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MINIMUM ENERGY, LIQUID HYDROGEN SUPERSONIC CRUISE VEHICLE STUDY

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

OCTOBER 1975

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16. Abstract

A study was performed to examine the potential of hydrogen-fueled supersonic vehicles designed for cruise at Mach 2.7 and at Mach 2.2. The aerodynamic, weight, and propulsion characteristics of a previously established design of a LH_2 fueled, Mach 2.7 Supersonic Cruise Vehicle (SCV) were critically reviewed and updated. The design of a Mach 2.2 SCV was established on a corresponding basis.

These baseline designs were then studied to determine the potential of minimizing energy expenditure in performing their design mission, and to explore the effect of fuel price and noise restriction on their design and operating performance. The baseline designs of LH_2 fueled aircraft were then compared with equivalent designs of Jet A (conventional hydrocarbon) fueled SCV's.

Use of liquid hydrogen for fuel for the subject aircraft provides significant advantages in performance, cost, noise, pollution, sonic boom, and energy utilization.

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FOREWORD

This is the final report of a study of hydrogen-fueled supersonic cruise vehicles performed under contract NAS 2-8781 for NASA - Ames Research Center, Moffett Field, California. The report presents documentation of the substance of work performed during the period 21 April through 17 October 1975.

The study was performed within the Advanced Design Division of the Science and Technology Organization at Lockheed-California Company, Burbank, California. G. Daniel Brewer was study manager and Robert E. Morris was project engineer. Other principal investigators were

Samuel J. Smyth	design
E. L. Bragdon Roy L. Adamson	propulsion
Robert D. Elliott	aerodynamics
Jerry J. Rising	stability and control
Roger N. Jensen	weights
Randy S. Peyton	vehicle synthesis

Mr. Charles Castellano of the Advanced Vehicle Concepts Branch of NASA - Ames Research Center, was technical monitor for the work.

All computations in this analysis were performed in U.S. Customary units and then converted to S.I. units.

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LIST OF SYMBOLS

A	=	Fuselage Cross Sectional Area
A_C	=	Capture Area
APU	=	Auxiliary Power Unit
AR	=	Aspect Ratio
AST	=	Advanced Supersonic Technology
BL	=	Buttock Line
Btu	=	British Thermal Unit
C_D	=	Drag Coefficient
$C_{D_{CORR}}$	=	Correction Drag Coefficient
C_{D_F}	=	Friction Drag Coefficient
$C_{D_{HT}}$	=	Drag Coefficient - Horizontal Tail
C_{D_i}	=	Induced Drag Coefficient
$C_{D_{TRIM}}$	=	Trim Drag Coefficient
$C_{D_{VT}}$	=	Drag Coefficient - Vertical Tail
C_{D_W}	=	Zero Lift Wave Drag Coefficient
$C_{D_{WING}}$	=	Drag Coefficient - Wing
C_L	=	Lift Coefficient
C_P	=	Specific Heat
C_V	=	Nozzle Velocity Coefficient

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LIST OF SYMBOLS (Continued)

CO	=	Carbon Monoxide
CO_2	=	Carbon Dioxide
D	=	Drag
D_{COMP}	=	Compressor Diameter
DBT	=	Duct Burning Temperature
DHTF	=	Duct Heating Turbofan
D_{MAX}	=	Nacelle Diameter
D_{NOZ}	=	Nozzle Diameter
DOC	=	Direct Operating Cost
F_n, F_N	=	Net Thrust
FAA	=	Federal Aviation Authority
FAR	=	Federal Aviation Regulation
$FN_{L.O.}$	=	Net Thrust at Lift-off
FN_{SLS}	=	Uninstalled Thrust at Sea Level Static
FPR	=	Fan Pressure Ratio
F.S.	=	Fuselage Station
GH_2	=	Gaseous Hydrogen
h	=	Altitude
HP	=	High Pressure
IOC	=	Indirect Operating Cost
L_{ENG}	=	Engine Length
L_{INLET}	=	Inlet Length
LH_2	=	Liquid Hydrogen

LIST OF SYMBOLS (Continued)

LP	=	Low Pressure
L/D	=	Lift/Drag
M	=	Mach
M_y	=	Moment - Y axis
M.A.C.,c	=	Mean Aerodynamic Chord
MEW	=	Manufacturers Empty Weight
MLG	=	Main Landing Gear
n_z	=	Load Factor - Z axis
NLG	=	Nose Landing Gear
NO_X	=	Oxides of Nitrogen
OEW	=	Operating Empty Weight
q	=	Dynamic Pressure
R	=	Range
SCV	=	Supersonic Cruise Vehicle
SFC	=	Specific Fuel Consumption
SN	=	Cross Sectional Area
S_{REF}	=	Reference Wing Area
S_W	=	Wing Area
t/c	=	Wing Thickness/Chord Ratio
TIT	=	Turbine Inlet Temperature
T.O.	=	Takeoff
TOGW	=	Takeoff Gross Weight
T/W	=	Thrust/Weight

LIST OF SYMBOLS (Continued)

V_{app}	=	Landing Approach Speed
$\bar{V}$	=	Tail Volume Coefficient
V_{KEAS}	=	Equivalent Air Speed (Knots)
W	=	Weight
W_G, GW	=	Gross Weight
$\dot{w}_f$	=	Fuel Flow Rate
W/S	=	Wing Loading
ZFW	=	Zero Fuel Weight
α	=	Angle of Attack
α_{WPR}	=	Angle of Attack - Wing Peference Plane
Δ	=	Increment
δ_{LE}	=	Flap Deflection - Leading Edge
δ_{TE}	=	Flap Deflection - Trailing Edge
ρ	=	Density

MINIMUM ENERGY, LIQUID HYDROGEN-SUPERSONIC CRUISE VEHICLE STUDY

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SUMMARY

This study was a re-examination of the design and performance potential of hydrogen-fueled supersonic cruise vehicles. The original study was conducted by Lockheed-California Company for NASA - Ames Research Center in 1973, and was reported in NASA CR 114718 (Reference 1).

The work reported herein involved updating the design of the Mach 2.7 LH_2 fueled SCV from the previous study, establishing a new design of a Mach 2.2 LH_2 fueled SCV, and comparing these two aircraft with conventionally (Jet A) fueled SCV designs which had been developed to identical guidelines under the "Advanced Technology Applied to Supersonic Cruise Aircraft" program (Reference 2). In addition, the potential for minimizing the energy utilization of both designs of LH_2 fueled SCV's was explored, as was the sensitivity of their performance capability to variation of numerous design parameters.

The results of this study confirmed the findings of the original investigation that use of liquid hydrogen as fuel in supersonic cruise transport aircraft, compared with Jet A, leads to significant advantages in performance, size, weight, energy consumption, cost and noise. The advantages previously established for LH_2 fueled SCV's relative to lower sonic boom overpressure and drastic reduction of noxious exhaust products were not re-evaluated in the present study because there were no differences in the vehicle designs which would lead to changes in these conclusions.

The following data compare some of the characteristics of aircraft designed to carry 234 passengers plus cargo, 7778 km (4200 nmi) at the cruise speeds indicated. The aircraft were selected using minimum gross weight as the criterion.

			MACH 2.7				MACH 2.2			
			JET A		LH_2		JET A		LH_2	
Gross weight	kg	(lb)	345,720	(762,170)	179,130	(394,910)	305,320	(673,110)	170,970	(376,920)
Operating empty weight	kg	(lb)	143,980	(317,420)	111,240	(245,235)	131,890	(268,720)	102,630	(226,260)
Total fuel weight	kg	(lb)	179,510	(395,750)	45,670	(100,675)	161,200	(355,390)	46,110	(101,660)
Thrust per Engine	N	(lb)	386,470	(86,890)	234,940	(52,820)	374,250	(84,140)	237,640	(43,430)
Cost	$\$10^6$									
RDT&E			4345		3778		3297		3094	
Production Aircraft			61.5		45.5		51.8		42.0	
Noise	EPNdB									
Sideline			108.0		104.0		108.0		106.7	
Flyover			108.0		102.2		108.0		104.7	
Energy Utilization	$\frac{kJ}{seat\ km}$	$\frac{Btu}{seat\ nmi}$	3522	(6189)	2551	(4483)	3227	(5672)	2608	(4583)

The comparison of LH_2 with Jet A fueled SCV's on the basis of direct operating costs is a function of the relative price of the two fuels. For example, with the Mach 2.7 SCV designs, when Jet A fuel costs 10.6¢/liter (40¢/gal), airlines could afford to pay $1.57 more per GJ ($1.65 more per million Btu) for LH_2 and still achieve equal DOC. If the price of Jet A is only 7.9¢/liter (30¢/gal), the differential for equal DOC with LH_2 fuel is reduced to $1.30/GJ ($1.37/10^6 Btu).

It was found that only minor saving in energy consumption could be realized by changing the design basis of the LH_2 SCV's. For example, for the Mach 2.7 aircraft, using minimum fuel weight as the selection criterion instead of minimum gross weight resulted in a 2.6 percent reduction of energy but a 4 percent increase in airplane cost. Minimum DOC is a good compromise selection criterion.

The most significant benefit of all to be realized from use of liquid hydrogen as the fuel for an advanced design of SCV is relief from dependency on a petroleum-based product which, by the time the new aircraft might become operational, could well be on the way to becoming unavailable for use as an aircraft fuel.

1. INTRODUCTION

The original conceptual design study to formally explore the feasibility, practicability, and potential advantages and/or disadvantages of using liquid hydrogen (LH_2) as fuel for an advanced design of supersonic transport was performed by Lockheed-California Company for NASA-Ames Research Center under Contract NAS 2-7732. The final report of that work was released as NASA CR 114718, dated January 1974 (Reference 1). It was concluded that LH_2 offered significant advantages over conventional hydrocarbon (Jet A) as a fuel for vehicles of this category.

The present study was performed to further explore the potential of LH_2 fueled supersonic cruise vehicles (SCV's). First, the Mach 2.7 design resulting from the original study was updated to incorporate changes in aerodynamic, propulsion, and structural weight input reflecting a more recent assessment of a feasible technology basis; second, a Mach 2.2 LH_2 SCV was designed on an equivalent technology basis to have the same payload/range capability; and third, several versions of each of these baseline aircraft were explored to investigate what potential there might be for minimizing energy expenditure in performing the design mission, and to investigate their design sensitivity to various parameters. Vehicles were designed for each Mach number to the following criteria:

- minimum gross weight at FAR 36 noise level
  at FAR 36 minus 5 EPNdB
  at FAR 36 minus 10 EPNdB

- minimum fuel weight

- minimum direct operating cost at LH_2 fuel costs of \$2, \$4, and \$6/10^6 Btu.

The baseline LH_2 aircraft resulting from this work, i.e., the minimum gross weight versions designed to meet FAR 36 noise constraints, were compared with equivalent designs of Jet A fueled Mach 2.7 and 2.2 SCV's from Task IV-2, Cruise Speed Selection Study (Reference 2) of the continuing SCV Technology Assessment Studies, Contract NAS 1-11940, performed for NASA-Langley Research Center by Lockheed-California Company.

Since the subject work is a "follow-on" to an earlier study, and uses the basic LH_2 airplane design concept developed and described in Reference 1, only the revisions to the design and the results derived therefrom are reported in full in this report. The reader interested in the background leading to derivation of the original airplane design concept should refer to NASA CR 114718 (Reference 1).

2. TECHNICAL APPROACH

There were two fundamental objectives of this work. One was to provide a direct comparison between LH_2 fueled and conventionally (Jet A) fueled supersonic cruise vehicles designed for cruise speeds of Mach 2.7 and 2.2. The second objective was to explore the potential of the LH_2 aircraft for minimizing utilization of energy in performing their design missions.

Because of the desire to compare new designs of LH_2 aircraft with existing Jet A designs, it was necessary to establish equivalency of technology base and ground rules. Accordingly, the first step of the present study was to obtain data on preferred designs of Jet A fueled Mach 2.7 and 2.2 SCV's from the work of Reference 2. These data are reproduced in Appendix A. Included are general arrangement drawings plus selected pages of ASSET (Advanced System Synthesis and Evaluation Technique) computer printout of both the CL 1607-5 (Mach 2.7) and the CL 1607-13 (Mach 2.2) Jet A fueled aircraft. Examination of these designs and review of the ground rules which served as a basis for their evolution resulted in the following changes in Guidelines for the subject study, relative to those used in the original evaluation of LH_2 supersonic transport aircraft (Reference 1):

- increased use of composite materials (see Table 1).
- limit landing approach speed to a maximum of 81.3 m/s (158 kts) equivalent airspeed at an aircraft weight equal to the takeoff gross weight reduced by the block fuel consumption. This is in lieu of a maximum landing field length of 2,900 m (9,500 ft) used in the Reference 1 study.
- aircraft cruise performance calculated for standard day plus 8°C (59°F + 14.4°F).

For convenience, the complete list of updated Guidelines used in the present study is presented in Table 2.

Following establishment of a consistent set of guidelines which would permit valid comparison of the subject LH_2 fueled SCV's with the designated Jet A fueled vehicles, preliminary sizing studies were carried out to determine approximate weights and dimensions of the projected LH_2 aircraft. The results of this preliminary analysis served as a starting point for the more rigorous design cycle which would produce the final vehicle configurations.

Preliminary configuration drawings of both the CL 1701-9 (Mach 2.7) and the CL 1701-10 (Mach 2.2) LH_2 aircraft were made based on the results of the sizing studies. As described in Section 3, Technology Modifications, the following detailed studies were then performed to provide a basis for definition of the final configurations:

- assessment of wave drag coefficients at selected speeds, plus evaluation of possible benefits from area ruling.

Table 1. Use of Composite Materials in Advanced Design SCV's

Component	Original Study of LH_2 SCV (Reference 1)		Advanced Technology Cruise Speed Study (Reference 2)	
	% Composite	% Component Wt Red	% Composite	% Component Wt Red
Wing	6.2	-15.5	6.2	-15.5
Tail	0	0	40	-19.1
Fuselage	3.6	- 6.25	34.3	- 9.8
Landing Gear	0	0	12	- 7.3
Nacelle	0	0	40	-11.9
Air Induction	0	0	30	- 5.0
Surface Controls	1.5	- 3.75	10	-10.0
Total	3.4		16.3	

- evaluation of stability and control requirements of both aircraft to determine tail sizes.
- generation of turbofan engine cycle characteristics for both cruise speeds using a complete representation of hydrogen/air combustion products.
- examination of structural and insulation requirements of the hydrogen tankage system to provide a realistic basis for determining tank wall thickness and insulation thickness.
- assessment of the effect the use of greater percentages of advanced composite materials would have on vehicle structural weight.

The results of these analyses, plus data from the preliminary sizing study, provided input to the ASSET (Advanced System Synthesis Evaluation Technique) computer program for parametric study of vehicle design and performance. Using ASSET, the following parameters were investigated to determine minimum gross weight, minimum fuel weight, and minimum DOC

Table 2. Basic Guidelines

Fuel - liquid hydrogen (assumed available at the airport)

Planform - NASA Arrow - wing

Initial Operational Capability - 1990

Use of advanced materials and technology postulated to be developed by 1985. (Composites comprise 16.3 percent of the total vehicle structural weight; see Table 1).

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 32.2°C (90°F) day and 304.8 m (1000 ft) airport altitude.

Approach Speed - 81.3 m/s (158 knots) equivalent airspeed.

Operability - compatible with Air Traffic Control Systems and general operating environment envisioned for 1990, including capability for Category III-A operations.

Aircraft Design Life - 50,000 flying hours.

Sonic Boom - no boom at ground level over populated areas.

Stability - control configured aircraft.

Direct Operating Cost:

- Modified 1967 ATA equations (international basis).
- 1973 dollars
- 600 aircraft production base
- Baseline fuel costs
  LH_2 = \$2.85/GJ (\$3/10^6 Btu = 15.48¢/lb)
  Jet A = \$1.90/GJ (\$2/10^6 Btu - 24.8¢/gal = 3.68¢/lb)

Payload - 22,226 kg (49,000 lb) = 234 passengers plus cargo allowance.

aircraft that satisfied the design mission requirements within the constraints imposed by airport performance and takeoff noise limitations.

- Maximum engine duct burning temperature (Max. DBT)
- Takeoff engine duct burning temperature (T.O.DBT)
- Noise abatement procedures such as power cutback
- Thrust/Weight Ratio (T/W)
- Wing Loading (W/S)

The following process was used to determine the optimum combination of design parameters for the subject aircraft: (See Section 4 for detail explanation)

1. For a specified Max DBT, aircraft designs were synthesized which satisfied the design mission requirements for a matrix of T/W and W/S combinations.

2. From the matrix of aircraft synthesized in Step 1, those aircraft which met the landing approach speed constraint were selected.

3. Using the aircraft selected in Step 2, the minimum T/W and T.O. DBT which satisfied the takeoff sideline and flyover noise limitations were determined.

4. The T/W and W/S combination which satisfied the landing approach speed, the sideline noise, and the flyover noise constraints, and which resulted in a minimum gross weight, minimum fuel weight, and/or minimum DOC aircraft, respectively, was identified. This was the optimum T/W and W/S combination corresponding to the Max DBT assumed in Step 1.

5. Using the T/W and W/S combination from Step 4, aircraft were synthesized which met the design mission requirements for a series of Max DBT's. The Max DBT that results in a minimum gross weight, minimum fuel weight, and/or minimum DOC aircraft was thus determined.

6. Using the Max DBT determined in Step 5, Steps 1 through 4 were repeated to optimize the T/W and W/S combination for the selected Max DBT.

The above steps to determine the optimum choice of maximum DBT were necessary because this parameter is so significant. It directly affects the engine physical size, thrust-to-weight ratio, engine cost, and the mission fuel consumption.

The effect of reduced noise levels on the selection criterion of minimum gross weight was examined with specific objectives of FAR 36, FAR 36 minus 5 EPNdB, and FAR 36 minus 10 EPNdB. Noise reduction was accomplished by throttling the engine (reducing duct temperature and exit velocity) and increasing the engine size (airflow) as required within practical limits which still permitted meeting the other mission constraints. No reoptimization of the engine cycle parameters, e.g., fan pressure ratio, selected for the basic aircraft (FAR 36 noise level), was made.

Aircraft point designs meeting FAR 36 and selected on the basis of minimum DOC were also defined for LH_2 fuel prices of $1.90, $3.80, and $5.70 per GJ ($2, $4, and $6 per million Btu's).

The minimum gross weight aircraft at both cruise Mach numbers which were designed to meet FAR Part 36 noise specification were used to compare with equivalent Jet A fueled reference aircraft. Those same aircraft were also used as a basis for establishing sensitivity of the design to variation of a number of parameters. The sensitivity of gross weight, DOC, price, and total fuel weight to the following parameters was determined:

- Design range
- Changes in empty weight before and after design freeze
- Noise constraints at FAR 36 minus 5 EPNdB and FAR 36 minus 10 EPNdB.

In addition, the minimum gross weight aircraft meeting FAR 36 were examined with regard to:

- DOC vs fuel cost
- Range and DOC vs change in SFC.
- Range and DOC vs change in drag count
- Range vs payload weight

3. TECHNOLOGY MODIFICATIONS

3.1 Aerodynamics

From the point-of-view of aerodynamics, redesign of the point-design configuration of the previous study (Reference 1) required an updating and refining of the aerodynamic data base. The basis for changing the aerodynamic data was the result of experience gained from the NASA-Langley Supersonic Cruise Vehicle System Study (Reference 2), and the Arrow-wing Structure Study (Reference 3). Continuing wind-tunnel tests at Langley, primarily low speed, also supplied additional information for updating the aerodynamic data base.

3.1.1 High Speed Characteristics

3.1.1.1 Wave Drag.—The wave drag of the parametric configuration elements used in the ASSET computer program to derive the point design aircraft configuration for the previous study (Reference 1) was calculated using the NASA-Langley wave drag program P7120. This program had the limitation of accepting only circular, uncambered fuselages and uncambered symmetrical wing airfoils. A newer, more sophisticated program for calculating wave drag (D2500) was also available from NASA-Langley at the time. It had the capability of handling non-circular cambered fuselages as well as twisted and cambered wings. However, due to its requirement for more accurate definition of the aircraft configuration, the D2500 program used approximately three times the computer time per case.

Test cases were run on an available design of Jet A fueled supersonic transport configuration using both the NASA wave drag programs to obtain a comparison. It was found the wave drag values were within 0.5 percent of each other. A penalty of two drag counts (0.0002) was arbitrarily added to all wave drag values calculated by the simpler program (P7120), and that program was then used throughout the previous study (Reference 1).

In the present study, the newer version of the wave drag program (D2500) was used exclusively. In addition to the increased accuracy which results from its use, the ability to treat non-circular fuselages enables the designer to define the fuselage cross-section profile in much more detail. The following paragraphs describe the procedure followed and present the data generated to define wave drag for the Mach 2.7 and 2.2 LH_2 fueled SCV designs.

3.1.1.2 CL 1701-9 Mach 2.7 LH_2 Design.—The Mach 2.7 SCV baseline wing was scaled from 976m^2 (10,500 ft^2) to the 678m^2 (7,300 ft^2) required for the LH_2 study in the form of a data set accessible from CADAM(TM) (Computer-graphics Augmented Design And Manufacturing). Digital Data on fuselage cross-sectional areas and centroids, horizontal and vertical tails, and engine nacelles were also obtained from CADAM.

Using a circular fuselage simulation, the above supplied data resulted in an assessment of $C_{D_W} = 0.00297$ at the design Mach number. Furthermore, the program predicted that with maximum fuselage area ruling, while maintaining the same maximum cross-sectional area and fuselage length, the wave drag could be reduced to $C_{D_W} = 0.00246$. This information was supplied in the form of a plot comparing the un-area ruled and full area-ruled fuselage cross-sectional areas versus length, Figure 1.

Unfortunately, as shown in the figure, the area-ruled option involved reduction of fuel tankage volume and excessive slimming of both the fuselage forebody and afterbody. Accordingly, after detailed consideration of the physical arrangement of the design it was determined that it was not possible

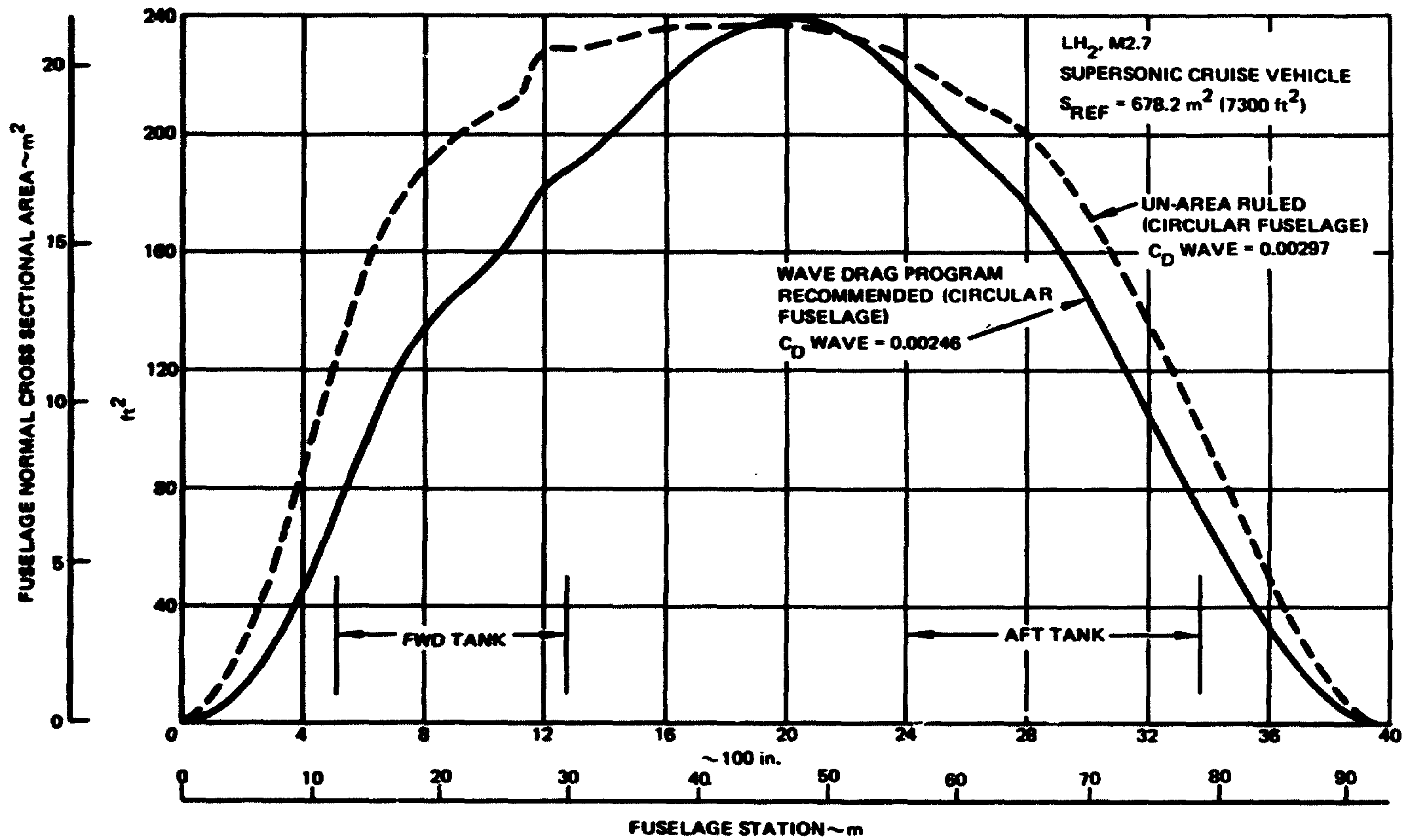

Figure 1. Effect of Fuselage Area Ruling on Wave Drag - M2.7 LH_2 Design.

to take advantage of fuselage area ruling within practical limitations of fuselage length and maximum diameter. Therefore the un-area ruled fuselage was taken as the M 2.7 LH_2 baseline, and 50th scale fuselage section drawings were generated using CADAM (see APPENDIX B). These drawings were digitized to produce a noncircular fuselage wave drag simulation shown isometrically as Figure 2.

Estimated total aircraft wave drag (Figure 3) and wave drag breakdown by component (Figure 4), along with their associated reference areas (Table 3) and wetted surface areas were supplied as input to the ASSET Program. Figure 5 is included to show the buildup of normal cross-sectional areas for the CL-1701-9 baseline.

Table 3. Component Reference Areas (M 2.7 LH_2 Design)

Wing:	Reference = 678m^2 (7,300 ft^2) Total Planform = 678m^2 (7,300 ft^2)
Fuselage:	Max. Cross-Sectional Area = 21.9m^2 (236 ft^2)
Nacelles:	Inlet Area = 7.41m^2 (79.80 ft^2) (4 Nacelles) Max. Cross-Sectional Area = 12.41m^2 (133.69 ft^2) (4 Nac.) Exhaust Area = 12.41m^2 (133.69 ft^2) (4 Nac.)
Vertical Wing Fins:	Area = 170m^2 (182.9 ft^2)/side
Vertical Fus. Fin:	Area = 13.41m^2 (144.3 ft^2)
Horizontal Stab:	Area (Inc. Carry Thru to BL 0) = 31.2m^2 (335.5 ft^2) Area (Exposed) = 20.8m^2 (224.0 ft^2)

A study of the effect of perturbations of fuselage length on wave drag was undertaken to develop sensitivity factors for the ASSET Program. Twenty-foot barrel sections were added to and removed from the mid-fuselage. Mach 2.7 and 1.2 were investigated for the Mach 2.7 aircraft design only. The results, shown in Figure 6, were applied to both the Mach 2.7 and the Mach 2.2 vehicles.

3.1.1.3 CL-1701-10 Mach 2.2 LH_2 Design.- The Mach 2.2 SCV baseline wing was scaled from 835 m^2 (9,000 ft^2) to the 535 m^2 (5,760 ft^2) required for the LH_2 study in the form of a data set from CADAM. Digital data on fuselage cross-sectional areas and centroids, horizontal and vertical tails, and engine nacelles were also obtained from CADAM.

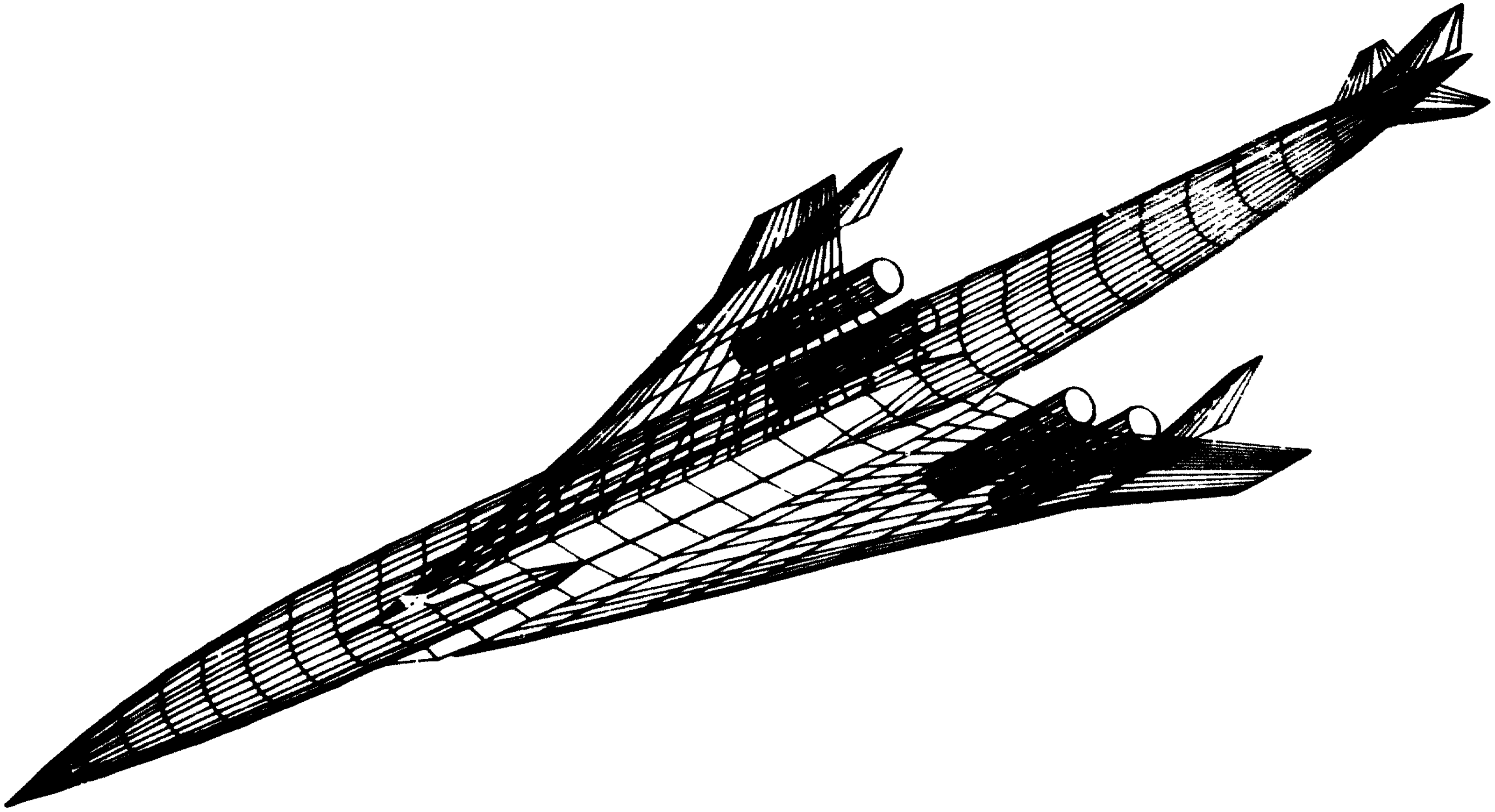

Figure 2. Isometric With Digitized Fuselage - M2.7 LH_2 Design

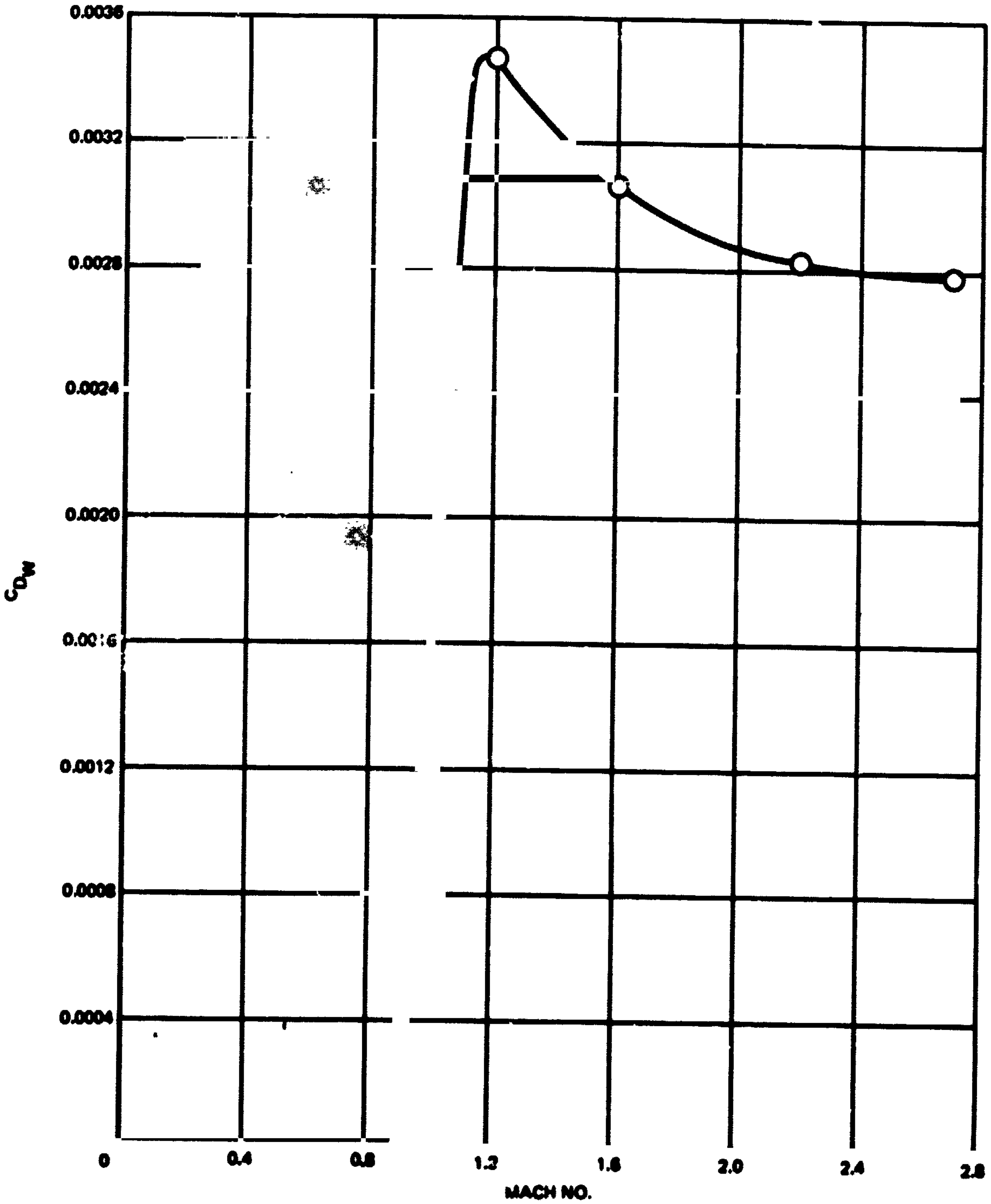

Figure 3. Total Wave Drag · M2.7 LH_2 Design.

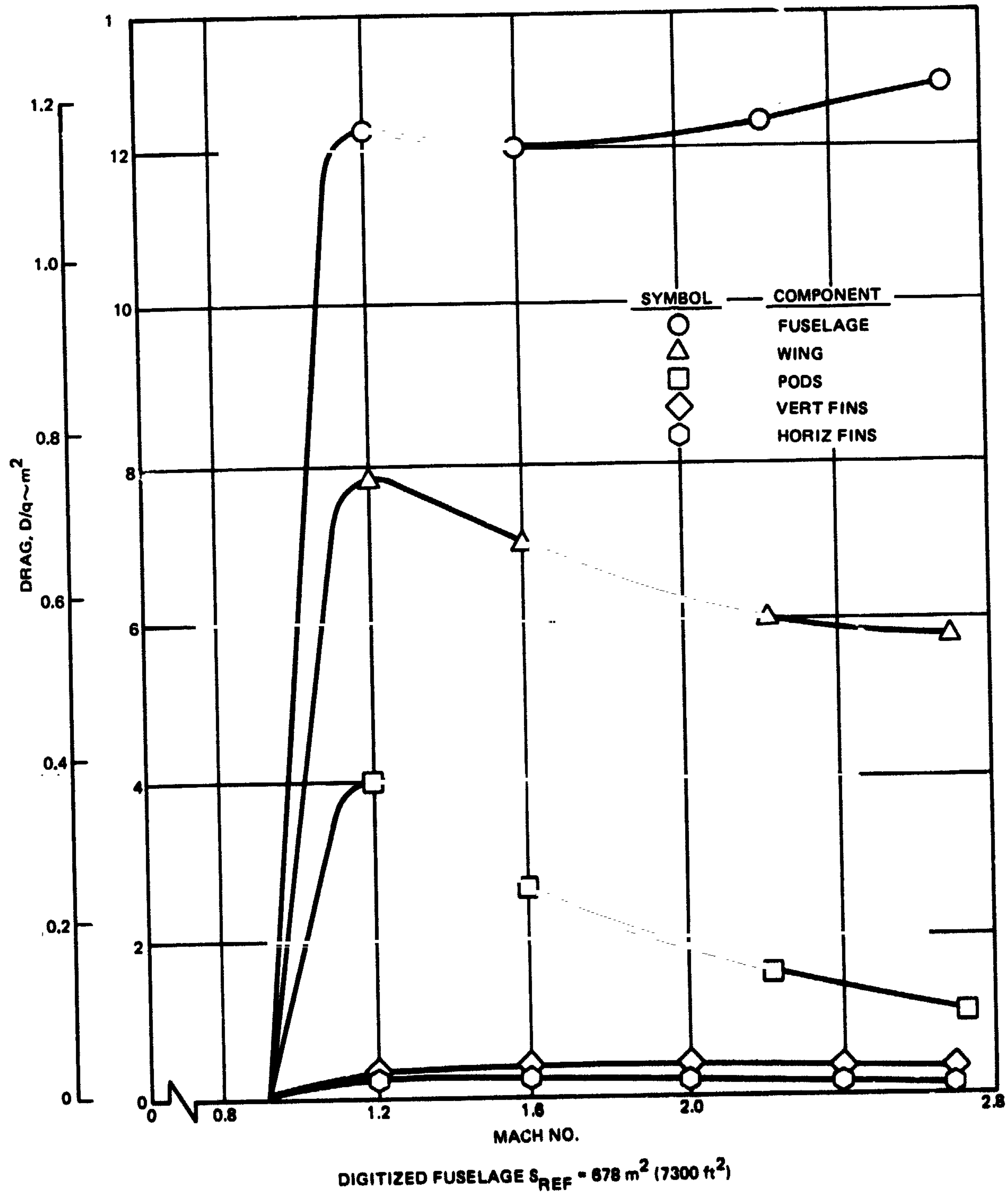

Figure 4. Component Wave Drag - M2.7 LH_2 Design.

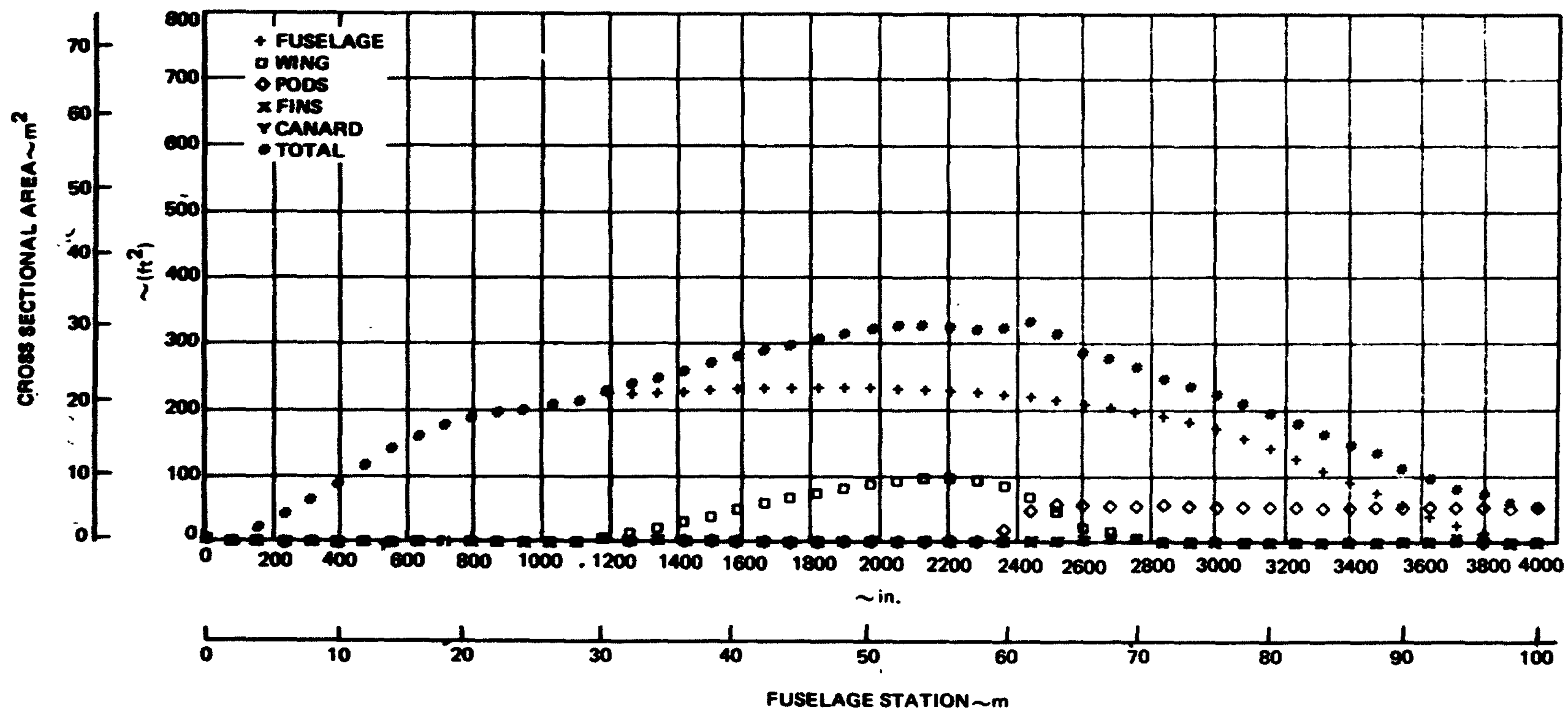

Figure 5. Cross-sectional Area Distribution - M2.7 LH_2 Design.

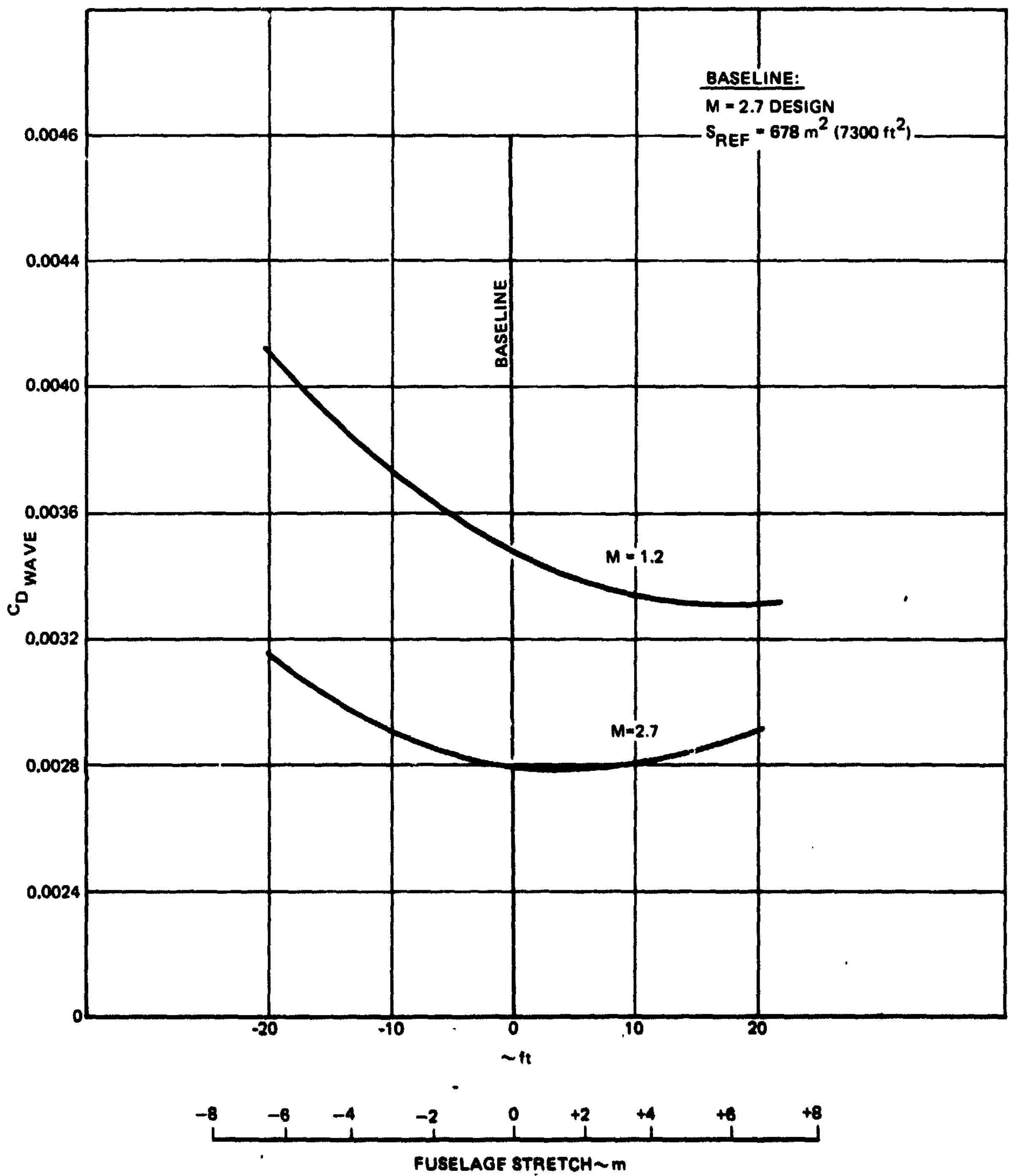

Figure 6. Effect of Fuselage Length on Wave Drag - M2.7 LH_2 Design.

Using a circular fuselage simulation, the above supplied data resulted in an assessment of C_{D_W} = 0.00355 at the design Mach number. Furthermore, the program predicted that with maximum fuselage area ruling the wave drag could be reduced to C_{D_W} = 0.00281 while maintaining the same maximum cross-sectional area and fuselage length. This information is shown in Figure 7 as a plot comparing the un-area ruled and full area-ruled fuselage cross-sectional areas versus length.

As in the case of the Mach 2.7 baseline, it was determined that area ruling was not feasible within fuel volume and overall fuselage length constraints. Therefore the un-area ruled fuselage was taken as the Mach 2.2 LH_2 baseline. Accordingly, 50th scale section drawings of those 15 fuselage sections different from the Mach 2.7 baseline were generated using CADAM (APPENDIX B). The balance of the 34 stations, fore and aft fuselage, are identical to the Mach 2.7 baseline. These sections were digitized to produce a non-circular fuselage wave drag simulation, shown isometrically in Figure 8.

Estimated total aircraft wave drag Figure 9, and wave drag breakdown by components, (Figure 10), along with their associated reference areas, (Table 4) and wetted surface areas, were supplied for use in the ASSET Program. Figure 11 shows the buildup of normal cross-sectional areas for the CL-1701-10 baseline aircraft.

Table 4. Component Reference Areas (Mach 2.2 LH_2 Design)

Wing:	Reference = 535m^2 (5,760 ft^2) Total Planform = 535m^2 (5,760 ft^2)
Fuselage:	Max Cross-Sectional Area = 22m^2 (236.8 ft^2)
Nacelles:	Inlet Area = 5.02m^2 (54.02 ft^2) (4 Nacelles) Max. Cross Sect. Area = 8.21m^2 (88.35 ft^2) (4 Nacelles) Exhaust Area = 8.21m^2 (88.35 ft^2) (4 Nacelles)
Vertical Wing Fins:	Area = 14.15m^2 (152.3 ft^2) (per side)
Vertical Fin-Fuselage:	Area = 18.16m^2 (195.5 ft^2)
Horizontal Stab:	Area (Incl. Carry Thru to BL 0) = 26.68m^2 (287.2 ft^2) Areas (Exposed) = 17.48m^2 (188.1 ft^2)

3.1.2 **Low Speed Aerodynamic Characteristics. — The low speed aerodynamic characteristics of the subject LH_2 aircraft are taken from the study of the equivalent Jet A aircraft (Reference 2). The data are presented in Figures 12 and 13 for Mach 2.7 design and Figures 14 and 15 for the Mach 2.2**

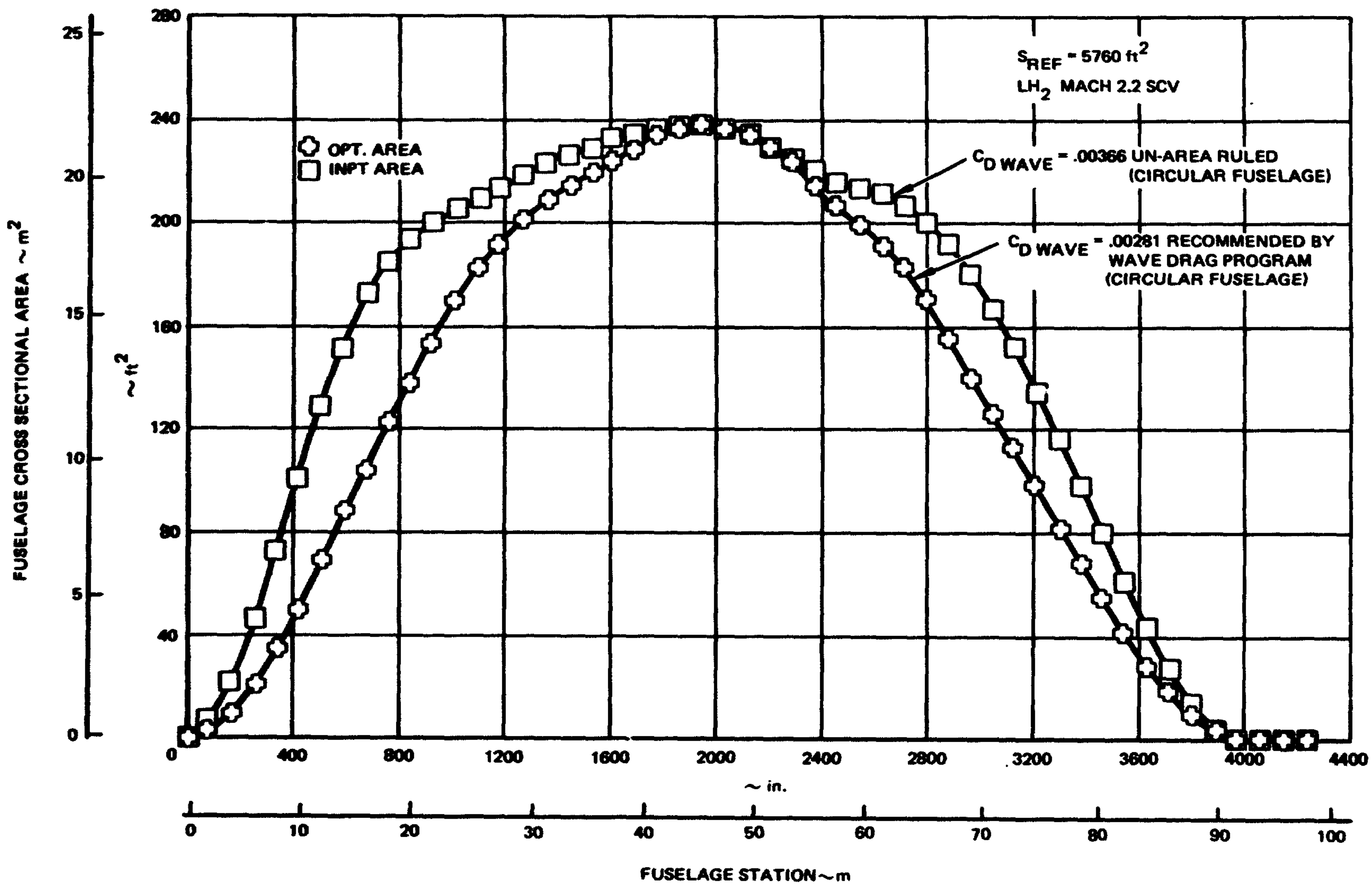

Figure 7. Effect of Fuselage Area Ruling on Wave Drag - M2.2 LH_2 Design.

