49.	5.6	21.1	0.	101211.	0.	9.09	0.567	0.0
CLIMB	34000.	0.989	419092.	21129.	30976.	490.	539.	23.8	44.9	0.	101208.	0.	6.34	0.596	2.38
CLIMB	69500.	3.194	397963.	806.	31783.	38.	577.	1.3	46.2	0.	101208.	0.	7.57	0.606	1.35
CRUISE	74500.	3.200	397157.	52731.	84514.	3383.	3960.	109.8	156.0	0.	-101201.	0.	7.68	0.598	1.25
DECEL	77500.	3.200	344425.	28.	84542.	43.	4003.	1.5	157.5	0.	101501.	0.	7.65	-0.376	1.15
DESCENT	77500.	2.789	344397.	264.	84806.	185.	4187.	13.9	171.3	0.	101501.	0.	7.66	-0.149	1.93
CRUISE	77500.	3.200	344135.	182.	84988.	12.	4200.	0.4	171.7	0.	-101201.	0.	7.65	0.600	1.13
CRUISE	5000.	0.414	343951.	708.	85695.	0.	4200.	5.0	176.7	0.	-101101.	0.	9.45	0.234	0.0
RESET	0.	0.0	343243.	0.	85695.	0.	4200.	0.0	176.7	0.	0.	0.	0.0	0.0	0.0
RESET	0.	0.0	343243.	0.	85695.	-4200.	0.	*****	0.0	0.	0.	0.	0.0	0.0	0.0
RESERVE	0.	0.0	343243.	5999.	91694.	0.	0.	0.0	0.0	0.	0.	0.	0.0	0.0	0.0
CLIMB	0.	0.200	337245.	927.	92621.	2.	2.	0.6	0.6	0.	101211.	0.	8.06	0.524	0.0
CLIMB	1500.	0.505	336318.	4624.	97245.	33.	35.	4.4	5.0	0.	101211.	0.	8.49	0.565	0.0
CRUISE	37000.	0.900	331695.	4176.	101421.	145.	180.	16.9	21.8	0.	-101201.	0.	9.17	0.413	0.0
DESCENT	37000.	0.900	327518.	135.	101556.	48.	228.	6.9	28.7	0.	101501.	0.	8.49	-0.168	0.0
CRUISE	37000.	0.900	327382.	900.	102456.	32.	260.	3.7	32.4	0.	-101201.	0.	9.15	0.412	0.0
CRUISE	15000.	0.503	326483.	4105.	106561.	0.	260.	30.0	62.4	0.	-101101.	0.	9.60	0.243	0.0

TOGRWT= 428939.3 FUEL A=106562.7 FUEL R=106560.5

PRODUCTION

	PRODUCTION YEARS 1	2	3	4	5	6	7	8	9	10	TOTAL
AIRFRAME	1050.10	972.93	1069.31	1170.81	1269.27	1174.79	1108.28	1057.74	1017.42	984.14	10874.79
ENGINEERING											
HOURS	3802.	3275.	3466.	3683.	3893.	3528.	3273.	3080.	2926.	2800.	33727.
LABOR RATE	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	
OVERHEAD RATE	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	
TOTAL	66.05	56.89	60.20	63.97	67.61	61.28	56.85	53.50	50.83	48.64	585.83
TOOLING											
HOURS	4563.	3930.	4159.	4420.	4671.	4234.	3927.	3696.	3512.	3360.	40472.
LABOR RATE	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	6.09	
OVERHEAD RATE	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	12.36	
TOTAL	84.19	72.52	76.73	81.54	86.18	78.11	72.46	68.19	64.79	62.00	746.70
MANUFACTURING											
HOURS	38025.	32753.	34659.	36830.	38925.	35280.	32728.	30798.	29264.	28003.	337266.
LABOR RATE	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	602.31	518.81	548.99	583.39	616.57	558.83	518.41	487.84	463.55	443.57	5342.29
QUALITY CONTROL											
HOURS	7605.	6551.	6932.	7366.	7785.	7056.	6546.	6160.	5853.	5601.	67453.
LABOR RATE	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.29	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	129.36	111.43	117.91	125.30	132.42	120.02	111.34	104.77	99.56	95.27	1147.38
MATERIAL											
RAW AND PURCH	50.43	67.39	85.23	102.65	119.64	116.97	114.97	113.37	112.05	110.92	993.62
PURCHASED EQUIP	93.66	125.15	158.28	190.63	222.18	217.23	213.51	210.55	208.10	206.00	1845.29
TOTAL	144.10	192.53	243.51	293.27	341.82	334.19	328.48	323.93	320.15	316.92	2838.91
MISCELLANEOUS											
HOURS	1521.	1310.	1386.	1473.	1557.	1411.	1309.	1232.	1171.	1120.	13491.
LABOR RATE	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	5.12	
OVERHEAD RATE	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	10.72	
TOTAL	24.09	20.75	21.96	23.34	24.66	22.35	20.74	19.51	18.54	17.74	213.69
ENGINES	150.50	176.83	211.50	244.91	276.80	264.26	255.07	247.85	241.95	236.97	2306.64
AVIONICS	6.00	9.00	12.00	15.00	18.00	18.00	18.00	18.00	18.00	18.00	150.00
PROFIT	157.52	145.94	160.40	175.62	190.39	176.22	166.24	158.66	152.61	147.62	1631.22
INSUR.+TAXES	105.01	97.29	106.93	117.08	126.93	117.48	110.83	105.77	101.74	98.41	1087.48
WARRANTY	52.51	48.65	53.47	58.54	63.46	58.74	55.41	52.89	50.87	49.21	543.74
TOTAL FLYAWAY	1521.63	1450.64	1613.61	1781.96	1944.85	1809.48	1713.83	1640.91	1582.59	1539.34	16598.84