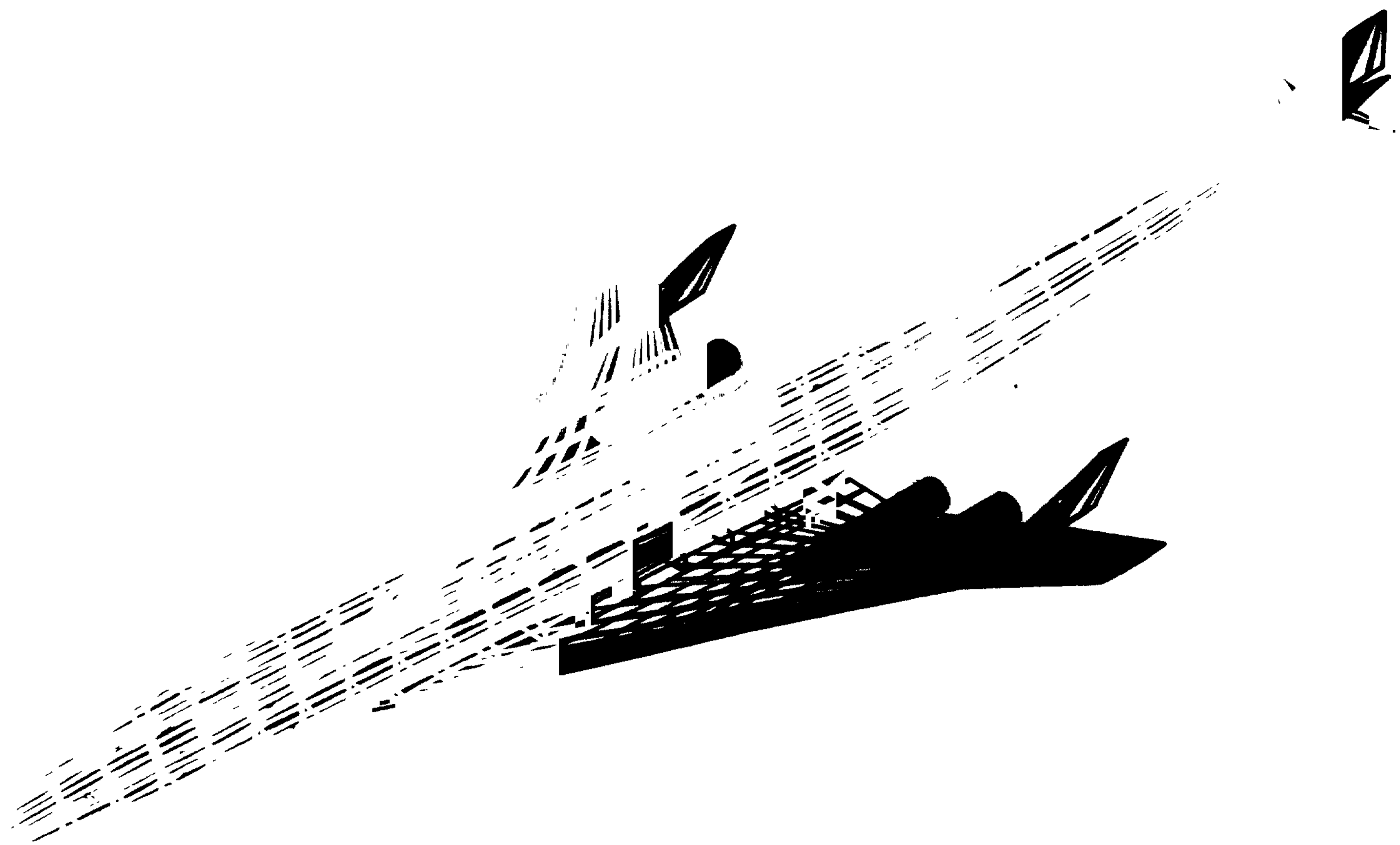

Figure 8. Isometric With Digitized Fuselage - M2.2 LH_2 Design

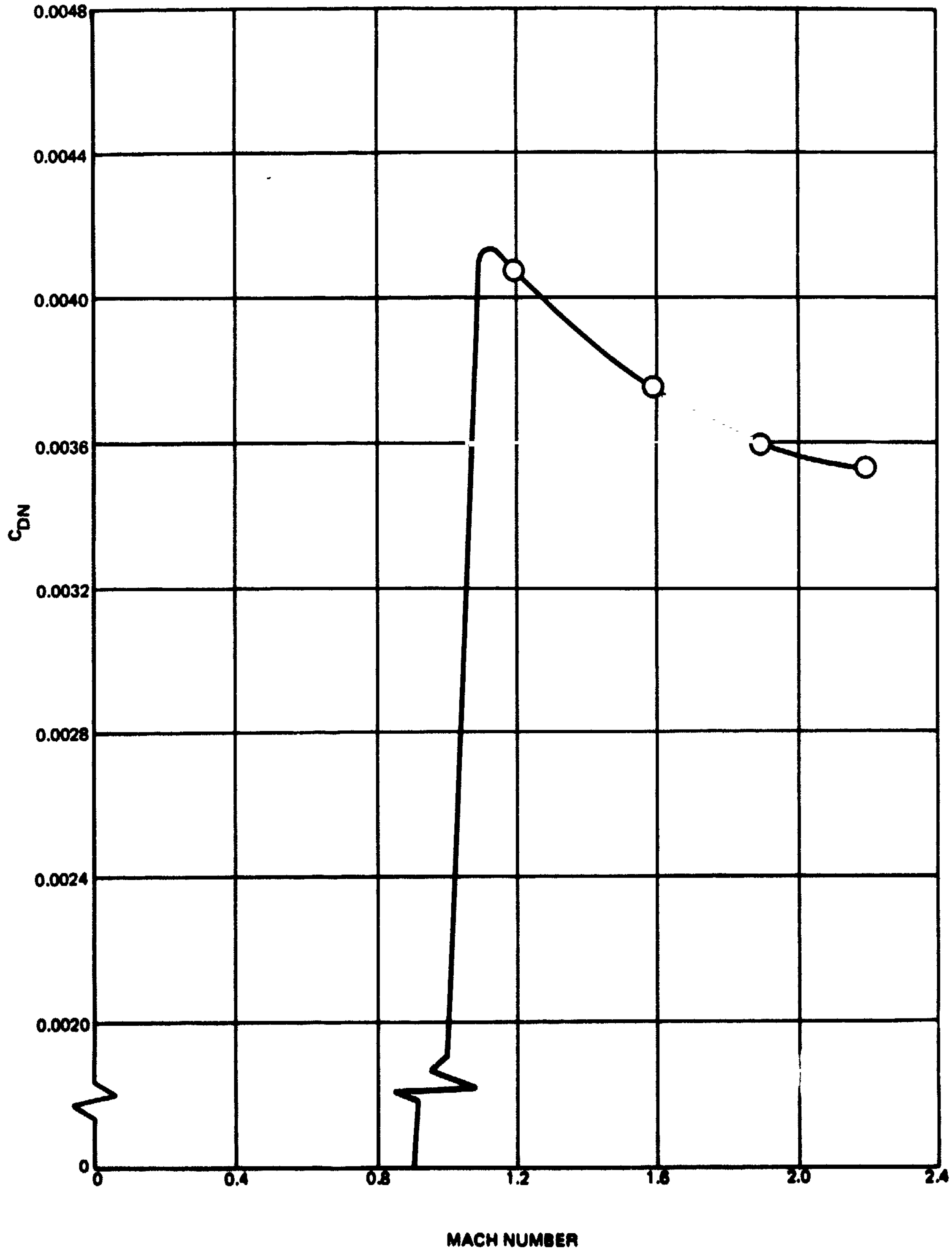

Figure 9. Total Wave Drag - M2.2 LH_2 Design.

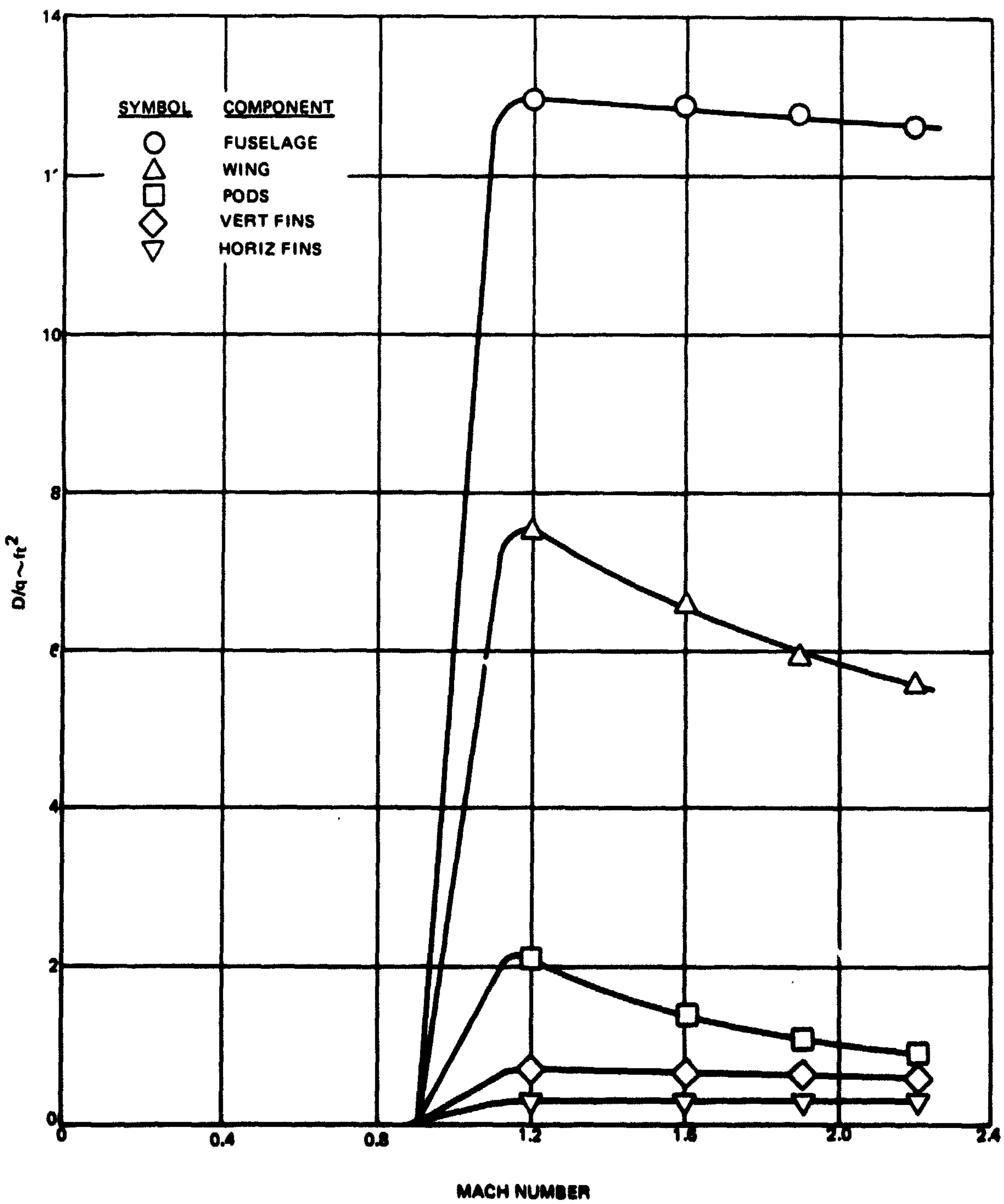

Figure 10. Component Wave Drag - M2.2 Design.

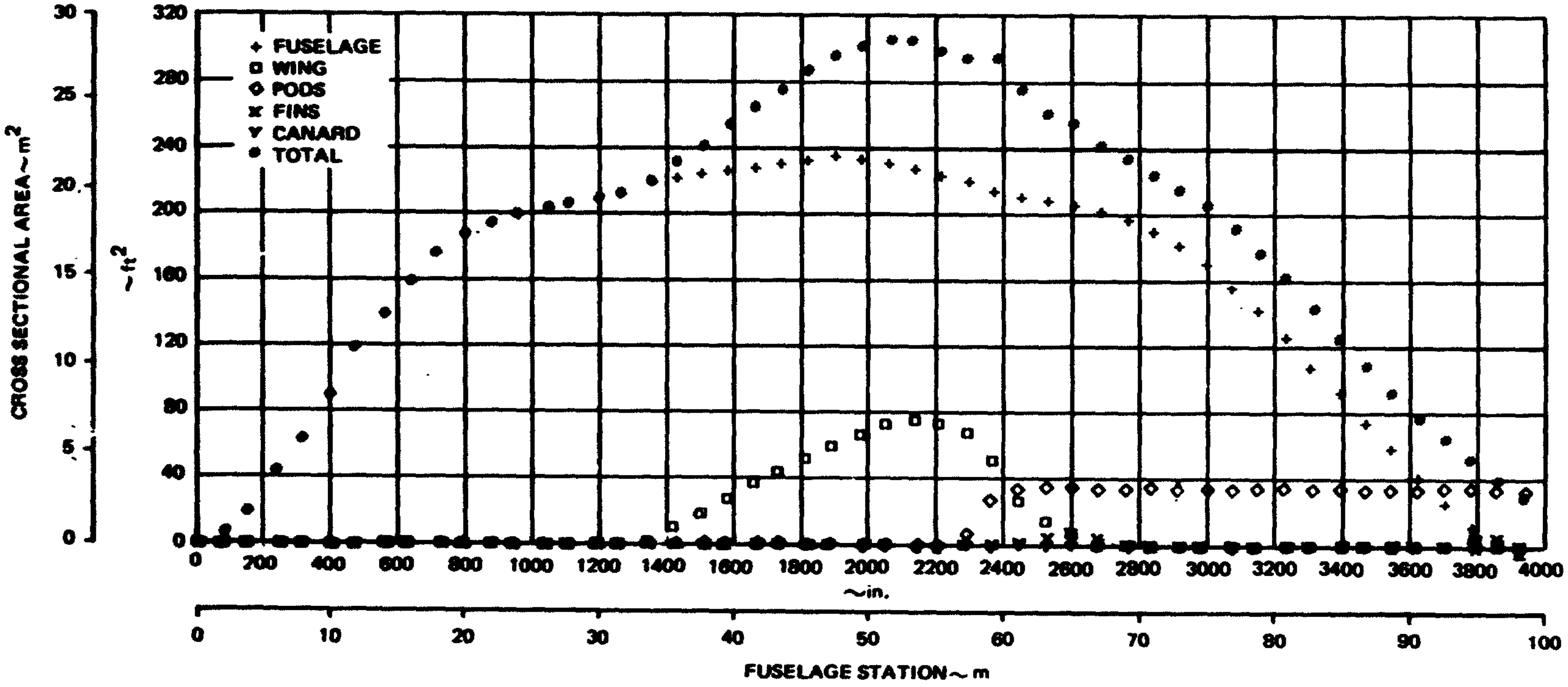

Figure 11. Cross Sectional Area Distribution - M2.2 LH_2 Design.

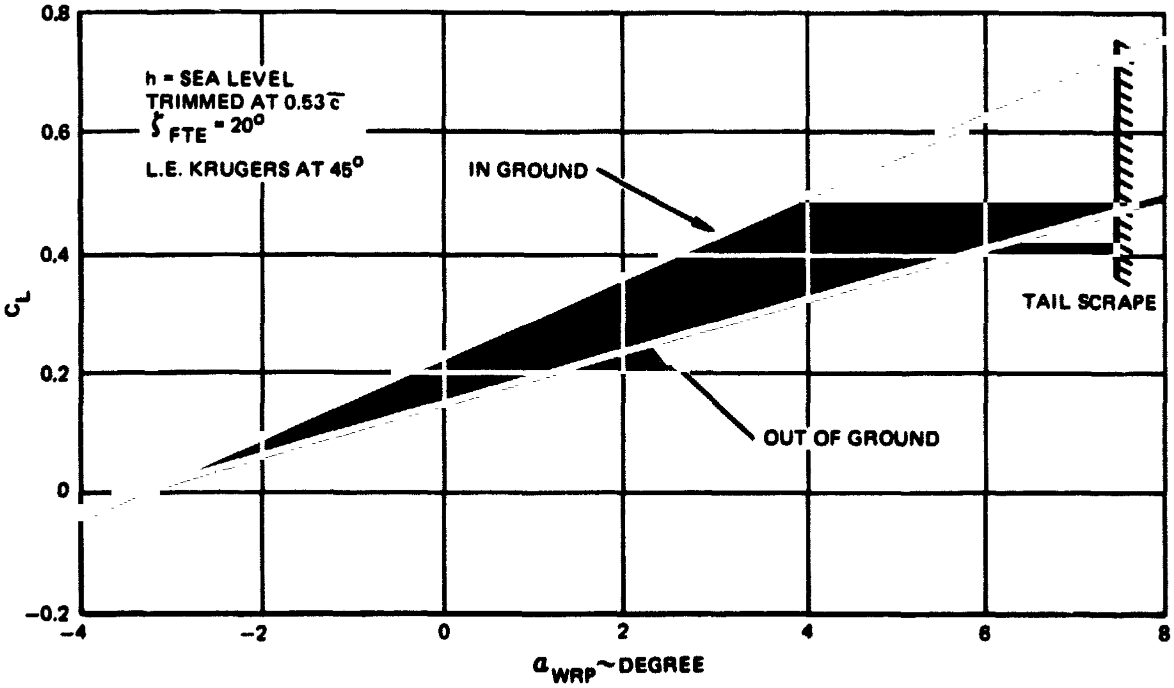

Figure 12. Low Speed Lift Characteristics - M2.7 LH_2 SCV.

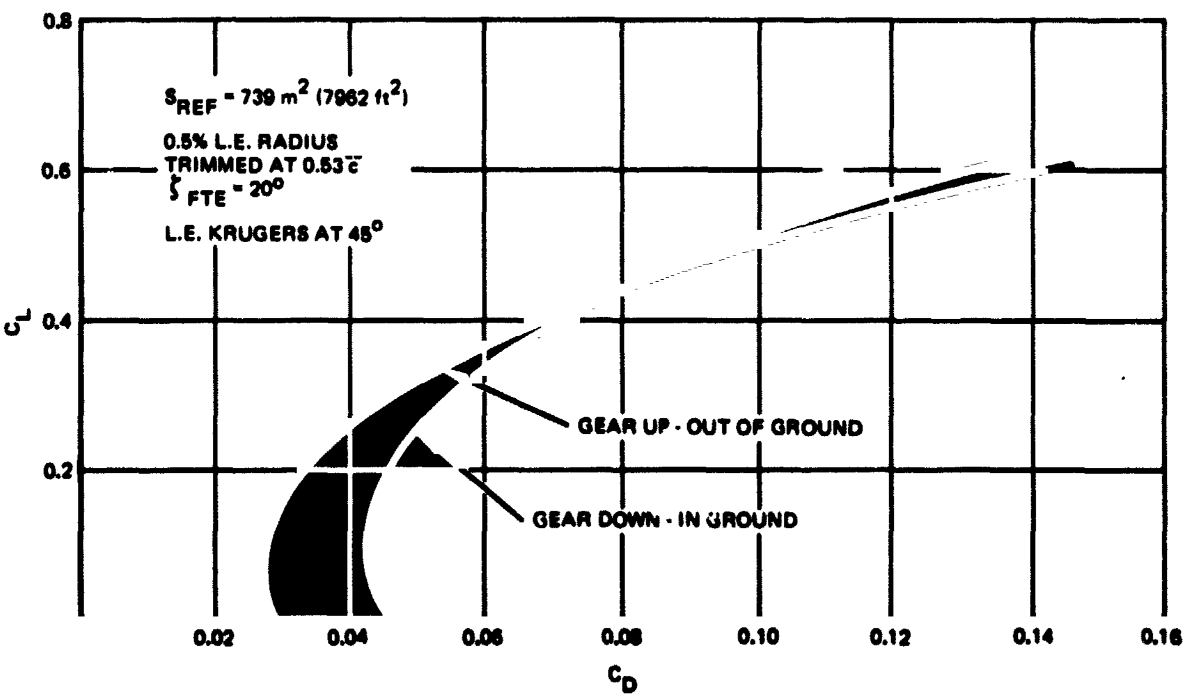

Figure 13. Low Speed Drag Polars - M2.7 LH_2 SCV.

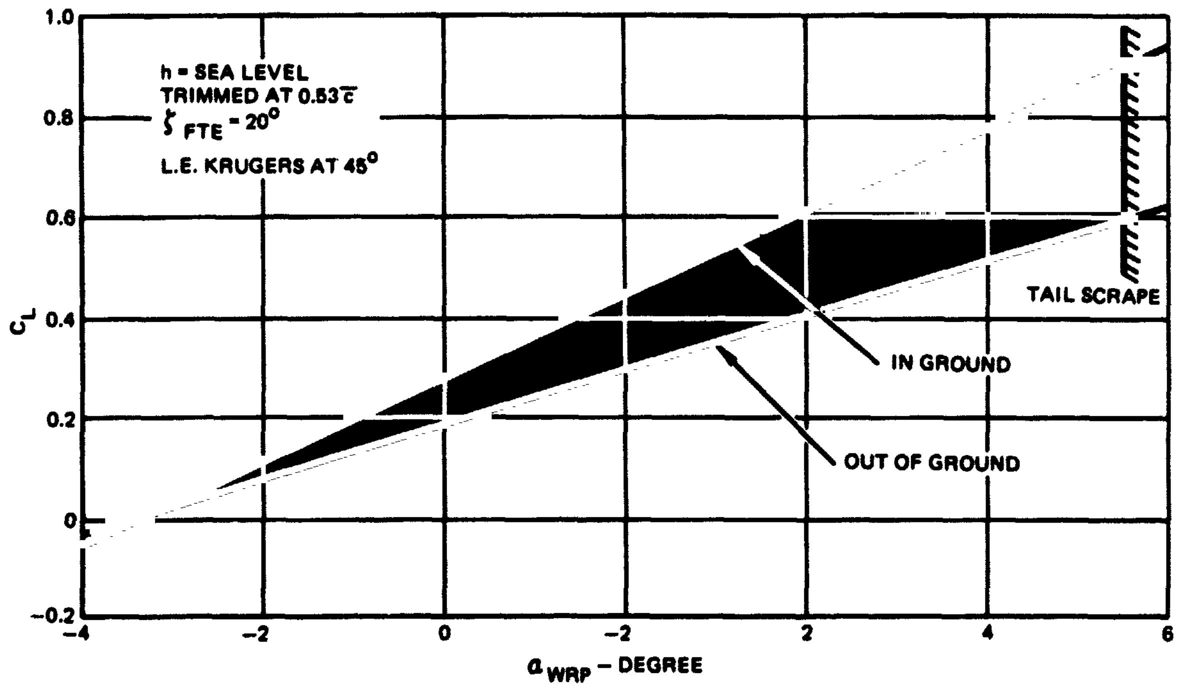

Figure 14. Low Speed Lift Characteristics - M2.2 LH_2 SCV.

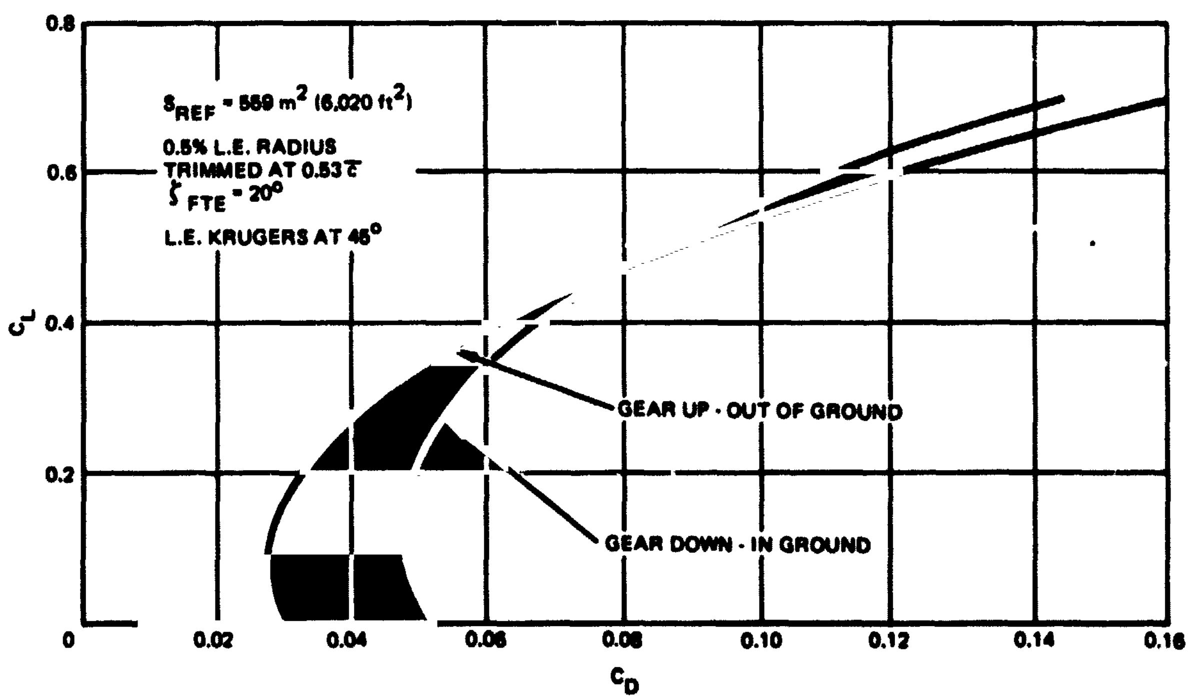

Figure 15. Low Speed Drag Polars - M2.2 LH_2 SCV.

design. Figures 12 and 14 show the low speed lift characteristics of the two aircraft, and Figures 13 and 15, the low speed drag polars. Both characteristics are presented for in-ground effect (gear down) and out-of-ground effect (gear up). The low speed lift characteristics of the LH_2 fueled aircraft are identical to those of their Jet A counterparts.

3.1.3 Stability Analysis. — The concept of relaxed static stability (RSS) has been used to size the horizontal and vertical tails for the study configurations. In the case of the horizontal tail, RSS allows movement of the center-of-gravity (c.g.) aft of the aerodynamic center in order to eliminate the supersonic trim drag penalty. The aft c.g. location is limited by the requirement to retain sufficient control power to ensure adequate handling qualities and also by the position which offers the most benefit in terms of drag reduction. The vertical tail is sized for a critical engine-out control condition, instead of being sized for supersonic directional stability. This allows a reduction in vertical tail area thereby somewhat reducing cruise drag. Both longitudinal and directional stability are provided by a stability augmentation system.

3.1.3.1 Horizontal Tail. — Detailed aerodynamic analysis of the M = 2.7 Jet A fueled configuration showed that minimum drag is achieved when the c.g. is located such that there is an upload on the horizontal tail. (See Reference 3). These data were obtained from NASA wind-tunnel tests of the SCAT-15F configuration modified to the full-scale vehicle. The data show that minimum drag occurs when the lift coefficient on the horizontal tail is approximately equal to the wing-body lift coefficient. Based on this relationship, the estimated optimum cruise c.g. location for the LH_2 configuration is $0.51\bar{c}$. This takes into consideration the relatively larger body diameter and forebody length of the LH_2 airplane which moves the aerodynamic center forward about $0.01\bar{c}$.

An airplane lacking inherent static stability and/or pitch-down tendency at the stall must be provided with active envelope limiting as a component of the longitudinal stability augmentation system. The margin of control power required to prevent a disastrous pitch excursion places an aft limit on c.g. position. The severity of the c.g. constraint depends on the size of the control power margin retained.

The analysis in this report is based on the premise that the aft c.g. control power requirement, over and above that required for trim, is highest for the approach and landing task since this is where pilot workload is high and also where the vehicle tends to be the most unstable. The magnitude of the control power required was determined from a statistical study of the stall recovery characteristics of three current jet aircraft. Stall time histories from C-5A, L-1011 and S-3A flight tests were examined to determine the stall recovery pitch acceleration commanded by the pilot for cases where less than full throw was used; this tends to define a recovery acceleration which feels comfortable to the pilot. The total pitch acceleration results from the combined effects of inherent stick-fixed pitch down tendency plus the incremental nose-down input commanded by the pilot to attain satisfactory progress of recovery. Pilots have found the longitudinal characteristics of stall recovery to be acceptable for all three airplanes. In reducing the data for the three configurations, a value of pitch acceleration was determined for each which represented the maximum in 90% of the cases analyzed. It was found that these values correlated as a function of pitch inertia with decreasing pitch acceleration for increasing pitch inertia. From this correlation a stall recovery of -0.08 rad/sec^2 was selected as being representative for the LH_2 configuration.

The horizontal tail is an all-moving surface with geared flap achieving a C_{Lmax} of +1.2; the tail lift effectiveness is based on results of tests in the Calac low-speed wind tunnel.

The forward c.g. limit was established by the nose-wheel lift-off requirement. Conditions for nose-wheel lift-off were determined in accordance with FAA tentative specifications for supersonic transports (Reference 4). This specification requires nose-wheel lift-off 3 seconds before rotation speed is reached. Calculations were based on a nose-wheel lift-off speed of 287 km/hr (155 kts) for the M = 2.7 design and 254 km/hr (137 kts) for the M = 2.2 design.

The aft c.g. limit was defined by the requirement to achieve a nose-down acceleration of 0.08 rad/sec^2 at landing approach V_{min} where V_{min} is defined by the speed margin required to pull 0.5 "g" at the minimum landing approach speed. This is the definition of V_{min} given in Reference 4. V_{min} was thus defined to be 231.5 km/hr (125 kts) for both the M = 2.7 and 2.2 designs.

The horizontal tail sizing summary is presented in Figure 16 for the M = 2.7 design (CL 1701-7). Figure 16 shows a volume coefficient requirement of 0.073 for the c.g. range of $0.480\bar{c}$ to $0.543\bar{c}$. Note that the landing gear should be located at least $0.054\bar{c}$ aft of the most aft c.g. to prevent tip-up at brake release with full thrust; the tip-up gear distance margin was based on a thrust-to-weight ratio for zero payload and full fuel. For the M = 2.2 design (CL 1701-10), Figure 17 shows a horizontal tail volume coefficient of 0.102 to be suitable for the c.g. range of $.485\bar{c}$ to $.546\bar{c}$. Because of the higher thrust-to-weight ratio for the M = 2.2 design, the landing gear should be located at least $0.063\bar{c}$ aft of the aft c.g.

3.1.3.2 Vertical Tail. The fuselage mounted vertical tail was sized in accordance with the landing approach minimum control speed specified for the Concorde. The requirement for the Concorde is to control loss of one outboard engine at minimum approach speed minus 9.26 km/hr (5 kts) with takeoff thrust on the remaining engines.

The vertical tail was assumed to be an all-moving surface with geared rudder achieving a C_{Lmax} of 0.9. 20% of the vertical tail C_L was reserved for possible dynamic over control and stability augmentation system requirements.

Vertical tail size requirements are presented in Figure 18. For the M = 2.7 design, the figure shows a volume coefficient requirement of 0.034 at 275 km/hr (148.5 kts). For the M = 2.2 design, a vertical tail volume

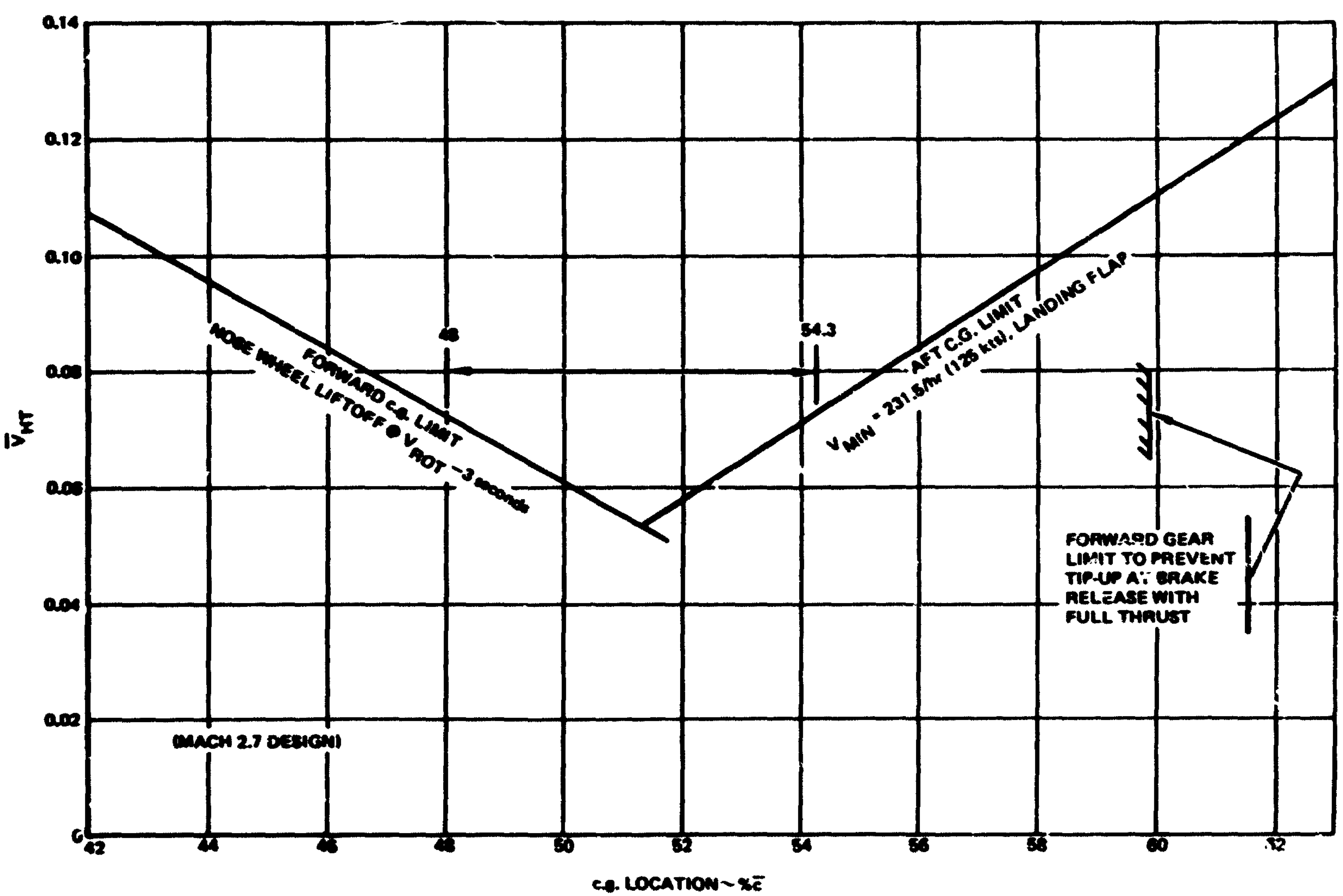

Figure 16. Horizontal Tail Size Requirements - M2.7 LH_2 SCV.

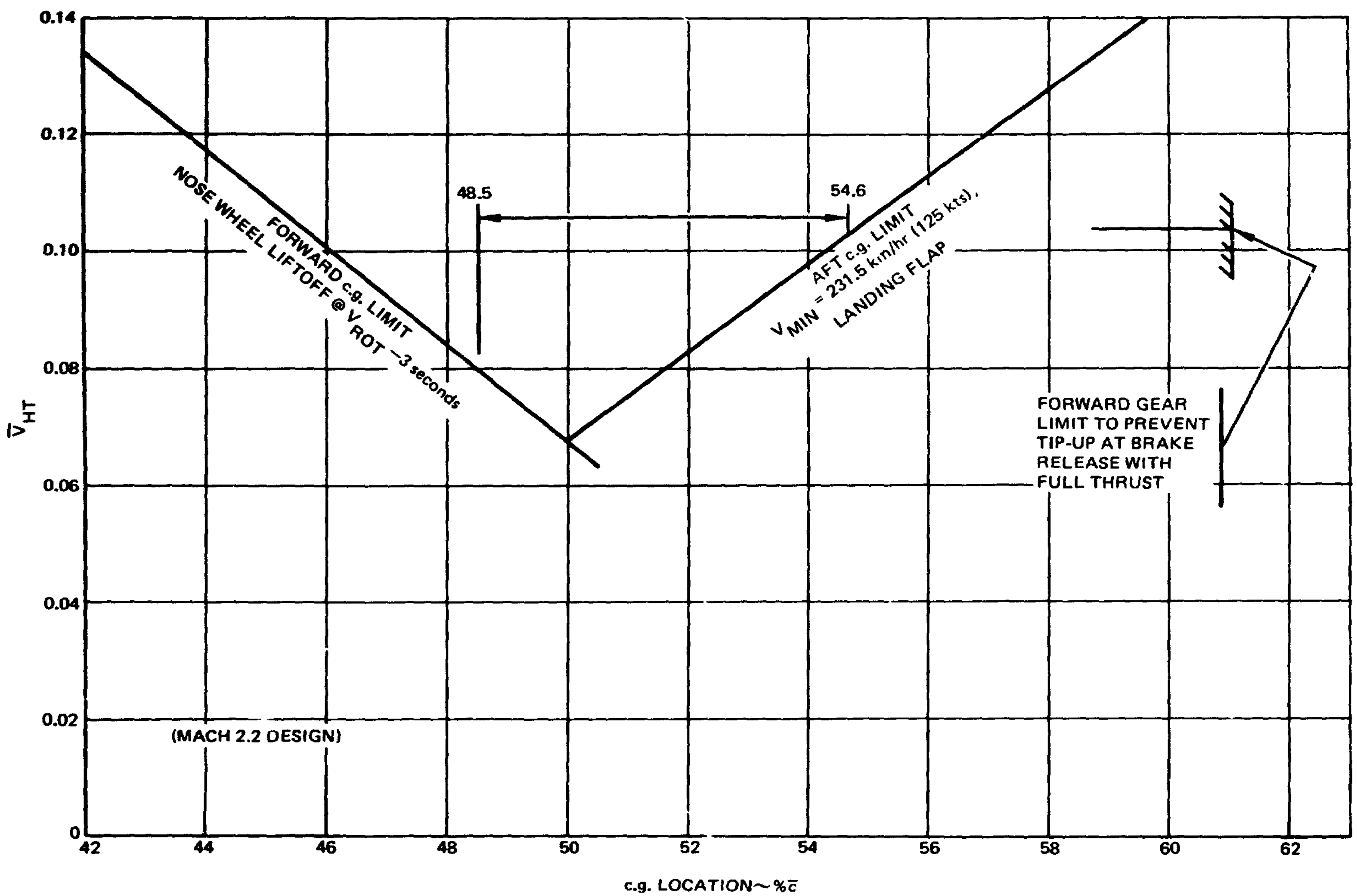

Figure 17. Horizontal Tail Size Requirements, M2.2 LH_2 SCV.

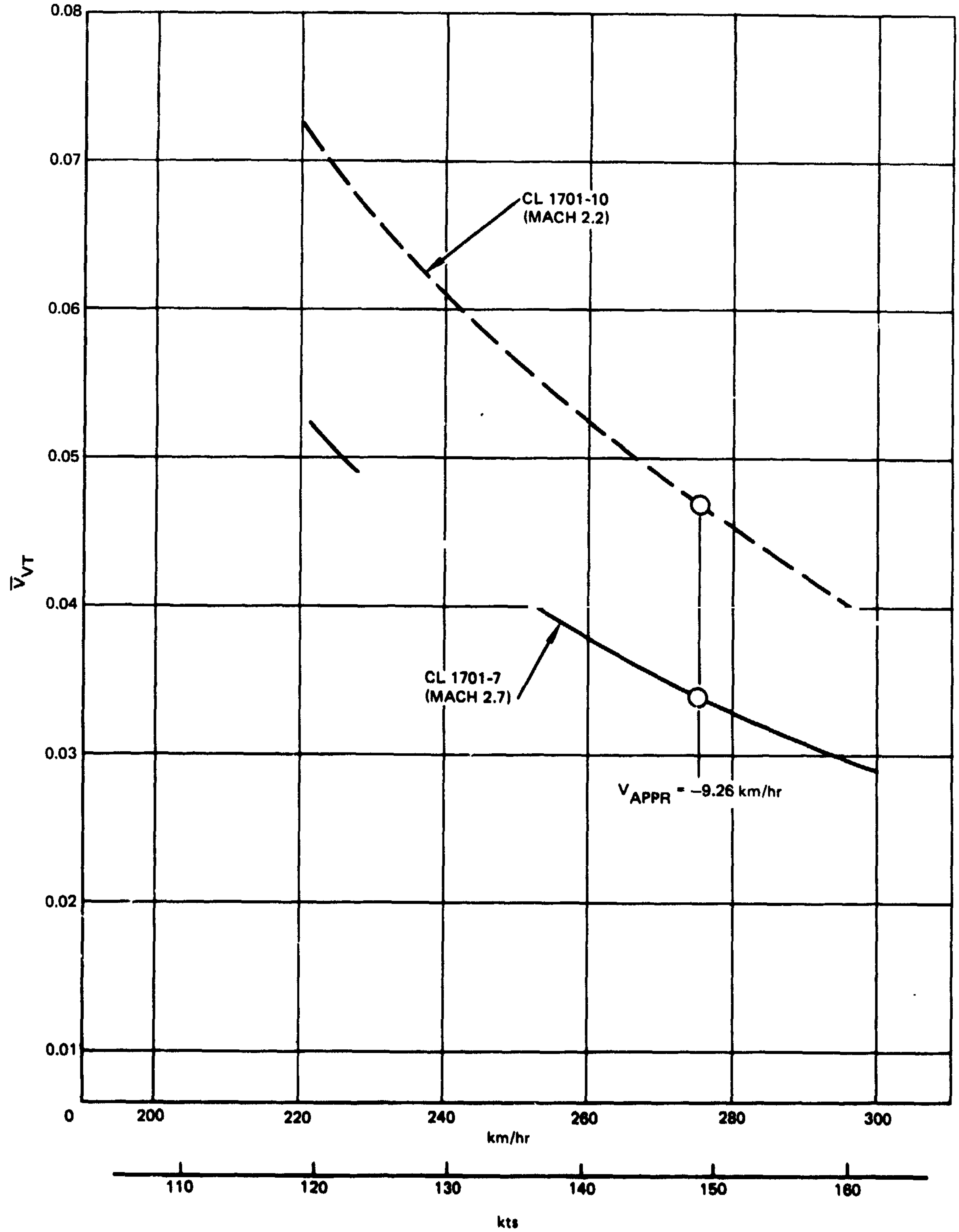

Figure 18. Vertical Tail Size Requirements - M2.7 and 2.2 LH_2 SCV.

coefficient 0.047 is required. The larger volume coefficient for the M = 2.2 design results from the thrust-to-weight ratio being somewhat higher and the wing reference area being smaller.

3.1.3.3 Induced Drag. - Tables 5 and 6 show the drag due-to-lift of the Mach 2.7 and Mach 2.2 aircraft. The Mach 2.7 data was derived from wind-tunnel data, while the Mach 2.2 was obtained from the 2.7 using a correction factor obtained from comparing the results of running both the Mach 2.7 and 2.2 in the Lockheed VORLAX program (Reference 5).

3.1.3.4 Miscellaneous Drag. - The trim drag used for the Mach 2.2 and 2.7 aircraft is presented in Figure 19 which assumes the vehicle's c.g. to be at 50 percent M.A.C. using the appropriate mission C_L for each Mach number.

Figure 20 shows the correction drag used to account for the difference between the predicted (analytical) and the actual high speed wind tunnel model test results.

3.2 Propulsion

The engine cycles examined in this study were duct heating turbofan engines (DHTF) with standard day cruise Mach Numbers of 2.7 and 2.2. All of the installation performance factors (i.e., ram recovery, inlet drags, nozzle coefficients, air bleed, and horsepower extraction) are identical to those utilized in the Reference 1 study.

3.2.1 Mach 2.7 Turbofan

3.2.1.1 Cycle Selection. - The cycle optimization studies completed for the LH_2-1 Mach 2.7 DHTF study of Reference 1 are applicable to this study and were therefore not repeated, i.e., the LH_2-2 Mach 2.7 DHTF engine data is rerun a of the same cycle. Table 7 lists the cycle parameters chosen for this engine. The engine performance for the previous study was computed with a fuel lower heating value applicable to hydrogen fuel, however, because a subroutine which describes the combustion products of hydrogen and air was not operational at the time at Lockheed, the properties of the exhaust gases were computed as if they were products of hydrocarbon and air based on Reference 6. Slight errors were, therefore, introduced in the turbine and nozzle calculations due to incorrect values of molecular weights and specific heats. Subsequent to that analysis Lockheed completed the development of a subprogram which computes the equilibrium gas properties of undisassociated products of combustion of hydrogen and air from the individual species present using thermodynamic values from Reference 6. The analysis of the LH_2-2 Mach 2.7 DHTF engine reported herein uses the new hydrogen air products subprogram and therefore the engine performance is more accurately defined.

3.2.1.2 Performance Characteristics. - No large difference was found during the present work, compared with the original analysis of the LH_2-1 Mach 2.7 DHTF engine; however, the installed performance characteristics of the reanalysis are slightly better over the entire engine operating envelope. Since the changes are all in the same direction (increased thrust and decreased SFC) the performance of the engine of Reference 1 was slightly conservative.

Table 5. Mach 2.7 LH_2 SCV Induced Drag

C_L	.00000 .31650	.05280 .42210	.10550	.15830	.21100	.26380
MACH NO						
0.23	.00053 .03112	.00000 .06024	.00158	.00485	.01066	.01962
0.40	.00053 .03017	.00000 .05834	.00127	.00475	.01023	.01867
0.60	.00053 .02891	.00000 .05634	.00105	.00454	.00981	.01772
0.80	.00032 .02775	.00000 .05486	.00105	.00422	.00939	.01720
0.90	.00053 .02680	.00000 .05317	.00112	.00411	.00907	.01656
0.93	.00053 .02616	.00000 .05169	.00115	.00411	.00897	.01625
0.95	.00053 .02564	.00000 .05106	.00118	.00411	.00886	.01593
0.98	.00053 .02543	.00000 .05085	.00121	.00411	.00886	.01582
1.00	.00053 .02543	.00001 .05106	.00124	.00411	.00886	.01582
1.05	.00052 .02574	.00003 .05212	.00131	.00433	.00897	.01593
1.10	.00050 .02646	.00005 .05376	.00140	.00443	.00917	.01641
1.20	.00048 .02870	.00010 .05760	.00158	.00485	.01013	.01804
1.40	.00040 .03471	.00021 .06583	.00200	.00591	.01245	.02247
1.60	.00032 .04030	.00032 .07427	.00253	.00696	.01477	.02648
1.80	.00021 .04536	.00042 .08261	.00295	.00802	.01699	.03007
2.00	.00011 .05043	.00053 .09263	.00338	.00918	.01920	.03344
2.20	.00000 .05549	.00063 .10044	.00380	.01034	.02152	.03714
2.30	.00000 .05824	.00071 .10687	.00411	.01097	.02268	.03882
2.55	.00000 .06464	.00088 .11937	.00468	.01268	.02580	.04332
2.70	.00000 .06815	.00096 .13114	.00506	.01361	.02754	.04610

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Table 6. Mach 2.2. LH_2 SCV Induced Drag

C_L	.00000 .31650	.05280 .36930	.10550 .42210	.15830	.21100	.26380
MACH NO						
0.23	.00043 .02540	.00000 .03676	.00129 .04916	.00396	.00869	.01601
0.40	.00041 .02362	.00000 .03420	.00099 .04568	.00372	.00801	.01462
0.60	.00040 .02197	.00000 .03199	.00080 .04282	.00345	.00746	.01347
0.80	.00040 .02093	.00000 .03056	.00080 .04138	.00318	.00708	.01297
0.90	.00040 .02031	.00000 .02983	.00086 .04030	.00312	.00688	.01256
0.93	.00040 .02011	.00000 .02951	.00089 .03972	.00316	.00689	.01248
0.95	.00041 .01987	.00000 .02912	.00091 .03959	.00319	.00687	.01235
0.98	.00042 .01996	.00000 .02941	.00094 .03993	.00323	.00696	.01243
1.00	.00042 .02013	.00001 .02965	.00097 .04043	.00326	.00702	.01253
1.05	.00042 .02078	.00002 .03092	.00104 .04208	.00349	.00724	.01286
1.10	.00042 .02176	.00004 .03216	.00114 .04421	.00364	.00754	.01350
1.20	.00041 .02439	.00008 .03560	.00134 .04896	.00413	.00861	.01533
1.40	.00037 .03110	.00019 .04405	.00181 .05899	.00529	.01116	.02014
1.60	.00029 .03748	.00029 .05230	.00235 .06907	.00648	.01374	.02463
1.80	.00020 .04258	.00040 .05901	.00277 .07753	.00753	.01594	.02822
2.00	.00010 .04792	.00050 .06667	.00321 .08802	.00872	.01825	.03178
2.20	.00000 .05355	.00061 .07391	.00367 .09692	.00998	.02077	.03584

Table 7.- Liquid Hydrogen Duct Heating Turbofan Cycle Characteristics (SLS Uninstalled)

Design Cruise Mach No.	2.2	2.7
Engine Type	DHTF	DHTF
Corrected Airflow $W\sqrt{\theta}/\delta$	400 kg/sec(880 lb/sec)	465 kg/sec(1026 lb/sec)
Fan Pressure Ratio	4.0	3.0
Compressor Pressure Ratio	6.25	8.33
Overall Pressure Ratio	25.0	25.0
Nozzle Velocity Coefficient (Duct)	0.981	0.981
Nozzle Velocity Coefficient (Primary)	0.981	0.981
Max Turbine Inlet Temperature	1922°K(3460°R)	1922°K(3460°R)
Max Duct Burning Temperature	1367°K(2460°R)	1367°K(2460°R)
Fuel Heating Value	119430kJ/kg(51590BTU/Lb)	119430kJ/kg(51590BTU/Lb)
Peak Fan Polytropic Efficiency	0.9	0.9
Peak Compressor Polytropic Eff.	0.915	0.915
HP Turbine Adiabatic Efficiency	0.92	0.92
LP Turbine Adiabatic Efficiency	0.91	0.91
Primary Burner Efficiency	1.0	1.0
Duct Burner Efficiency	*	*
Primary Burner Pressure Loss Ratio	0.060	0.060
Duct Burner Pressure Loss Ratio	*	*
Primary Nozzle Pressure Loss Ratio	0.005	0.005

* Variable based on Burner Temperature Rise

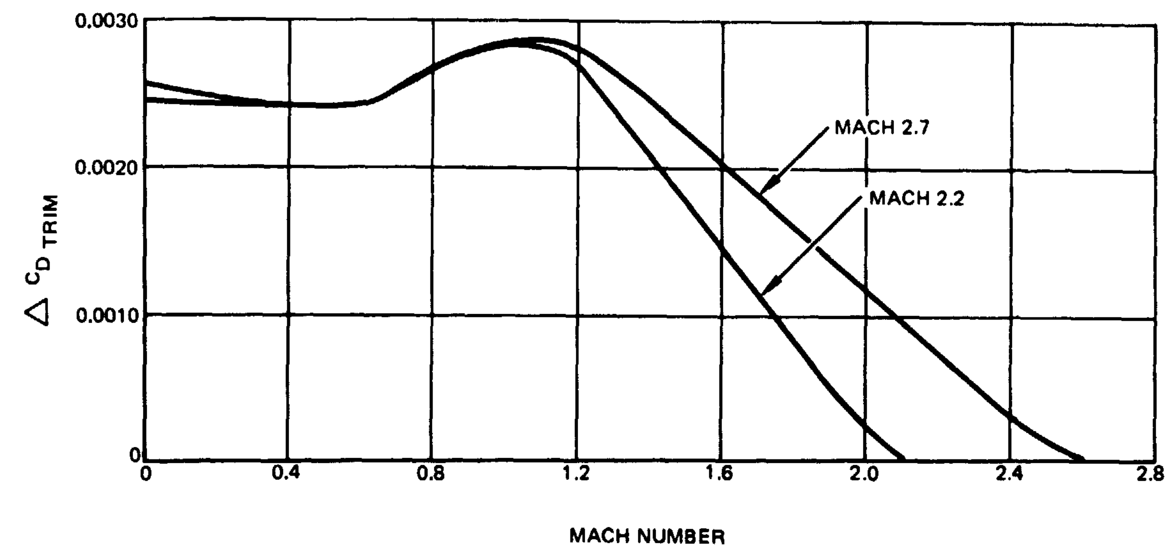


Figure 19. Configuration Trim Drag - LH_2 SCV.

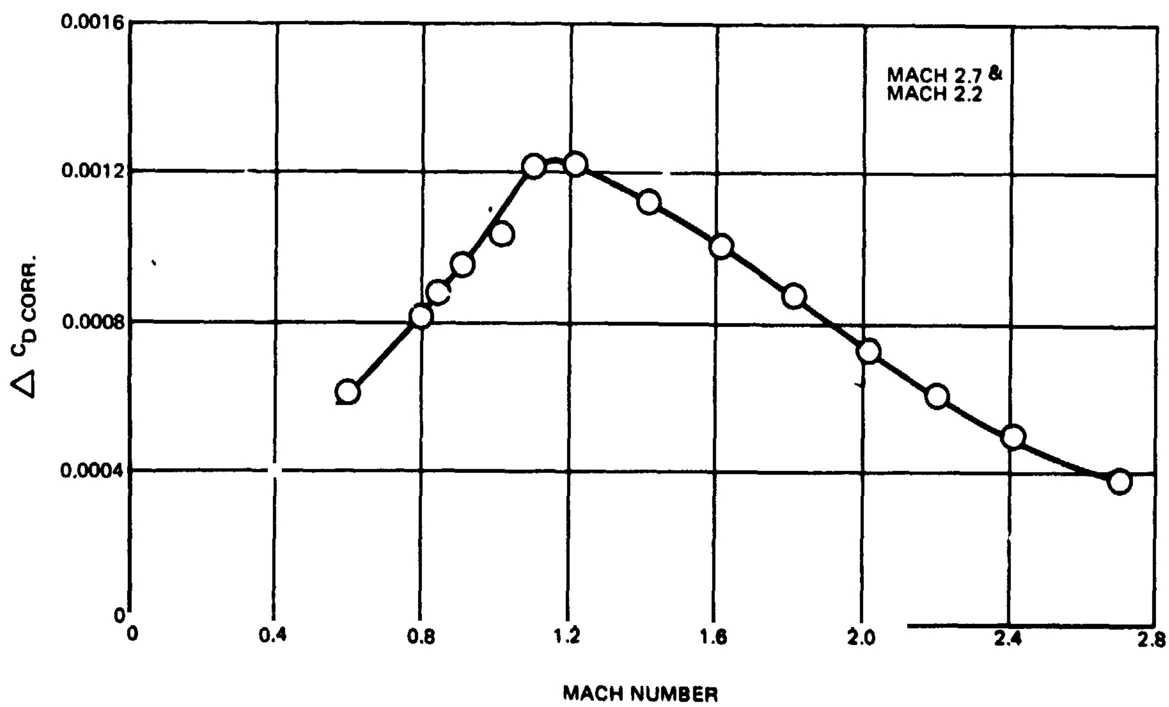


Figure 20. Correction Drag - LH_2 SCV.

Installed flight performance characteristics of the LH_2-2 Mach 2.7 study engine are shown in Figures 21 through 26 for the engine size based on Table 7. Figure 21 shows one of the many duct-heating temperature-limited engine operating schedules, for the U.S. Std. atmosphere +15°C (59°F + 27°F), which were evaluated at takeoff to meet the various noise constraints. It should be noted that the climb and cruise performance of Figures 22 through 26 are based on U.S. Std. atmosphere +8°C (59°F + 14.4°F) rather than the U.S. Std. atmosphere of Reference 1. This is to facilitate a direct comparison with the Jet A fueled aircraft from the Langley SCV System Studies (Reference 2). Figures 22 and 23 are shown as examples of the many duct-heating, temperature-limited engine operating schedules which were evaluated for optimum climb performance, from the standpoint of: (1) minimum gross weight, (2) minimum fuel weight, (3) minimum DOC, and (4) minimum noise.

3.2.1.3 Physical Characteristics. The internal flow path engine configuration and engine dimensions of the LH_2-2 Mach 2.7 DBTF sized in Table 7 are unchanged from those presented in Reference 1. Nacelle configuration, dimensions and scaling data for the Table 7 engine with cruise duct-heating temperature of 1367°K (2460°R) are shown in Figure 27. The variation in engine size and weight with maximum climb duct augmentation temperature is shown in Figure 28.

3.2.1.4 Noise Considerations. The engine size was selected to meet aircraft liftoff thrust requirements, and to also satisfy the low noise limits, by restricting duct-burning temperatures, for example, to 644°K (1160°R) for the FAR 36 minus 10 EPNdB limits. The cycle turbine energy is split so that the gas generator noise is lower than the noise goals and, therefore, a noise supressor is only required for the fan exhaust. Figure 29 was used for estimates of sound suppressor effectiveness at the point of aircraft liftoff. The suppressor effectiveness is plotted vs relative jet velocity which is the difference between jet velocity and aircraft velocity. These data were used to establish the thrust size and takeoff power setting for noise limited engine operation for both the Mach 2.7 and the Mach 2.2 cruise engines. The same noise suppressor effectiveness was used in establishing performance of the Jet A fueled SCV's (Reference 2).

3.2.2 Mach 2.2 Turbofan

3.2.2.1 Cycle Selection. - The Mach 2.2 engine cycle was based on a previously optimized Lockheed Mach 2.2 DHTF study engine, the Jet A fueled BSTF 2.2, used in the Langley SCAR studies (Reference 2). The optimizing studies previously made for the Langley program are applicable and therefore were not repeated. The LH_2 Mach 2.2 engine was computed with the hydrogen-air subroutine and is thermodynamically consistent with the Mach 2.7 cruise vehicle engine, see Table 7.

3.2.2.2 Performance Characteristics. - Installed flight performance characteristics of the LH_2 Mach 2.2 engine are shown on Figures 30 through 35. Figure 30 show one of the many duct-heating temperature engine operating schedules for the U.S. Std. atmosphere +15°C (59°F + 27°F) which were evaluated at takeoff to meet the various noise requirements. Figure 31 and 32

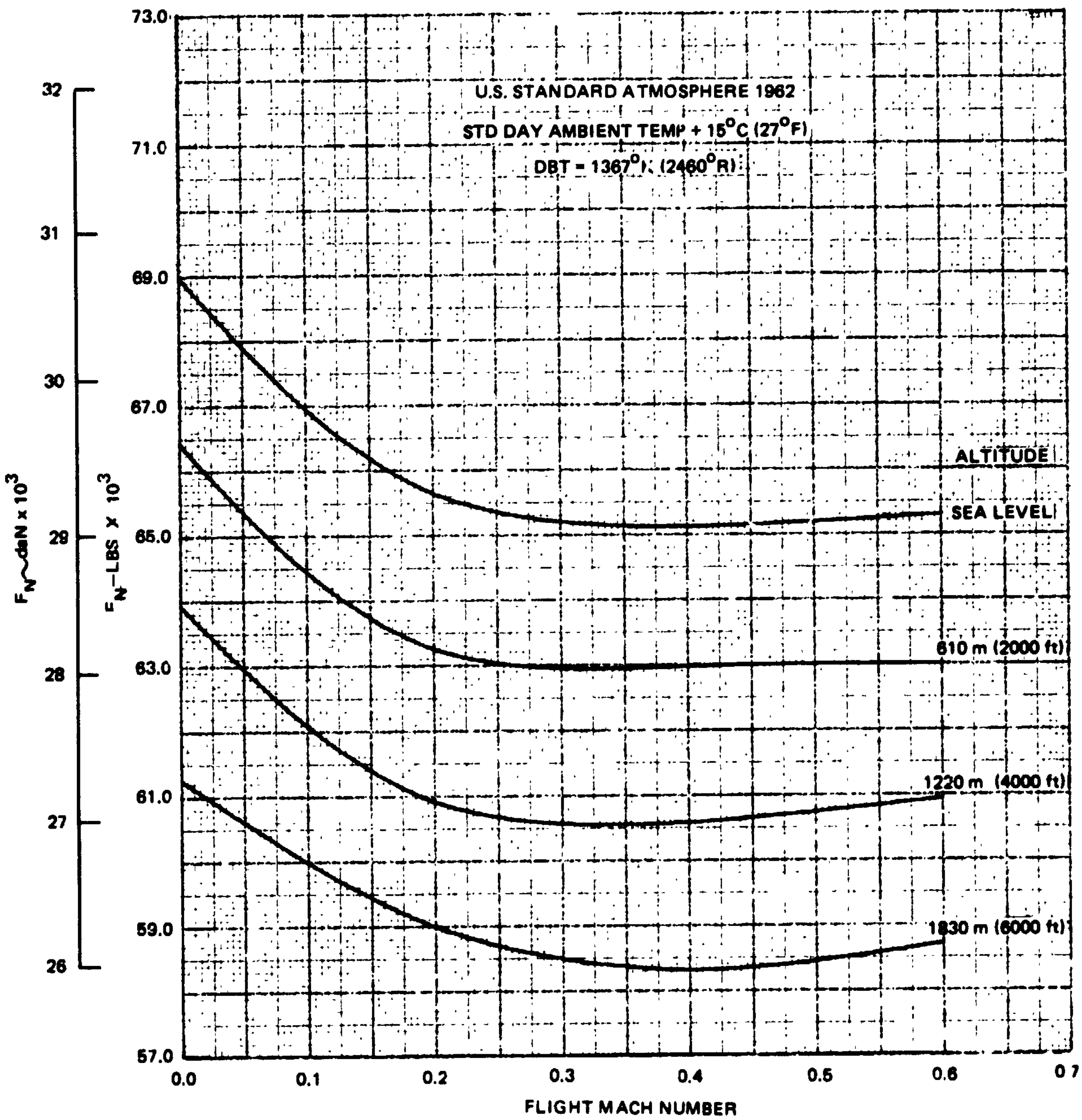


Figure 21. Installed Thrust - Takeoff Power
Mach 2.7 Engine.

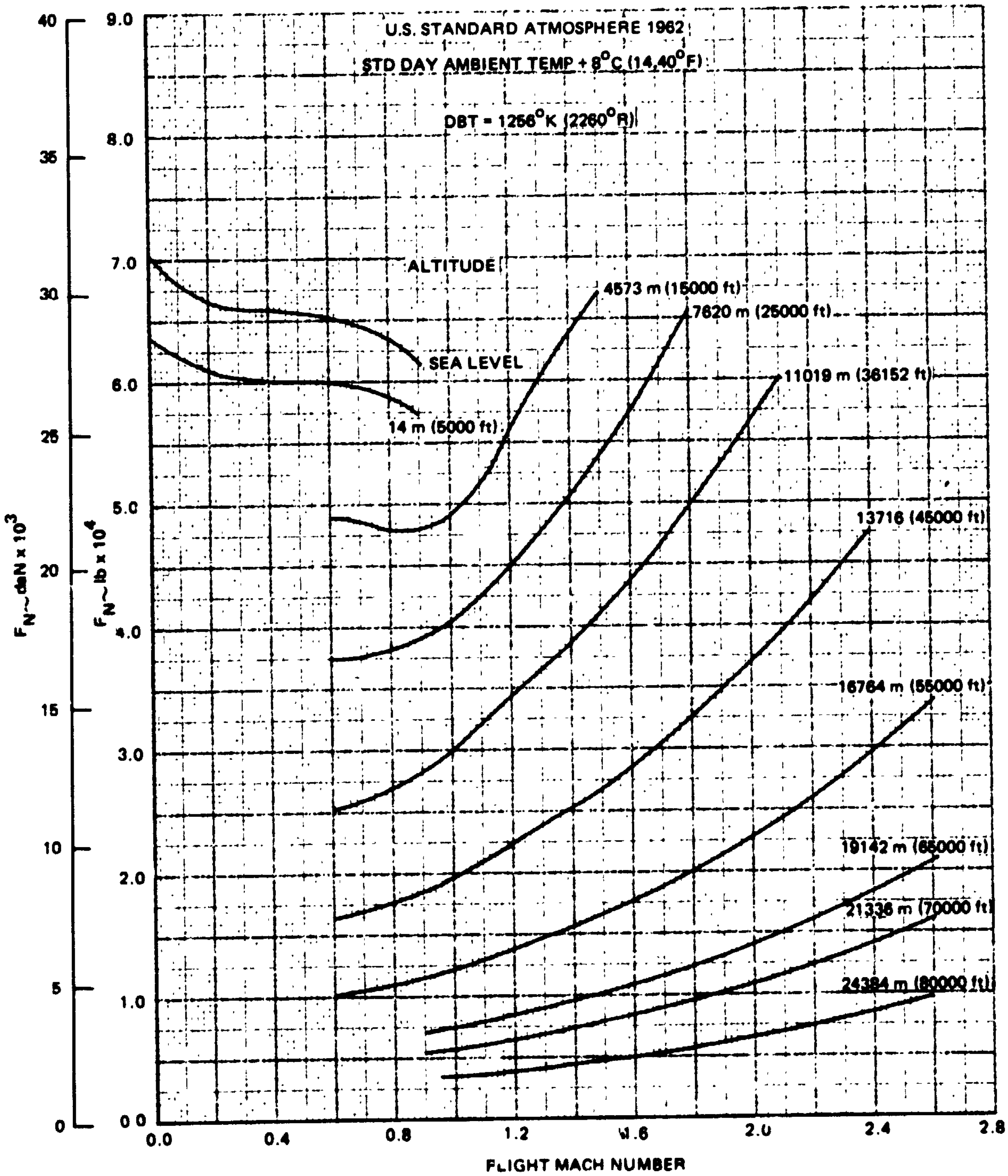


Figure 22. Installed Thrust - Hot Day Mission
Mach 2.7 Engine.

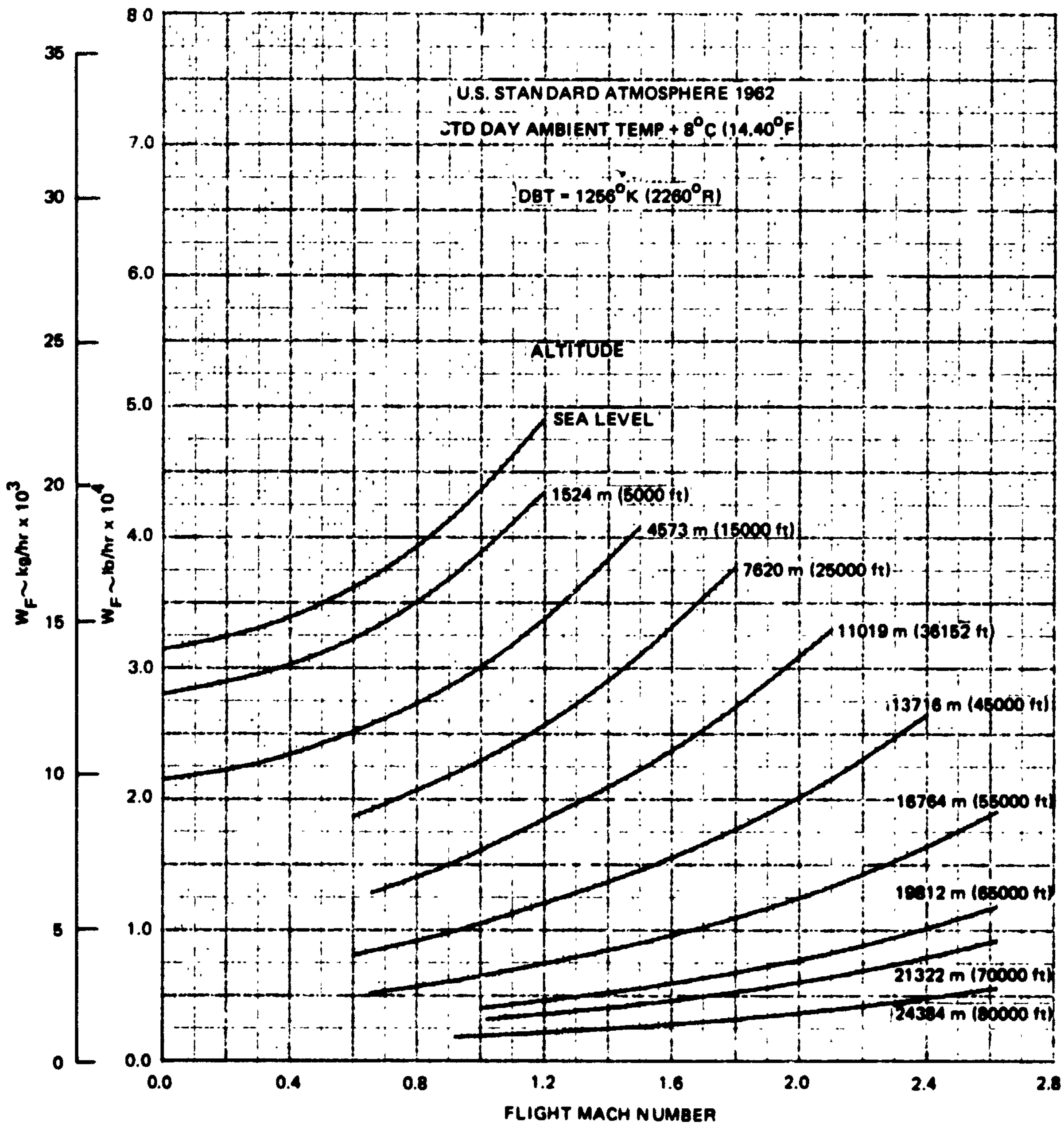


Figure 23. Installed Fuel Flow - Ho Day Mission
Mach 2.7 Engine.

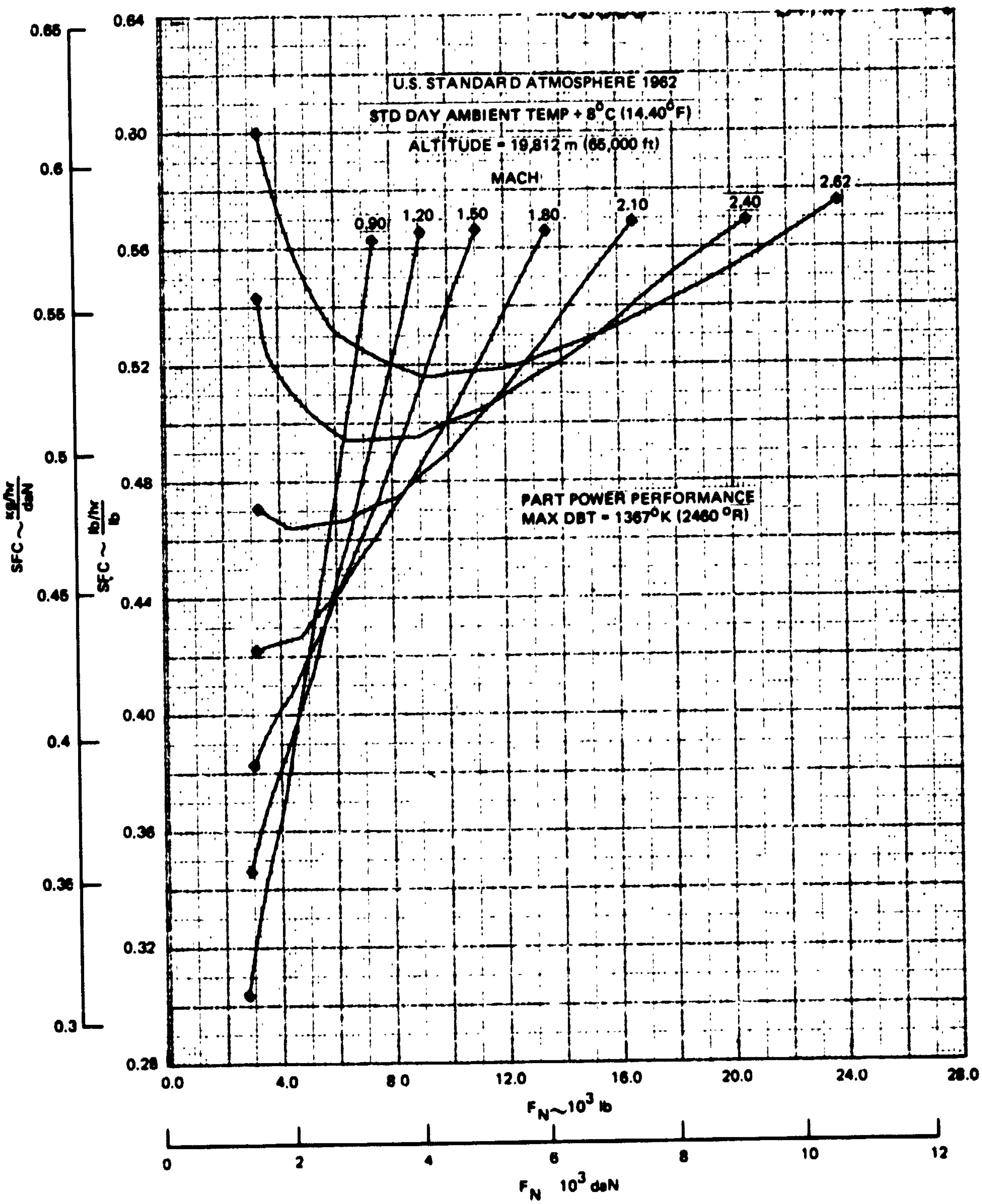


Figure 24. Installed Performance - Mach 2.7 Engine 19,812 m (65,000 ft).

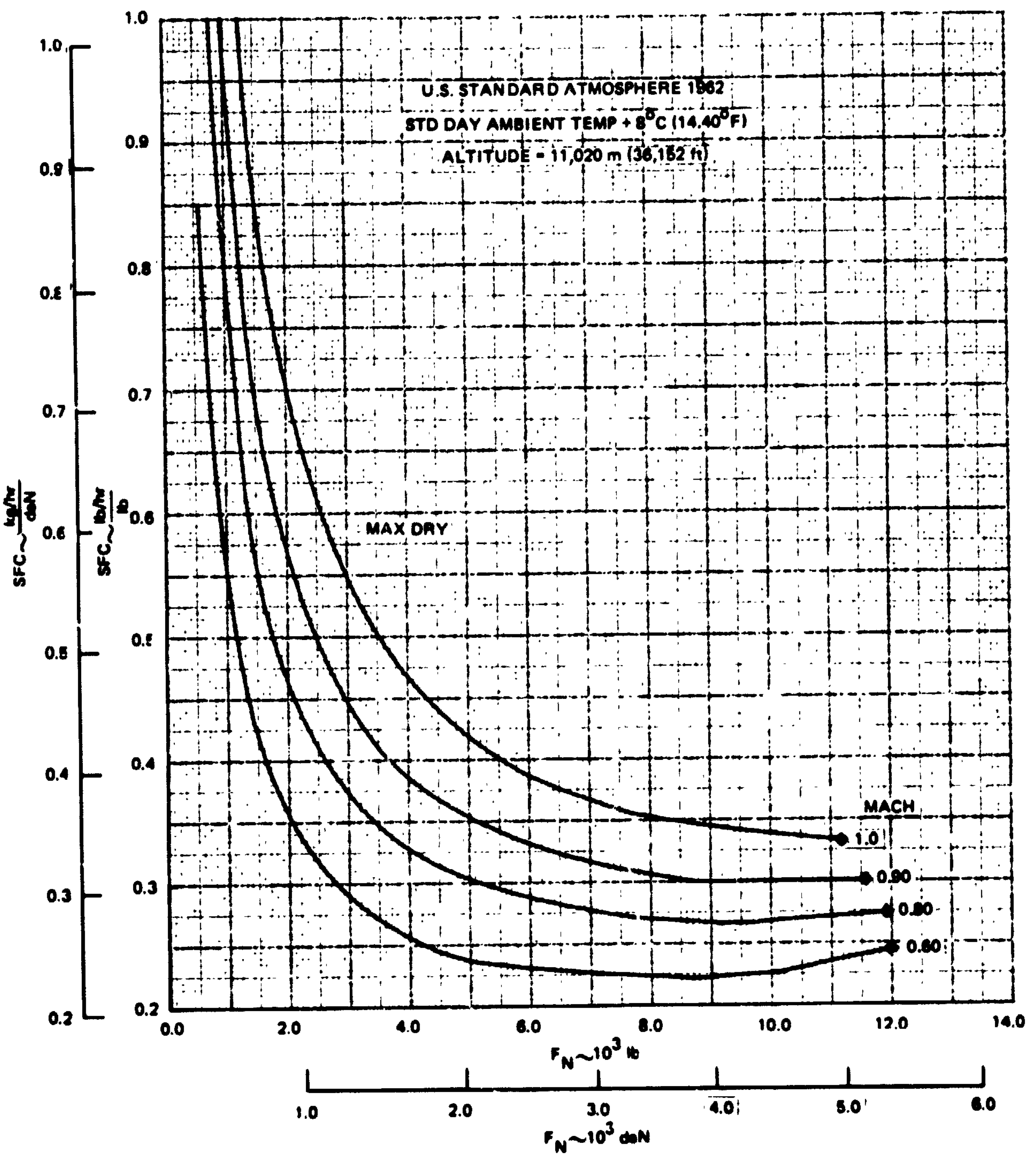


Figure 25. Installed Performance - Mach 2.7 Engine 11,019 m (36,152 ft).

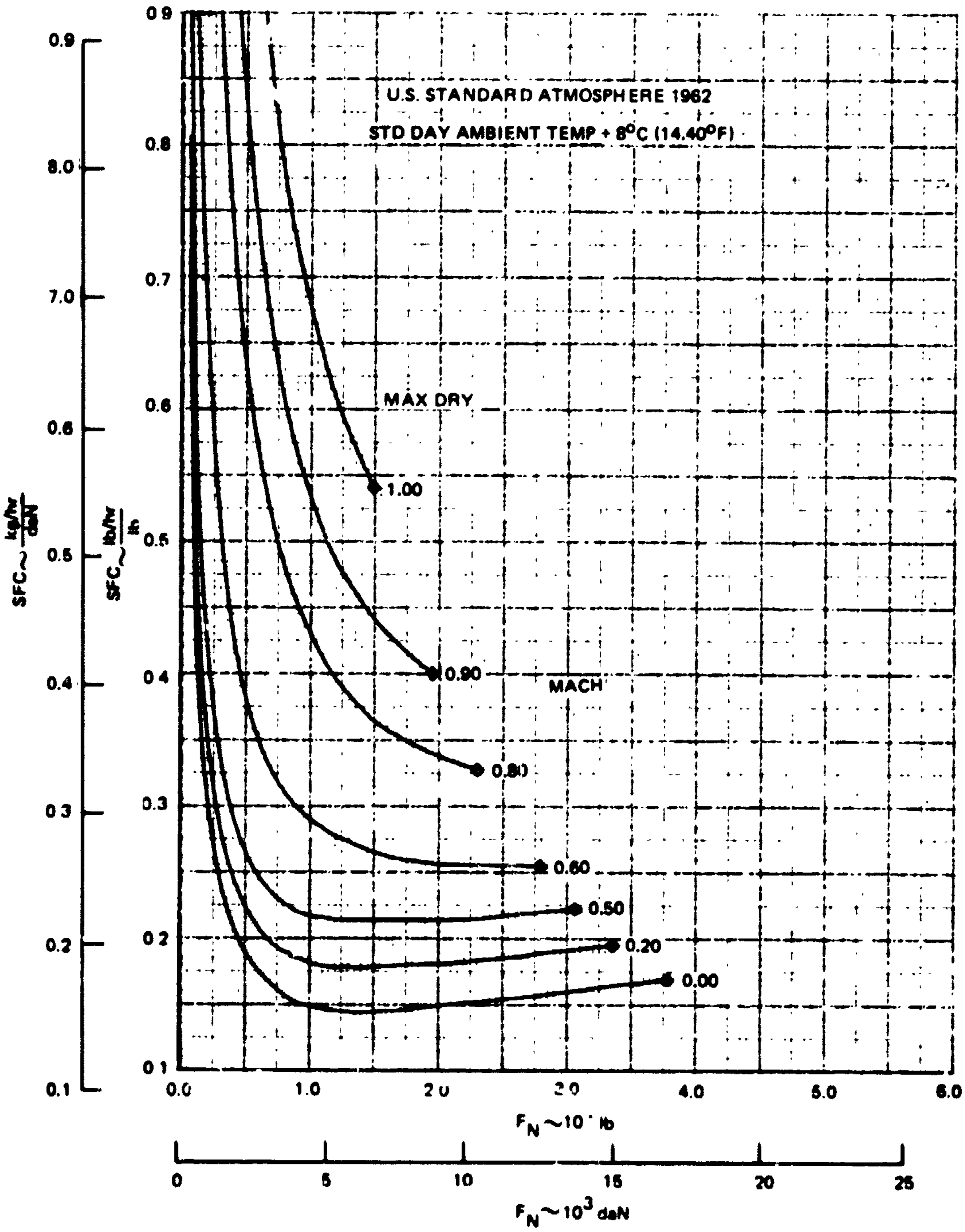
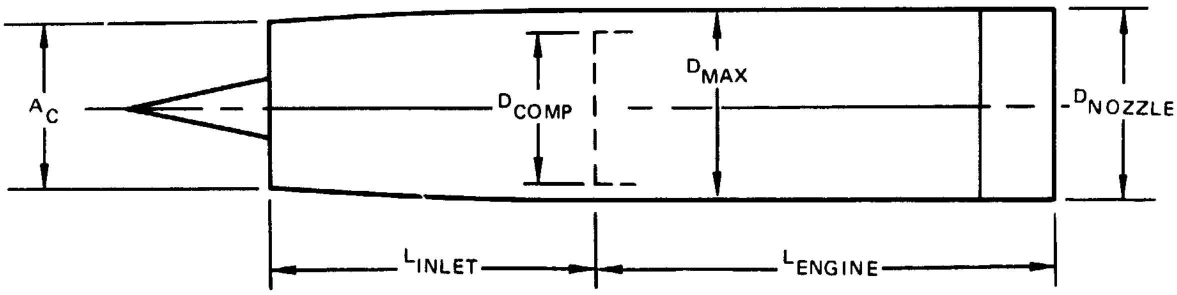


Figure 26. Installed Performance · Mach 2.7 Engine
1524 m (5,000 ft).

PARAMETER	REFERENCE VALUE	
CRUISE MACH NO	2.7	2.2
F_N SLS MAX	37600 da N (84500 lb)	35100 da N (78900 lb)
A_C	3.18 m^2 (34.2 ft^2)	1.90 m^2 (20.4 ft^2)
D_{COMP}	2.069 m (81.47 in.)	1.618 m (63.71 in.)
D_{MAX}	2.604 m (102.5 in.)	1.987 m (78.22 in.)
D_{NOZZLE}	2.604 m (102.5 in.)	1.987 m (78.22 in.)
L_{ENG}	6.782 m (267 in.)	5.466 m (215.3 in.)
WEIGHT	5260 kg (11600 lb)	4900 kg (10800 lb)

* INCLUDES REVERSER AND SUPPRESSOR



$$DIAM = DIAM_{REFERENCE} \left(\frac{F_{NSLS}}{F_{NSLS_{REF}}} \right)^{0.5}$$

$$L_{ENG} = L_{ENG\ REFERNCE} \left(\frac{F_{NSLS}}{F_{NSLS\ REF}} \right)^{0.35}$$

$$WEIGHT = WEIGHT_{REFERENCE} \left(\frac{F_{NSLS}}{F_{NSLS\ REF}} \right)$$

$$L_{INLET} = D_{COMP} \times 2.56$$

Figure 27. Duct Heating Turbofan Nacelle Dimensions and Scaling Data.

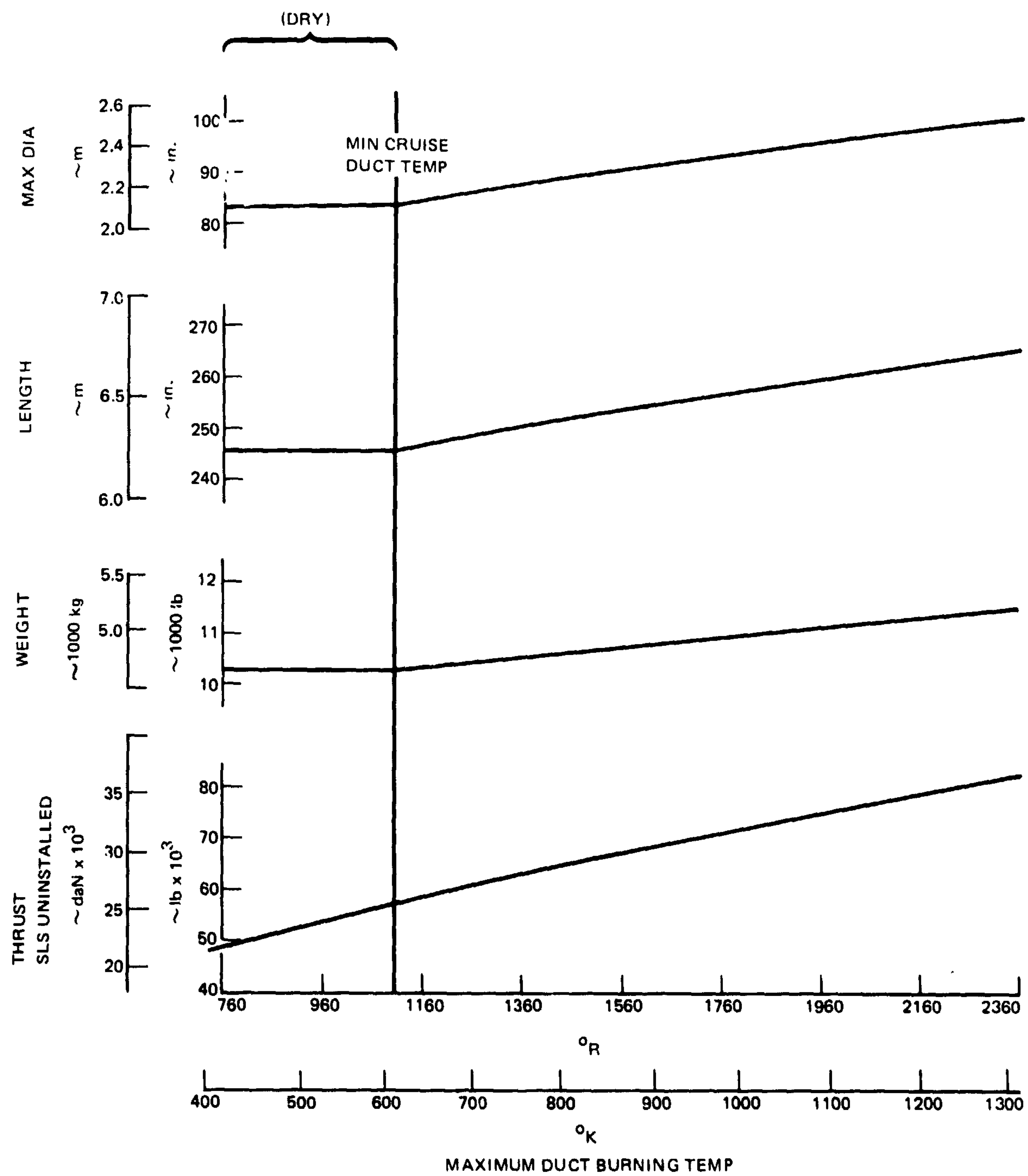


Figure 28. M2.7, LH_2 DBTF Engine Size vs max Duct Burning Temperature.

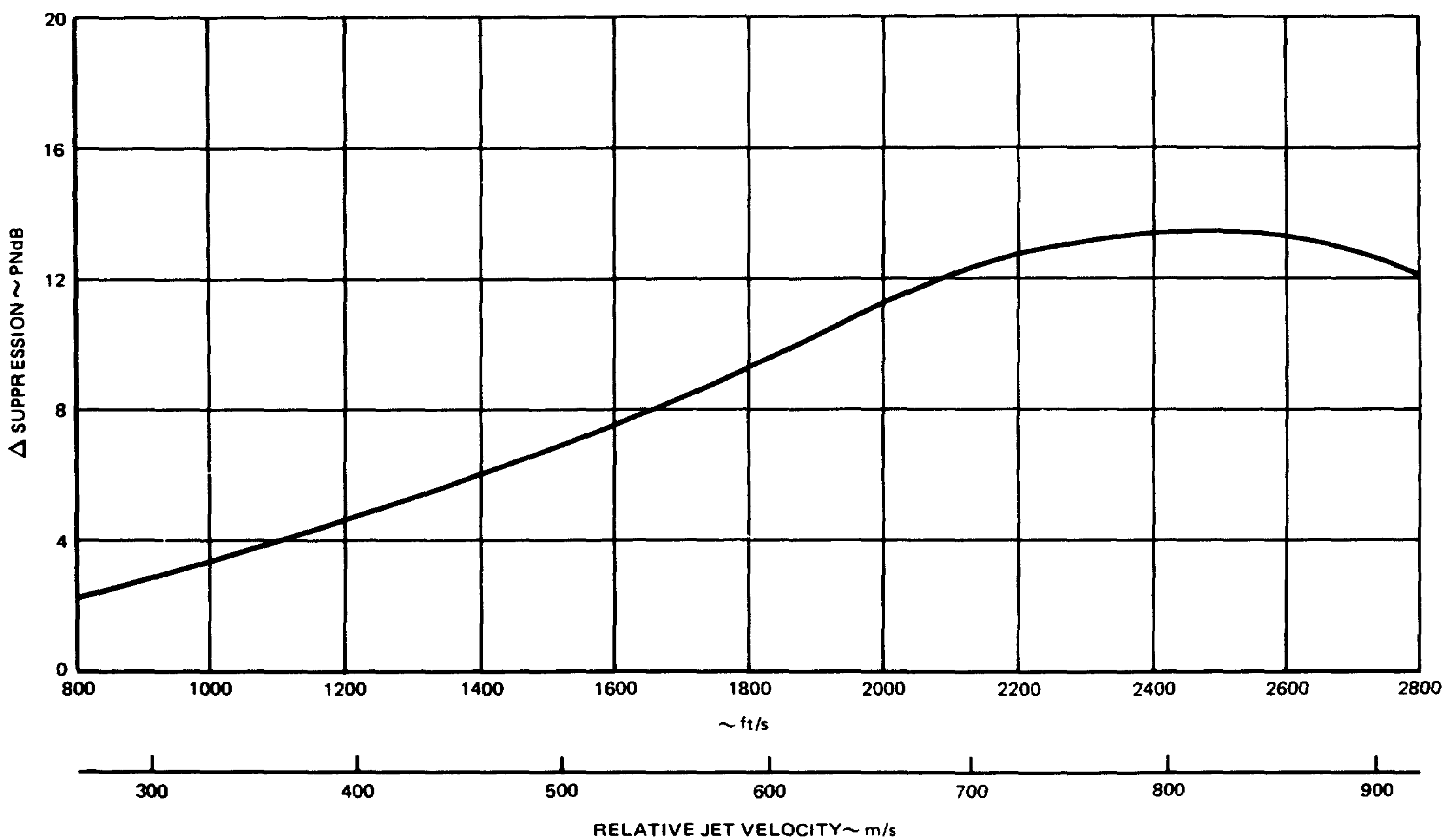


Figure 29. Jet Noise Suppressor Performance Envelope.

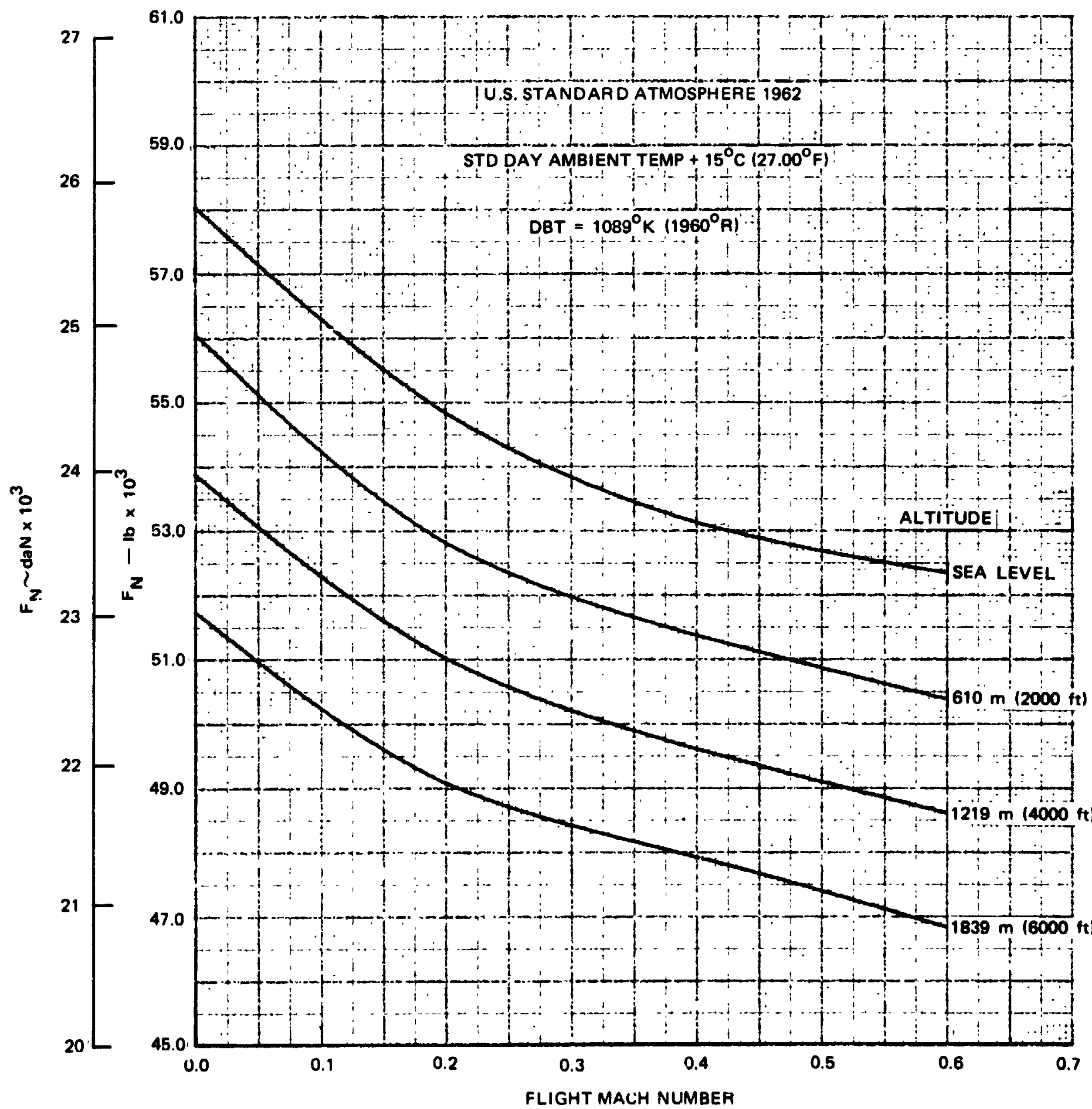


Figure 30. Installed Thrust - Takeoff Power
Mach 2.2 Engine.

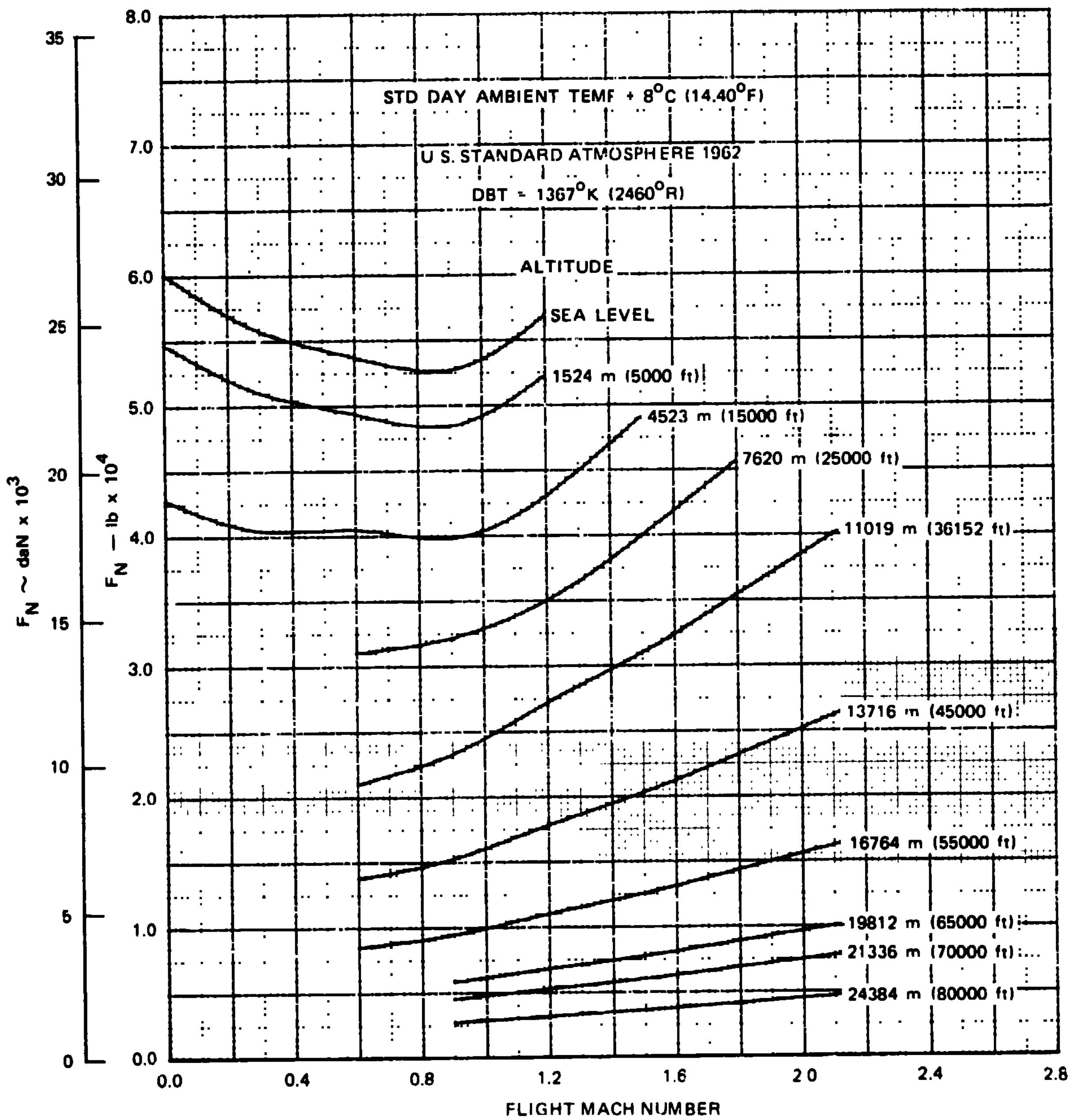


Figure 31. Installed Thrust - Hot Day Mission
Mach 2.2 Engine.

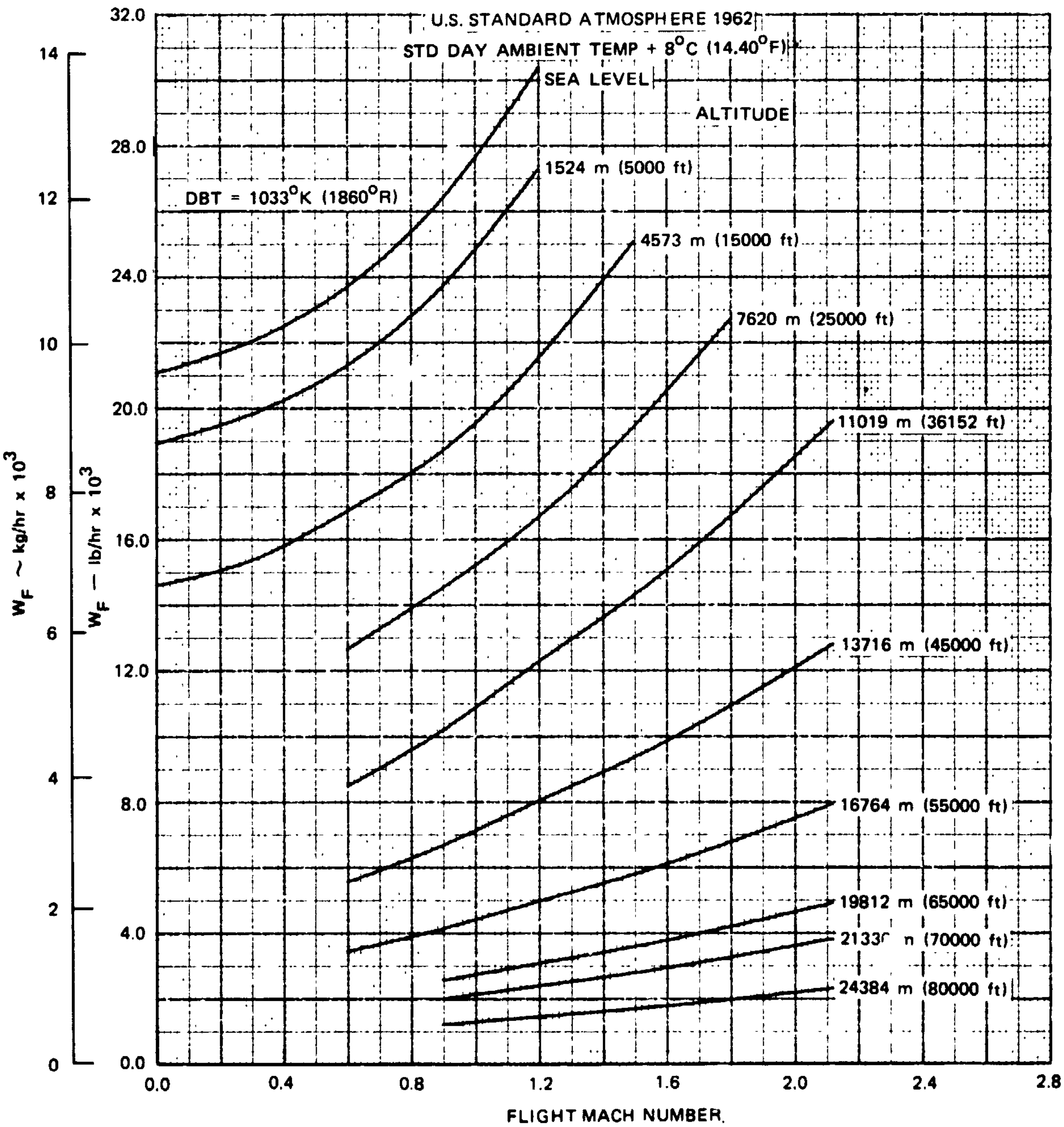


Figure 32. Installed Fuel Flow - Hot Day Mission Mach 2.2 Engine.

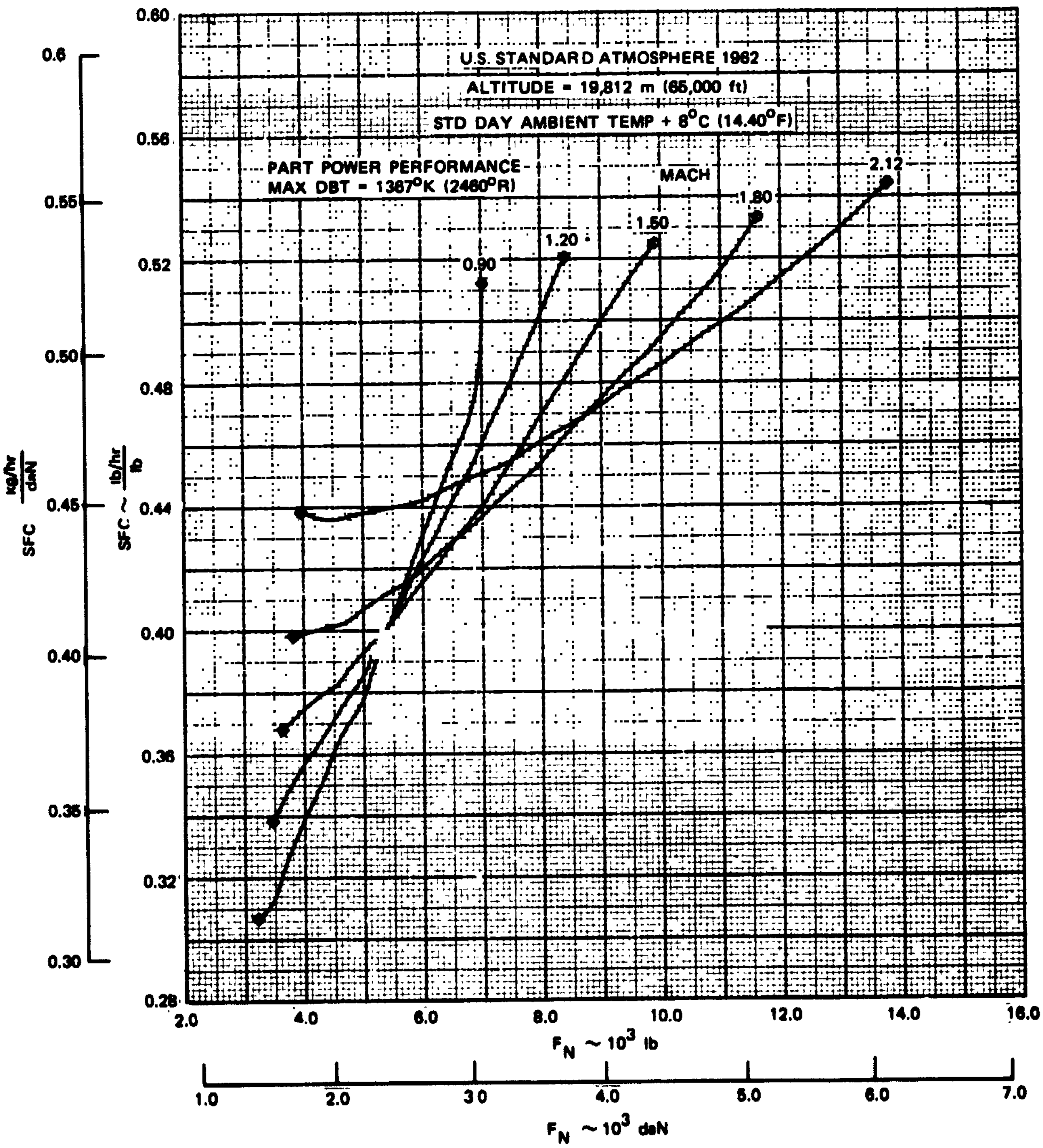


Figure 33. Installed Performance - Mach 2.2 Engine
19,812 m (65,000 ft).