COST SUMMARY

RDT AND E

	TOTAL*
PROTOTYPE AIRCRAFT	764.66
DESIGN ENGINEERING	1395.00
DEVELOPMENT TEST ARTICLES	357.88
FLIGHT TEST	146.80
ENGINE DEVELOPMENT CRUISE	939.34
ENGINE DEVELOPMENT LIFT	0.0
AVIONICS DEVELOPMENT	0.0
MAINTENANCE TRAINER DEVEL	0.0
OPERATOR TRAINER DEVELOP	0.0
DEVELOPMENT TOOLING	1068.86
SPECIAL SUPPORT EQUIPMENT	15.30
DEVELOPMENT SPARES	111.42
TECHNICAL DATA	24.10
TOTAL RDTE	4843.55

INVESTMENT

	TOTAL*	PER PROD A/C**
PRODUCTION AIRCRAFT	16598.84	55329.48
PRODUCTION ENGINEERING	0.0	0.0
MAINTENANCE TRAINERS	0.0	0.0
OPERATOR TRAINERS	0.0	0.0
PRODUCTION TOOLING	332.86	1109.54
SPECIAL SUPPORT EQUIPMENT	829.94	2766.47
PRODUCTION SPARES	2368.21	7894.03
TECHNICAL DATA	100.65	335.50
TOTAL INVESTMENT	20230.50	67435.00

DIRECT OPERATIONAL COST (DOC)

	C/SM***	PERCENT
FLIGHT CREW	0.08481	4.61193
FUEL AND OIL	0.77302	42.03610
INSURANCE	0.14082	7.65750
DEPRECIATION	0.45308	24.63802
MAINTENANCE	0.38722	21.05646
TOTAL DOC	1.83894	100.000

INDIRECT OPERATIONAL COST (IOC)

	C/SM***	PERCENT
SYSTEM	0.00361	0.44778
LOCAL	0.10885	13.49766
AIRCRAFT CONTROL	0.00513	0.63593
CABIN ATTENDANT	0.06103	7.56839
FOOD AND BEVERAGE	0.02109	2.61551
PASSENGER HANDLING	0.13656	16.93324
CARGO HANDLING	0.00849	1.05257
OTHER PASSENGER EXPENSE	0.33550	41.60289
OTHER CARGO EXPENSE	0.00278	0.34445
GENERAL + ADMINISTR.	0.12340	15.30156
TOTAL IOC	0.80643	100.000

MISC. DATA

RANGE (ST. MILES)	4833.20
BLOCK SPEED (MPH)	1512.40
FARE ($)	248.73
FLEET SIZE	12.46
PRODUCTION BASIS	300.00
REV.PASSENG.(MIL.PER YR)	1.81
AVER. CARGO PER FLIGHT	2000.00
FLIGHT PER A/C PER YEAR	1126.51

RETURN ON INVESTMENT (ROI)

TOTAL REVENUE PER YEAR *	469.74
TOTAL EXPENSE PER YEAR *	420.01
TOTAL INVESTMENT * INCL. FACILITIES	1041.57
ROI BEFORE TAXES	9.55
ROI AFTER TAXES	4.97