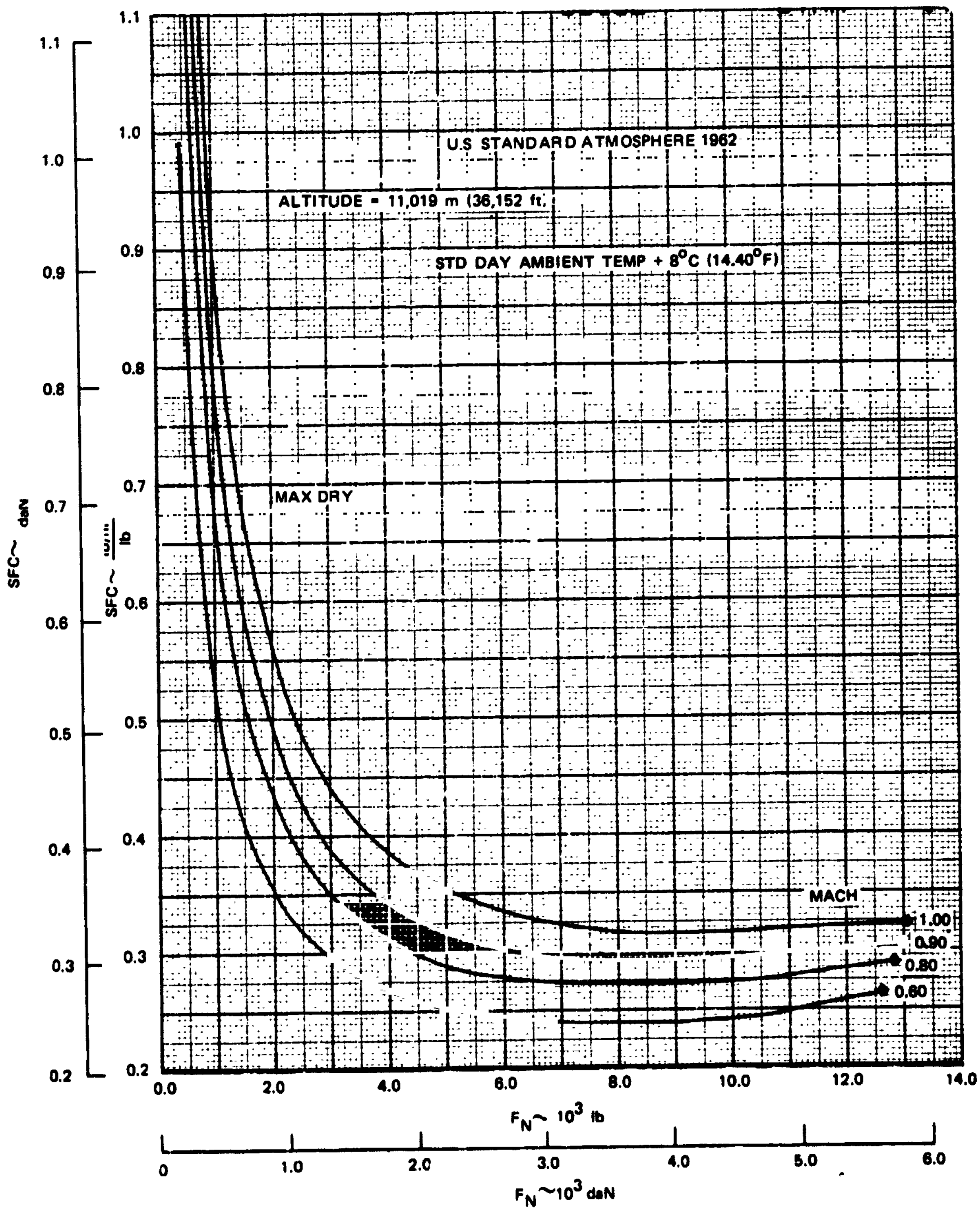


Figure 34. Installed Performance - Mach 2.2 Engine
11,019 m (36,152 ft).

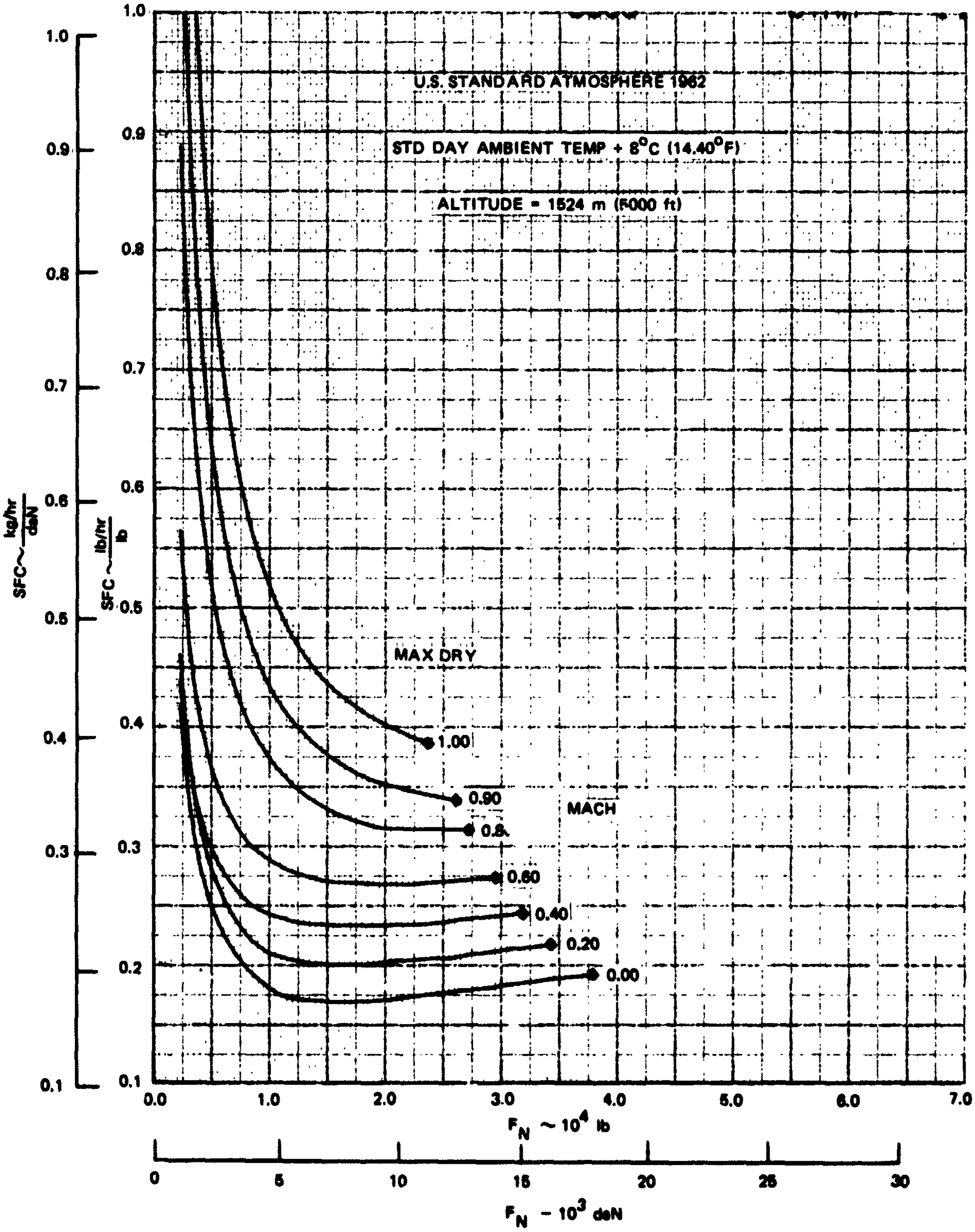


Figure 35. Installed Performance - Mach 2.2 Engine 1524 m (5,000 ft).

depict one of the climb schedules studied for the U.S. Std. atmosphere +8°C (59°F + 14.4°F). Figures 33 through 35 show supersonic and subsonic cruise performance for the U.S. Std. atmosphere +8°C (59°F + 14.4°F).

3.2.2.3 Physical Characteristics. - Nacelle configuration, dimensions and scaling data for the Table 7 engine with a cruise duct-heating temperature of 1367°K (2460°R) are shown in Figure 72. The variations of engine size and weight with maximum climb duct augmentation temperature are shown in Figure 36.

3.2.2.4 Noise. - The noise considerations for the Mach 2.2 DHTF are similar to those of the Mach 2.7 DHTF.

3.3 LH_2 Tank and System Design

3.3.1 Tank Weight. - Based on work performed under Reference 7, the tank fatigue allowable stress level of 275,800 kPa (40,000 psi) used in the previous (Reference 1) study on the aluminum shell structure was reduced to 206,850 kPa (30,000 psi) commensurate with an aircraft design life of 50,000 hours. Table 8 compares the effect on the tank component weights, total weight, and weight fractions including both the forward and aft tanks. Due to similar loads and design conditions, the same tank weight fraction was used for both the Mach 2.2 and 2.7 aircraft.

3.3.2 Tank Sizing. - Allowances for sizing the tanks are listed in Table 9. The volume required is based on the "as built" (warm) condition and includes the other allowances listed. The fluid expansion to a 138 kPa (20 psia) pressure is assumed to occur after filling from a ground storage facility at an equilibrium temperature corresponding to 103 kPa (15 psia). The effective density is used to calculate the tank volume required using the total mission fuel determined in the ASSET computer program.

3.3.3 Fuel System - The fuel system functional operation remains the same as described in Section 4.1.3.3 of Reference 1.

3.3.4 Tank Insulation. - A study was made to determine a preferred thickness of cryogenic insulation for tanks of both aircraft. Figures 37 and 39 show the weight of the insulation, the hydrogen boil off, and the heat shield in cumulative fashion, all as functions of insulation thickness for the Mach 2.7 and 2.2 aircraft, respectively. The arrow indicates the insulation thickness for minimum weight to be slightly over 76 mm (3 in) in both cases. Figures 38 and 40 consider the economic tradeoffs involved in the problem. They show the cost in thousands of dollars per day for flying 10.2 hours per day

Table 8. LH_2 Integral Tank Weight

Tank Component	Fatigue Stress Level			
	275,800 kPa	40,000 psi	206,850 kPa	30,000 psi
Shell	3,946 kg	8,700 lb	5,253 kg	11,580 lb
Ends	608	1,350	817	1,800
Frames	894	1,970	962	2,120
Baffles/bulkheads	367	810	367	810
Crosstie	408	900	489	1,078
Transition trusses	1,134	2,500	1,134	2,500
Crack stoppers	109	240	145	320
Contingency (10%)	753	1,660	916	2,020
Total (both tanks)	8,224	18,130	10,083	22,228
Fuel weight	41,958	92,500	41,958	92,500
Tank weight fraction $\left(\frac{W_{tank}}{W_{LH_2}}\right)$	0.196	0.196	0.2403	0.2403

using a baseline cost of LH_2 of \$2.85/GJ (\$3/10^6 Btu). Again, insulation thickness is the independent variable. The lower curve on both charts is the difference in airframe amortization over a 50,000 hour life cycle as affected by aircraft size and weight changes due to carrying insulation of various thicknesses, and including consideration of the corresponding losses hydrogen due to boil off. A portion of the gaseous hydrogen boiled off in flight is required to maintain design pressure in the tank. The cost of the GH_2 lost through venting is indicated by the difference in the flight boil off segment. The minimum operating cost is achieved with 127 mm (5 in.) of insulation thickness in both aircraft designs.

Table 9. LH_2 Tank Sizing Allowances

(Allowances in percent of as-built volume)	M2.2	M2.7
Tank chill-down contraction	0.90	0.90
Structure allowance	0.52	0.52
Equipment allowance	0.08	0.08
Fluid expansion 104 to 138 kPa (15 to 20 psia)	1.70	1.70
Ullage after expansion	0.30	0.30
Unusable	0.30	0.30
Pressurant gas	1.77	1.77
Vented boil-off	1.38	1.63
Total allowance	6.95	7.20
Effective density of Tanked LH_2 $= \dfrac{\text{density at 15 psia}}{1 + \dfrac{\text{total allowance \%}}{100}} \left(\dfrac{\text{lb}}{\text{ft}^3}\right)$	4.1272	4.118

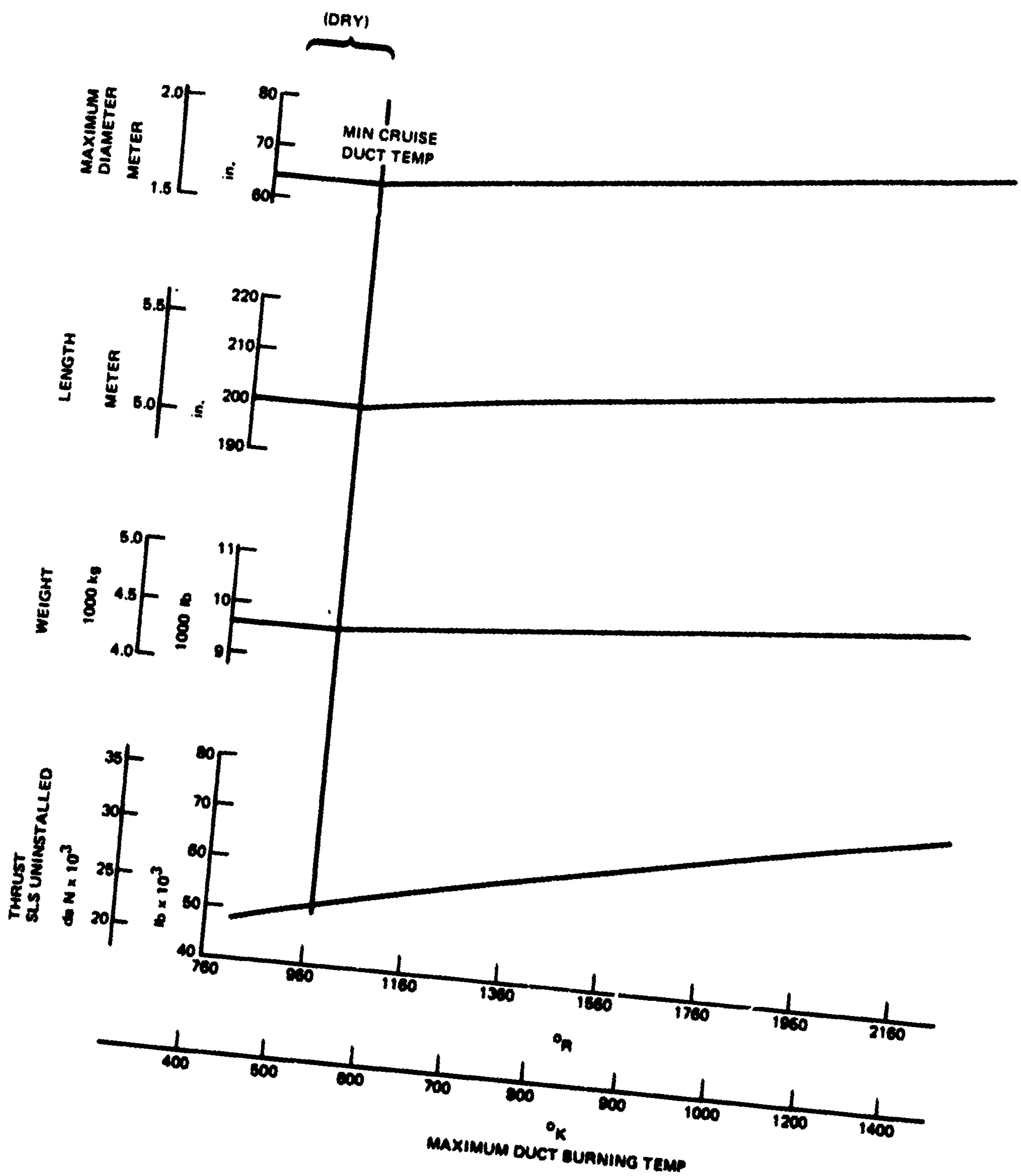


Figure 36. M2.2, LH_2 DBTF Engine Size vs max Duct Burning Temperature.

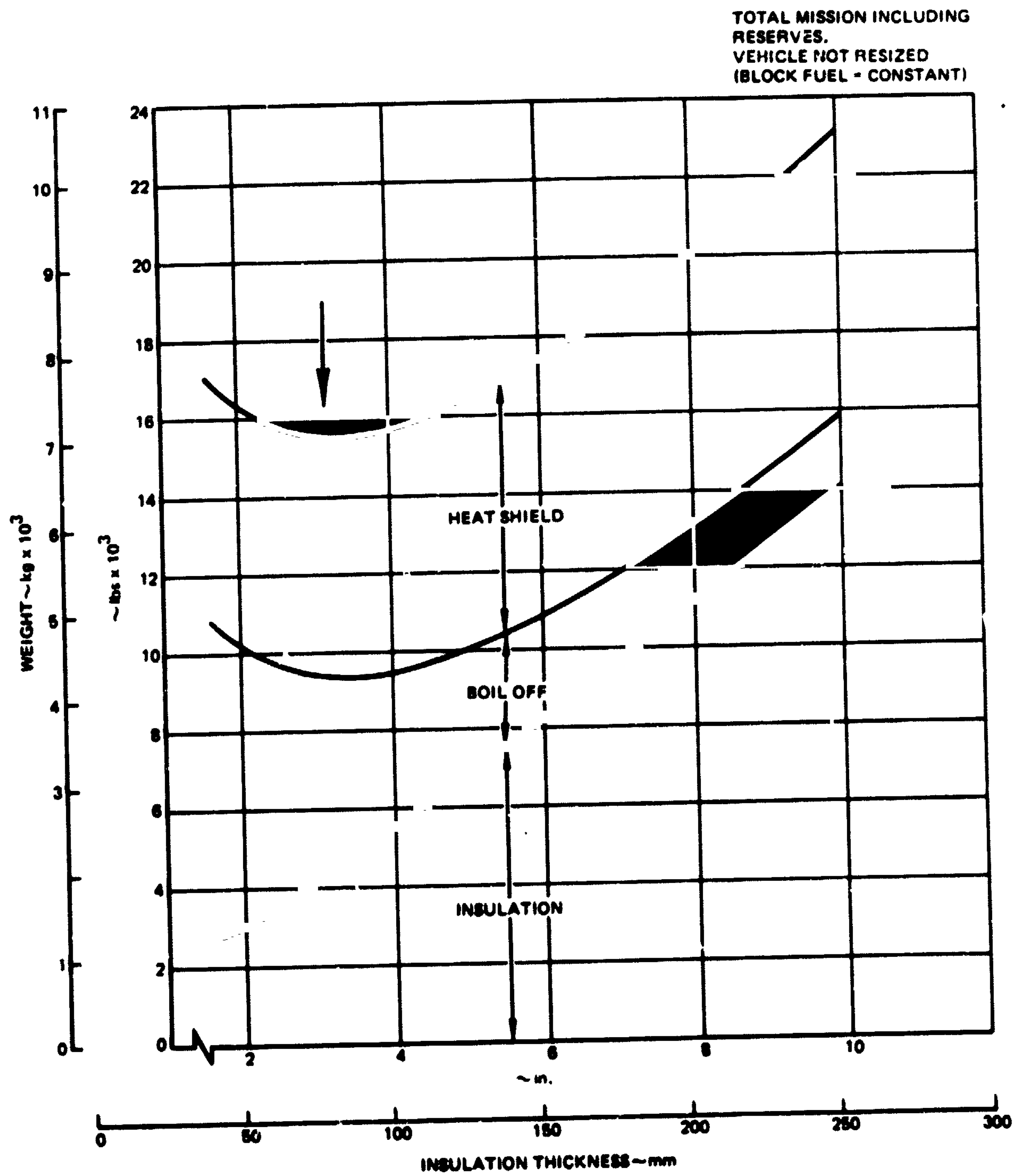


Figure 37. Insulation Thickness vs Weight for M2.' SCV.

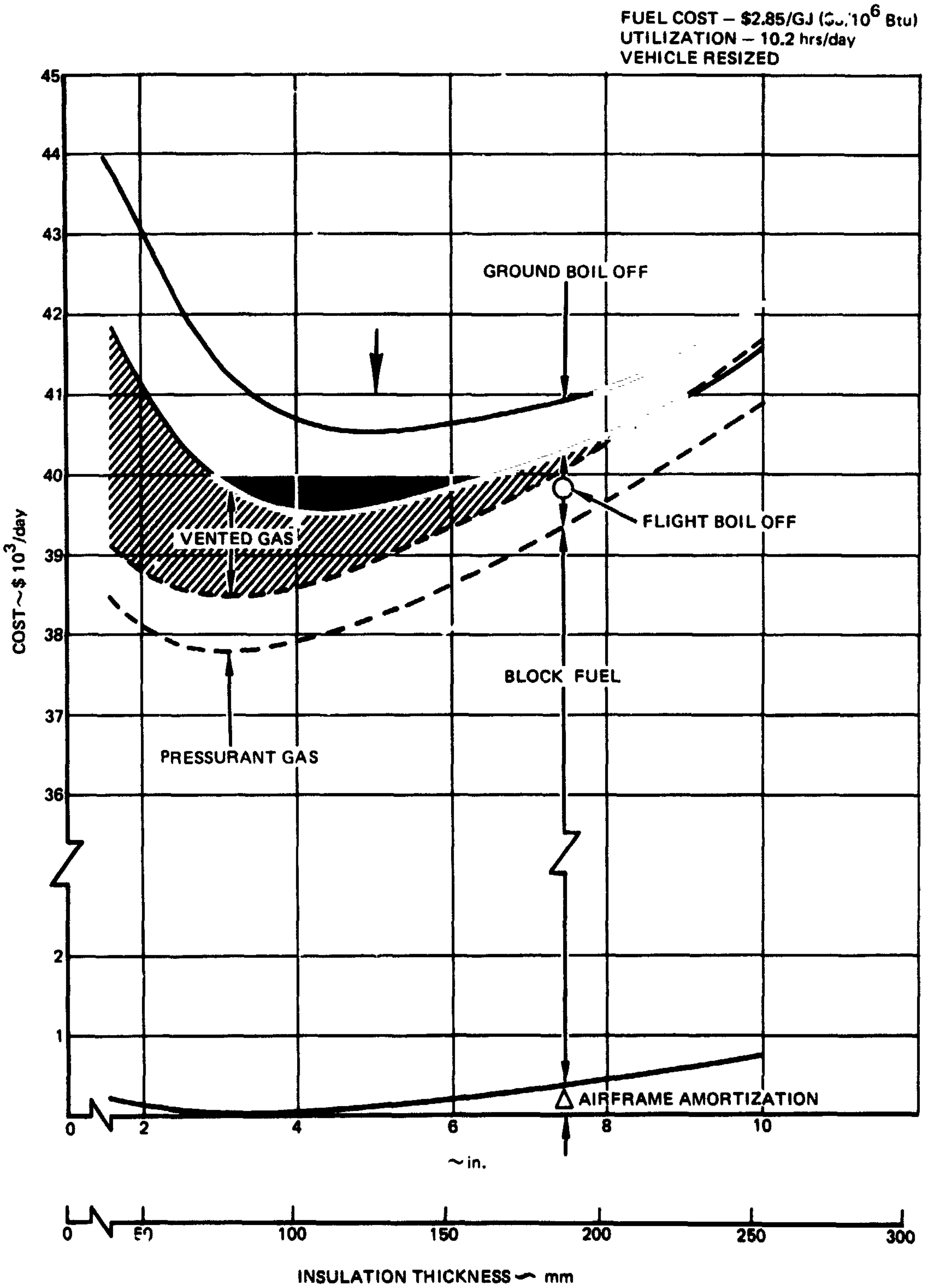


Figure 38. Insulation Thickness vs Cost for M2.7 SCV.

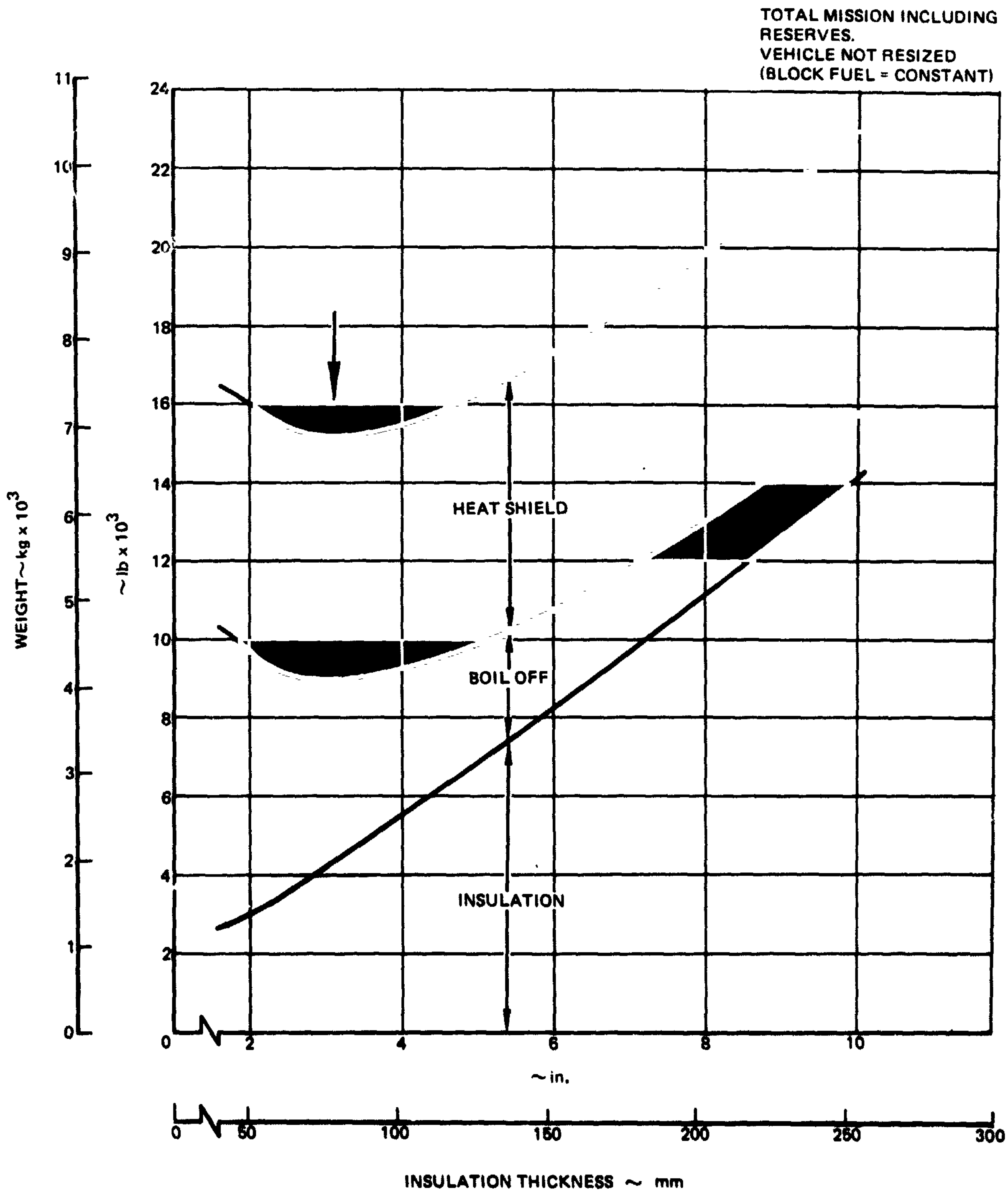


Figure 39. Insulation Thickness vs Weight for M2.2 SCV.

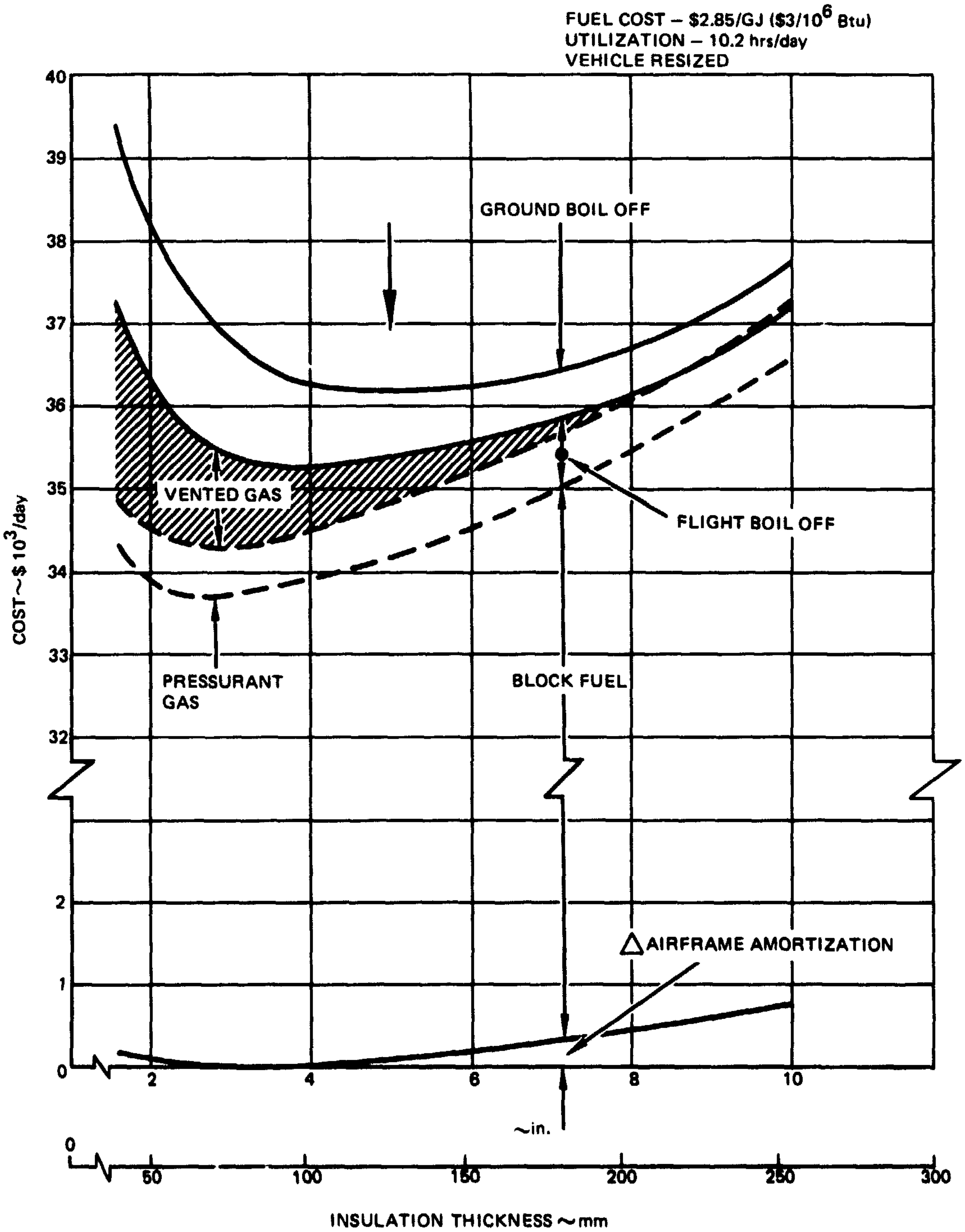


Figure 40. Insulation Thickness vs Cost for M2.2 SCV.

3.3.5 <u>LH_2 Fuel Losses</u> - A summary of the total fuel losses for the Mach 2.2 and 2.7 aircraft is shown in Table 10. The boil-off losses are based on an insulation thickness of 5 inches as described in the previous paragraph. The fuel quantities listed are for the minimum gross weight aircraft meeting FAR 36 and described in subsequent sections.

Table 10. LH_2 Unusable Fuel and Boil-off Losses

		M2-2	M2.7
Unusable - Tank	%	0.30	0.30
Unusable - Lines	%	0.40	0.40
Pressurant Gas	%	1.77	1.77
Vented Boil-off	%	1.38	1.63
Total	%	3.85	4.10
Total Mission Fuel	kg (lb)	46,113 (101,660)	45,668 (100,679)
Total Unusable and Boil-Off*	kg (lb)	1,775 (3,914)	1,873 (4,128)
Total Fuel	kg (lb)	47,888 (105,574)	47,541 (104,807)

*Included in "STANDARD ITEMS" in ASSET Weight statement.

3.4 Weight Parameters

As stated in the Technical Approach, Section 2, the philosophy for this study was to make the design and technology basis for the subject LH_2 vehicles identical to that used for the Jet A fueled designs from the "Cruise Speed Study" of Contract NAS1-11940 (Reference 2). This resulted in greater use of advanced composites than that for the "LH_2 AST Concept Study" of Contract NAS 2-7732 (Reference 1). Greater use of advanced composites results in significant weight reductions for each structural component as shown in Table 1 (Section 2). The percentage use of composites in the LH_2 fueled Mach 2.7 and 2.2 airplane designs of the present study was the same as that shown in Table 1 for the Advanced Technology Cruise Speed Study airplane.

Two other differences incorporated in the design of the LH_2 aircraft during the present study, which were significant changes from the design of the original study, were the result of 1) a decrease in allowable stress of the 2219 aluminum alloy used in design of the LH_2 tanks, and 2) an increase in thickness of the rigid, closed-cell plastic foam used for cryogenic insulation on the external surfaces of the tanks. The basis for these changes is discussed in Section 3.3. Briefly, the allowable stress was reduced to reflect use of more realistic fatigue allowables, and the increase in cryogenic insulation thickness was the result of consideration of economic factors in a tradeoff of insulation weight with the amount of gaseous hydrogen lost through boil-off. Both of these changes obviously increase the inert weight of the aircraft and thus affect the improvement resulting from increased use of composites.

The various weight parameters employed to represent the weight of the wing of the subject LH_2 fueled aircraft were derived primarily from the results of the Arrow-Wing Structures Study (AWSS) of Reference 3. That study was an analytical investigation performed to provide data to support the selection of the best structural concepts for the design of a near-term Jet A fueled Mach 2.7 supersonic cruise aircraft wing and fuselage primary structure. To arrive at proper projections for airframe structural mass for advanced technology supersonic cruise aircraft, similar designs using graphite- and boron-polyimide composites were evaluated. Data derived from that study, plus the design parameters and requirements identified below, were used to develop primary wing structure weights for the LH_2 fueled supersonic cruise aircraft of the present study.

The importance of the various interactive parameters that influence the design of a Jet A fueled supersonic cruise aircraft with a prescribed arrow-wing configuration employing chordwise-stiffened, beaded surface panels and submerged composite reinforced spar caps were identified as follows:

(1) The wing design parameters, Figure 41, were categorized by three distinct zones:

- The tip structure was stiffness critical and sized to meet the flutter requirements.

- The aft box and the more highly loaded portion of the forward box structures were strength-designed to transmit the wing spanwise and chordwise bending moments and shears.

- The forward box structural-sizing resulted in surface panels and substructure components with active minimum gage constraints. Foreign object damage was the governing criteria for selection of minimum gage.

(2) The design conditions which displayed the maximum inplane surface panel loads are identified in Figure 42. An exception is the tip structure which was stiffness critical for the Mach 1.85 condition. Although the start-of-cruise condition (Mach 2.7) has the highest value of inplane loading, combination with the appropriate pressure

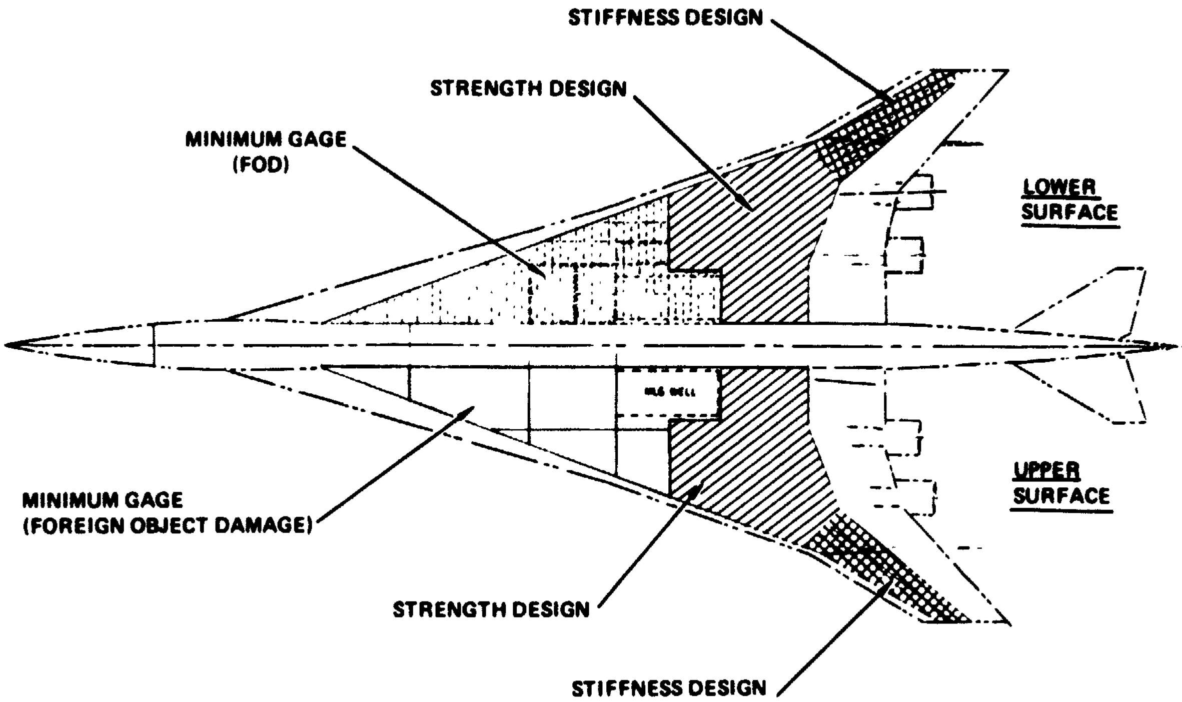


Figure 41. Wing Design Parameters.

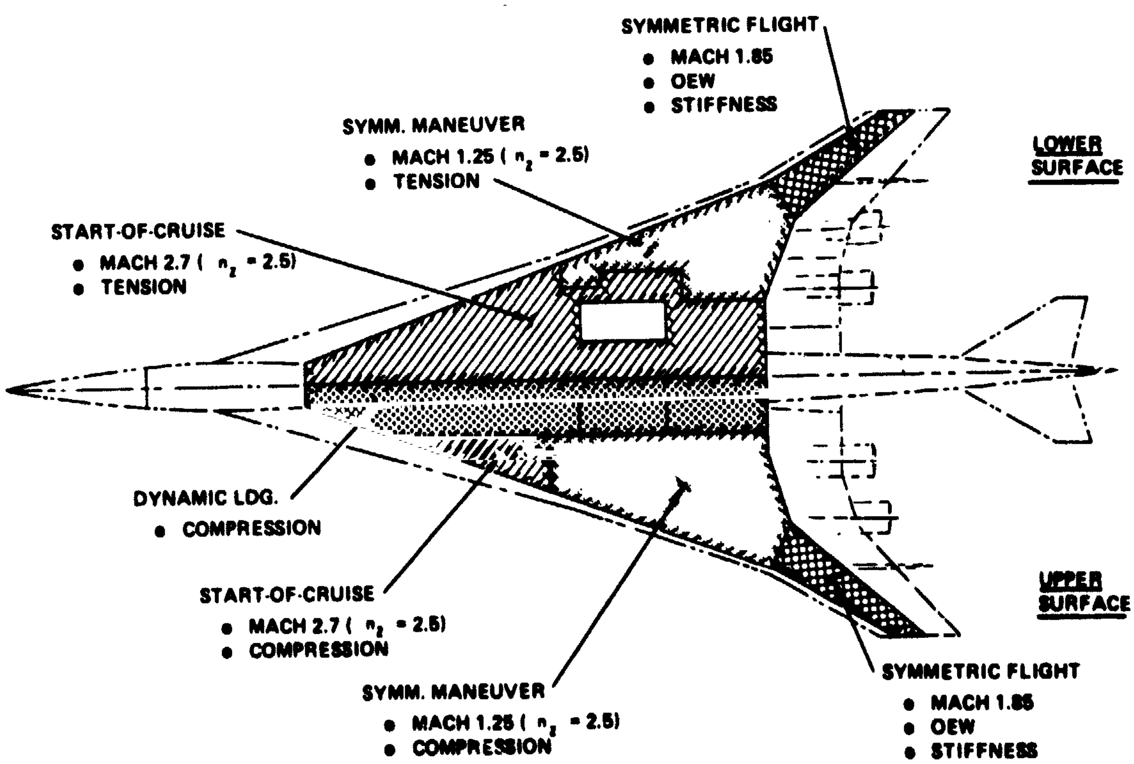


Figure 42. Critical Wing Design Conditions.

loads results in the symmetric maneuver condition at Mach 1.25 designing the wing in the forward and aft box region.

(3) **The Jet A fueled supersonic cruise aircraft displayed critical** loads at Mach numbers wherein the structural temperatures do not influence the design appreciably. Although Figure 43 indicates a major area of the wing lower surface impacted by the thermal environment, analysis of the surface panels and substructure using the applicable load-temperature environment results in the symmetric maneuver condition at Mach 1.25 as the critical design condition. The resulting designs, however, were very similar in geometry and structural mass. The upper surface in the forward box was constrained by the minimum gage criteria.

For the LH_2 fueled Mach 2.7 aircraft, several distinct changes are evident. The wing loading is reduced from 3.29 kPa (68.7 lb/ft^2) for the Jet A fueled aircraft (Reference 3) to 2.56 kPa (53.5 lb/ft^2) for the LH_2 fueled aircraft (Reference 1), the wing area of the LH_2 design is much smaller, and the LH_2 fuel is carried in fuselage tanks, in lieu of in the wing. This minimizes the beneficial effect of inertial relief; however, the surface panels for the LH_2 fueled aircraft are near-minimum gage since the effect of fuel pressures which had a strong influence on surface panel sizing for the Jet A fueled aircraft no longer apply.

The wing tip region of the LH_2-fueled aircraft is presumed to be stiffness-critical, thus the unit weights were considered invariant.

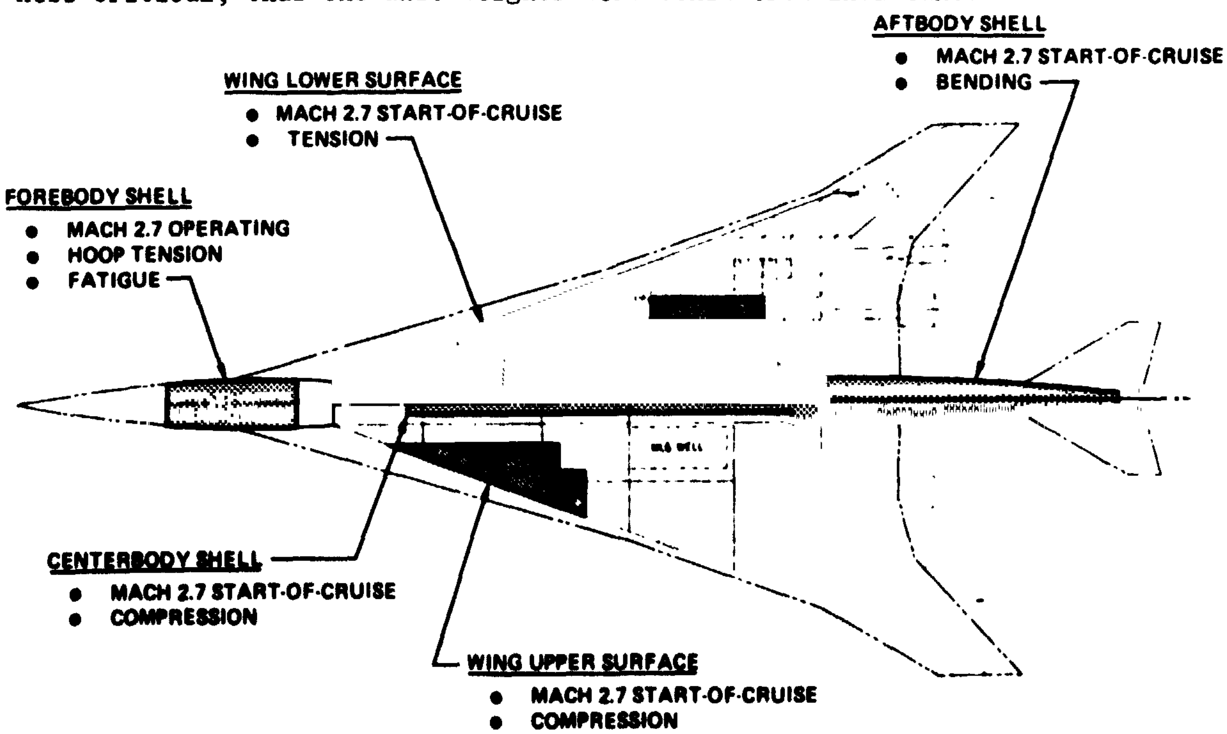


Figure 43.- Mach 2.7 Thermal Environment Considerations

Summarized in Table 11 are the pertinent parameters associated with the primary wing box structure weights. A comparison of the AWSS results (Jet A fueled) with the LH_2 fueled aircraft is shown. Only the substructure unit weights in the forward box are reduced from the Jet A fueled weights by the wing loading ratio (53.5/68.7 = 0.78) since the panel structure is near-minimum gage. For the aft box/transition substructure a factor of 0.65 is applied. This further reduction is warranted, over the wing loading change, because of the greatly reduced span of the LH_2-fueled aircraft. A comparison of relative surface spanwise loads is shown in Figure 44.

The following features are common to both the Jet A and LH_2 fueled SCV's compared in this study insofar as weight representation is concerned:

- No APU
- No allowance for customer options
- Weight of cargo containers included in payload
- Very austere furnishings weight allowance.

The items which account for the significant weight differences between the reference Jet A and the LH_2 fueled aircraft of this study are shown in the following list. The provisions made to account for the weight difference required for the LH_2 vehicles are defined.

- Body - Added 6% of Jet A body weight for double-deck passenger cabin and two extra pressure bulkheads. [The weight increment (ΔW) is approximately 1089 kg (2400 lb).]
- Landing Gear - 180" extended strut length of main landing gear in lieu of 160" to provide for an adequate scrape angle with a longer body. [$\Delta W \approx$ 499 kg (1100 lb).]
- Engine - M2.7 LH_2 engine weight/thrust (SLS) = 0.13751 lb/lb in lieu of 0.142859 lb/lb for the Jet A design
  - M2.2 LH_2 engine weight/thrust (SLS) = 0.13498 lb/lb in lieu of 0.142859 lb/lb for Jet A
- Fuel System - Added 80% to weight of comparable Jet A fuel system for insulation and/or vacuum tubing around fuel lines. [$\Delta W \approx$ 907 kg (2000 lb).]
- Integral LH_2 Tanks - Added 0.0958 x weight of LH_2 fuel (WLH_2) for tank ends and support structure (included in "body" weight). [ΔW = 4375 kg (9645 lb) for min. W_G M2.7 aircraft.]
- Unusable Fuel and Boiloff (% of usable fuel weight)
  M2.7 = 4.10; M2.2 = 3.85, in lieu of 0.89 for Jet A design

Table 11. Wing Box Weight Comparisons
(S. I. Units)

Wing Region	Forward Box		Aft Box/Trans.		Tip Box		Σ Wing Box		Total Wing	
Reference	AWSS	LH_2	AWSS	LH_2	AWSS	LH_2	AWSS	LH_2	AWSS	LH_2
Area (m^2)	384	242	198	125	88	52	670	419	1015	639
Upper Surface Panel(B)										
t_o (mm)	0.381	0.381	0.660	0.381	1.575	1.575	-	-		
t_i (mm)	0.330	0.330	0.584	0.330	1.575	1.575	-	-		
$\bar{t}_u$ (mm)	0.838	0.838	1.473	0.838	3.327	3.327	-	-		
Lower Surface Panel(B)										
t_o (mm)	0.508	0.508	0.508	0.508	1.905	1.905	-	-		
t_i (mm)	0.381	0.381	0.483	0.381	1.905	1.905	-	-		
$\bar{t}_\ell$ (mm)	1.041	1.041	1.219	1.041	3.886	3.886	-	-		
$\Sigma \bar{t}_u + \bar{t}_\ell$ (mm)	1.879	1.879	2.692	1.879	7.213	7.213	-	-		
WEIGHTS										
PANEL (kg/m^2)	8.35	8.35	11.91	8.35	31.93	31.93	12.50	11.28		
SUBSTR. (kg/m^2)	10.98	8.54	22.26	14.50	6.74	6.74	13.77	10.11		
Σ (kg/m^2)	19.33	16.89	34.17	22.85	38.67	38.67	26.27	21.39	(A) 40.47	(A) 33.49
$W_{BOX}^{(A)}$ (kg)	9,335	5,156	8,511	3,592	4,485	2,676	22,332	11,423	41,089	21,412

Note (A) Includes nonoptimum factor
(B) t_o = outer skin thickness; t_i = inner skin thickness; $\bar{t}_u$ and $\bar{t}_\ell$ = equivalent thickness of upper and lower panels

Table 11. Wing Box Weight Comparisons (Continued)
(U.S. Customary Units)

Wing Region	Forward Box		Aft Box/Trans.		Tip Box		Σ Wing Box		Total Wing	
Reference	AWSS	LH_2	AWSS	LH_2	AWSS	LH_2	AWSS	LH_2	AWSS	LH_2
Area (ft^2)	4137	2607	2132	1342	947	564	7216	4513	10923	6880
Upper Surface Panel (B)										
t_o (in)	0.015	0.015	0.026	0.015	0.062	0.062	-	-		
t_i (in)	0.013	0.013	0.023	0.013	0.062	0.062	-	-		
$\bar{t}_u$ (in)	0.033	0.033	0.058	0.033	0.131	0.131	-	-		
Lower Surface Panel (B)										
t_o (in)	0.020	0.020	0.020	0.020	0.075	0.075	-	-		
t_i (in)	0.015	0.015	0.019	0.015	0.075	0.075	-	-		
$\bar{t}_\ell$ (in)	0.041	0.041	0.048	0.041	0.153	0.153	-	-		
$\Sigma\ \bar{t}_u + \bar{t}_\ell$ (in)	0.074	0.074	0.106	0.074	0.284	0.284	-	-		
WEIGHTS										
PANEL (lb/ft^2)	1.71	1.71	2.44	1.71	6.54	6.54	2.56	2.31		
SUBSTR. (lb/ft^2)	2.25	1.75	4.56	2.97	1.38	1.38	2.82	2.07		
Σ (lb/ft^2)	3.96	3.46	7.00	4.68	7.92	7.92	5.38	4.38	(A) 8.29	(A) 6.86
$W_{BOX}^{(A)}$ (lb)	20,580	11,366	18,764	7,918	9,888	5,900	49,232	25,184	90,584	47,205

Note (A) Includes nonoptimum factor
(B) t_o, outer skin thickness; t_i, inner skin thickness; $\bar{t}_u$ and $\bar{t}_\ell$ = equivalent thickness of upper and lower panels

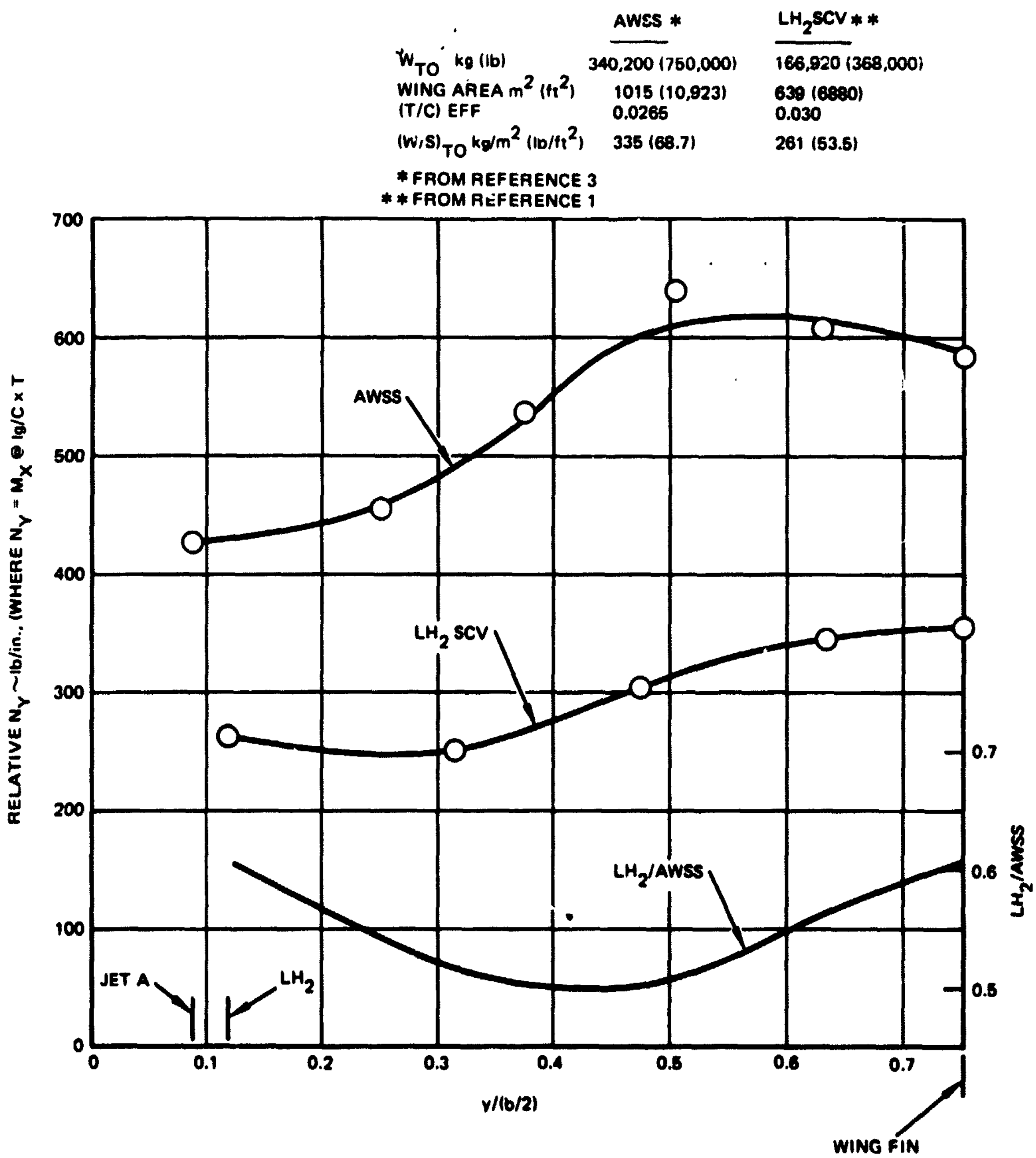


Figure 44. Comparison of Surface Loads, AWSS vs LH_2 SVC.

4. AIRCRAFT SYNTHESIS

Previous sections have described the basic propulsion, aerodynamic, and weight inputs to the ASSET aircraft synthesis program. This section describes the logic used in generation of the parametric data and selection of the final aircraft.

The optimization of an aircraft to meet the takeoff performance, noise, and landing approach speed constraints involves consideration of the following independent variables:

1. Wing loading

2. Thrust-to-weight

3. Optimum duct burning temperature used in take-off, climb and cruise.

In addition, the noise constraints of FAR 36, FAR 36 minus 5 dB, and FAR 36 minus dB, require selection of the take-off thrust level, the altitude, and amount of power cut back that are necessary. Further limitations are imposed by FAR 36, Section C36-7 which states that, for four-engined airplanes, no cutback is allowed below 213 m (700 ft) and that a minimum climb gradient of zero must be maintained with one engine out at this reduced power level. The highest thrust level possible during the ground run, while meeting the maximum sideline noise constraint, increases the altitude at the 6.48 km (3.5 n. mi.) flyover measuring point. Selection of the best balance between the takeoff thrust level and the power cut back, minimizes both sideline and flyover noise levels as the aircraft climbs out of the ground attenuation effect. Since the subject LH_2 fueled aircraft weighs less than 272,160 kg (600,000 lb), the allowable noise levels are a function of the gross weight and require an iterative procedure to meet the desired noise levels for each predicted weight. The procedure and results of the take-off noise analysis are further described for both the Mach 2.7 and 2.2 aircraft in following sections.

Table 12 illustrates the ten steps involved in the final selection process. In most cases, the sample curves shown are repeated in the following sections for both the Mach 2.7 and 2.2 aircraft.

5. MACH 2.7 AIRCRAFT

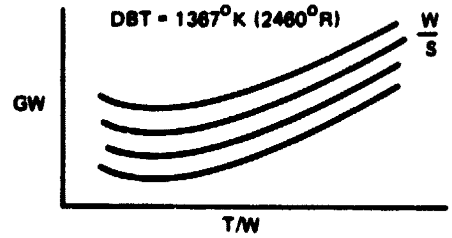
5.1 Configuration Description

The general arrangement of the liquid hydrogen (LH_2) fueled Mach 2.7 minimum gross weight airplane is shown in Figure 45. It is basically the same configuration as described for the CL 1701-7-1 airplane in Reference 1, the final report of the original study of LH_2 fueled supersonic transport aircraft

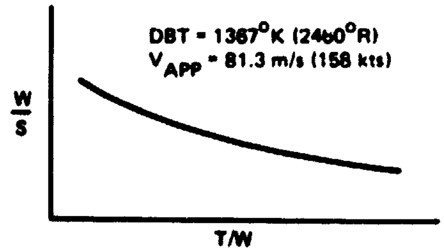
Table 12
PARAMETRIC STUDY LOGIC

(1.) SYNTHESIZE AIRCRAFT FOR SPECIFIED T/W AND W/S MATRIX THAT SATISFY FOLLOWING REQUIREMENTS:

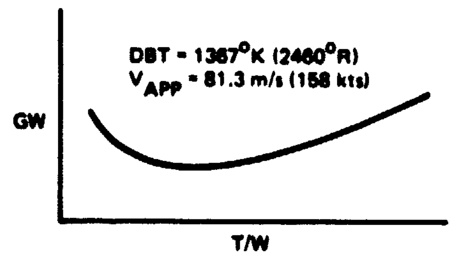
RANGE = 4200 n.mi.
PAYLOAD 49000 lbs
MAX DBT = 2460°R



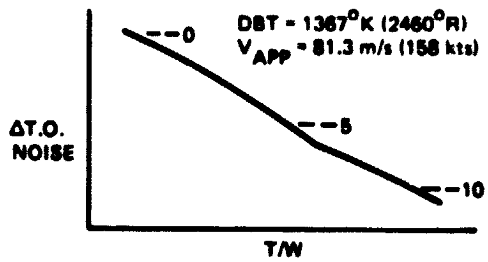
(2.) DETERMINE W/S REQUIRED FOR V_{APP} = 158 kts USING LANDING PERFORMANCE FROM (1).



(3.) SYNTHESIZE AIRCRAFT FROM (1) THAT SATISFY APPROACH CONSTRAINT, V_{APP} = 158 kts, USING W/S FROM (2).



(4.) DETERMINE MINIMUM TAKEOFF NOISE LEVELS FOR AIRCRAFT FROM (3).



(5.) SYNTHESIZE AIRCRAFT THAT SATISFY TAKEOFF NOISE CONSTRAINTS USING T/W FROM (4) AND W/S FROM (2).
T.O. NOISE = FAR 36
T.O. NOISE = FAR 36 – 5 EPNdB
T.O. NOISE = FAR 36 – 10 EPNdB

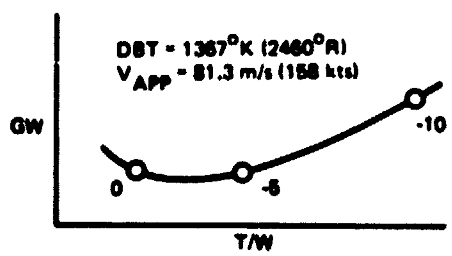
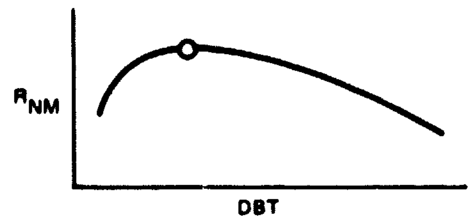
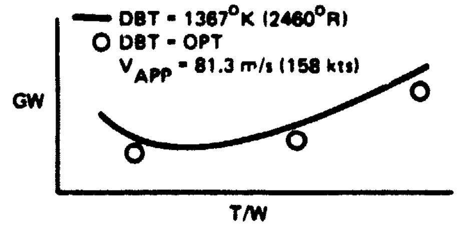


Table 12
PARAMETRIC STUDY LOGIC (Continued)

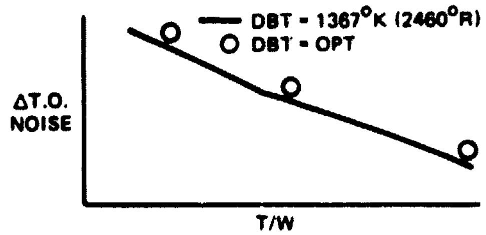
(6.) DETERMINE MAX DBT FOR AIRCRAFT FROM (5) THAT MAXIMIZE RANGE



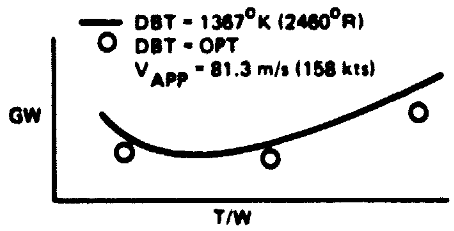
(7.) SYNTHESIZL AIRCRAFT USING T/W FROM (4), DBT FROM (6), AND W/S FOR V_{APP} = 81.3 m/s (158 kts)



(8.) REFINE T/W REQUIRED TO MEET TAKEOFF NOISE CONSTRAINTS USING AIRCRAFT FROM (7).

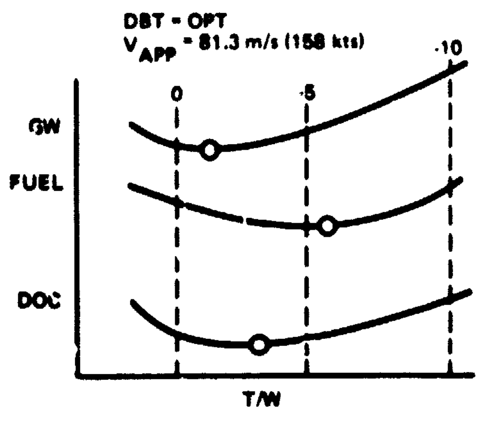


(9.) SYNTHESIZE AIRCRAFT USING T/W FROM (8), DBT FROM (6), AND W/S FOR V_{APP} = 81.3 m/s (158 kts)



(10.) SYNTHESIZE AIRCRAFT THAT SATISFY FOLLOWING REQUIREMENTS:

- MINIMUM TOGW, NOISE FAR 36
- MINIMUM FUEL, NOISE FAR 36
- MINIMUM DOC, NOISE FAR 36
- T.O. NOISE = FAR 36 – 5 EPNdB
- T.O. NOISE = FAR 36 – 10 EPNdB



performed by Lockheed-California Company for NASA-Ames Research Center. As illustrated in Figure 46, passengers are carried in the same double-deck arrangement located amidships as in the previous design. Liquid hydrogen fuel is contained in insulated, integral tanks located both forward and aft of the passenger compartment. The double-lobe cross section of the fuselage, which was found to be structurally and volumetrically efficient for both passenger seating and fuel containment purposes, was retained.

Except for the hydrogen tanks, the fuselage is basically conventional skin/stringer/frame type construction using titanium alloy reinforced with boron-polyimide in critical areas. The floor between the upper and lower passenger compartments is located between the cusps of the double-lobe cross section where it also serves as a tension tie to counteract the unbalanced pressure load between the two sides of the pressurized cabin.

The integral fuel tanks, which serve as both fuel containers and fuselage structure, are a welded structure of 2219 aluminum skin, stiffened with integral longitudiral stringers, and stabilized with circumferential frames. Approximately every 5.08 m (200 in.) along the length of the tank there is a diaphragm baffle to control fuel slosh. An aluminum-bonded honeycomb sandwich panel located between the cusps of the double-lobe tanks, similar to the floor in the passenger compartment, is used to react the unbalanced pressure loads and also to serve as a walk-way for routine inspection and maintenance of the tank. The tank ends are modified elliptical shapes to minimize the interconnect distance between the tanks and the adjacent structure. The interconnect structure is a truss framework using tubes made of fiberglass reinforced with boron filament.

The tank thermal protection system is a little different from that used in the previous study (see Section 3.3 for a detailed description). Basically it consists of a layer of closed-cell foam material bonded to the tank exterior surfaces for cryogenic insulation, and a fiberglass/polyimide honeycomb core faced with graphite/Kevlar/polyimide surfaces to serve the combined functions of heat shield and damage-resistant external surface of the airplane.

The wing has the arrow-planform and section prescribed by NASA-Langley Research Center for all of their supersonic cruise vehicle advanced technology studies. The structural arrangement is identical to that evolved from the study of Reference 3, modified to account for the

- smaller wing area,
- lower wing loading, and
- elimination of fuel in the wing (no load relief)

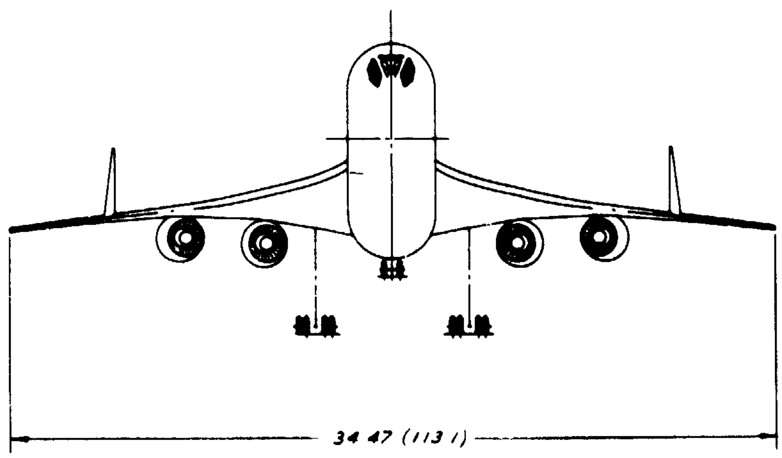
which are characteristic of the differences between Jet A and LH_2 fueled aircraft. The thin, flexible, highly swept and cambered wing is carried as a continuous structure under the fuselage, except at the forward apex.

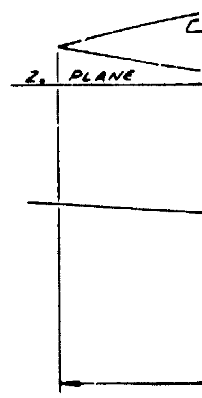
CHARACTERISTICS	WING	HORIZ TAIL	FUS VERT. TAIL	WING VERT. TAIL (EA)
AREA M² (SQ FT)	73874 (7952)	34 09 (367)	17.74 (191.0)	15.39 (165 7)
ASPECT RATIO	1.607	1.707	0.517	0 517
SPAN M (FT)	34.47 (113.1)	7.62 (25.0)	3 02 (9.9)	2.83 (9.3)
ROOT CHORD M(IN)	47 63 (1875 2)	7.29 (287 2)	9 53 (375 2)	9 09 (358 0)
TIP CHORD M(IN)	5 49 (216.0)	1.64 (64.6)	2 19 (86 3)	1.82 (71.6)
TAPER RATIO	0.1135	0.225	0 23	0 20
MAC M (IN)	29.29 (1153.3)	5 06 (199 4)	6.62 (260.69)	6 26 (246 6)
SWEEP-RADIAN (DEG)	1 292 (74)	1.058 (60.64)	1.190 (68.2)	1 281 (73 42)
RADIAN (DEG)	1.236 (70.84)	—	—	—
RADIAN (DEG)	1.047 (60)	—	—	—

DESIGN GROSS WEIGHT - 179,133 KG. (394,914 LBS)

POWER PLANT - SCV LH₂ M2.7 DUCT BURNING TURBOFAN
UNINSTALLED THRUST - 234,939 NEWTONS (52,819 LBS)

PASSENGERS - 234





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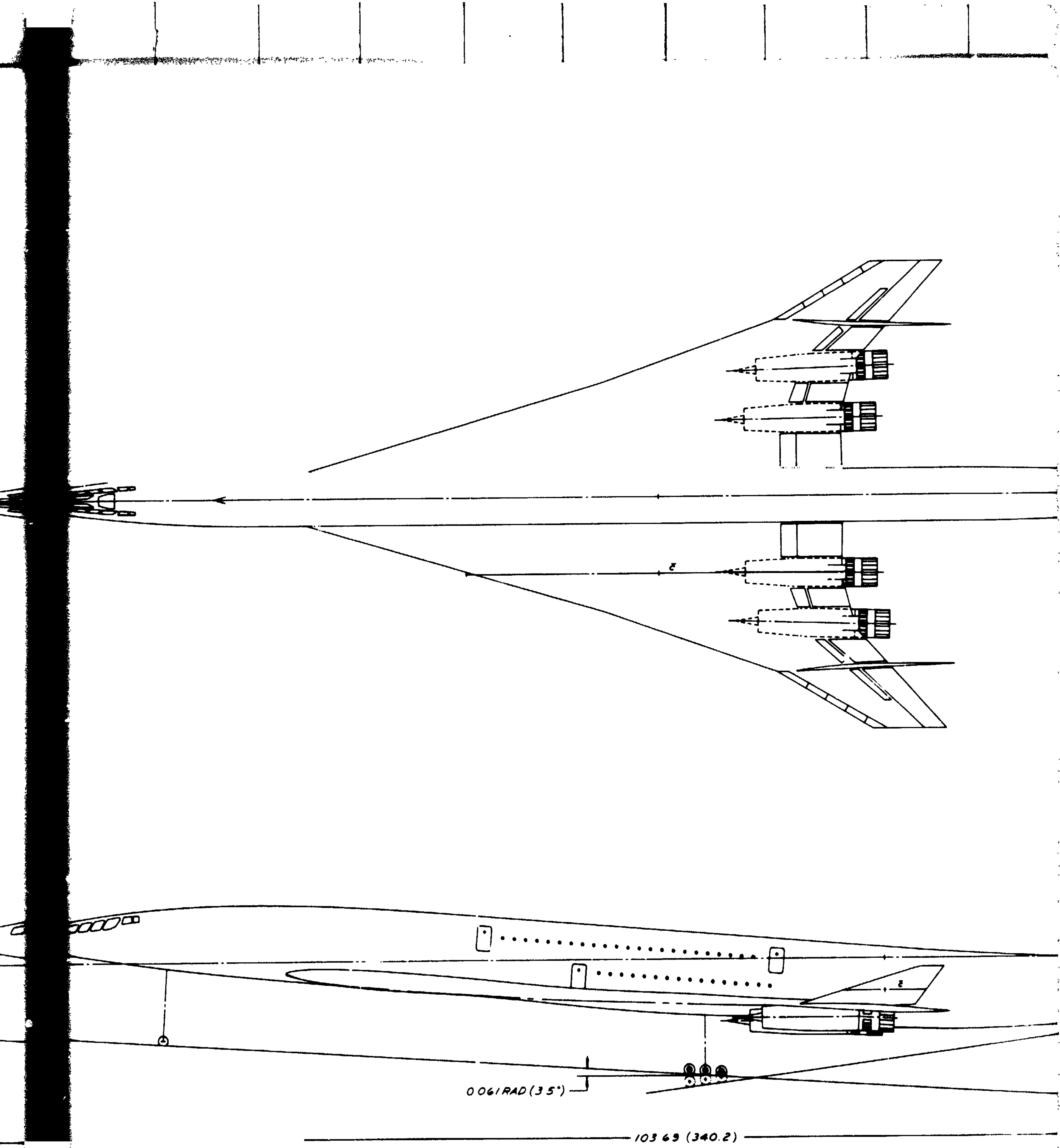
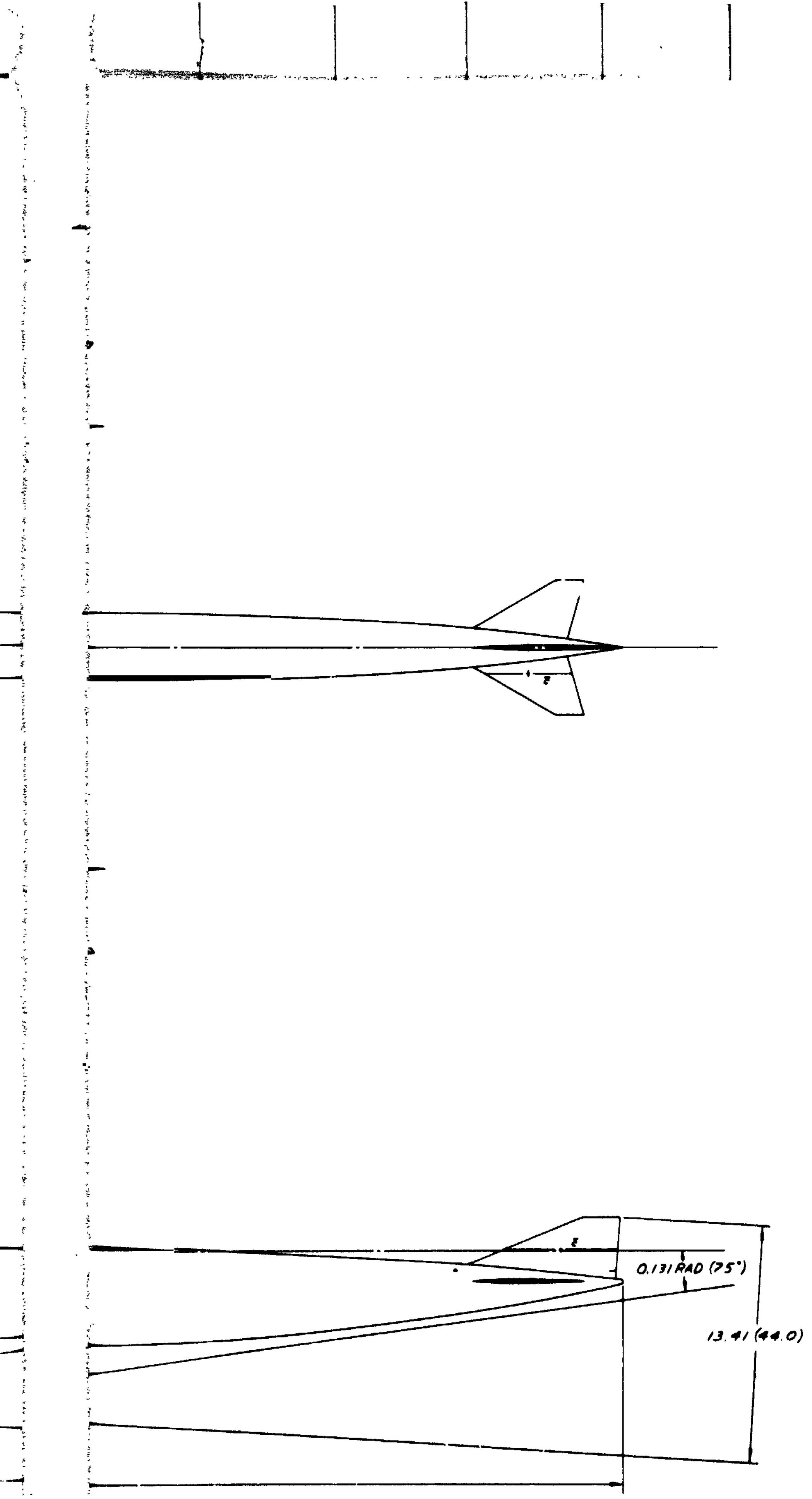
FOLDOUT FRAME

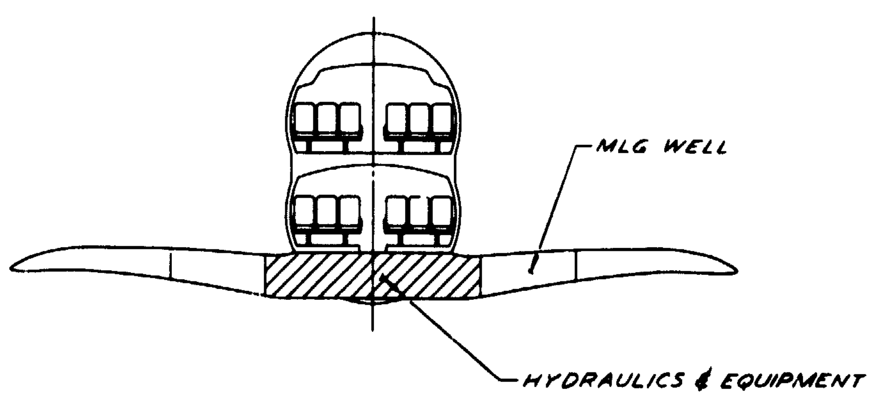


FOLDOUT FRAME 2



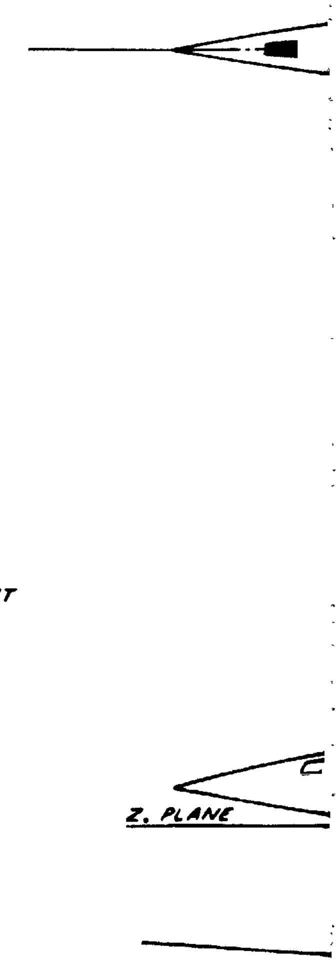
2. DIM. IN METERS (FEET), OR NOTED
1. THIS DESIGN DEVELOPED ON COMPUTER GRAPHICS, DWG NO CL 1701-11, 3V1, 3V2, & 3V3
NOTE:

Figure 45. General Arrangement - M2.7 LH_2 SCV



SECTION A-A

SCALE: 1/50



FOLDOUT FRAME 1

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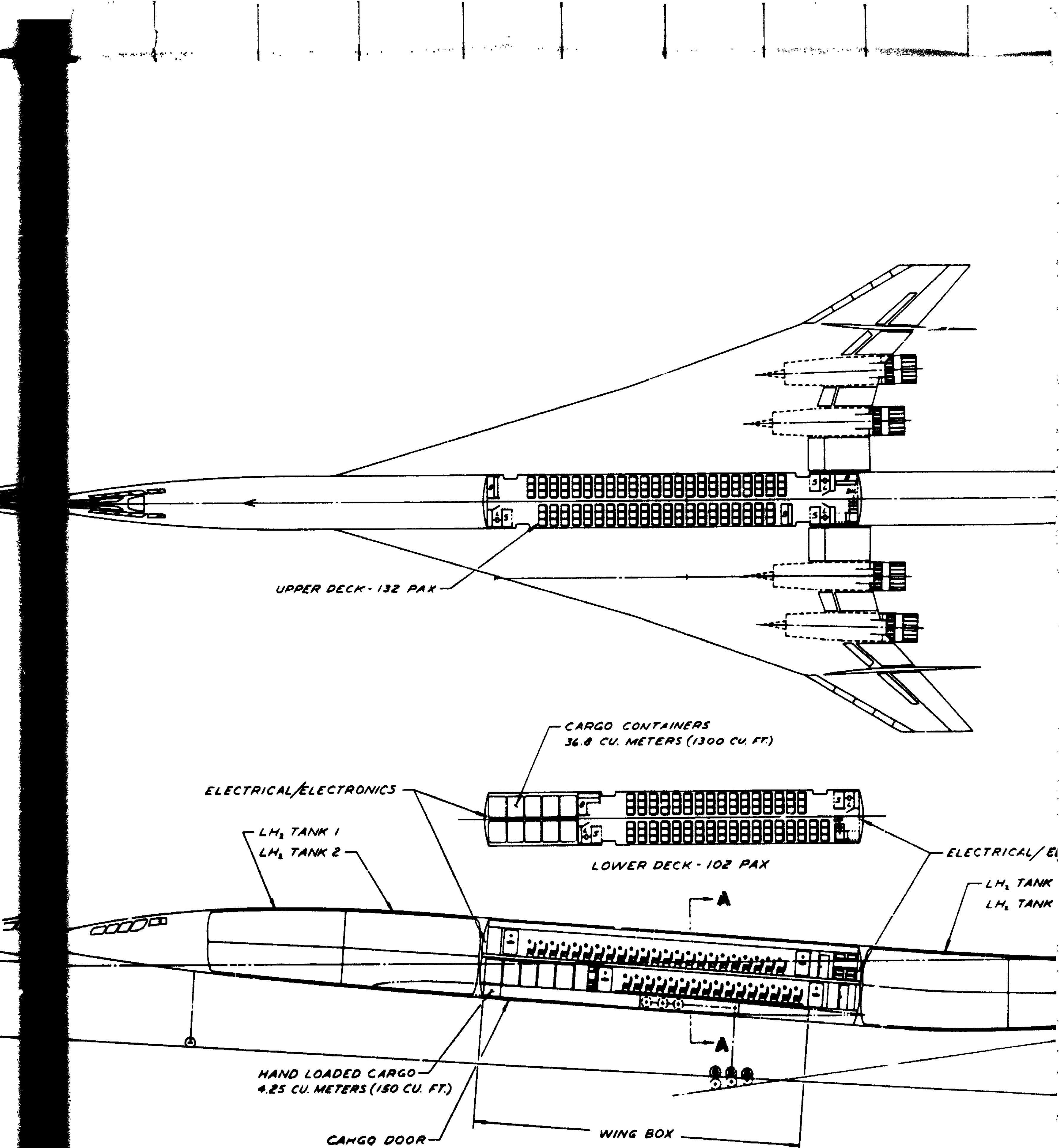
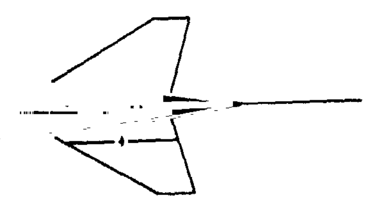
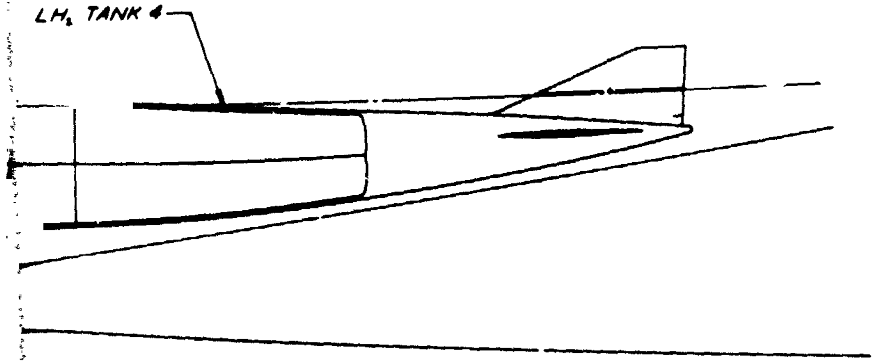

UPPER DECK - 132 PAX
CARGO CONTAINERS
36.8 CU. METERS (1300 CU. FT.)
ELECTRICAL/ELECTRONICS
LH₂ TANK 1
LH₂ TANK 2
LOWER DECK - 102 PAX
ELECTRICAL/E
LH₂ TANK
LH₂ TANK
A
A
HAND LOADED CARGO
4.25 CU. METERS (150 CU. FT.)
CARGO DOOR
WING BOX





NOTE:

1. THIS DESIGN DEVELOPED ON COMPUTER GRAPHICS, DWG. NO CL 1701-11, IP1, IP2, & IP3
2. S - COATS/STORAGE
   L - LAVATORIES
   B - BUFFET

Figure 46. Interior Arrangement - M2.7 LH_2 SCV

FOLDOUT FRAME 3

The wing skin is titanium alloy and its structural framework is a series of spanwise beams located approximately 20 in. apart throughout the main load-carrying area. The beams are extruded titanium alloy spar caps, reinforced with boron-polyimide, to which are welded titanium tubes to form a trusswork. The outer wing panels are a titanium faced, titanium core, aluminum brazed honeycomb.

The empennage structure is similar to that of the wing outer panels.

Flight control and high lift devices for the CL 1701-9-1, as shown in Figure 45, are the same as those for the CL 1701-7-1 aircraft from Reference 1. 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 for high speed directional stability. The outer wing includes ailerons for roll control at low speed 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 turbo-fan engines, each with 234,940 N (52,820 lb) of uninstalled thrust, are mounted in underwing pods. The engines are equipped with axisymmetric inlets and thrust reversers.

5.2 Parametric Data Results

Figures 47 and 48 show the original matrix of 40 aircraft in terms of gross weight and fuel consumption for various thrust-to-weights and wing loadings with a maximum duct burning temperature (DBT) of 1367° K (2460° R). The dashed line indicates the locus of those aircraft meeting the maximum approach speed of 81.3 m/s (158 KEAS) which is determined by the block fuel consumption and the take-off wing loading. Figure 49 shows the effect of various fuel prices on DOC for aircraft meeting the 81.3 m/s constraints and indicates a very slight shift in the optimum T/W for minimum DOC. From these plots three preliminary aircraft meeting FAR 36 can be selected; one each for minimum gross weight, minimum fuel and minimum DOC.

Takeoff jet noise was determined for a parametric family of aircraft having engines designed for a maximum duct burning temperature of (1367° K (2460° R) and wing sized to meet the landing approach speed limit. The predicted noise does not include the effects of aerodynamic, burner, compressor or fan noise sources. The jet noise suppression used for the analysis was taken from Figure 29. The thrust setting at brake release, the thrust setting at cutback, the aircraft height at cut-back used for takeoff noise abatement, and the maximum climb and cruise DBT's are presented in Figure 50 for parametric

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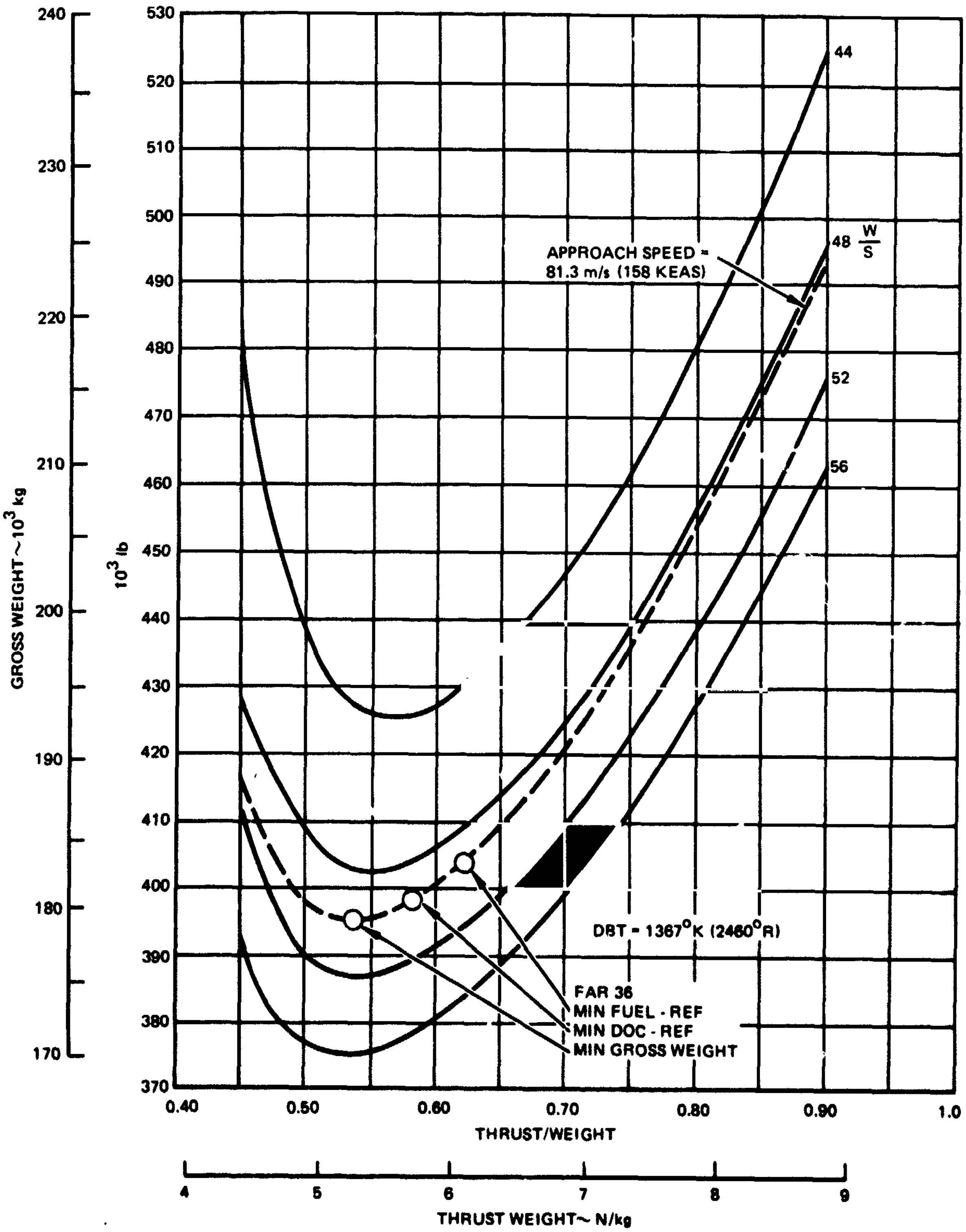

Figure 47. T/W versus Gross Weight - M2.7 LH_2 SCV.

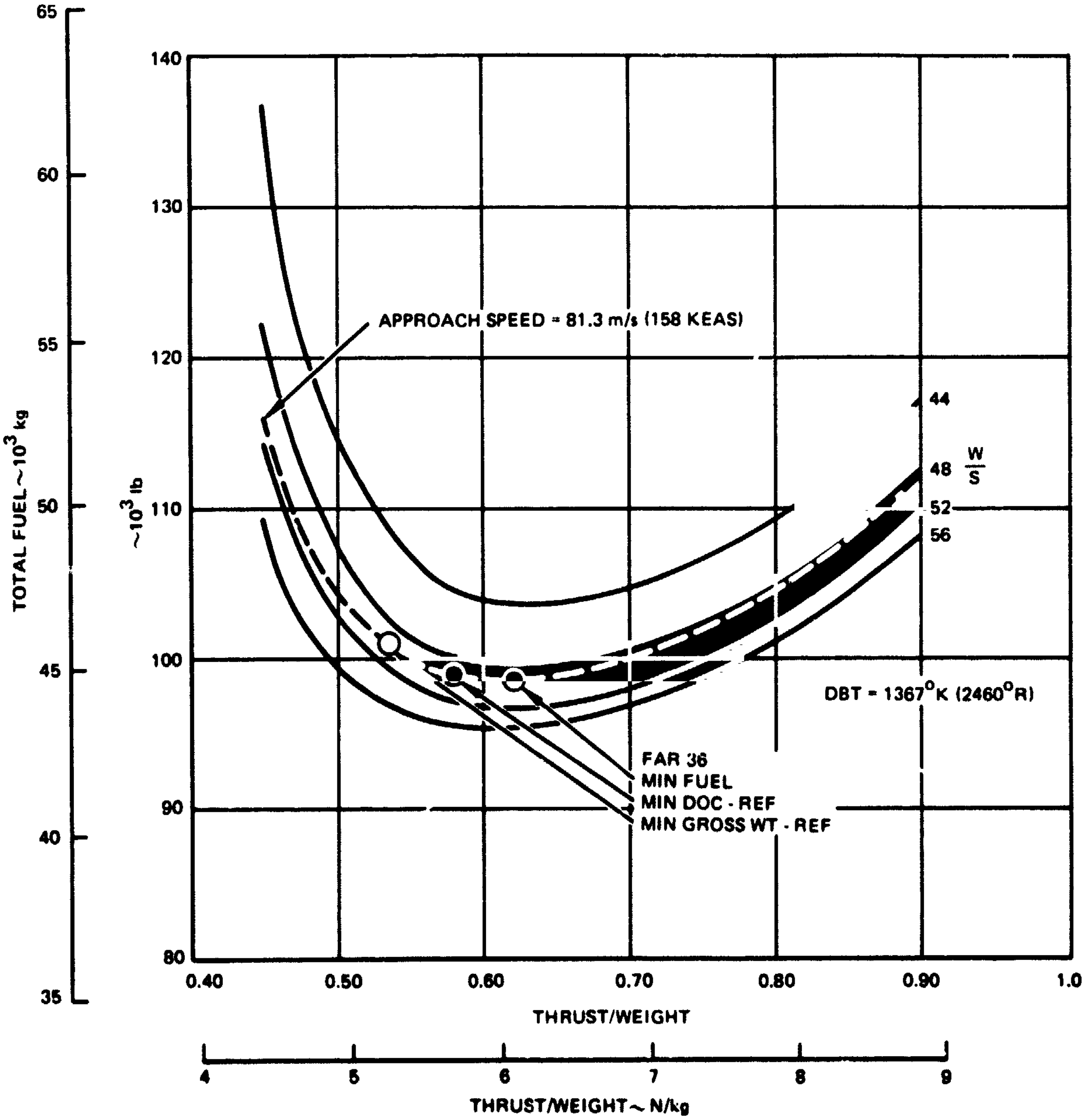

Figure 48. T/W versus Total Fuel Weight - M2.7 LH_2 SCV.

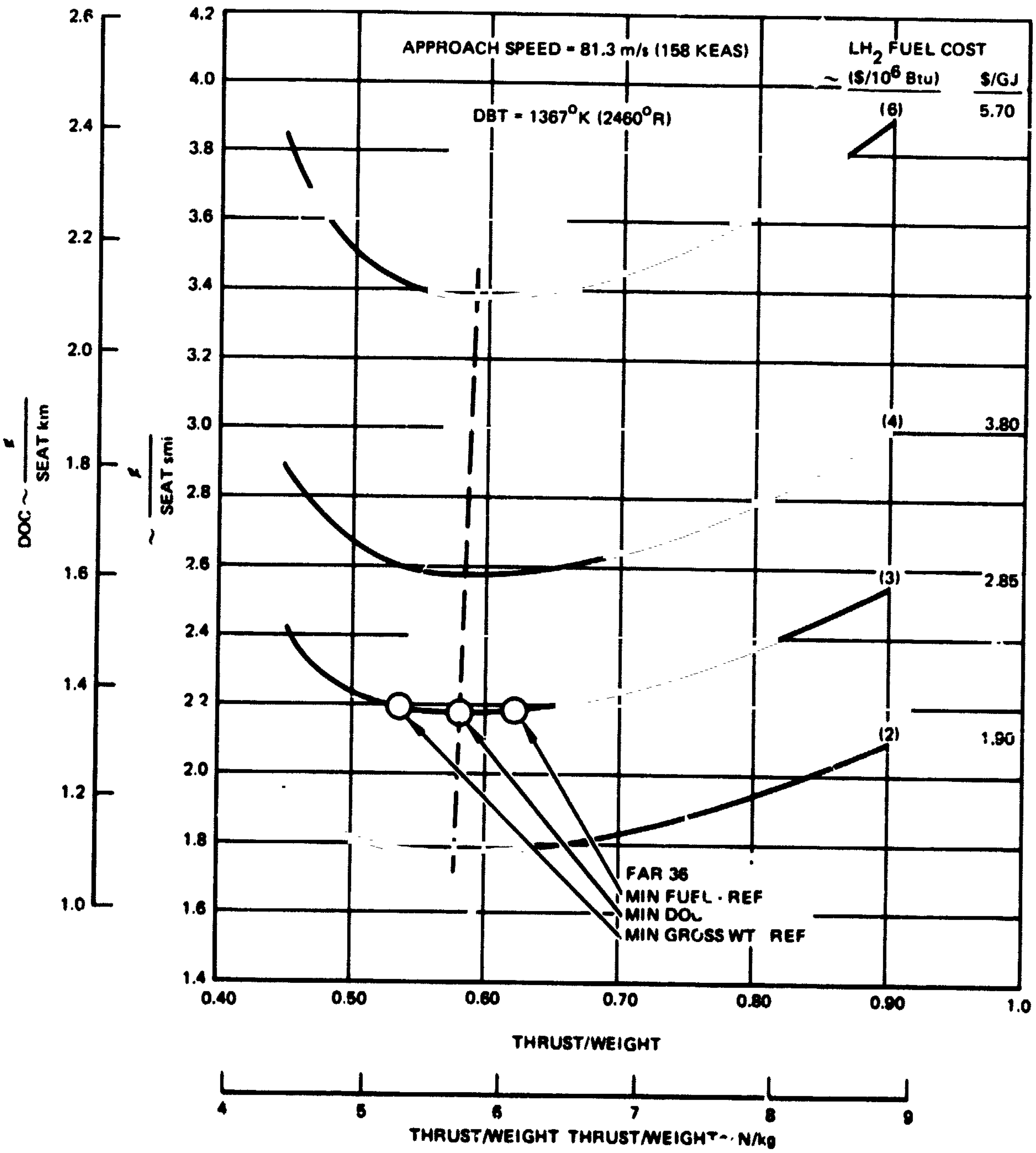

Figure 49. DOC versus LH_2 Fuel COst - M2.7 LH_2 SCV.

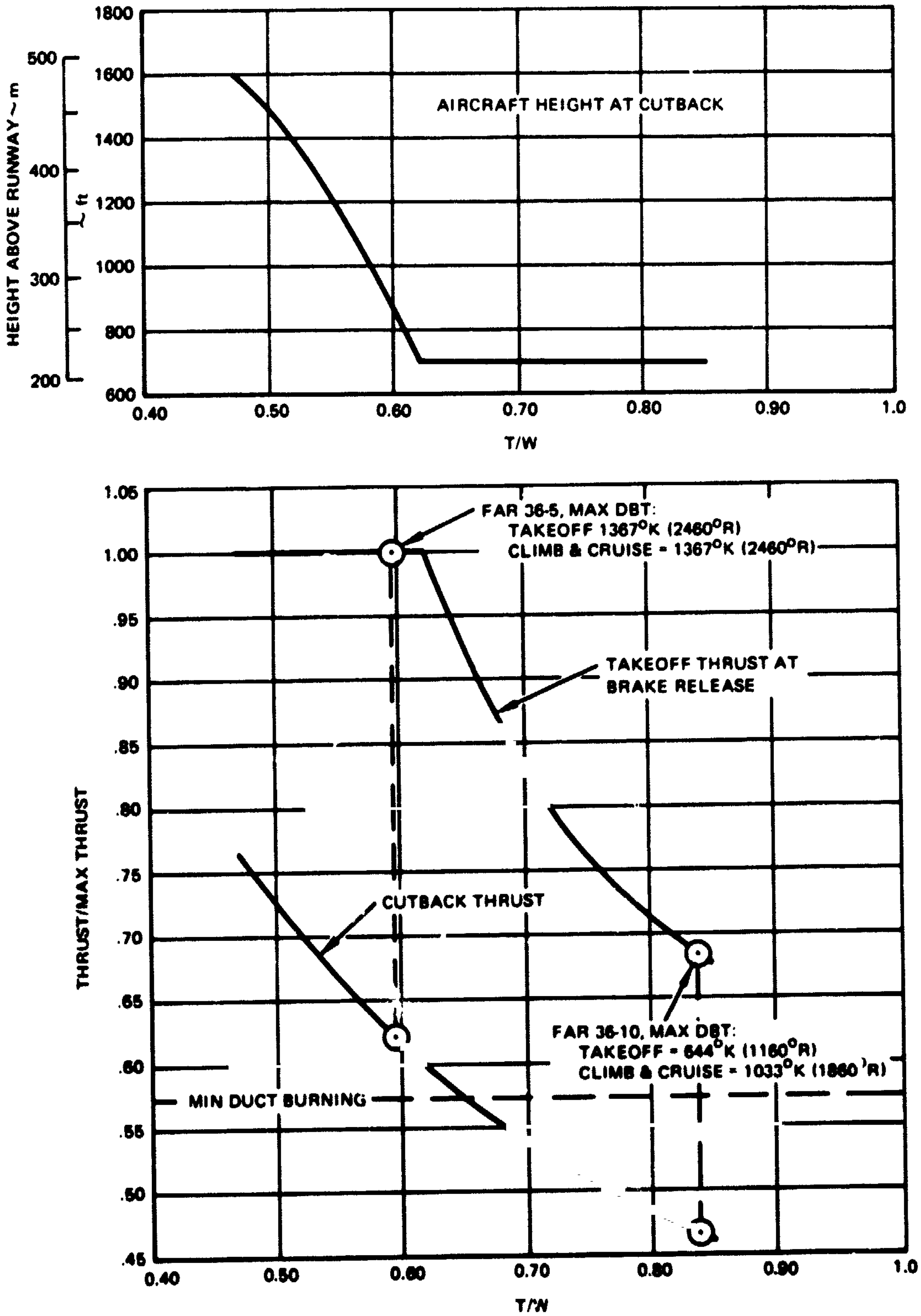

Figure 50. Takeoff Noise Abatement Procedure - M2.7 LH_2 SCV.

family of aircraft. The thrust setting at cut-back provides a zero climb gradient with one engine inoperative. For aircraft with T/W less than 0.62, maximum thrust was used prior to cut-back. The cut-back height was selected to match the FAR 36 sideline and flyover noise decrements. For aircraft with T/W greater than 0.62, the height at cut-back was held at 213 m (700 feet) and the thrust setting at brake release was adjusted to match the FAR 36 sideline and flyover noise decrements. The resulting matched FAR 36 sideline and flyover noise decrements are presented in Figure 51 for the parametric family of aircraft. The discontinuity in the curve reflects the discontinuance of duct burning. Below the break, no duct burning occurs.

Figure 52 shows the range of the FAR 36-10 aircraft (T/W = 0.834) when the mission is flown with verious levels of maximum duct burning temperature (DBT). From this plot, the optimum DBT of 1033° K (1860° R) was selected and the aircraft resized to a 7783 km (4200 n.mi.) range to produce the final FAR 36-10 aircraft.

Table 13 summarizes the characteristics of the final five selected aircraft. The sensitivity of the Mach 2.7 aircraft to noise reduction shows that up to 5 EPNdB can be met with essentially no penalty in terms of the critical DOC parameter and with very little increase in gross weight or aircraft price. A reduction to -10 EPNdB will penalize the gross weight by 14 percent, the DOC by 10 percent and the price by 19 percent relative to the -5 EPNdB aircraft because of the high thrust (engine weight) required to allow the power cutback (47%) necessary to meet the noise constraint. Comparison of the data also indicates that there is very little difference between the first four aircraft. The maximum gross weight spread is only 2.2 percent while the block fuel consumption is 2.6 percent. DOC spread is less than 0.8 percent with the minimum DOC aircraft a good compromise between minimum fuel and minimum gross weight. It should be noted that no power cutback or throttling during ground run was required of these aircraft to meet or better the standard FAR Part 36 noise constraint. Cutback and throttling is required, however, for the minus 10 dB constraint and the optimum DBT is reduced to 644° K (1160° R).

Examination of Table 13 shows the thrust to weight ratios selected for minimum gross weight (0.535) and minimum DOC (0.580) provide aircraft which are quieter than FAR 36 by -2.75 and -4.43 EPNdB, respectively. Therefore, the FAR 36 constraint is not critical for aircraft selected to these two criteria. Minimum fuel weight is critical with regard to FAR 36-5 and the thrust-to-weight required (0.620) exceeds that noise specification by 0.95 EPNdB. Aircraft designed to meet FAR 36-10 are noise critical and require a thrust-to-weight of 0.838 in order to allow the power cutback necessary to meet the -10 EPNdB constraint.

Figure 53 shows the c.g. travel of the Mach 2.7 aircraft with the desired c.g. at 51 percent of the M.A.C in the mid-cruise weight range.

5.3 Sensitivity Analysis

The minimum gross weight, FAR 36, Mach 2.7 LH_2 vehicle was perturbed on the basis of range, empty weight, SFC, drag, and payload to determine its

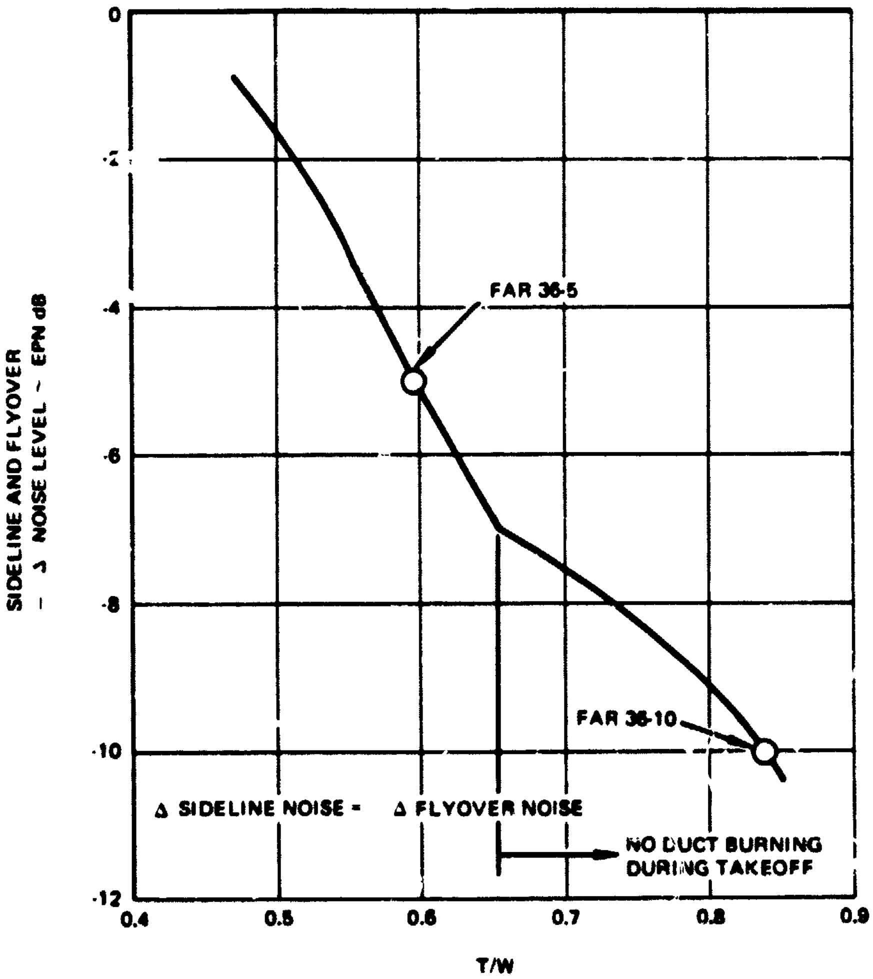

Figure 51. FAR 36 Takeoff Noise Decrement - M2.7, LH_2 SCV.