* - MILLIONS OF DOLLARS
** - 1000 OF DOLLARS PER PRODUCTION A/C
*** - CENTS PER SEAT MILE

RESEARCH DEVELOPMENT TEST AND EVALUATION (RDTE)

	DEVELOPMENT AND DESIGN	CONTRACTOR TEST AND EVALU	DEVELOPMENT AIRCRAFT	TOTAL RDT AND E
AIRFRAME	2159.87	438.85	541.22	3139.94
ENGINEERING				
HOURS	69835.	11040.	2708.	83583.
LABOR RATE	8.17	8.17	8.17	8.17
OVERHEAD RATE	9.20	9.20	9.20	9.20
TOTAL	1213.04	191.76	47.04	1451.84
TOOLING				
HOURS	46919.	2257.	4513.	53689.
LABOR RATE	6.09	6.09	6.09	6.09
OVERHEAD RATE	12.36	12.36	12.36	12.36
TOTAL	946.83	41.63	83.27	1071.74
MANUFACTURING				
HOURS		9026.	18053.	27079.
LABOR RATE		5.12	5.12	5.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		142.98	285.96	428.94
QUALITY CONTROL				
HOURS		1805.	3611.	5416.
LABOR RATE		6.29	6.29	6.29
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		30.71	61.42	92.12
MATERIAL				
RAW AND PRCHSD		9.12	18.24	27.35
PURCHASED EQUIP		16.93	33.87	50.80
TOTAL		26.05	52.10	78.15
MISCELLANEOUS				
HOURS		361.	722.	1083.
LABOR RATE		5.12	5.12	5.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		5.72	11.44	17.16
ENGINES	939.34		59.28	998.63
AVIONICS	0.0		2.00	2.00
PROFIT(AIRFRAME)	323.98	65.83	81.18	470.99
INSUR.+TAXES			54.12	54.12
WARRANTY			27.06	27.06
SUBTOTAL	3423.20	504.68	764.86	4692.74
OTHER ITEMS				150.82
TOTAL (RDTE)				4843.55

APPENDIX B

AERODYNAMIC HEATING ANALYSIS

Inviscid Flow Field Determination:

Local flow properties (pressure, temperature, velocity) at all examined locations on the airplane external surface are calculated by the equations of compressible flow theory as in Reference 1. Freestream air properties are obtained from the vehicle flight profile and from the United States Standard (1962) Atmosphere tables (Reference 2).

The specification of flow properties at the boundary layer edge requires knowledge of either the local flow deflection angle or the local pressure coefficient. In this case, local flow angles were obtained from airplane configuration drawings, and provided, with the vehicle angle of attack, a fairly good approximation of local flow properties at the boundary layer edge. This technique was only selected because the aerodynamic analysis usually used to determine pressure distribution was unavailable at that time. Subsequent checks showed no significant inaccuracies. Pressure coefficients were calculated for various Mach numbers and angles of attack for a grid of surface points on the wing by calculating from the flow angles, surface pressure distributions to match the load conditions of the airframe.

A typical calculation procedure for local flow properties is shown in Table 1. The equations are for a wedge (flat plate) in supersonic flow, and are applicable to all wing, fin, and fuselage areas (excluding conical sections at nose and tail). Temperature dependence of air properties is included in all calculations. Real gas effects are included for all supersonic flow field calculations and for heat transfer calculations above Mach 3. The air property charts of Reference 3 and 4 are used, either in tabular form for interpolation or as functional curve fits.

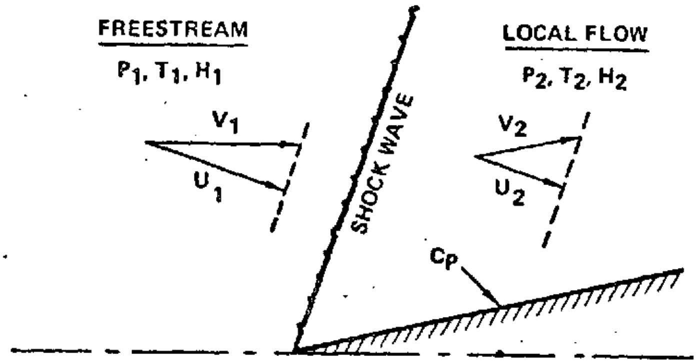
Heat Transfer Coefficients:

The following procedures are used to calculate heat transfer coefficients for aerodynamic heating:

- Laminar flow heat transfer is computed using the Blasius skin friction formula with the Eckert reference enthalpy formula to calculate reference conditions and the Colburn-Reynolds analogy to obtain the heat transfer coefficient.

TABLE 1. LOCAL FLOW ON A SUPERSONIC WEDGE

SKETCH:



NOTE:

1. SUBSCRIPT ($_1$) INDICATES FREESTREAM; ($_2$) INDICATES BOUNDARY LAYER EDGE
2. f_n (X, Y) ARE CURVE FIT OR TABULATED FUNCTIONS FOR THE GIVEN AIR PROPERTY VERSUS THE VARIABLES X AND Y

GIVEN:

P_1	FREESTREAM PRESSURE
T_1	FREESTREAM TEMPERATURE
M_1	VEHICLE MACH NUMBER
C_P	LOCAL PRESSURE COEFFICIENT
R	AIR GAS CONSTANT

FREESTREAM:	$\rho_1 = P_1/(R \cdot T_1)$	DENSITY
	$\gamma_1 = f_1(T_1, P_1)$	SPECIFIC HEAT RATIO
	$V_1 = M_1 \cdot \sqrt{P_1/(\gamma_1 \cdot \rho_1)}$	VELOCITY
	$H_1 = f_2(T_1, P_1)$	ENTHALPY
LOCAL:	$\xi = P_2/P_1 = 1 + \frac{\gamma_1}{2} C_P M_1^2$	STATIC PRESSURE RATIO
	$U_1 = V_1 \cdot \sqrt{(6\xi + 1)/(7M_1^2)}$	NORMAL VELOCITY COMPONENT
	$U_2/U_1 = 1 + \frac{P_1}{\rho_1 U_1^2}(1 - \xi)$	NORMAL VELOCITY RATIO
	$P_2 = \xi \cdot P_1$	LOCAL STATIC PRESSURE
	$H_2 = H_1 + \frac{1}{2}(U_1^2 - U_2^2)$	LOCAL STATIC ENTHALPY
	$T_2 = f_3(H_2, P_2)$	LOCAL STATIC TEMPERATURE
	$V_2 = \sqrt{V_1^2 - U_1^2 + U_2^2}$	LOCAL VELOCITY

- Turbulent flow heat transfer is computed using the Spalding and Chi skin friction theory, with a linear Crocco integration through the boundary layer to account for real gas effects in the compressible transformation, and the Colburn-Reynolds analogy to obtain the heat transfer coefficient.

Flow transition is assumed to occur at a local Reynolds number of one million, which for the present configuration and flight profile means that turbulent flow exists over all surfaces but the first foot or two of the fuselage nose and wing leading edge.

The calculation procedures for heat transfer coefficient have been included in computer subroutines for direct callout in the temperature calculation program. Use is made of standard atmosphere tables, the vehicle flight profile, and tabulated pressure coefficient data to calculate automatically the local flow field and the heat transfer coefficient at the airplane surface point being analyzed.

The local convective heat flow to the skin is

$$\frac{q_{conv}}{A} = h(T_r - T_w)$$

where h is the heat transfer coefficient, T_w is the skin temperature, and T_r is the recovery temperature. The recovery temperature, also called the adiabatic wall temperature, is the temperature the skin would reach in the absence of any other heat transfer at the surface. Recovery temperature is determined for real gas calculations from the recovery enthalpy, H_r, defined as

$$H_r = H_2 + \left(r\ V_2^2/2.\right)$$

H_2 and V_2 are evaluated at the boundary layer edge during the local flow calculation. The recovery factor, r, is defined as the ratio of recovery enthalpy increase (over local static enthalpy increase, or

$$r = \frac{H_r - H_2}{H_T - H_2}$$

The recovery factor is approximated well by the square root of Prandtl number for laminar flow, and by the cube root of Prandtl number of turbulent flow. T_r is found from real gas tables as a function of H_r and the local static pressure, P_2.

The term "reference condition" refers to evaluation of a property at a reference temperature, T^*, and the local static pressure, P_2. T^* is determined for these analyses by the Eckert reference enthalpy method (Reference Item-6), which defines a reference enthalpy as

$$H^* = .5 \times H_w + .28 \times H_2 + .22 \times H_r$$

H_w is evaluated at T_w and P_2.