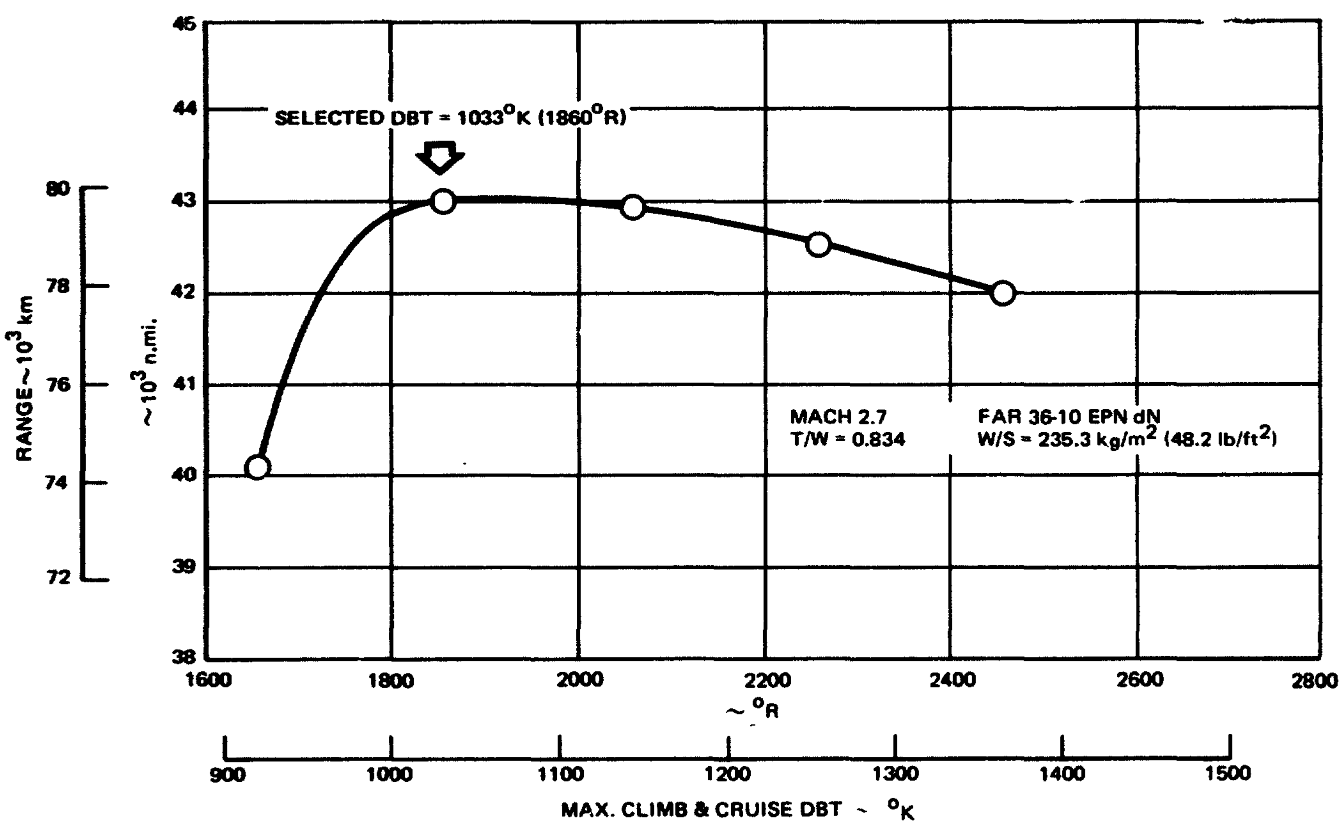

Figure 52. Selection of Maximum DBT for Optimum Climb and Cruise - M2.7 Far 36-10 LH_2 SCV.

C.2

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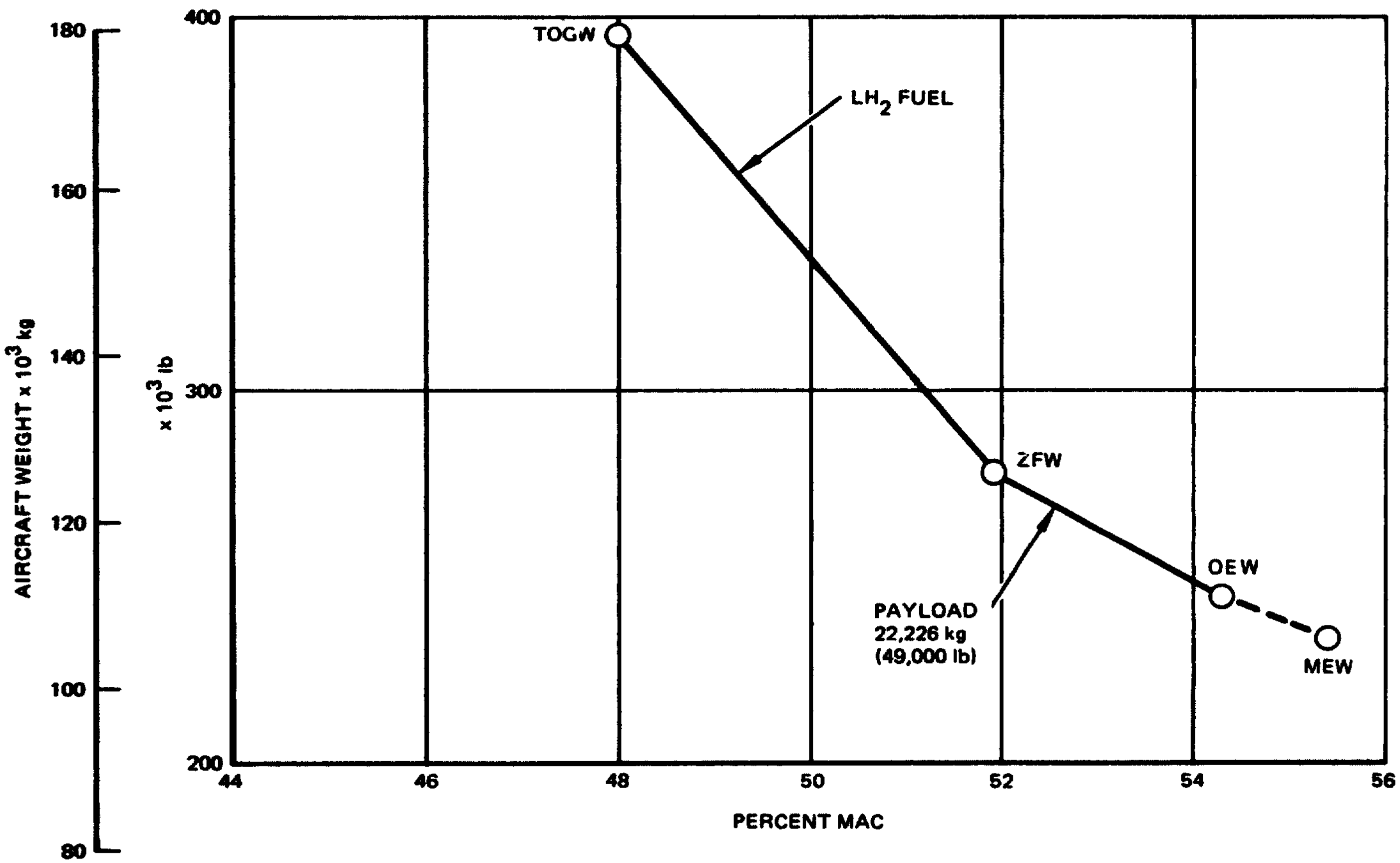

Figure 53. Center of Gravity Travel - M2.7 LH_2 SCV.

Table 13. Mach 2.7 LH_2 SCV - Aircraft Comparison
7783 km Range - 22,226 kg Payload
Fuel Cost = $2.85/GJ
(S.I. Units)

		Minimum Fuel Wt	Minimum DOC	Minimum Gross Weight		
				FAR 36	FAR 36-5	FAR 36-10
Gross Wt. - Ref	kg	182,990	180,400	179,130	181,265	206,550
Block Fuel Wt	kg	37,740	37,950	38,735	37,835	39,970
DOC	$\frac{¢}{\text{seat km}}$	1.356	1.350	1.360	1.351	1.487
Airplane Price	$\$10^6$	47.44	46.36	45.50	46.74	55.63
Wing Loading	kg/m^2	239.2	240.7	242.6	240.2	235.8
Thrust/Weight (DBT = 2460° R)	N/kg	6.080	5.687	5.246	5.835	8.217
Maximum DBT - Climb and Cruise	°R	2460	2460	2460	2460	1860
Maximum DBT - Takeoff	°R	2460	2460	2460	2460	1160
Wing Area	m^2	765	749	740	754	876
Span	m	35.1	34.7	34.5	34.8	37.5
Fuselage Length	m	102.8	103.0	103.7	102.9	105.3
Landing Approach Speed	m/s					
FAR T.O. Field Length	m	1597	1704	1853	1661	1740
FAR Landing Field Length	m	2387	2384	2377	2387	2411
Average Cruise L/D		7.53	7.49	7.42	7.51	7.60
Average Cruise SFC	$\frac{kg}{hr}$/daN	0.572	0.578	0.585	0.576	0.542
Average Cruise Alt.	m	21,640	21,640	21,340	21,640	21,340
Structure Wt*	kg	66,980	65,970	65,430	66,310	75,873
Propulsion Wt**	kg	30,310	28,580	26,990	29,210	41,700
Equip. and Furn Wt	kg	13,710	13,690	13,690	13,700	13,970
Empty Wt	kg	111,010	108,250	106,110	109,210	131,535
Std. + Operating Items	kg	5,100	5,100	5,130	5,100	5,260
Operating Empty Wt	kg	116,100	113,350	111,240	114,310	136,790
Payload	kg	22,226	22,226	22,226	22,226	22,226
Zero Fuel Wt	kg	138,330	135,570	133,465	136,540	159,020
Total Fuel	kg	44,660	44,830	45,670	44,730	47,540
Take-off Gross Wt	kg	182,990	180,400	179,130	181,265	206,550
Sideline Noise $\frac{\text{Actual}}{\text{FAR 36}}$	EPNdB	$\frac{100.91}{106.86}$	$\frac{102.38}{106.81}$	$\frac{104.04}{106.79}$	$\frac{101.82}{106.82}$	$\frac{97.20}{107.20}$
Flyover Noise $\frac{\text{Actual}}{\text{FAR 36}}$	EPNdB	$\frac{99.19}{105.14}$	$\frac{100.60}{105.03}$	$\frac{102.23}{104.98}$	$\frac{100.06}{105.06}$	$\frac{96.01}{106.01}$
ΔNoise Reduction (from FAR 36)	EPNdB	-5.95	-4.43	-2.75	-5	-10
Energy Utilization	$\frac{kJ}{\text{seat km}}$	2485	2500	2551	2492	2632

*Includes LH_2 tank weight.
**Includes insulation and heat shield weight.

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Table 13. Mach 2.7 LH_2 SCV - Aircraft Comparison (Continued)
4200 n.mi. Range - 49,000 lb Payload
Fuel Cost = $3/10^6$ Btu (15.48¢/lb)
(U.S. Customary Units)

		Minimum Fuel Wt	Minimum DOC	Minimum Gross Weight		
				FAR 36	FAR 36-5	FAR 36-10
Gross Wt - Ref	lb	403,410	397,710	394,910	399,615	455,360
Block Fuel Wt	lb	83,190	83,670	85,390	83,410	88,110
DOC	¢/seat smi	2.182	2.173	2.189	2.175	2.392
Airplane Price	$\$10^6$	47.44	46.36	45.50	46.74	55.63
Wing Loading	lb/ft^2	49.0	49.3	49.7	49.2	48.3
Thrust/Weight (DBT = 2460° R)	-	0.620	0.580	0.535	0.595	0.838
Maximum DBT - Climb and Cruise	°R	2,460	2,460	2,460	2,460	1,860
Maximum DBT - Takeoff	°R	2,460	2,460	2,460	2,460	1,160
Wing Area	ft^2	8,235	8,067	7,962	8,121	9,432
Span	ft	115.	113.9	113.1	114.3	123.1
Fuselage Length	ft	337.4	337.9	340.2	337.6	345.6
Landing Approach Speed	KEAS	158	158	158	158	158
FAR T.O. Field Length	ft	5,240	5,590	6,080	5,450	5,710
FAR Landing Field Length	ft	7,830	7,820	7,800	7,830	7,910
Average Cruise L/D	-	7.53	7.49	7.42	7.51	7.60
Average Cruise SFC	lb/hr/lb	0.562	0.568	0.575	0.566	0.532
Average Cruise Alt	ft	71,000	71,000	70,000	71,000	70,000
Structure Wt*	lb	147,660	145,440	144,255	146,180	167,270
Propulsion Wt**	lb	66,830	63,020	59,500	64,390	91,920
Equip. and Furn Wt	lb	30,230	30,180	30,170	30,200	30,790
Empty Wt	lb	244,725	238,640	233,930	240,770	289,980
Std. + Operating Items	lb	11,240	11,240	11,300	11,240	11,585
Operating Empty Wt	lb	255,960	249,880	245,235	252,010	301,570
Payload	lb	49,000	49,000	49,000	49,000	49,000
Zero Fuel Wt	lb	304,960	298,880	294,235	301,010	350,570
Total Fuel	lb	98,450	98,830	100,675	98,610	104,740
Take-off Gross Wt	lb	403,410	397,710	394,910	399,615	455,360
Sideline Noise Actual/FAR 36	EPNdB	100.01/106.86	102.39/106.81	104.04/106.79	101.82/106.82	98.20/107.20
Flyover Noise Actual/FAR 36	EPNdB	99.19/105.14	100.60/105.03	102.23/104.98	100.06/105.06	96.01/106.01
ΔNoise Reduction (from FAR 36)	EPNdB	-5.95	-4.43	-2.75	-5	-10
Energy Utilization	Btu/seat nmi	4,368	4,393	4,483	4,379	4,626

*Includes LH_2 tank weight.
**Includes insulation and heat shield weight.

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sensitivity to each of these factors. Figures 54 through 59 show the results of these excursions, together with sensitivity factors at the design point, where appropriate, on gross weight, DOC, price, and total fuel weight.

Figure 54 examines the growth of the point design aircraft on the basis that the design mission range was increased. To accommodate the increased fuel required the fuselage was allowed to grow in length. In each case the vehicle is resized and the constraints of approach speed and noise held constant, Since the landing wing loading is held constant to meet the approach speed, the takeoff wing loading can be increased slightly as more mission fuel is consumed. FAR 36 allows increasing takeoff and flyover noise as gross weight is increased, which results in a slightly higher allowable jet velocity. The result is that the turbofan engine power can be reduced. More usable thrust allows a slight decrease in the installed thrust-to-weight ratio. This slightly increases the takeoff field length but it remains well within the 3,200 m (10,500 ft) constraint. The result of this study shows that the design range of the Mach 2.7 LH_2 vehicle can be greatly extended with a reasonable increase in gross weight (a 28 percent increase for a 2,224 km (1,200 n.mi.) range increment). For convenience, the sensitivity of each of the characteristics around the design point, indicated by the circle on the plots, is listed. For example, the plot of gross weight versus range indicates a growth of about 34 kg (74 lb) in gross weight would be required for every nautical mile increase in design range.

Figures 55 and 56 illustrate the effect of a change in empty weight as would be the case if equipment or structural weight were to increase or decrease from the original target weight. Two different situations were examined. In Figure 55, the assumption is that the vehicle design has not been frozen and the option exists to resize the vehicle to accomplish the original mission. This might be the case if, for example, the target wing weight were exceed by 4,536 kg (10,000 lb) at the original design gross weight. This causes a subsequent increase in fuel, propulsion, structure, etc. and finally a further increase in the wing itself to maintain the vehicle performance. The sensitivity or growth factor shown is about 1.38 kg (3.05 lb) of gross weight per kg (lb) of original empty weight change. The sensitivity of DOC, price and fuel required is also shown. Figure 56 assumes that the design gross weight has been frozen and that the fuel available (and fuel volume) must be adjusted to reflect the change in empty weight. The result is a change of about 0.153 km/kg (0.0374 n.mi. per pound) of empty weight change. DOC, and price and fuel sensitivities are also shown.

Figure 57 shows the effect of a uniform change in engine specific fuel consumption (SFC) on total range and DOC. In the range tradeoff the vehicle is not resized but flies at different ranges as the fuel consumption is varied. This is a significant sensitivity and allows an increase of 101 km (54.5 n.mi.) with each 1 percent decrease SFC. The DOC tradeoff is shown to be much less sensitive.

Figure 58 is simply the increase in range which would be possible if payload if off-loaded. The increase is about 0.066 km/kg (0.0162 n.mi. per lb) of payload. When no payload is carried, the airplane has a range capability

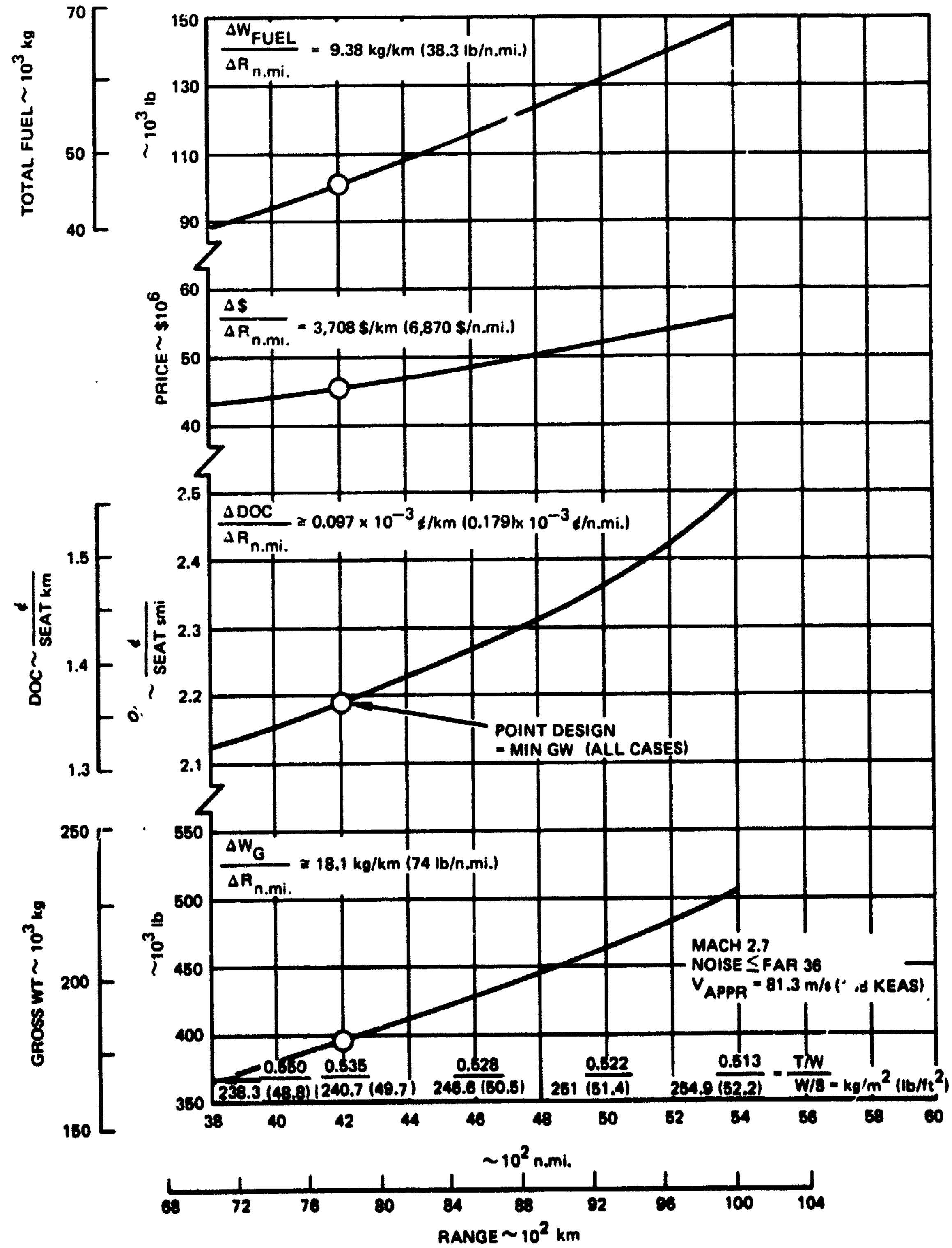

Figure 54. Range Sensitivity - M2.7 LH_2 SCV.

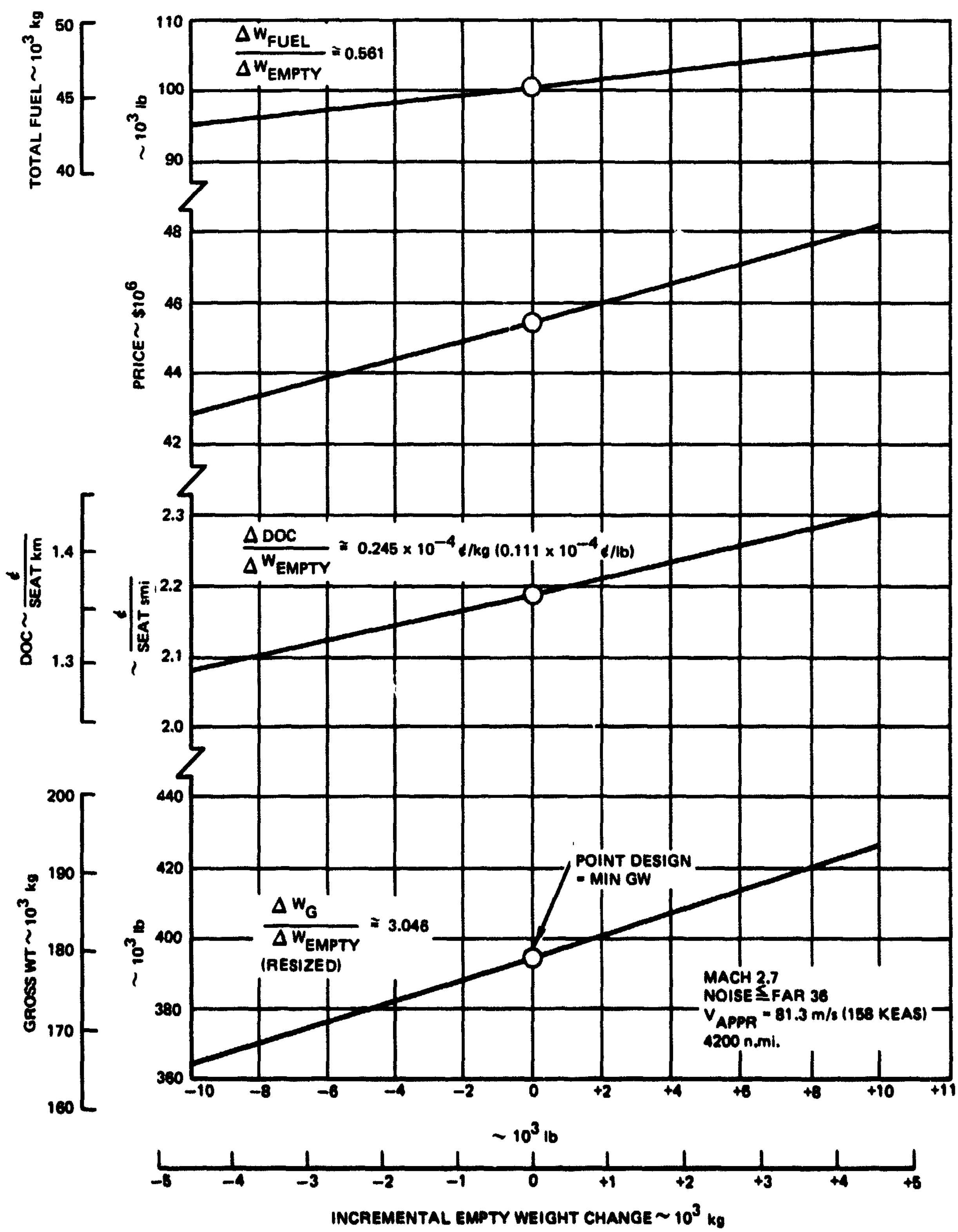

Figure 55. Empty Weight Change Sensitivity - M2.7 LH_2 SVC (aircraft resized).

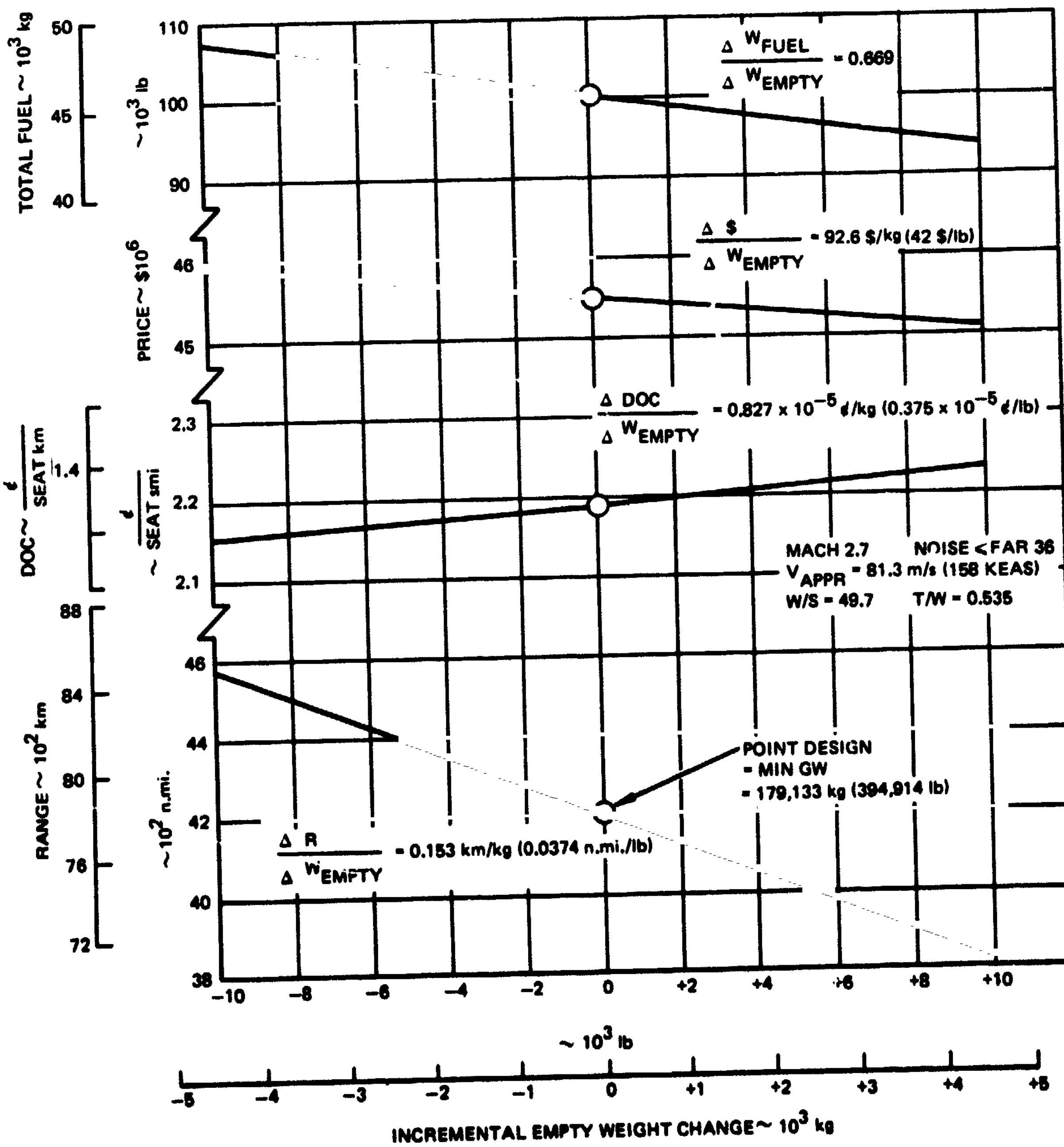

Figure 56. Empty Weight Change Sensitivity - M2.7 LH_2 SCV (constant gross weight).

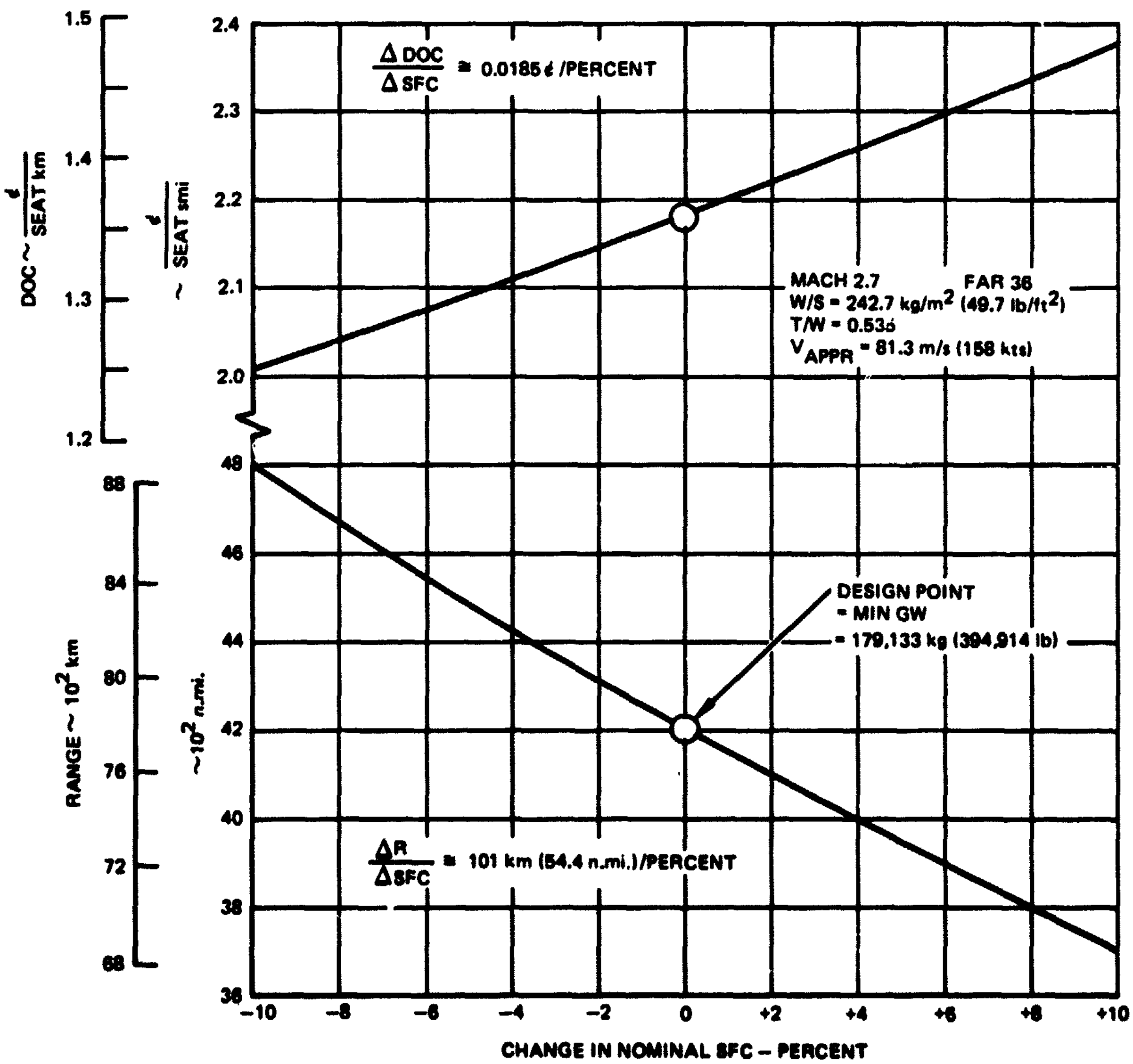

Figure 57. Sensitivity to SFC - M2.7 LH_2 SCV.

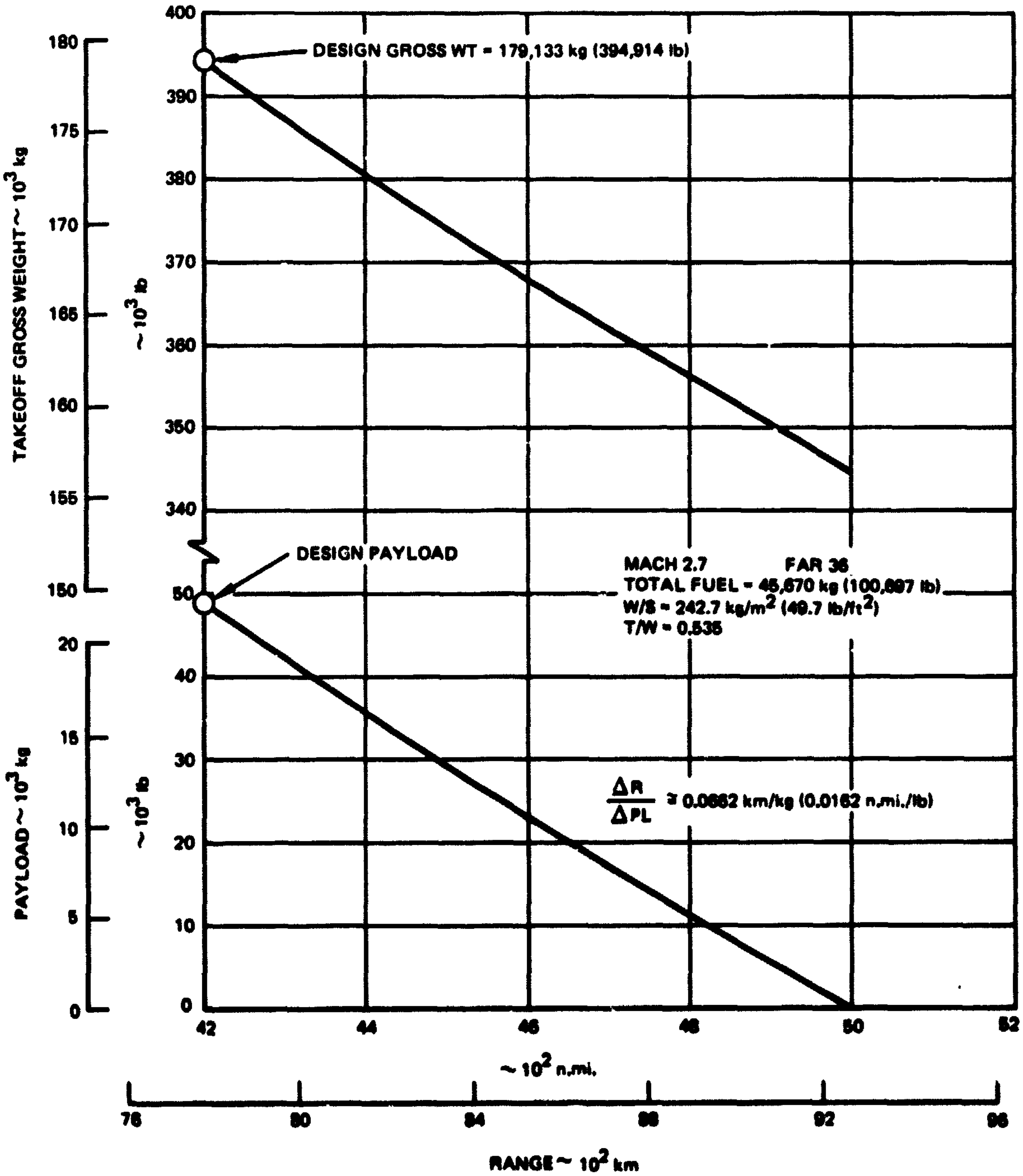

Figure 58. Range Sensitivity to P :load - M2.7 LH_2 SCV.

of 9,256 km (4,995 n.mi.). It should be noted that as designed, the point design vehicle is fuel volume limited and no additional fuel can be added as the payload is reduced as is the case for the conventional, hydrocarbon fueled aircraft. In the real world, the advisability of carrying extra tankage to increase flexibility would be a matter of route structure and economics. The method of construction of the vehicle would allow enlargement of the tanks by a simple fuselage plug within the limits of aircraft strength and the wing area selected.

Of equal importance to engine specific fuel consumption is the drag level. Figure 59 shows a change of about 105.6 km (57 n.mi.) distance and 0.021¢ in DOC for each drag count. The analysis assumed that the change in nominal drag was applied uniformly to the zero-lift frag at all Mach numbers. For reference, the nominal drag level of the M2.7 LH_2 SCV in cruise is about 157 counts.

5.4 Comparison With Jet A Design

One of the study objectives was to provide designs of LH_2 fueled SCV's which could be compared directly with equivalent hydrocarbon (Jet A) fueled versions. The ground rules of the subject study were modified to provide a comparable basis for design with the Jet A fueled SCV developed under Contract NAS1-12288 (Reference 2). Table 14 presents a number of relevant factors to compare characteristics of aircraft designed to carry a payload of 22,226 kg (49,000 lb) (234 passengers) 7778 km (4200 n.mi.) and cruise at Mach 2.7. They are designed to the same technology state-of-the-art, defined by the work of Reference 2 as that which is presumed to be available for start of hardware development in 1985.

As seen in the Table, the LH_2 SCV gross weight is approximately 52 percent of the Jet A fueled design. This leads to lower airline operating costs for a variety of reasons, e.g., wheels, tires, and brakes, all sized as functions of gross weight, which are among the most significant maintenance cost items. Low gross weight also minimizes ground handling problems and cost of equipment. In addition, low gross weight also means smaller engines since engines basically are sized to provide the thrust/weight ratio needed to meet takeoff field length requirements, modified as needed to also meet noise limitations. Smaller engines mean lower initial cost as well as lower maintenance costs.

Operating empty weight is 80 percent that of the Jet A vehicle. This reflects a significant reduction of empty weight which need not be either manufactured (at an average cost of about \$85/kg (\$200/lb) for typical supersonic transport aircraft) or lifted and accelerated to cruise conditions on every flight for the life of the aircraft. These results also lead to airline operating economies.

One of the most interesting items observed in the table is the fact that there is a factor of 3.93 difference in the total fuel weight required by the two aircraft. However, the ratio of the average specific fuel consumption (SFC) values during cruise listed in the table is only 2.61. It might be

Table 14. Comparison of Mach 2.7 Jet A and LH_2 Fueled SCV's

Fuel			Jet A		LH_2		Ratio Jet A/LH_2
Payload	kg	(lb)	22,226	(49,000)	22,226	(49,000)	
Range	km	(n.mi.)	7,783	(4200)	7,783	(4,200)	
Cruise Speed (Std. day + 8° C)		Mach		2.62		2.62	
Takeoff Gross Weight	kg	(lb)	345,720	(762,170)	179,130	(394,910)	1.93
Operating Empty Weight	kg	(lb)	143,980	(317,420)	111,240	(245,235)	1.29
Fuel Weight, Block	kg	(lb)	149,960	(330,590)	38,735	(85,390)	3.88
Total	kg	(lb)	179,510	(395,750)	45,670	(100,675)	3.93
Fuel Volume	m^3	(ft^3)	237	(8,380)	692	(24,450)	2.92
Wing Area	m^2	(ft^2)	1031	(11,094)	739	(7,952)	1.39
Wing Loading (W/S) Takeoff	kg/m^2	(lb/ft^2)	335.4	(68.7)	242.6	(49.7)	
Landing	kg/m^2	(lb/ft^2)	189.9	(38.9)	189.9	(38.9)	
Span	m	(ft)	40.7	(133.5)	34.4	(113)	1.18
Overall Length	m	(ft)	90.5	(297)	103.7	(340.2)	0.87
Lift/Drag (cruise)				8.65		7.42	1.17
Specific Fuel Consumption (cruise)	kg/hr/daN	((lb/hr)/lb)	1.528	(1.501)	0.585	(0.575)	2.61
Thrust/Weight (SLS)	N/kg	-	4.472	(0.456)	5.246	(0.535)	
Thrust Per Engine	N	(lb)	386,470	(86,890)	234,940	(52,820)	1.64
FAR Takeoff Field Length	m	(ft)	2893	(9,490)	1853	(6,080)	1.56
FAR Landing Field Length	m	(ft)	2432	(7,980)	2377	(7,800)	1.02
Landing Approach Speed	m/s	(KEAS)	81.3	(158)	81.3	(158)	
Weight Fractions		Percent					
Fuel				52.0		25.5	
Payload				6.4		12.4	
Structure				25.9		36.5	
Propulsion				9.9		15.1	
Equipment and Operating Items				5.8		10.5	
Energy Utilization	$\frac{kJ}{seat\ km}$	$\frac{Btu}{seat\ n.mi.}$	3522	(6,189)	2531	(4,483)	1.38

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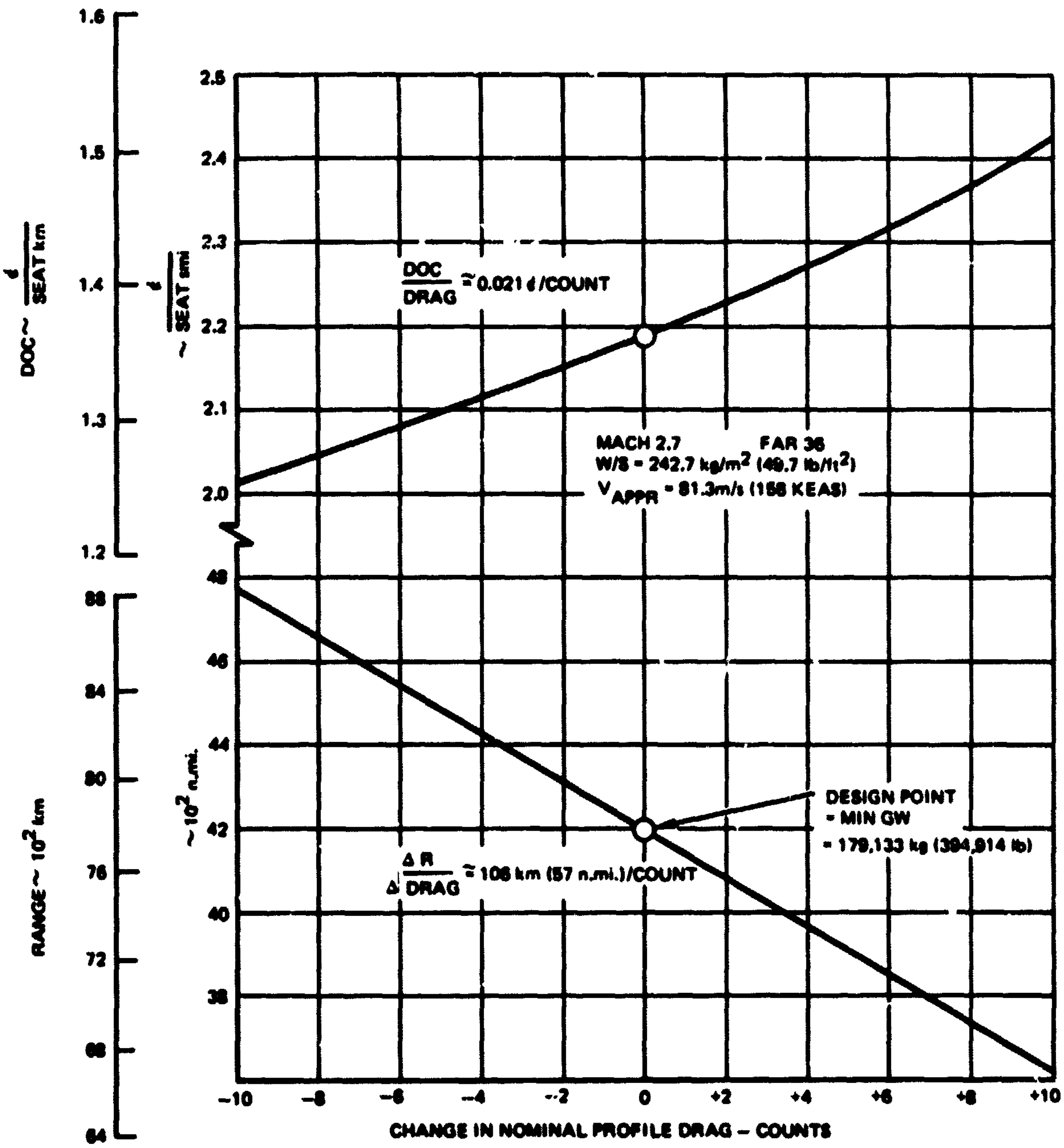


Figure 59. Sensitivity to Drag Level - M2.7 LH_2 SCV.

expected that the same ratio should apply for both parameters. The fact that there is a higher ratio for the fuel weights than there is for the SFC's, is largely accounted for by the greatly reduced weight which must be lifted and accelerated by the hydrogen fueled aircraft. This reduced weight consists of not only the inert weight factor mentioned above, but also the much lighter fuel load. The reduced fuel load is mainly attributable to the SFC ratio; however, it is also favorably affected by the consideration that because the vehicle is lighter to begin with, for a given L/D it will require less thrust to overcome drag, therefore it will consume proportionately less fuel. It is seen that the L/D for the LH_2 aircraft is lower by almost 14 percent, but its average weight in cruise is lower by approximately 50 percent, thus leading to the favorable effect on fuel required during the flight.

Examination of the physical characteristics of the aircraft shows the LH_2 SCV to be longer, have a shorter span and a much smaller wing. The wing loading is much lower at takeoff but the same at landing. The thrust per engine is 61 percent that of the Jet A, but the thrust loading (uninstalled total thrust, sea level static, standard day condition, divided by gross weight) is higher.

Another factor of interest to compare the relative desirability of the two aircraft is energy expended per available seat mile. The Jet A SCV uses 38 percent more Btu available seat mile than does the LH_2 vehicle, viz., 6189 Btu versus 4483 Btu per seat mile. It should be noted that neither of these numbers includes the energy required to produce the fuels, nor to transport them to the airport. Both values represent just the energy contained in the fuel required by the respective aircraft to accomplish the given mission.

Table 15 is a comparison of the group weight statement of both aircraft and shows the penalty paid for LH_2 fuel tanks and insulation.

Table 16 lists some pertinent cost data for comparision of the two types of aircraft. The costs are expressed in terms of 1973 dollars, calculated on the bases noted. The LH_2 SCV aircraft is almost $16 million cheaper than the comparable Jet A airplane in production, and development is estimated to cost 670 million dollars less due largely to the lower airframe weight and use of smaller engines. Direct Operating Cost (DOC) is strongly influenced by the cost of the fuel. The values of DOC shown in the table are based on fuel costs which were arbitrarily specified for both fuels. In September 1973, Jet A sold for approximately 3.17¢/liter (12¢/gal). By September 1975, the price had risen to 7.55¢/liter (28.6¢/gal) for domestic and 10.03¢/liter (36.6¢/gal) for bonded fuel, used by international carriers. The cost of LH_2 produced in large quantities is variously quoted at prices from $2.37 to $4.74/GJ ($2.50 to $5.00 per 10^6 Btu = 12.9 to 25.8¢/lb) delivered to the airport.

Figure 60 presents a plot of DOC for each type of aircraft as a function of fuel cost. This shows that a 9.7¢/liter (36.6¢/gal) for jet fuel the airlines could afford to pay $4.26/GJ ($4.49 per 10^6 Btu) for LH_2. It is

Table 15. Group Weight Statement - Mach 2.7 Jet A and LH_2 SCV's

	Jet A		LH_2 (Min. W_G)	
	kg	lb	kg	lb
Take-Off Weight	(345,721)	(762,171)	(179,133)	(394,914)
Fuel Available	179,513	395,752	45,668	100,679
Zero Fuel Weight	166,208	(366,419)	183,465	(294,235)
Payload	22,226	49,000	22,[illegible]26	49,000
Operating Weight	143,981	(317,419)	(111,239)	(245,235)
Operating Items	2,479	5,466	2,439	5,376
Standard Items	2,420	5,334	2,698	5,927
Empty Weight	(139,082)	(306,619)	(106,112)	(233,932)
Wing	47,391	104,477	26,244	57,858
Tail	4,262	9,337	2,345	5,170
Body	17,773	39,181	25,354 ①	55,895 ①
Landing Gear	14,069	31,017	8,080	17,812
Surface Controls	3,842	8,471	2,089	4,600
Nacelle and Engine Section	2,263	4,989	1,325	2,920
Propulsion	(34,365)	(75,761)	(26,991)	(59,503)
Engines	22,522	49,651	13,178	29,053
Thrust Reversal (in engines)	-	-	-	-
Air Induction System	8,674	19,123	4,967	10,951
Fuel System	2,438	5,375	8,222 ②	18,127 ②
Engine Controls and Starter	732	1,613	622	1,372
Instruments	561	1,237	500	1,102
Hydraulics	2,630	5,799	1,363	3,004
Electrical	2,069	4,562	2,160	4,761
Avionics	863	1,903	863	1,903
Furnishings and Equipment	5,228	11,526	5,228	11, 526
Environmental Control System	3,764	8,297	3,573	7,877
Auxiliary Gear	0	0	0	0

① **Includes: 10,997 kg (24,243 lb) of fuel tankage and interconnect structure.**

② **Consists of: 3,388 kg (7,470 lb) insulation**
2,772 kg (6,111 lb) heat shield
2,062 kg (4,546 lb) fuel system

Table 16. Cost Comparison: Jet A Versus LH_2 Mach 2.7 SCV's
(Refer to Table 14 for vehicle data)

Costs*		Aircraft			
		Jet A		LH_2	
RDT&E	$\$10^6$				
Engine		1,001		876	
Airframe		3,344		2,902	
Total		4,345		3,778	
Production Aircraft, each	$	61,408,000		45,500,000	
Direct Operating Cost (DOC)		¢/seat km	¢/seat sm	¢/seat km	¢/seat sm
Flight Crew		0.059	0.095	0.062	0.099
Fuel and Oil		0.682	1.098	0.766	1.233
Insurance		0.096	0.154	0.075	0.120
Depreciation		0.308	0.496	0.240	0.387
Maintenance		0.275	0.442	0.217	0.350
Total		1.420	2.285	1.360	2.189
Indirect Operating Cost (IOC)		0.559	0.900	0.522	0.840

*Basis for Costs:

- 1973 dollars
- production of 600 aircraft
- passenger load factor = 0.55
- aircraft utilization = 3,600 hrs/year
- fuel cost: Jet A = \$1.90/GJ (\$2/10^6 Btu = 24.8¢/gal = 3.68¢/lb).
  LH_2 = \$2.85/GJ (\$3/10^6 Btu = 15.48¢/lb).

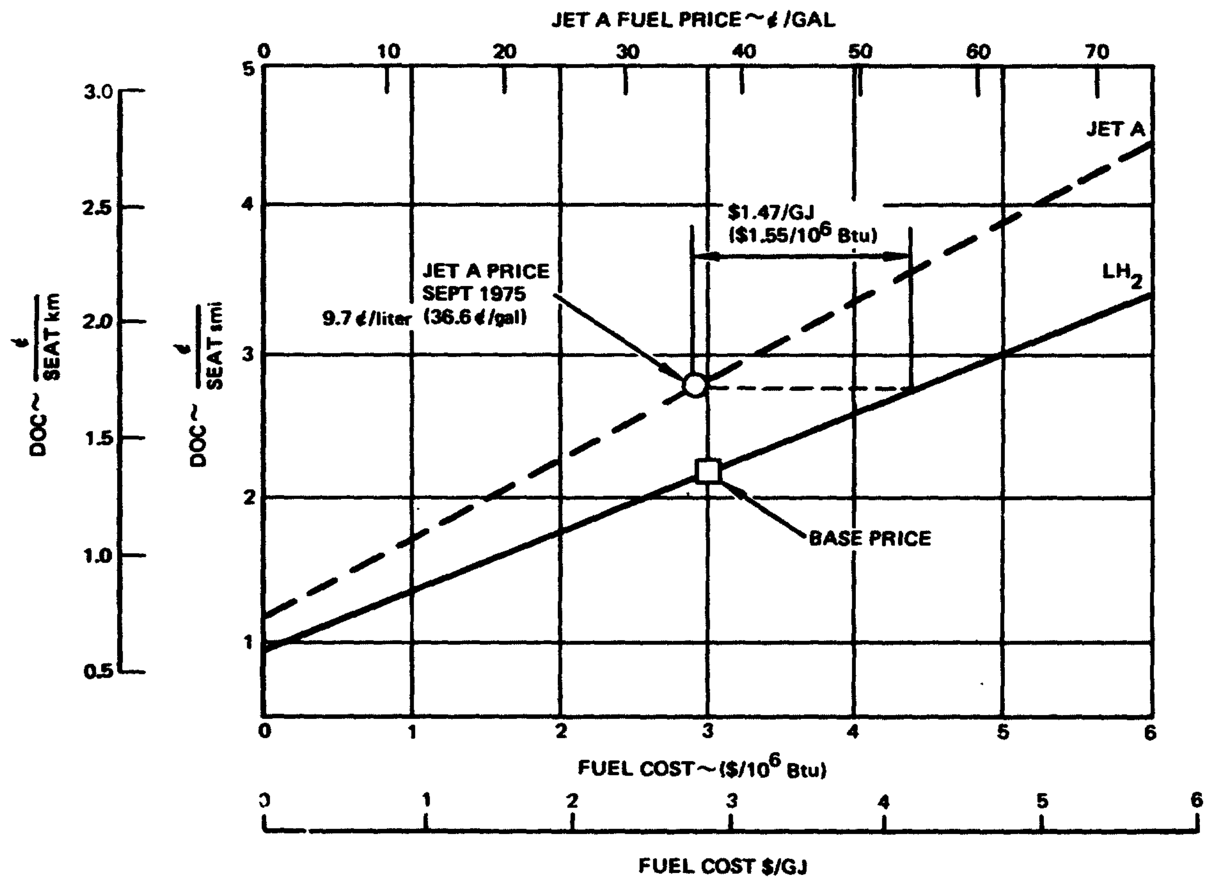


Figure 60. DOC versus Fuel Cost - M2.7 SCV's.

significant that this comparison, favorable as it is to the hydrogen aircraft, does not include consideration of cost advantages resulting from the lower maintenance requirements and the longer life anticipated for components on engines fueled with liquid hydrogen.