The heat transfer coefficient is evaluated through calculation of a local Stanton number, St, defined as

$$St = \frac{h}{\rho c_p V_2}$$

Density, ρ, is evaluated at the reference condition for the Eckert reference enthalpy method (laminar flow), and at the local boundary layer edge condition for the Spalding and Chi method (turbulent flow). Specific heat, c_p, is approximated for real gas effects by substitution of a ratio of enthalpy difference to temperature difference, or

$$c_p = \frac{H_r - H_w}{T_r - T_w}$$

The procedure to determine the local Stanton number involves calculation of the local skin friction coefficient, C_f, and use of modified Reynolds analogy of the form

$$St = \frac{C_f}{2} R_{AF}$$

where R_{AF} is the Reynolds analogy factor. The R_{AF} selected for both laminar and turbulent flow is the Colburn-Reynolds analogy factor,

$$R_{AF} = (Pr^*)^{-2/3}$$

where Pr* is the Prandtl number evaluated at the reference condition. This form of the Reynolds analogy factor was found to give the best prediction of heat transfer when the Spalding and Chi theory was used for turbulent flow (see Reference 6).

The skin friction coefficient for laminar flow is based on the Blasius equation,

$$C_f = .664/(Re^*)^{0.5}$$

The Reynolds number, Re*, for this equation is the local Reynolds number based on distance from the leading edge, with air properties evaluated at the reference condition.

The skin friction coefficient for turbulent flow is based on a numerical curve fit of the incompressible flow formulas of Spalding and Chi (Reference 7) performed by White and Christoph (Reference 8),

$$C_{f,\ inc} = 0.225/(\log_{10} Re_x)^{2.32}$$

which agrees with the Spalding and Chi formulas within 0.5 percent. Re_x is the local Reynolds number based on distance from start of turbulence. The transformation to compressible flow is made by use of the transformation functions, F_C and F_{Rx}, to give

$$F_C\, C_f = C_{f,\ inc}$$

where $C_{f,\ inc}$ is evaluated at a modified Reynolds Number, F_{Rx}, R_{ex}.

The Spalding and Chi expressions for the transformation functions are

$$F_C = \left[\int_0^1 \left(\frac{\rho}{\rho_2}\right)^{0.5} d\left(\frac{V}{V_2}\right) \right]^{-2}$$

$$F_{Rx} = \left(\frac{T_2}{T_w}\right)^{.702} \left(\frac{T_r}{T_w}\right)^{.772} / F_C$$

For a perfect gas, the ratios ρ/ρ_2 and V/V_2 may be expressed in compatible terms and the integral solved for an explicit definition of F_C (see References 7 and 8). For a real gas, Pearce (Reference 9) recommends substitution of enthalpy for temperature in the F_{Rx} equation,

$$F_{Rx} = \left(\frac{H_2}{H_w}\right)^{.702} \left(\frac{H_r}{H_w}\right)^{.772} / F_C$$

and definition of enthalpy variation through the boundary layer based on a linear form of the Crocco expression,

$$H = H_w + (H_r - H_w) \times (V/V_2) - (H_r - H_2) \times (V/V_2)^2$$

The density variation, $\rho(h,P)$, is obtained from real gas curves, and the integral in the F_C expression is evaluated by a five-point Gaussian quadrature. The resulting compressible, turbulent skin friction coefficient is used directly in the Stanton number equation to determine the local turbulent heat transfer coefficient.

REFERENCES - APPENDIX B

1. "Equations, Tables, and Charts for Compressible Flow", NACA Report 1135, 1953

2. "U.S. Standard Atmosphere, 1962", U.S. Government Printing Office, Washington, D.C., December 1962

3. Hansen, C.F., "Approximations for the Thermodynamic and Transport Properties of High Temperature Air", NASA TR R-50, 1959

4. Keenan, J. H., and Kaye, J., "Gas Tables," Wiley & Sons, 1945

5. Eckert, E. R. G., "Survey of Boundary Layer Heat Transfer at High Velocities and High Temperature", WADC TR 59-624, 1960

6. Johnson, C. B., and Boney, L. R., "A Simple Integral Method for the Calculation of Real-Gas Turbulent Boundary Layers with Variable Edge Entropy", NASA TN D-6217, 1971

7. Spalding, D. B. and Chi, S. W., "The Drag of a Compressible Turbulent Boundary Boundary Layer on a Smooth Flat Plate With and Without Heat Transfer", J. Fluid, Mech., Vol. 18, pt 1, January 1964, pp 117-143

8. White, F. M., and Cristoph, G. H., "A Simple New Analysis of Compressible Turbulent Two-Dimensional Skin Friction Under Arbitrary Conditions", AFFDL-TR-70-133, 1971

9. Pearce, B. E., "Method of Spalding and Chi Modified for Real Gas Effects and Determination of Virtual Origin", Aerospace Corp., 1969