An indicated by the divergent lines representing DOC for the two fuels on Figure 60, the difference in fuel cost which produces equivalent DOC varies with cost of the fuels. The following expression can be used for the subject Mach 2.7 SCV aircraft to calculate the differential which can be paid for LH_2, over the selected price of Jet A, to provide parity in direct operating cost.

$$\Delta C_{LH_2} = 0.335\ C_{JA} + 0.560$$

where ΔC_{LH_2} = cost increment in $/$10^6$ Btu permitted for LH_2 to produce equal DOC.

C_{JA} = cost assigned for Jet A fuel in $/$10^6$ Btu.

6. MACH 2.2 AIRCRAFT

6.1 Configuration Description

The general arrangement of the LH_2 fueled Mach 2.2 minimum gross weight airplane is shown in Figure 61. Fundamentally the design is identical to the Mach 2.7 aircraft described in Section 4.1. The only differences are those prescribed by the aerodynamic requirements of cruising at Mach 2.2 instead of 2.7. Accordingly, the wing and tail surfaces have less sweep, a higher aspect ratio, and smaller areas. The wing area is smaller for the Mach 2.2 design because the higher aspect ratio leads to better low speed lift characteristics and the wing loading can be increased.

Overall dimensions and significant geometric characteristics of the Mach 2.2 LH_2 fueled airplane are shown on the general arrangement drawing, Figure 61. The inboard profile, Figure 62, illustrates the passenger seating arrangement and shows the same relationship between the passenger compartment and the liquid hydrogen tanks previously described. In fact, the Mach 2.7 and 2.2 airplane designs are identical insofar as the fuselage is concerned, except for a small difference in length. Since the Mach 2.2 airplane requires slightly more fuel, the fuel tanks are a total of 0.3048 m (one foot) longer. The higher fuel consumption of the Mach 2.2 compared to the Mach 2.7 is largely attributable to the lower cruise efficiency of the 2.2 design as indicated by the cruise parameter M(L/D)/SFC value of 30.5 compared to 34.8 for the Mach 2.7.

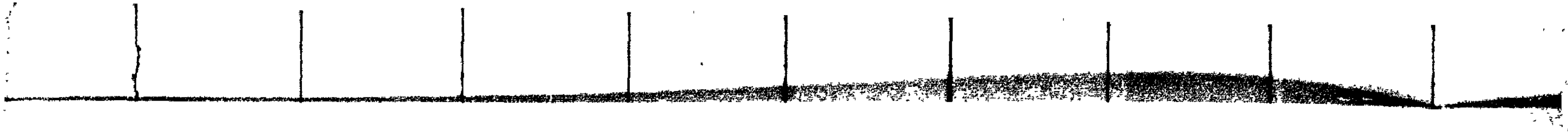
6.2 Parametric Data Results

Figures 63 and 64 show the original matrix of 40 aircraft in terms of gross weight and fuel consumption for various thrust-to-weight ratios and wing loadings with a maximum duct burning temperature (DBT) of 1367° K (2460° R). The dashed line indicates the locus of those aircraft meeting the maximum approach speed of 81.3 m/s (158 KEAS) which is determined by the block fuel consumption and the take-off wing loading. Figure 65 shows the effect of various fuel prices on DOC for aircraft meeting the 81.3 m/s (158 KEAS) constraint and indicates a very slight shift in the optimum T/W for minimum DOC. From these three plots preliminary aircraft meeting FAR 36 can be selected; one each for minimum gross weight, minimum fuel and minimum DOC. It should be pointed out that this selection of aircraft is based on a maximum DBT of 1367° K (2460° R). Subsequent optimization of this temperature and the final aircraft are presented in the following sections.

As with Mach 2.7 aircraft, takeoff jet noise was determined for a parametric family of aircraft having engines designed for a maximum duct burning temperature of 1367° K and wing sized to meet the 81.3 m/s landing approach speed limit. The jet noise suppression used for the analysis was taken from Figure 29. The thrust setting at brake release, the thrust setting at cut back, and the aircraft height at cut back used for takeoff noise abatement and the maximum climb and cruise DBT's are presented in Figure 66 for a parametric family of aircraft. The thrust setting at cut back provides a zero climb gradient with one engine inoperative. For aircraft with T/W less than 0.643, maximum thrust was used prior to cut back. The cut-back height was selected to match the FAR 36 sideline and flyover noise decrements. For aircraft with T/W greater than 0.643, the height at cut back was held at 213.4 m (700 feet) and the thrust setting at brake release was adjusted to match the FAR 36 sideline and flyover noise decrements.

The resulting matched FAR 36 sideline and flyover noise decrements are presented in Figure 67 for the parametric family of aircraft. The discontinuity at T/W = 0.652 occurs when the thrust at cut back corresponds to the minimum duct burning thrust setting.

Figure 68 shows the range of the FAR 36-5 and -10 aircraft when the mission is flown with various levels of maximum duct burning temperature (DBT). From this plot, the optimum DBT's of 1034° K (1860° R) and 811° K (1460° R) were selected and the aircraft resized to a 7783 km (4200 n. mi.) range to produce the final design. The figure shows a range gain of (280 n. mi.) for the FAR-10 airplane compared to (100 n. mi.) for the -5. This is due to the fact that the higher thrust-to-weight of the -10 design (1.1) permits climb and cruise at a lower duct burning temperature than the -5, resulting in a lower SFC. For example, in cruise the SFC of the -10 is 0.46 (kg/hr)/daN (0.452 (lb/hr)/lb) compared to 0.495 (0.486) for the -5. This effect is shown in Figure 33, indicating the reduction in SFC as the thrust (DBT) is reduced at Mach 2.12.

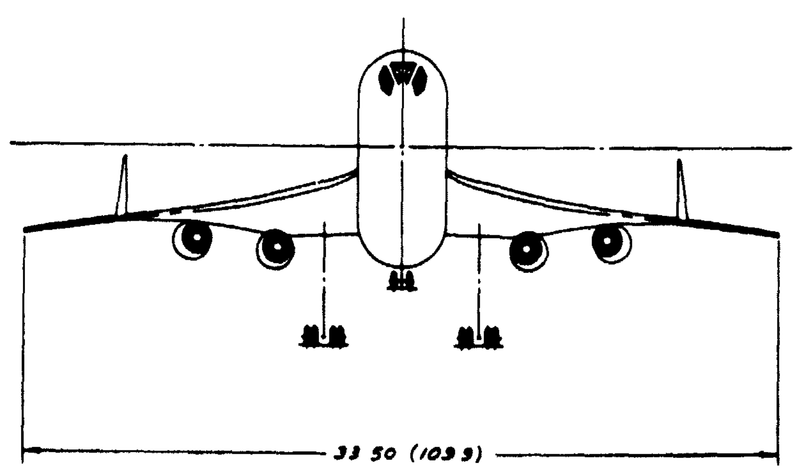


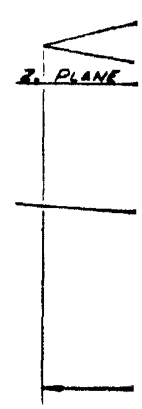
CHARACTERISTICS	WING	HORIZ TAIL	FUS. VERT TAIL	WING VERT TAIL (EA)
AREA M² (SQ FT)	559 28 (6020 1)	27.29 (293 8)	18 23 (196.2)	29 58 (318 4)
ASPECT RATIO	2 058	1.707	0 517	0.517
SPAN M (FT)	33.50 (109 9)	6 74 (22.1)	3 08 (10 1)	2.77 (9.1)
ROOT CHORD M(IN)	36 84 (1450.4)	6 53 (257 0)	9 65 (380 1)	8.92 (351.1)
TIP CHORD M(IN)	4 30 (169 2)	1.47 (57 8)	2.22 (87.4)	1.78 (70.2)
TAPER RATIO	0.1167	0.225	0.23	0.20
MAC M(IN)	22 54 (887 5)	4 53 (178 4)	6 71 (264 3)	6.14 (241 9)
SWEEP- RADIAN (DEG)	1.225 (70 20)	0.988 (56.64)	1.190 (68 2)	1 281 (73 42)
RADIAN (DEG)	1.154 (66 11)	—	—	—
RADIAN (DEG)	0 910 (52 15)	—	—	—

DESIGN GROSS WEIGHT - 170,970 KG. (376,917 LBS.)

POWER PLANT - SCV LH, M2 2 DUCT BURNING TURBOFAN
UNINSTALLED THRUST - 237,643 NEWTONS (53,427 LBS.)

PASSENGERS - 234





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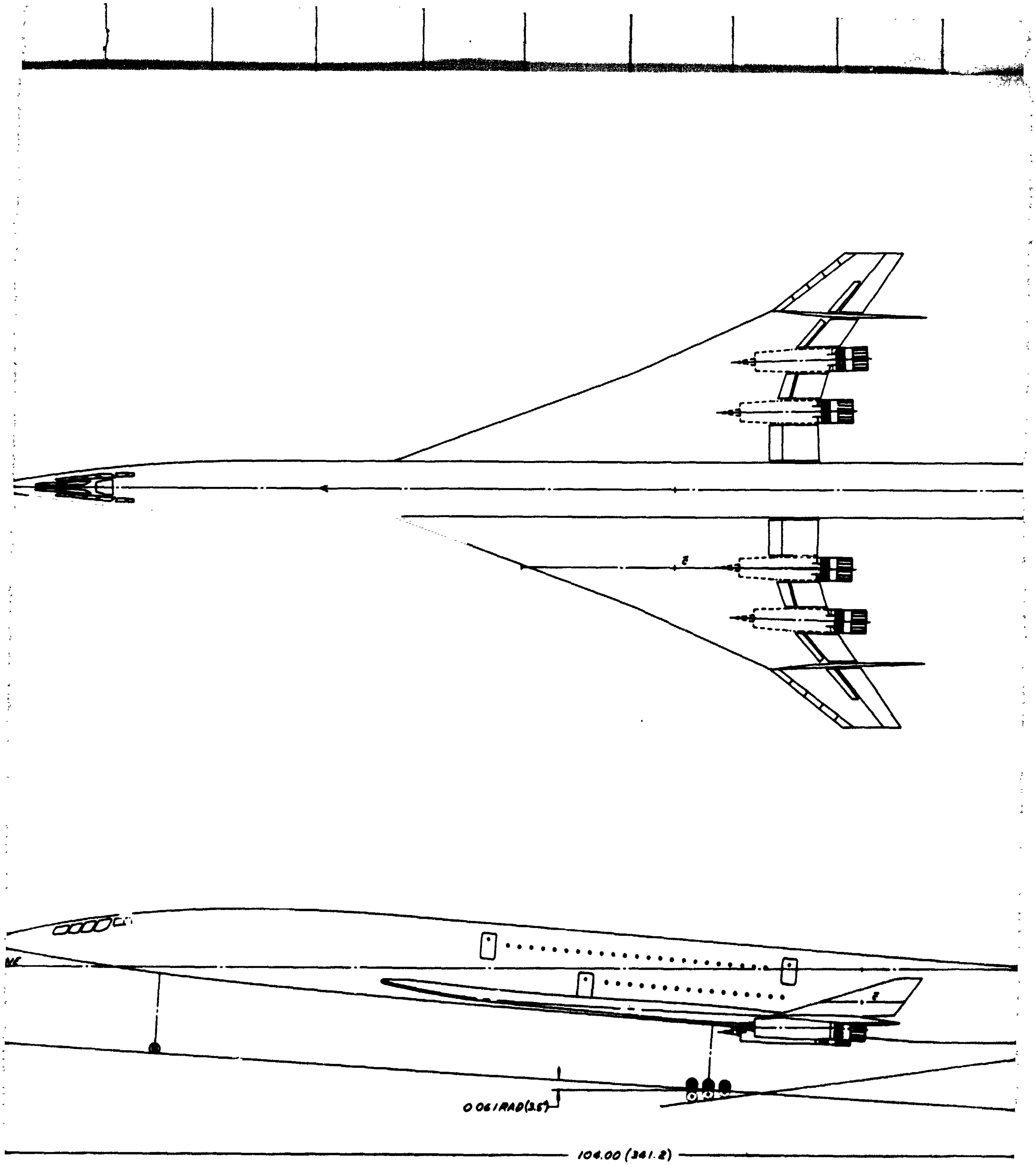

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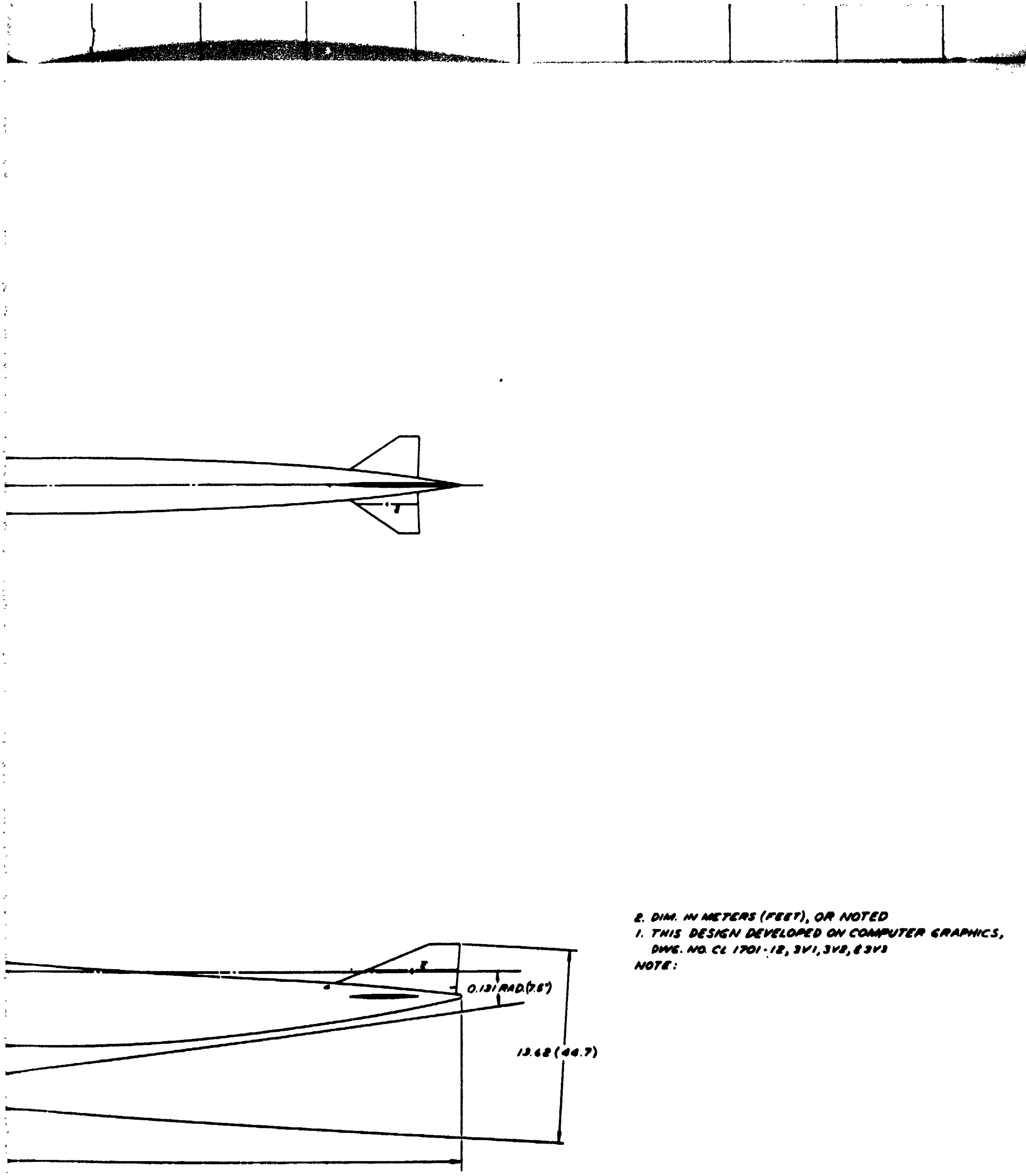

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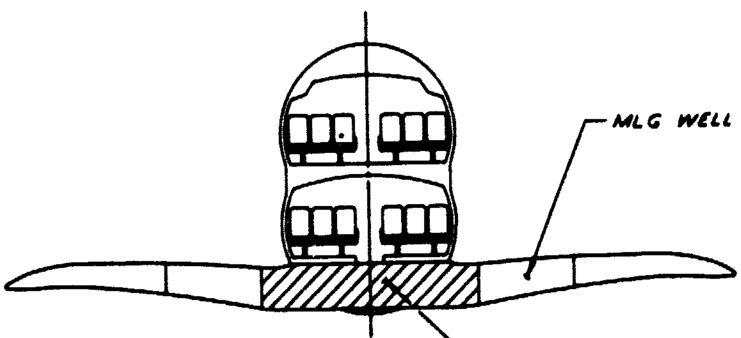
1. THIS DESIGN DEVELOPED ON COMPUTER GRAPHICS,
DWG. NO. CL 1701-12, 3V1, 3V2, & 3V3
NOTE:

Figure 61. General Arrangement - M2.2 LH_2 SCV



SECTION A-A

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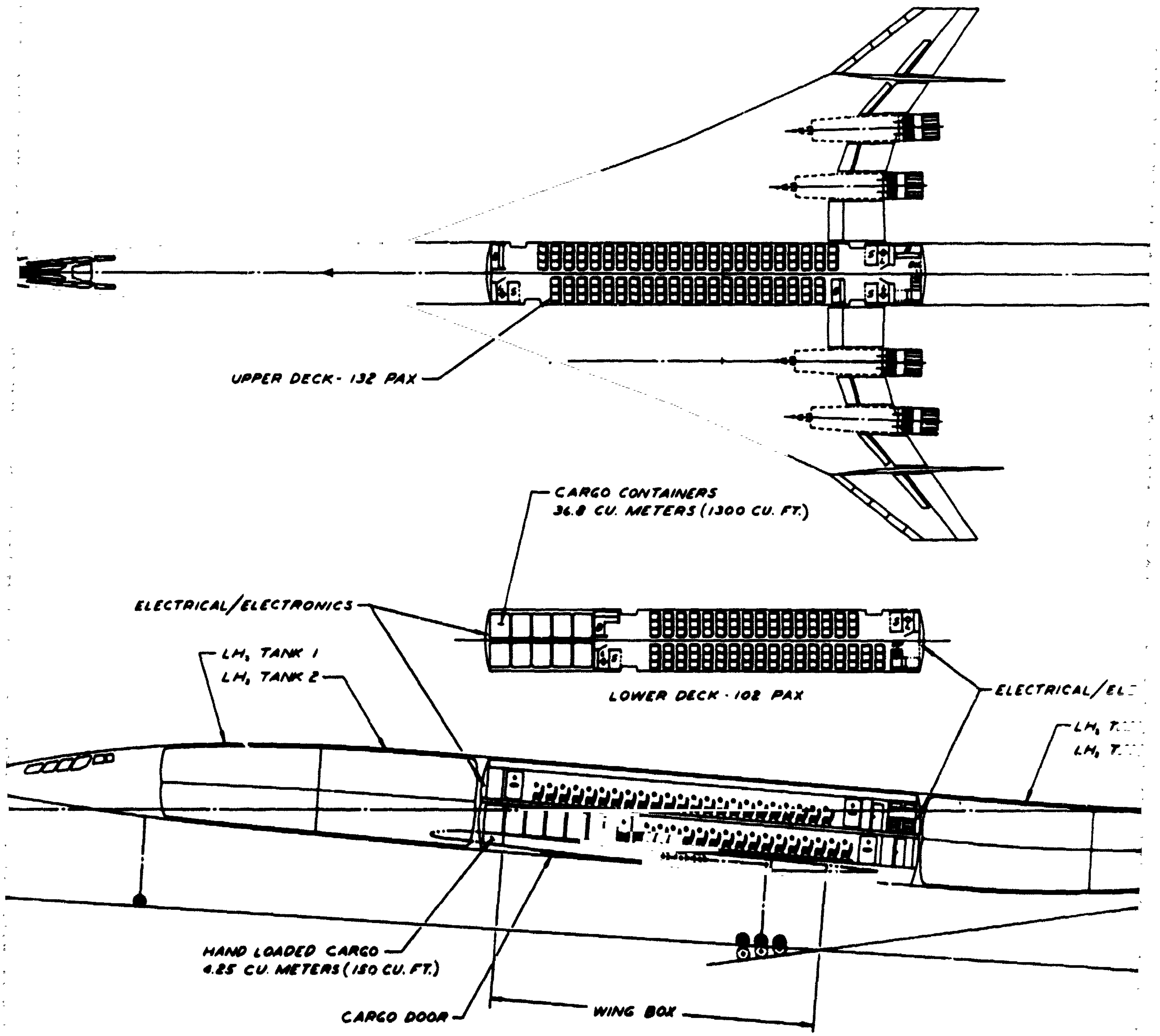
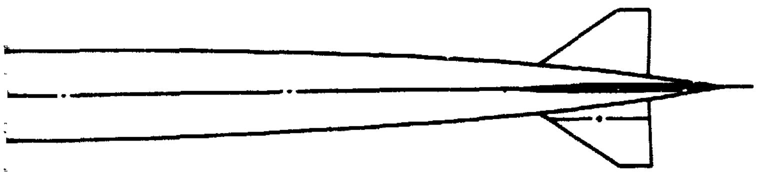
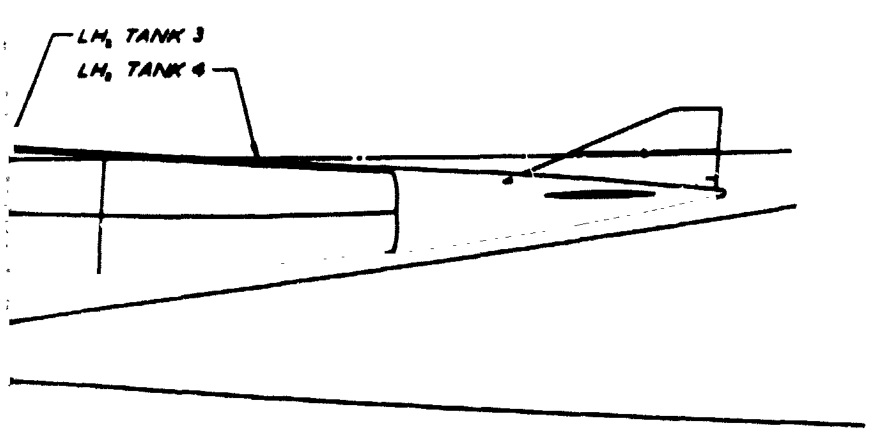

UPPER DECK - 132 PAX
CARGO CONTAINERS
36.8 CU. METERS (1300 CU. FT.)
ELECTRICAL/ELECTRONICS
LH_2 TANK 1
LH_2 TANK 2
LOWER DECK - 102 PAX
HAND LOADED CARGO
4.25 CU. METERS (150 CU. FT.)
CARGO DOOR
WING BOX





B - BUFFET
L - LAVATORIES
2. S - COATS/STORAGE
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1. THIS DESIGN DEVELOPED ON COMPUTER GRAPHICS,

Figure 62. Interior Arrangement - M2.2 LH_2 SCV

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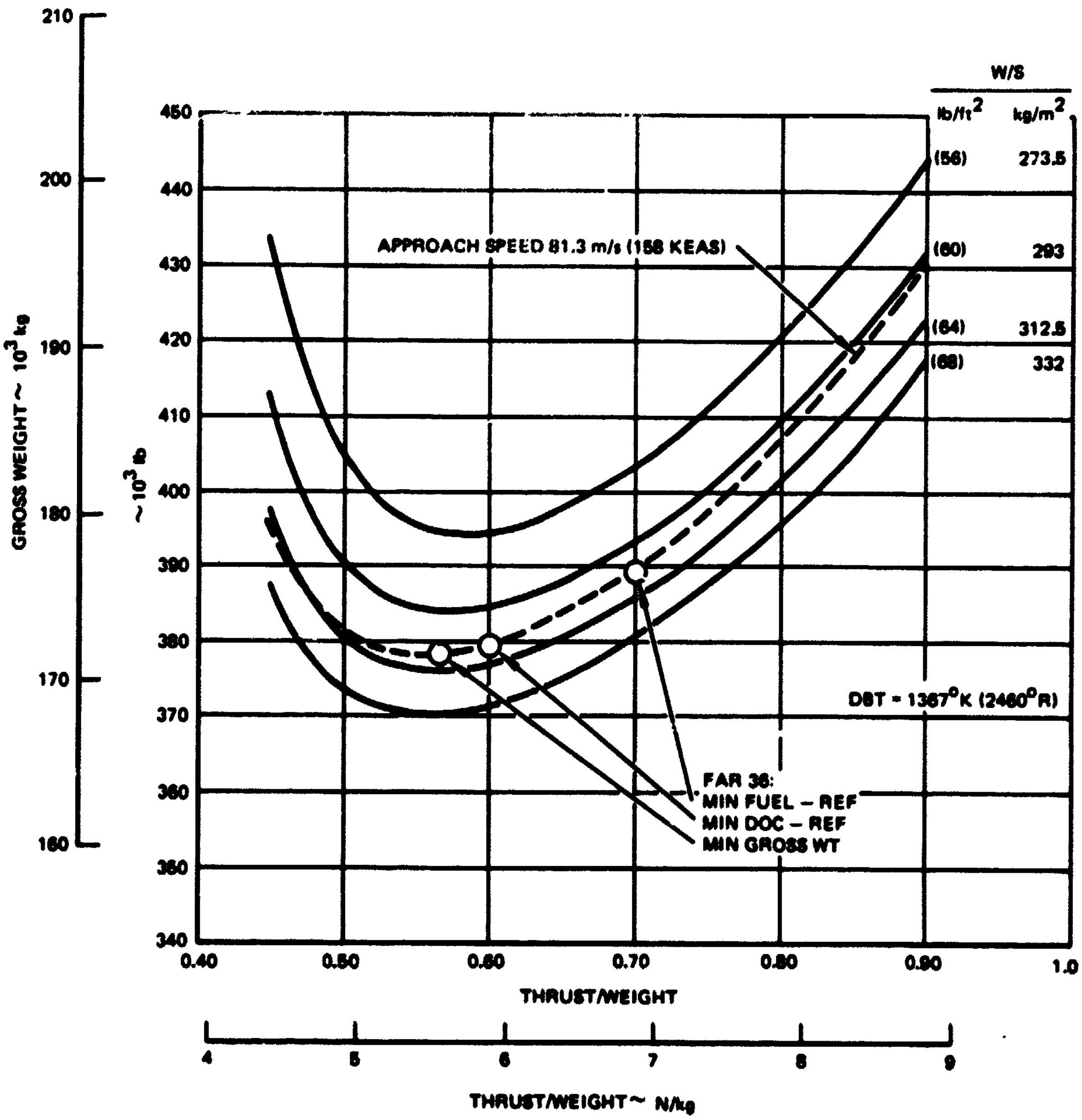

Figure 63. T/W vs Gross Weight - M2.2, LH_2 SCV.

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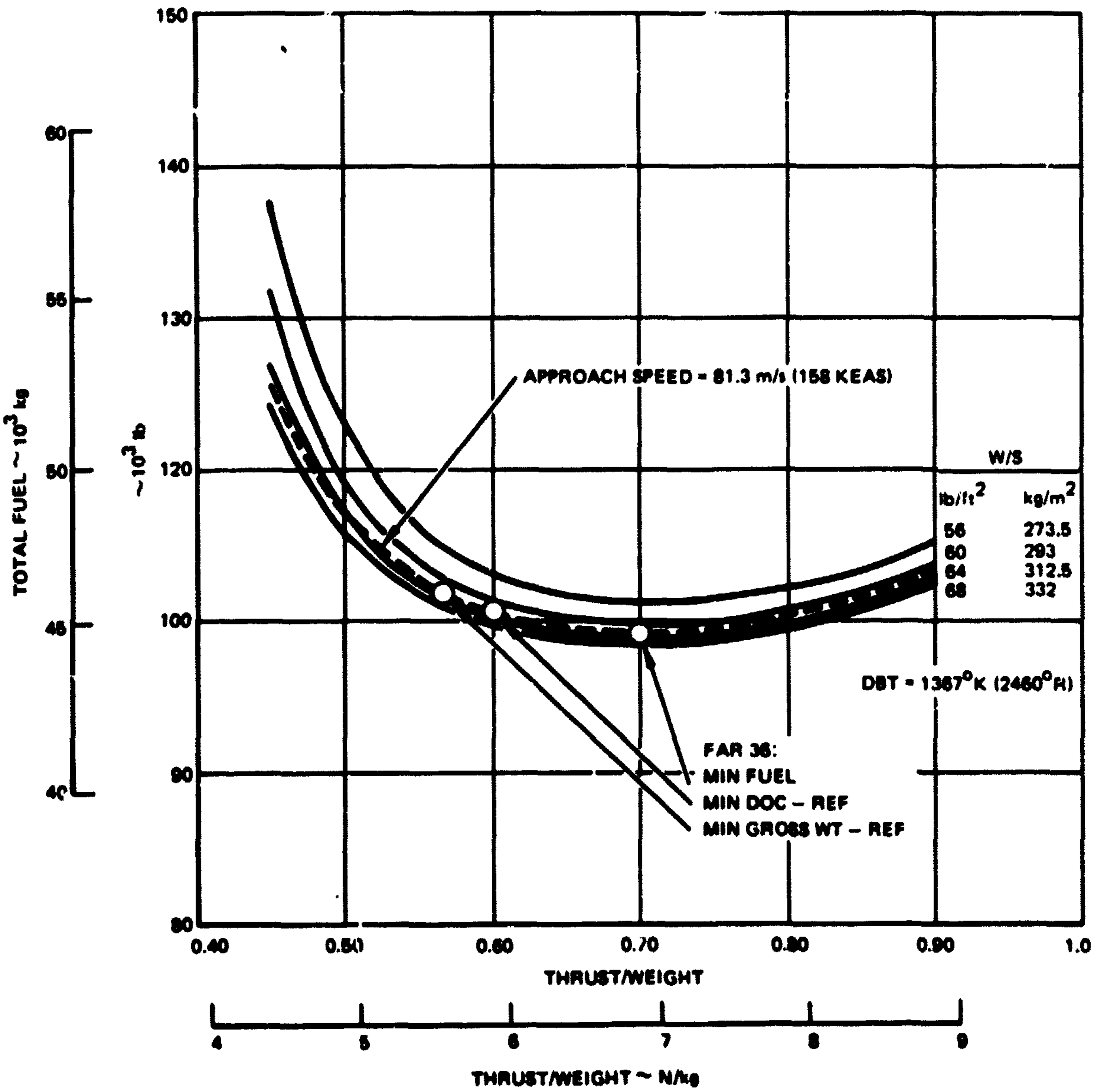

Figure 64. T/W vs Total Fuel - M2.2 LH_2 SCV.

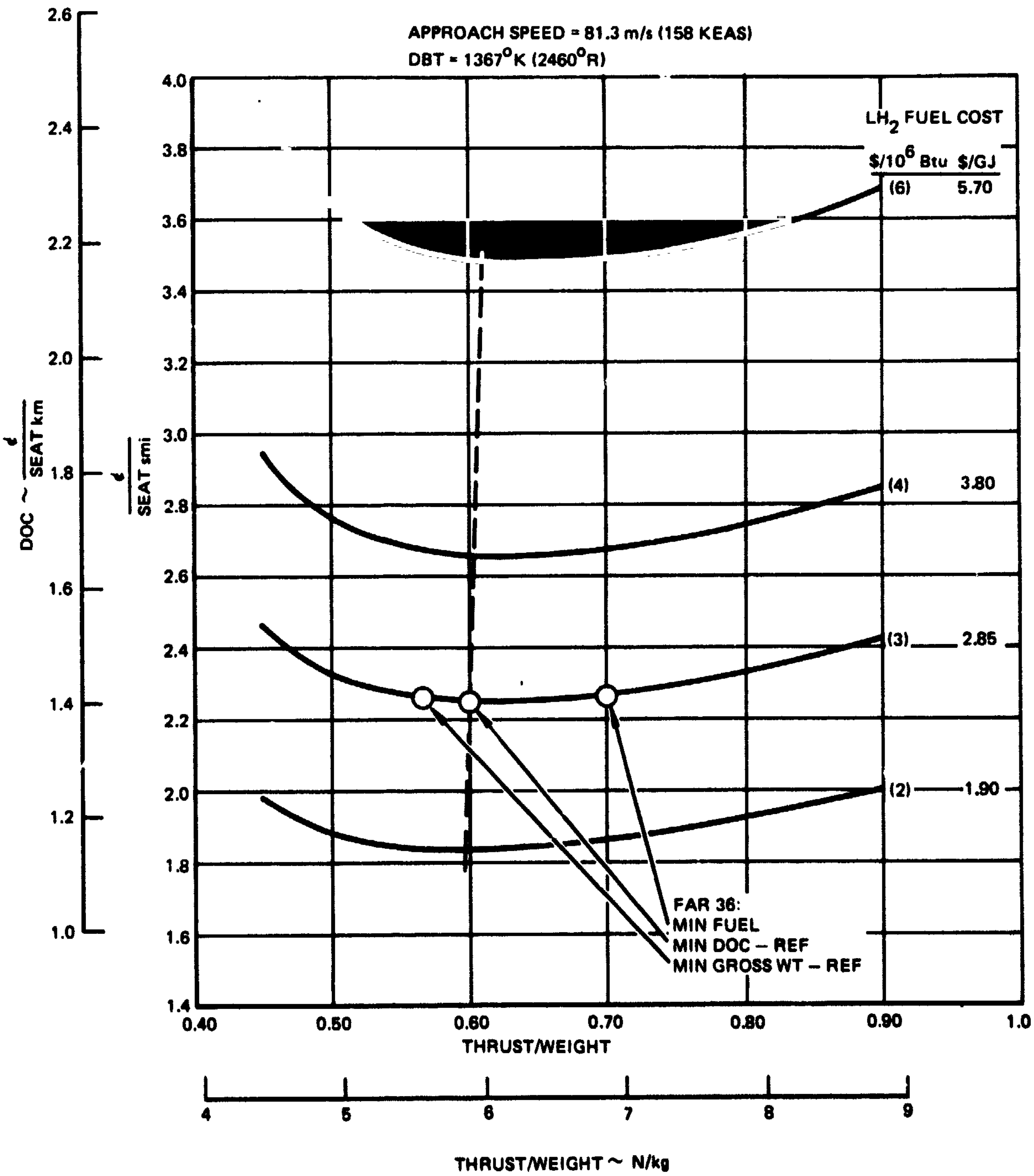

Figure 65. DOC vs Fuel Cost - M2.2 LH_2 SCV.

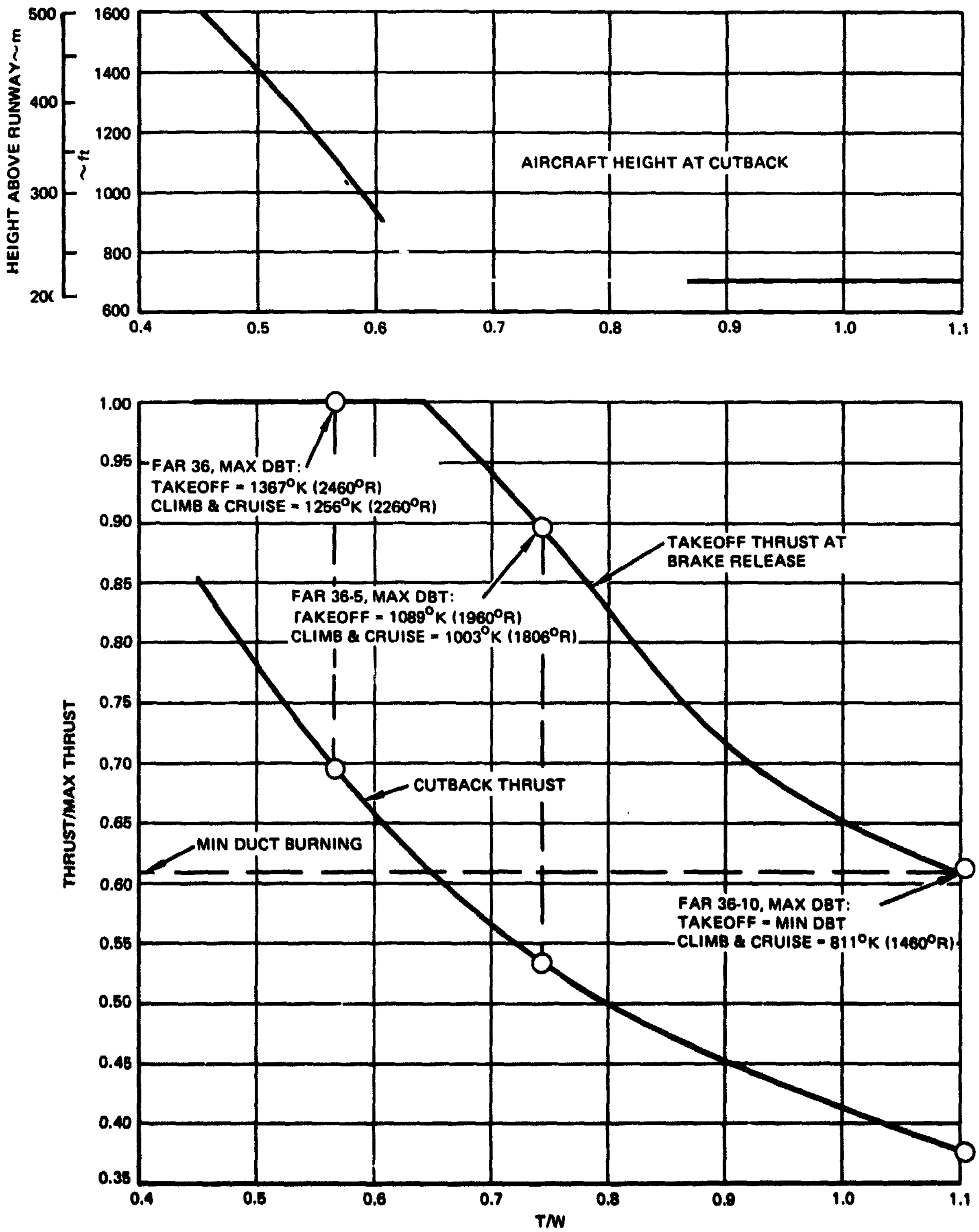

Figure 66. Takeoff Noise Abatement Procedure - M2.2 LH_2 SCV.

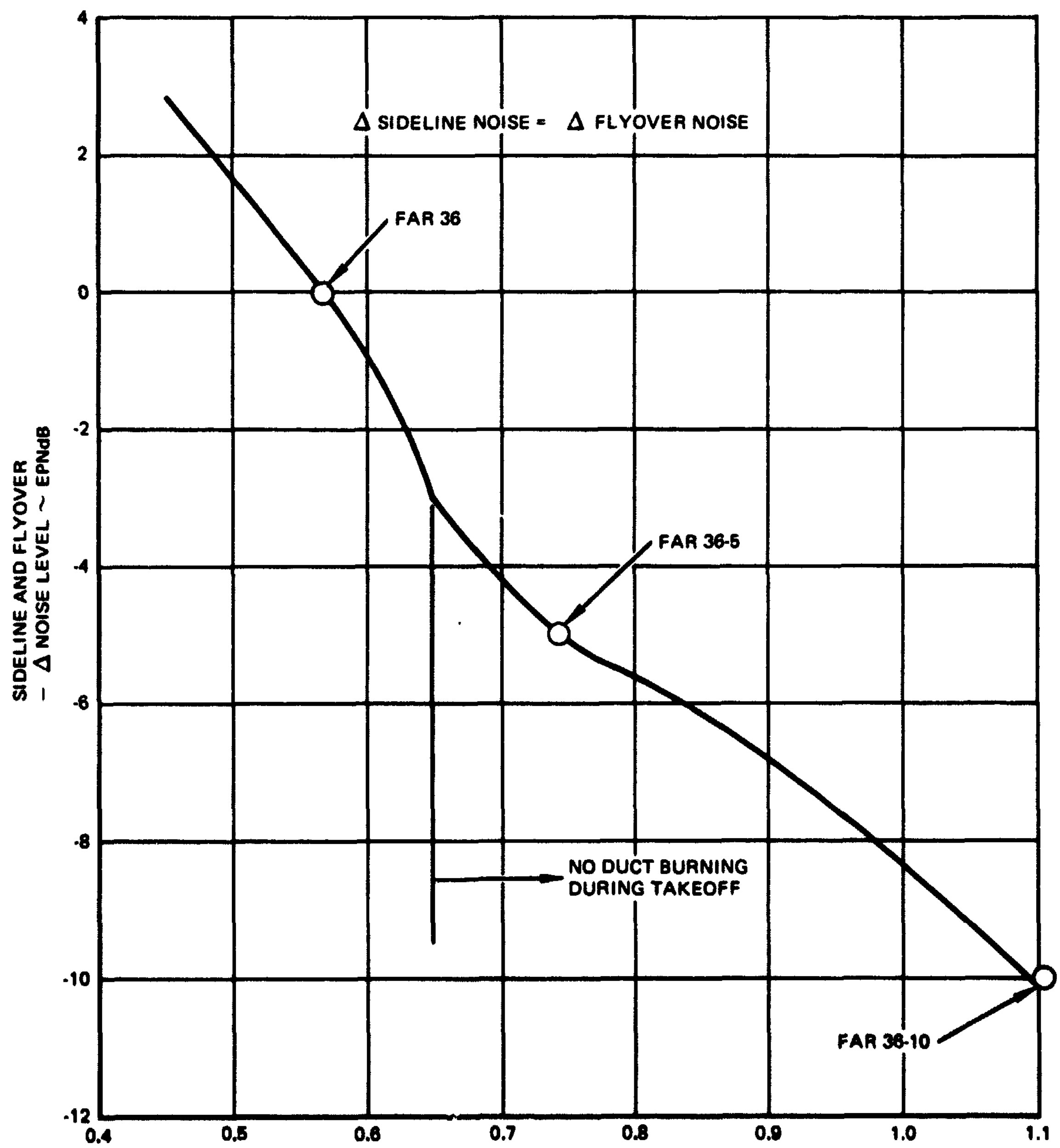

Figure 67. FAR 36 Takeoff Noise Decrement - M2.2 LH_2 SCV.

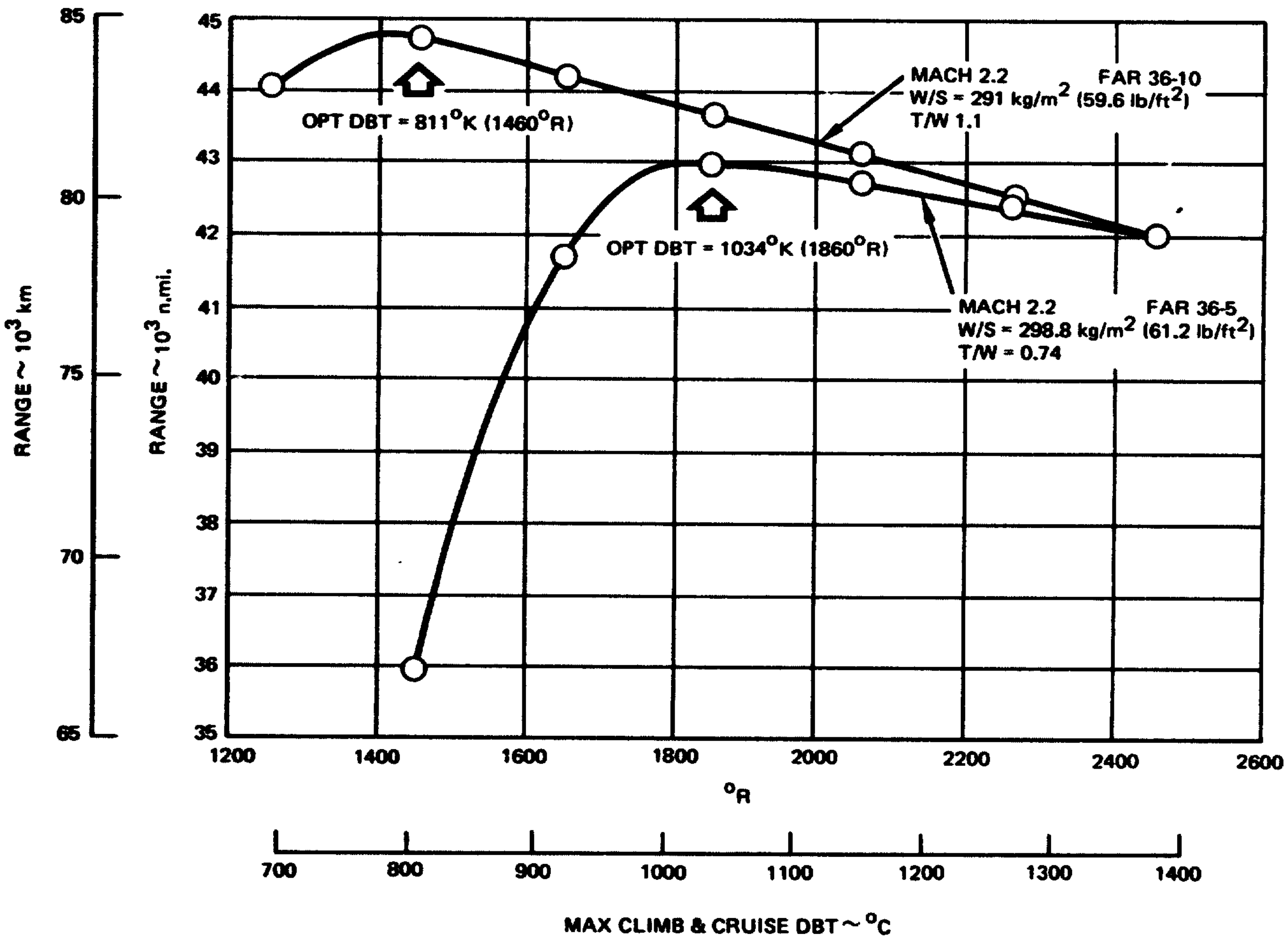

Figure 68. Selection of max DBT for Optimum Climb and Cruise - M2.2, Far -5 and -10, LH_2 SCV.

Table 17 summarizes the characteristics of the five selected final designs of Mach 2.2 aircraft. The aircraft designed for minimum fuel weight coincidentally was found to be 5 EPNdB quieter than permitted by the FAR 36 specification. This aircraft shows a slight decrease in DOC, and increases of 2.9% and 6.9%, respectively, for gross weight and price, relative to the FAR 36 aircraft. A reduction to -10 EPNdB will penalize the DOC by 15.3%, the gross weight by 24.7% and the price by 35.7% relative to the FAR 36 aircraft. This aircraft required very large engines, a T/W = 1.106, in order to allow the power cutback needed to meet the noise constraint. Power reduction during takeoff was required for both the FAR -5 and -10 aircraft, and a reduction in the maximum design DBT (2460° R) used in climb and cruise for all aircraft. The aircraft designed to minimize DOC is a good compromise considering price, noise, energy utilization and DOC.

Comparison of the sensitivity to noise reduction of the Mach 2.7 and 2.2 aircraft indicates that the Mach 2.7 minimum gross weight aircraft with a noise reduction level of -2.75 EPNdB, had a growth of 15.3% in gross weight to meet the -10 noise constraint, or a growth of 2.1% per EPNdB. The Mach 2.2 had a growth of 24.7% in going from 0 to -10 EPNdB or 2.47% per EPNdB, only slightly more than the Mach 2.7. It is reiterated that the engine characteristics of the subject aircraft were not reoptimized to provide the required noise reductions. Reoptimization could possibly reduce this penalty by either reducing fan pressure ratio or going to a variable cycle approach. Although beyond the scope of this study, such an exercise would be required to minimize the -10 EPNdB noise penalty.

Figure 69 shows the C.G. travel of the Mach 2.2 aircraft indicating the desired C.G. at 51% during mid cruise.

6.3 Sensitivity Analysis

The minimum weight, FAR 36, Mach 2.2 vehicle was perturbed on the basis of range, empty weight, SFC, drag, and payload to determine its sensitivity to each of these factors. Figures 70 thru 71 show the results of these excursions, together with approximate sensitivity factors where appropriate on gross weight, DOC, price, and total fuel weight.

Figure 70 examines the growth of the point design aircraft on the basis that the design mission range was increased. To accommodate the increased fuel required the fuselage was allowed to grow in length. In each case the vehicle is resized and the constraints of approach speed and noise held constant. Since the landing wing loading is held constant to meet the approach speed, the takeoff wing loading can be increased slightly as more mission fuel is consumed. FAR 36 allows increasing takeoff and flyover noise as gross weight is increased which results in a slightly higher allowable jet velocity. The result is that the turbofan engine power can be reduced. More usable thrust allows a slight decrease in the installed thrust-to-weight. This slightly increases the takeoff field length but it remains well within the 3,200 m (10,500 ft) constraint. The result of this study shows that the

Table 17. Mach 2.2 LH_2 SCV - Aircraft Comparison
(S.I. Units)

7783 km Range - 22,226 kg Payload
Fuel Cost = $2.85/GJ

		Minimum Fuel Wt	Minimum DOC	Minimum Gross Weight		
				FAR 36	FAR 36-5	FAR 36-10
Gross Wt - Ref	kg	175,920	172,070	170,970	Same as min W_F	213,210
Block Fuel Wt	kg	37,540	38,180	39,600		40,960
DOC	¢/seat km	1.399	1.389	1.404		1.619
Airplane Price	$10^6	44.90	43.12	42.01		57.02
Wing Loading	kg/m^2	298.8	301.7	305.6		291.0
Thrust/Weight (DBT-1366°C)	N/kg	7.286	6.374	5.560		10.845
Maximum DBT - Climb and Cruise	°R	1860	2060	2260		1460
Maximum DBT-Takeoff	°R	1960	2410	2460		Min. DBT
Wing Area	m^2	589	570	559		733
Span	m	34.8	34.3	33.9		38.9
Fuselage Length	m	102.3	102.7	104.0		106.5
Landing Approach Speed	m/S					
FAR T.O. Field Length	m	1384	1430	1611		1372
FAR Landing Field Length	m	2489	2420	2411		2473
Average Cruise L/D	-	7.43	7.35	7.25		7.85
Average Cruise SFC	kg/hr/daN	0.495	0.513	0.531		0.460
Average Cruise Alt	M	18288	18288	17983		18898
Structure Wt*	kg	61,130	59,710	59,240		74,460
Propulsion Wt**	kg	30,710	27,750	25,600		49,090
Equip. and Furn. Wt	kg	12,780	12,750	12,750		13,170
Empty Wt	kg	104,620	100,210	97,600		136,720
Std. + Operating Items	kg	4,980	4,985	5,030		5,235
Operating Empty Wt	kg	109,600	105,190	102,630		141,960
Payload	kg	22,226	22,226	22,226		22,226
Zero Fuel Wt	kg	131,820	127,420	124,860		164,180
Total Fuel	kg	44,090	44,650	46,110		49,025
Take-off Gross Wt	kg	175,910	172,070	170,970		231,210
Sideline noise Actual/FAR 36	EPNdB	101.74/106.74	103.63/106.68	106.66/106.66		97.30/107.30
Flyover noise Actual/FAR 36	EPNdB	99.85/104.85	101.64/104.69	104.65/104.65		96.24/106.24
Δ Noise Reduction (from FAR 36)	EPNdB	-5	-3.05	0		-10
Energy Utilisation	kJ/seat km	2472	2514	2608		2697

*Includes LH_2 tank weight.
**Includes insulation and heat shield weight.

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Table 17. Mach 2.2 LH_2 SCV - Aircraft Comparison (Continued)
(U.S. Customary Units)

4200 n.mi. Range - 49,000 lb Payload
Fuel Cost = $3/10^6 Btu (15.48¢/lb)

		Minimum Fuel Wt	Minimum DOC	Minimum Gross Weight		
				FAR 36	FAR 36-5	FAR 36-10
Gross Wt - Ref	lb	387,825	379,340	376,920	Same as Min W_F	470,030
Block Fuel Wt	lb	82,760	84,180	87,300	↕	90,290
DOC	¢ / seat smi	2.251	2.236	2.259		2.605
Airplane Price	$10^6	44.90	43.12	42.01		57.02
Wing Loading	lb/ft²	61.2	61.8	62.6		59.6
Thrust/Weight (DBT=2460°R)	-	0.743	0.650	0.567		1.106
Maximum DBT - Climb and Cruise	°R	1,860	2,060	2,260		1,460
Maximum DBT - Takeoff	°R	1,960	2,410	2,460		Min. DBT
Wing Area	ft²	6,640	6,134	6,020		7,892
Span	ft	114.3	112.4	111.3		127.5
Fuselage Length	ft	335.5	337.1	341.2		349.5
Landing Approach Speed	KEAS	158	158	158		158
FAR T.O. Field Length	ft	4,540	4,690	5,285		4,500
FAR Landing Field Length	ft	7,970	7,940	7,910		8,115
Average Cruise L/D	-	7.43	7.35	7.25		7.85
Average Cruise SFC	lb/hr/lb	0.486	0.504	0.522		0.452
Average Cruise Alt	ft	60,000	60,000	59,000		62,000
Structure St*	lb	134,760	131,640	130,600		164,150
Propulsion Wt**	lb	67,710	61,170	56,440		108,225
Equip. and Furn. Wt	lb	28,180	28,110	28,120		29,040
Empty Wt	lb	230,650	220,920	215,170		301,410
Std. + Operating Items	lb	10,970	10,990	11,090		11,540
Operating Empty Wt	lb	241,620	231,910	226,260		312,950
Payload	lb	49,000	49,000	49,000		49,000
Zero Fuel Wt	lb	290,620	280,910	275,260		361,950
Total Fuel	lb	97,210	98.440	101,660		108,080
Take-off Gross Wt	lb	387,825	379,340	376,920		470,030
Sideline noise Actual / FAR 36	EPNdB	101.74 / 106.74	103.63 / 106.68	106.66 / 106.66		97.30 / 107.30
Flyover noise Actual / FAR 36	EPNdB	99.85 / 104.85	101.64 / 104.69	104.65 / 104.65		96.24 / 106.24
Δ Noise Reduction (from FAR 36)	EPNdB	-5	-3.05	0		-10
Energy Utilization	Btu / seat n.mi.	4345	4419	4583	↓	4740

*Includes LH_2 tank weight.
**Includes insulation and heat shield weight.

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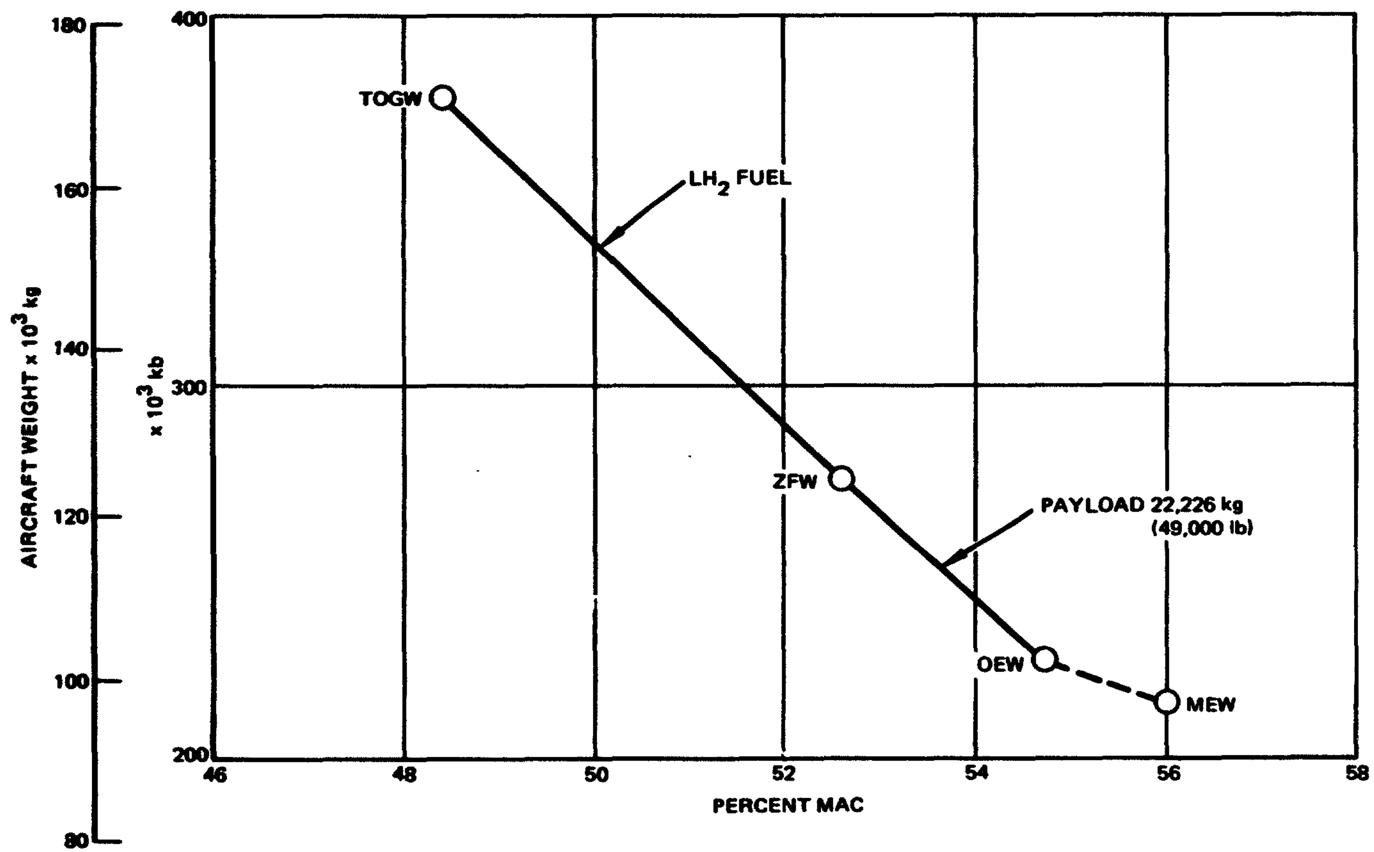

Figure 69. Center of Gravity Travel - M2.2 LH_2 SCV.

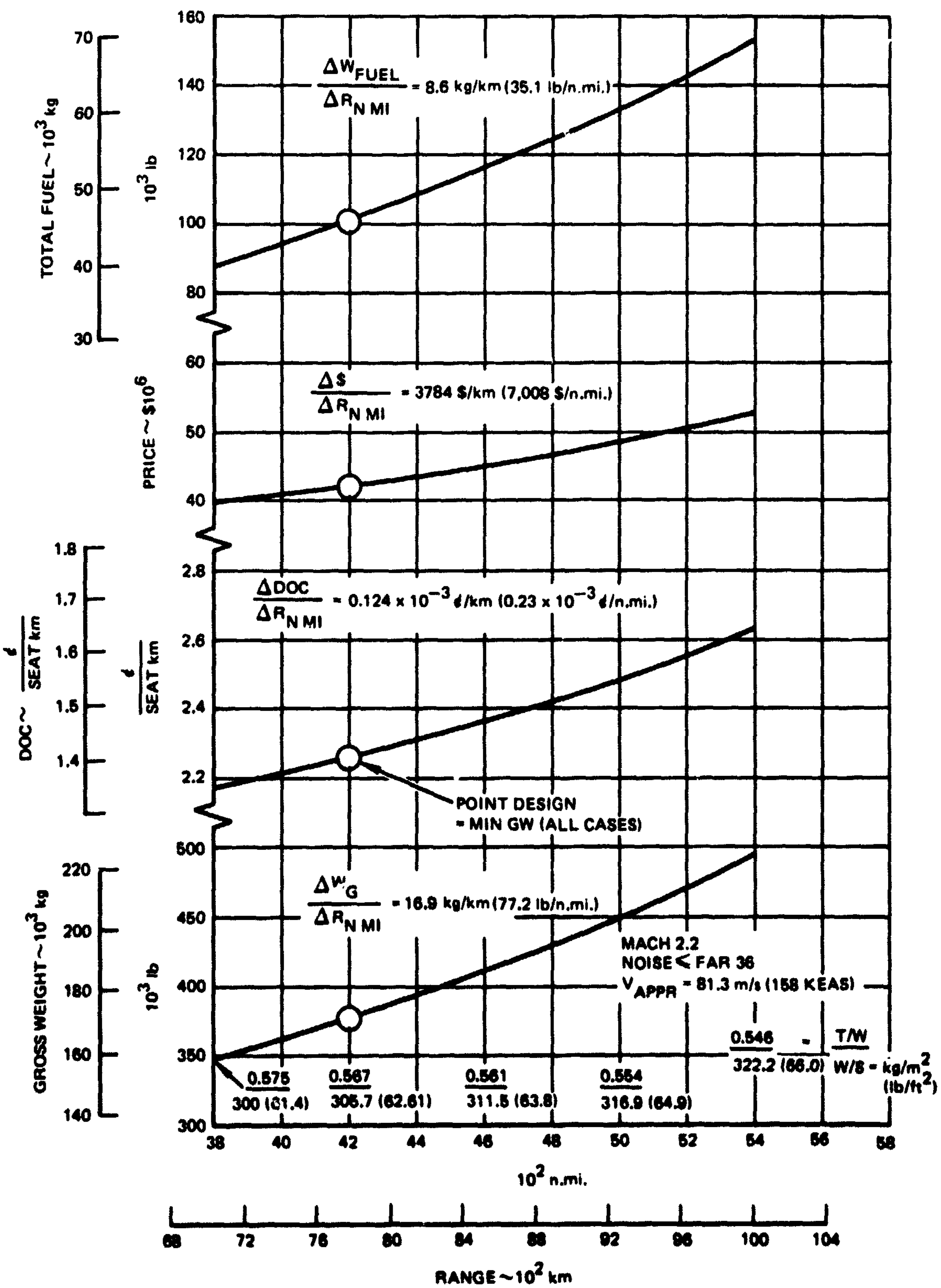

Figure 70. Range Sensitivity - M2.2 LH_2 SCV.

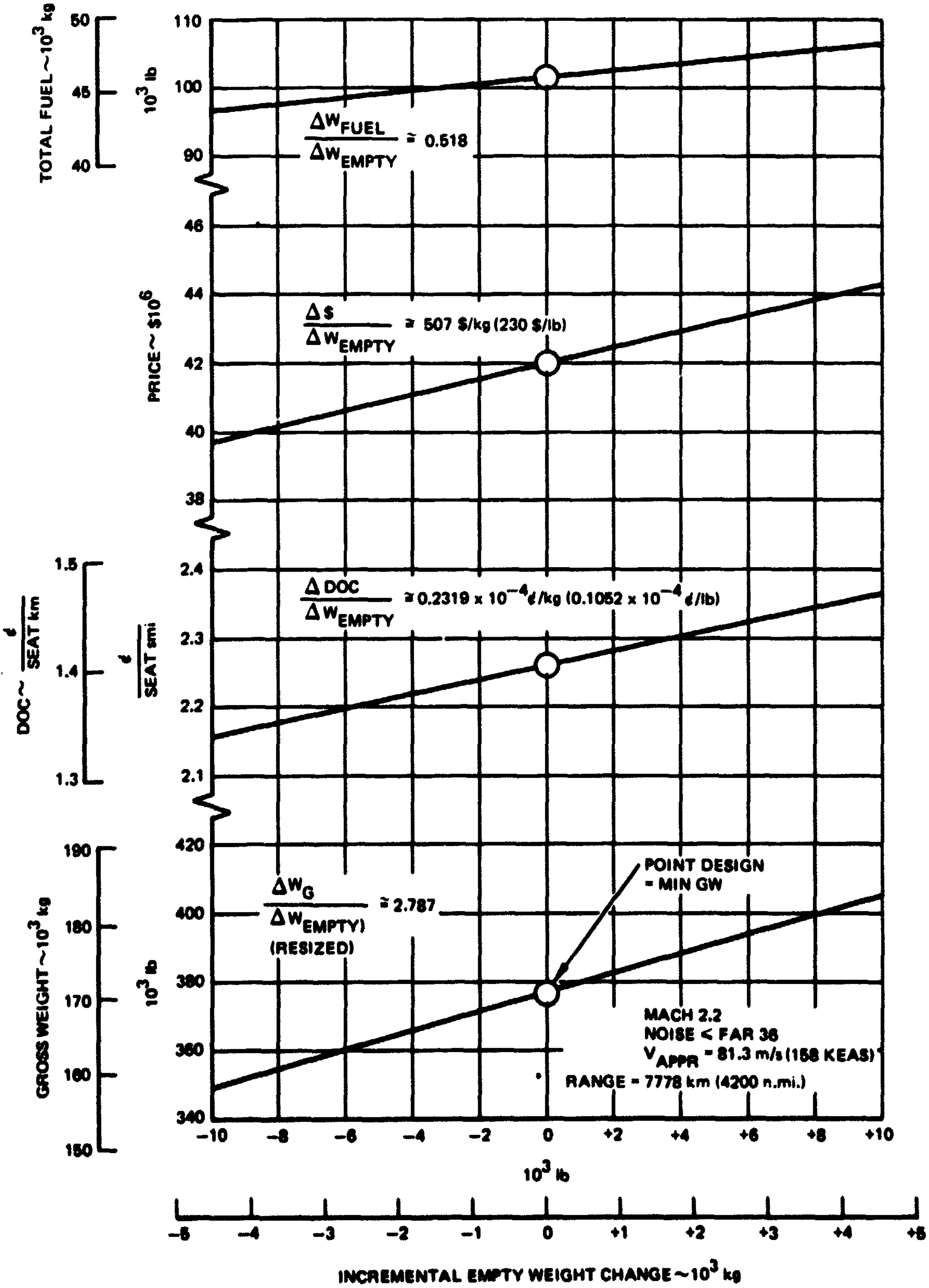

Figure 71. Empty Weight Change Sensitivity - M2.2 LH_2 SCV (aircraft resized).

design range of the Mach 2.2 LH_2 vehicle can be greatly extended with only a reasonable increase in gross weight (a 31 percent increase for a 1,200 n. mi. range increment). For convenience, the sensitivity of each of the characteristics around the design point, indicated by the circle on the plots, is listed. For example, the plot of gross weight vs range indicates a growth of about 77.2 lb in gross weight would be required for every nautical mile increase in design range.

Figures 71 and 72 illustrate the effect of a change in empty weight as would be the case if equipment or structural weight were to increase or decrease from the original target weight. Two different situations were examined. In Figure 71, the assumption is that the vehicle design has not been frozen and the option exists to resize the vehicle to accomplish the original mission. This might be the case if, for example, the target wing weight were exceeded by 4536 kg (10,000 lb) at the original design gross weight. This causes a subsequent increase in fuel, propulsion, structure, etc., and finally a further increase in the wing itself to maintain the vehicle performance. The sensitivity or growth factor shown is about 1.27 kg (2.79 lb) of gross weight per pound of original empty weight change. The sensitivity of DOC, price and fuel required is also shown. Figure 72 assumes that the design gross weight has been frozen and that the fuel available (and fuel volume) must be adjusted to reflect the change in empty weight. The result is a change of about 0.034 n. mi. per pound of empty weight change. DOC, price and fuel sensitivities are also shown.

Figure 73 shows the effect of a uniform change in engine specific fuel consumption (SFC) on total range and DOC. In the range tradeoff the vehicle is not resized but flies at different ranges as the fuel consumption is varied. This is a significant sensitivity and allows an increase of 50 n. mi. with each 1 percent decrease SFC. The DOC tradeoff is shown to be much less sensitive.

Figure 74 is simply the increase in range which would be possible if payload is off-loaded. The increase is about 0.067 km/kg (0.0163 n. mi. per lb) of payload. It should be noted that as designed, the point design vehicle is fuel volume limited and no additional fuel can be added as the payload is reduced as is the case for the conventional, hydrocarbon fueled aircraft. In the real world, the advisability of carrying extra tankage to increase flexibility would be a matter of route structure and economics. The method of construction of the vehicle would allow enlargement of the tanks by a simple fuselage plug within the limits of aircraft strength and the wing area selected.

Of equal importance to engine specific fuel consumption is the drag level. Figure 75 shows a change of about 77.6 km (41.9 n. mi.) distance and 0.0145 ¢ in DOC for each drag count. The analysis assumed that the change in nominal drag was applied uniformly to the zero-lift drag at all Mach numbers.

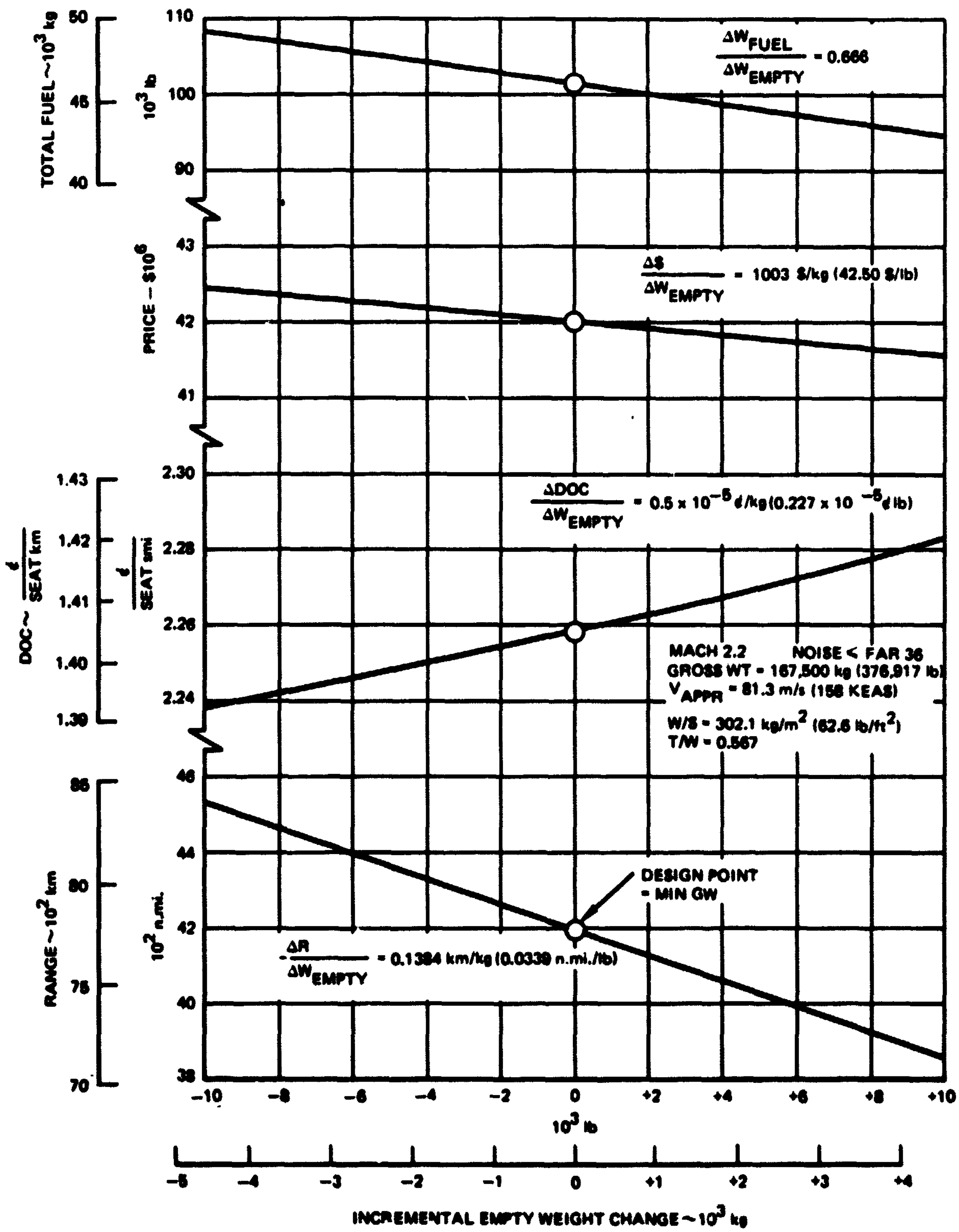

Figure 72. Empty Weight Change Sensitivity - M2.2 LH_2 SCV (constant gross weight).

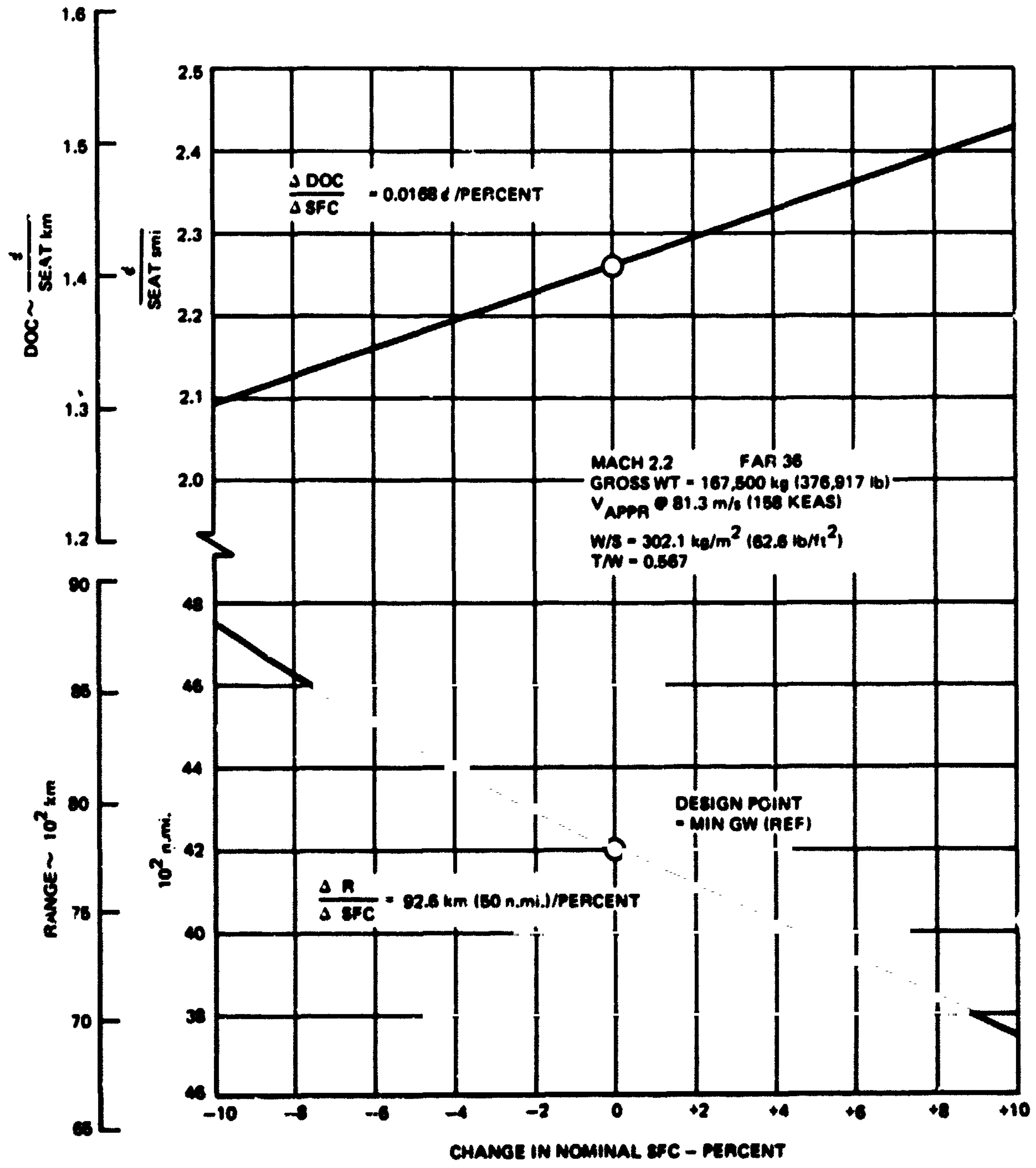

Figure 73. Sensitivity to SFC - M2.2 LH_2 SVC.

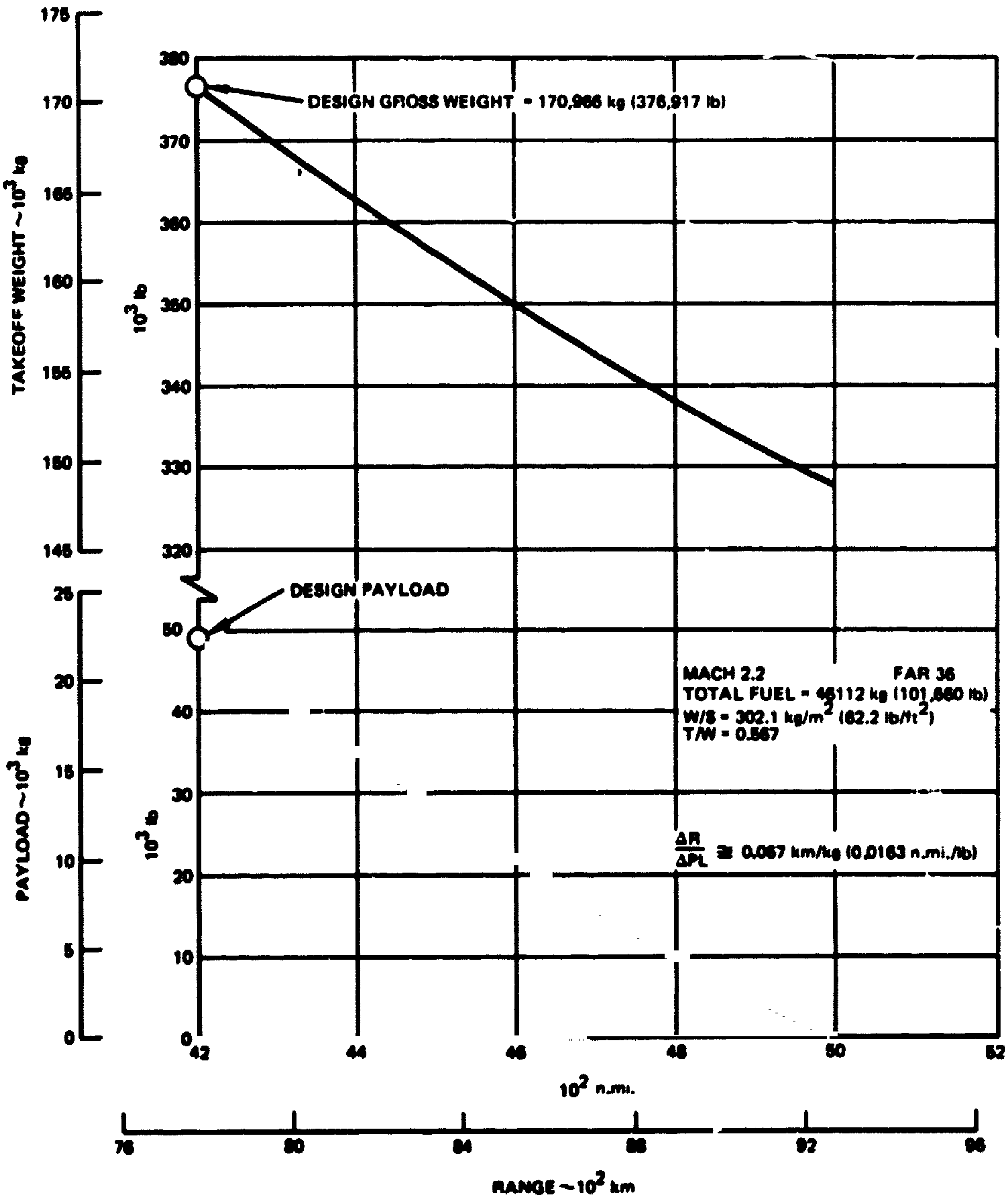

Figure 74. Range Sensitivity to Payload - M2.2 LH_2 SCV.

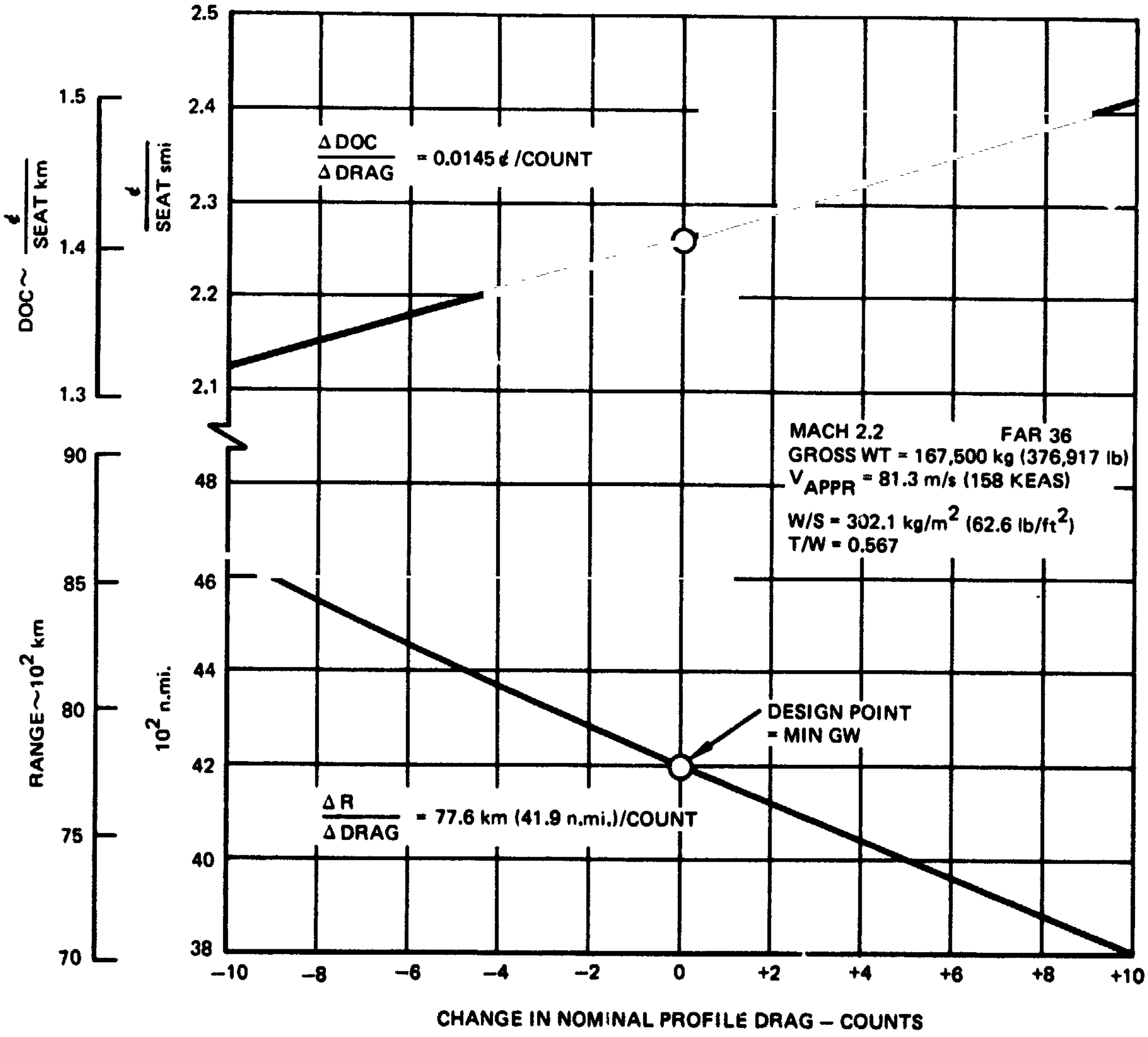

Figure 75. Sensitivity to Drag Level - M2.2 LH_2 SCV.

6.4 Comparison with Jet A Design

As with the Mach 2.7 aircraft, Table 18 presents a comparison of the minimum gross weight LH_2 Mach 2.2 aircraft to the Jet A vehicle designed to the same technology and ground rules. The LH_2 SCV gross weight is 56 percent, and the operating empty weight is 84 percent that of the Jet A SCV. The total fuel required differs by a factor of 3.50 while the SFC ratio is 2.47.

Another factor of interest to compare the relative desirability of the two aircraft is energy expended per available seat mile. The Jet A SCV uses 24 percent more Btu/available seat mile than does the LH_2 design, viz., 3,227 kJ/seat km (5,672 Btu/seat n.mi.) vs 2,608 kJ/seat km (4,583 Btu per seat mile). It should be noted that neither of these numbers includes the energy required to produce the fuels, nor to transport them to the airport. Both values represent just the energy contained in the fuel required by the respective aircraft to accomplish the given mission.

Table 19 is a comparison of the group weight statement of both aircraft and shows the penalty paid for LH_2 fuel tankage and insulation.

Table 20 lists some pertinent cost data for comparison of the two types of aircraft. The costs are expressed in terms of 1973 dollars, calculated on the bases noted. The LH_2 SCV aircraft is $9.8 million cheaper than the comparable Jet A airplane in production, and development is estimated to cost 200 million dollars less.

Figure 76 presents a plot of DOC for both aircraft as a function of fuel cost. This shows that for the subject Mach 2.2 aircraft, at 9.7¢/liter (36.6¢/gal) for jet fuel the operators could afford to pay $3.72/GJ ($3.92 per 10^6 Btu) for LH_2. The general expression to represent the cost differential which can be paid for LH_2 over the cost of Jet A fuel, and still produce parity of DOC for the Mach 2.2 SCV's is

$$\Delta C_{LH_2} = 0.197\ C_{JA} + 0.400$$

where cost of both fuels is expressed in $\$/10^6$ Btu.

The general comments made in Section 5.4 for the Mach 2.7 aircraft apply equally to the Mach 2.2 design.

Table 18. Comparison of Mach 2.2 Jet A and LH_2 Fueled SCV's

Fuel			Jet A		LH_2		Ratio Jet A/LH_2
Payload	kg	(lb)	22,226	(49,000)	22,226	(49,000)	
Range	km	(n.mi.)	7,783	(4,200)	7,783	(4,200)	
Cruise Speed (Std. day + 8°C)		Mach		2.12		2.12	
Takeoff Gross Weight	kg	(lb)	305,320	(673,110)	170,970	(376,920)	1.79
Operating Empty Weight	kg	(lb)	121,890	(268,720)	102,630	(226,260)	1.19
Fuel Weight, Block	kg	(lb)	137,420	(302,950)	39,600	(87,300)	3.48
Total	kg	(lb)	161,200	(355,390)	46,110	(101,660)	3.50
Fuel Volume	m^3	(ft^3)	209	(7,370)	697	(24,630)	3.34
Wing Area	m^2	(ft^2)	716	(7,702)	559	(6,020)	1.28
Wing Loading (W/S Takeoff	kg/m^2	(lb/ft^2)	426.7	(87.4)	305.6	(62.6)	
Landing	kg/m^2	(lb/ft^2)	234.3	(48)	234.3	(48)	
Span	m	(ft)	38.4	(126)	33.9	(111.3)	1.13
Overall Length	m	(ft)	90.5	(297)	104.0	(341.2)	0.87
Lift/Drag (cruise)				8.18		7.25	1.13
Specific Fuel Consumption (cruise)	(kg/hr)/daN	((lb/hr)/lb)	1.311	(1.288)	0.531	(0.522)	2.47
Thrust/Weight (SLS)	N/kg	-	4.90	(0.500)	5.56	(0.567)	
Thrust Per Engine	N	(lb)	374,250	(84,140)	237,640	(43,430)	1.58
FAR Takeoff Field Length	m	(ft)	2475	(8,120)	1611	(5,285)	1.54
FAR Landing Field Length	m	(ft)	2463	(8,080)	2411	(7,910)	1.02
Landing Approach Speed	m/sec	(KEAS)	81.3	(158)	81.3	(158)	
Weight Fractions	Percent						
Fuel				42.8		27.0	
Payload				7.3		13.0	
Structure				23.9		34.7	
Propulsion				9.9		15.0	
Equipment and Operating Items				6.1		9.3	
Energy Utilization	kJ/seat km	Btu/seat n.mi	3227	(5,672)	2608	(4,583)	1.24

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Table 19. Group Weight Statement - Mach 2.2
Jet A and LH_2 SCV's

	Jet A		LH_2 (Min. W_G)	
	kg	lb	kg	lb
Takeoff Weight	(305,322)	(673,108)	(170,970)	(376,917)
Fuel Available	161,204	355,388	46,113	101,660
Zero Fuel Weight	(144,118)	(317,721)	(124,857)	(275,257)
Payload	22,226	49,000	22,226	49,000
Operating Weight	(121,892)	(268,721)	(102,630)	(226,257)
Operating Items	2,476	5,459	2,440	5,378
Standard Items	2,247	4,953	2,591	5,713
Empty Weight	(117,169)	(258,309)	(97,599)	(215,166)
Wing	35,230	77,667	21,230	46,803
Tail	2,495	5,500	2,164	4,771
Body	16,784	37,002	24,868 ①	54,824 ①
Landing Gear	12,738	28,082	7,783	17,159
Surface Controls	3,859	8,507	2,005	4,420
Nacelle and Engine Section	1,986	4,379	1,192	2,627
Propulsion	(30,282)	(66,760)	(25,603)	(56,444)
Engines	21,809	48,080	13,085	28,847
Thrust Reversal (in engines)	-	-	-	-
Air induction System	5,420	11,949	3,259	7,185
Fuel System	2,331	5,138	6,634 ②	19,035 ②
Engine Con-rols and Starter	723	1,593	625	1,377
Instruments	521	1,148	497	1,096
Hydraulics	2,239	4,936	1,254	2,764
Electrical	1,973	4,349	2,087	4,600
Avionics	863	1,903	863	1,903
Furnishings and Equipment	5,228	11,526	5,228	11,526
Environmental Control System	2,971	6,550	2,826	6,229
Auxiliary Gear	0	0	0	0

1 Includes 11,081 kg (24429 lb) of fuel tankage and interconnect structure.

2 Consists of: 3,417 kg (7,533 lb) insulation
2,799 kg (6,171 lb) heat shield
2,418 kg (5,331 lb) fuel system

Table 20. Cost Comparison: Jet A vs LH_2 Mach 2.2 SCV's (Refer to Table 18 for vehicle data)

Costs*		Aircraft			
		Jet A		LH_2	
RDT&E	$\$10^6$				
Engine		866		805	
Airframe		2431		2289	
Total		3297		3094	
Production Aircraft, each	$	51,769,000		42,000,000	
Direct Operating Cost (DOC)		¢/seat km	¢/seat s mi	¢/seat km	¢/seat s mi
Flight Crew		0.069	0.111	0.070	0.113
Fuel and Oil		0.625	1.006	0.784	1.261
Insurance		0.093	0.149	0.078	0.125
Depreciation		0.297	0.478	0.250	0.402
Maintenance		0.266	0.428	0.222	0.357
Total		1.350	2.172	1.403	2.258
Indirect Operating Cost (IOC)		0.552	0.888	0.530	0.853

*Basis for Costs:

- 1973 dollars
- production of 600 aircraft
- passenger load factor = 0.55
- aircraft utilization = 3600 hr/year
- fuel cost: Jet A = 1.90/GJ ($\$2/10^6$ Btu = 24.8¢/gal = 3.68¢/lb)
  LH_2 = 2.85/GJ ($\$3/10^6$ Btu = 15.48¢/lb).

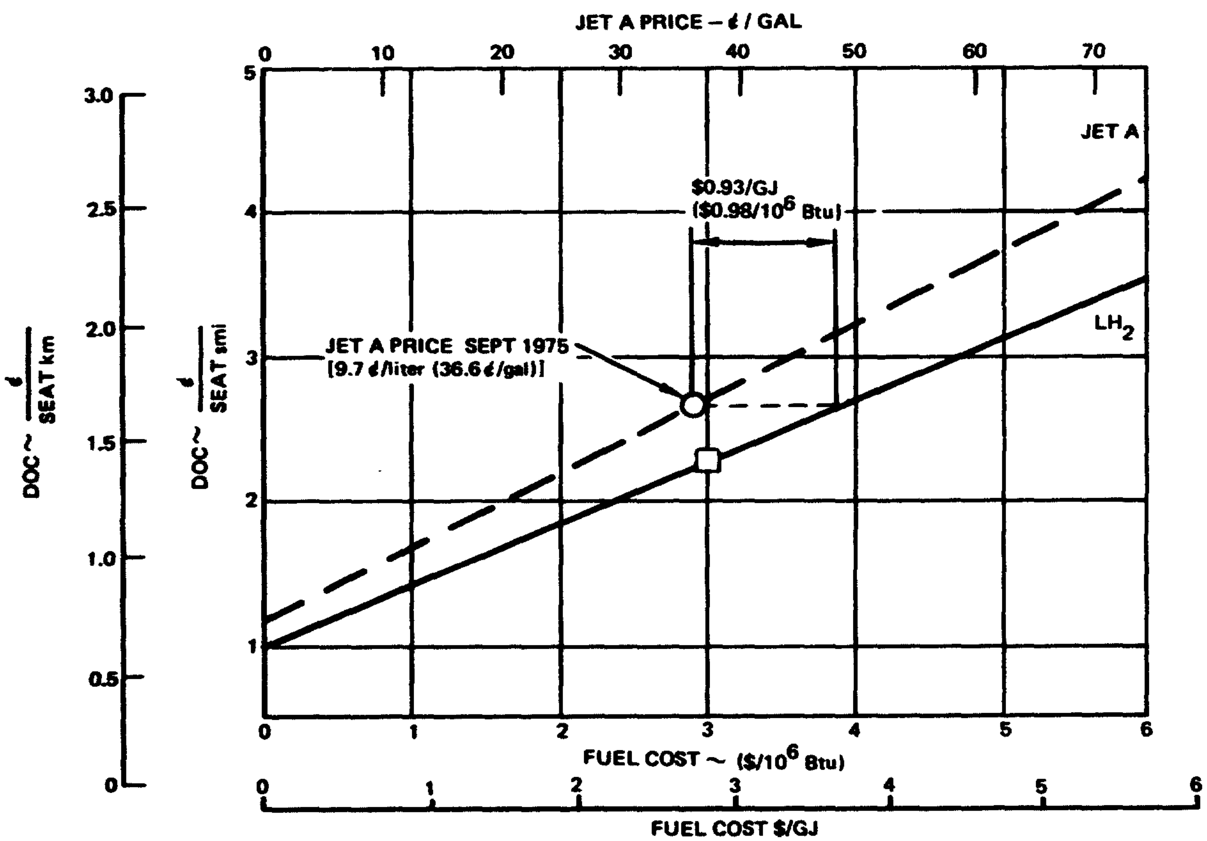

Figure 76. DOC vs Fuel Cost - M2.2 SCV's.

7. RESEARCH AND TECHNOLOGY RECOMMENDATIONS

Technology development required to permit initiation of development of a LH_2 fueled supersonic cruise transport aircraft is essentially as defined in the Final Report of the previous study (Reference 1). Modifications resulting from changes in the design of the baseline aircraft, plus further consideration of the technology requirements of LH_2 fueled aircraft in general, resulting from the study of subsonic transport aircraft (Reference 7), have been incorporated in Table 21, a summary of a program of recommended development for LH_2 SCV's.

In addition to the technology development listed in Table 21, a very significant event which should precede this program is an assessment of the impact the initiation of use of hydrogen as fuel for commercial transport aviation would have on society in general.* In this study a hypothetical but realistic scenario depicting the transition to hydrogen would be developed, and the economic ramifications, the institutional barriers and incentives, and the social dislocations and opportunities of all major stakeholder classes in society would be disclosed. Stakeholder classes whose participation in the evolutionary scenario would be described include the following:

- airlines
- aircraft manufacturers
- fuel suppliers
- airport operators
- consumers
- government regulators

While not classified as a "technology development," this study would provide important input and an order of priorities for the technical work. In addition it would acquaint, and hopefully convince, many stakeholders of the need for early conversion of commercial aviation to hydrogen fuel.

The technology development program shown in Table 21 relates specifically to the hydrogen peculiar items of the subject aircraft. It should be recognized that these items are in addition to the usual program of development associated with design of a new, advanced type of aircraft. Further, the problems of developing adequate supplies of liquid hydrogen for use at designated major airports around the country, and overseas, are not included.

*This study proposed by Stanford Research Institute, September 26, 1975.

Table 21. Technology Development Required for LH_2-Fueled SCV Aircraft

1. Duct-burning turbofan engines designed to operate efficiently on hydrogen fuel.

2. Lightweight cryogenic insulation, e.g., PVC or reinforced-polyurethane foam, which is impervious to air and which can be bonded to an aluminum tank. Must demonstrate an acceptable useful life.

3. Lightweight heat shield structural material having low thermal conductivity, e.g., fiberglass core, graphite/Kevlar/polyimide faced honeycomb sandwich, which is satisfactory for airline service.

4. Lightweight aluminum tankage, capable of withstanding airline service, plus exposure to cryogenic temperatures and attendant thermal stresses.

5. A satisfactory vent system for the LH_2 fueled aircraft.

6. An aircraft fuel feed system including pumps, valves, quantity sensors, heat exchanger, pressurization system and control, and vacuum-jacketed lines acceptable for airline service.

7. A ground supply and fuel handling system for use at airline terminals.

8. An acceptable specification and set of standards for handling liquid hydrogen in routine airline operation.

9. A flight demonstration program involving conversion of existing subsonic aircraft to LH_2 fuel and operation of the aircraft in extended use simulating airline operations. Purpose would be to learn practical aspects of handling LH_2 fuel and to demonstrate feasibility of using it in a commercial environment.

These latter problems are currently being addressed in other, separately funded studies being conducted for both NASA and ERDA.

8. CONCLUSIONS

This study has confirmed the findings of the original program (Reference 1) which investigated the potential of using liquid hydrogen as fuel in advanced designs of supersonic transport aircraft. Significant benefits can be realized in performance, size, weight, energy utilization, cost, noise, sonic boom, and environmental pollution. All this can be realized in addition to perhaps the most important benefit of all, relief from dependency on a petroleum product which, by the time an advanced design SCV might become operational, could be well on the way to becoming unavailable for use as an aircraft fuel.

The present study provided a more critical evaluation of the aerodynamic characteristics of the subject LH_2 fueled aircraft, compared to that of the original study (Reference 1). In particular, wave drag and aircraft stability and control requirements were more rigorously evaluated. In addition, the program included an updated assessment of the weight of the wing. It is a smaller structure compared with the equivalent Jet A fueled design, with compensating structural design conditions, i.e., lower wing loading but no load relief due to no fuel being carried in the wing. A third major difference in the designs of the LH_2 fueled SCV's of the present study and those of the original work was a more exact representation of the properties of hydrogen/air combustion in evaluation of performance of the turbofan engines.

The net result of these critical reviews of the design basis of the subject aircraft, compared with corresponding designs of Jet A fueled SCV's, was a very slight decrease in the weight advantage of the LH_2 aircraft. For example, for the M2.7 aircraft the operating empty weight ratio (Jet A/LH_2) is calculated to be 1.29 in the present study. Reference 1 showed this ratio to be 1.386 in the original study. The ratio of block fuel weights (Jet A/LH_2) is 3.88, it was 4.00. Vehicle gross weight ratios are 1.93, they were 2.04.

The analysis to determine the potential of minimizing energy expenditure in performing the baseline mission showed that only minor saving can be accomplished. The Mach 2.7 SCV designed for minimum fuel weight required 97 percent of the energy per seat kilometer of the version designed for minimum gross weight. In the case of the Mach 2.2 aircraft the minimum fuel weight version used 95 percent of the energy required by the minimum gross weight design.

At both cruise speeds the aircraft designed to provide minimum direct operating cost proved to be a good compromise between the alternate choices based on minimum gross weight and minimum energy. Varying fuel costs, within bounds of \$1.90/GJ (\$2 per 10^6 Btu) and \$5.70/GJ (\$6 per 10^6 Btu) for LH_2, produced very little difference in the choice of thrust-to-weight ratio which established the preferred aircraft design.

Hydrogen-fueled SCV aircraft can be designed to be 5 dB quieter than FAR Part 36 with no penalty in fuel expenditure. In fact, the Mach 2.7 LH_2 SCV designed for minimum fuel weight was nearly 6 dB quieter than the specification. The comparable M2.2 LH_2 SCV was exactly 5 dB quieter. On the other hand, designing either Mach 2.7 or 2.2 LH_2 SCV's to be 10 dB quieter than the specification involves sizeable penalties in gross weight, fuel weight, operating empty weight, aircraft price, and direct operating cost.

9. RECOMMENDATIONS

In view of the many attractive advantages, it is recommended that development of technology for LH_2 fueled supersonic transport aircraft be actively pursued. The following actions are recommended to further explore the potential of such aircraft and to establish technology feasibility.

- Perform detailed studies of a selected point-design aircraft to establish better definition of the design, including windtunnel testing.
- Build and test insulated model cryogenic tanks to determine their capability for withstanding thermal cycling under simulated structural loading conditions.
- Investigate thermal protection system concepts.
- Establish detailed design characteristics of the aircraft fuel system, including all significant components. Build breadboard model and run flow tests with cryogenic liquids, including liquid hydrogen.
- Study alternate configuration concepts of LH_2 SCV's which appear to have advantage.
- Study aircraft ground handling and refueling operations to establish specifications for equipment and procedures to assure safe, economical practices.
- Initiate a flight demonstration program based on conversion of two existing subsonic aircraft to LH_2 fuel, to learn the practical aspects of handling hydrogen in simulated airline operations.
- Assess the impact conversion of the air transport industry to LH_2 fuel would have on all affected aspects of U.S. society.

APPENDIX A

REFERENCE JET A FUELED SUPERSONIC CRUISE VEHICLES

A1 - Mach 2.7 Jet A SCV (CL 1607-5)

A2 - Mach 2.2 Jet A SCV (CL 1607-13)

SUMMARY NO. 165 ASSET PARAMETRIC ANALYSIS AUGUST 14 19--

AIRCRAFT MODEL —CL 1607-5
I.O.C. DATE —1990
DESIGN SPEED —SUPERSONIC

ENGINE I.D. -- 97000
SLS SCALE 1.0 = 78000
NUMBER OF ENGINES = 4.

WING QUARTER CHORD SWEEP = 6..50 DEG
WING TAPER RATIO = 0.0

1 W/S	49.7	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
2 T/W	0.450	0.0	0.0	0.0	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	1.61	0.0	0.0	0.0	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	2.65	0.0	0.0	0.0	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	762171	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
7 FUEL WEIGHT	395757	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
8 OP. WT. EMPTY	317611	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
9 ZERO FUEL WT.	366615	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
10 THRUST/ENGINE	88817	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
11 ENGINE SCALE	1.114	0.0	0.0	0.0	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	11094.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
13 WING SPAN	113.5	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	1112.0	0.0	0.0	0.0	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	473.3	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
16 BODY LENGTH	297.0	0.0	0.0	0.0	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—MILLION DOLLARS/AIRCRAFT																
17 FLYAWAY COST	6.346	0.0	0.0	0.0	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 AIRFRAME COST	78.441	0.0	0.0	0.0	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 ENGINE COST	1.136	0.0	0.0	0.0	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 AVIONICS COST	1.774	0.0	0.0	0.0	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—BILLION DOLLARS																
21 TOTAL INVEST.	6.900	0.0	0.0	0.0	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 TOTAL SYSTEM	4.143	0.0	0.0	0.0	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																

APPENDIX A1

Reference Jet A Airplane (CL 1607-5)

Mach 2.7 SCV

Range = 4200 n. mi.

Payload = 234 Passengers (49,000 lb)

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

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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	762171.	3600.	3600.	0.	0.	10.0	10.0	0.	-47101.	0.	0.0	0.567	0.0
POWER 2	0.	0.300	758571.	6359.	9959.	0.	0.	0.5	10.5	0.	47201.	0.	5.23	1.369	0.0
CLIMB	0.	0.300	752212.	7450.	17409.	5.	5.	1.1	11.6	0.	47201.	0.	7.51	1.444	0.0
CRUISE	5000.	0.413	744762.	4314.	21723.	0.	5.	4.0	15.6	0.	-47101.	0.	8.33	0.726	0.0
ACCEL	5000.	0.413	740448.	1824.	23547.	2.	6.	0.3	15.9	0.	47201.	0.	10.05	1.472	0.0
CLIMB	5000.	0.537	738624.	23323.	46870.	39.	45.	5.8	21.0	0.	47201.	0.	11.34	1.503	0.0
CLIMB	34000.	0.904	715301.	53168.	100038.	284.	329.	15.8	36.8	0.	47201.	0.	9.36	1.533	2.72
CLIMB	63000.	2.403	662133.	811.	100849.	7.	336.	0.3	37.1	0.	47201.	0.	8.77	1.563	1.96
CRUISE	64000.	2.420	661323.	224213.	325061.	3654.	3990.	143.2	180.3	0.	-47201.	0.	8.65	1.501	1.94
DECEL	73000.	2.420	437110.	94.	325155.	23.	4013.	0.9	181.2	0.	47501.	0.	6.48	-0.769	1.45
DESCENT	73000.	2.369	437016.	1265.	326420.	156.	4169.	13.3	194.5	0.	47501.	0.	5.21	-0.385	1.51
CRUISE	73000.	2.020	435751.	1546.	327967.	31.	4200.	1.2	195.7	0.	-47201.	0.	8.54	1.514	1.43
CRUISE	5000.	0.413	434206.	2625.	330592.	0.	4200.	5.0	200.7	0.	-47101.	0.	11.29	0.823	0.0
RESET	0.	0.0	431579.	0.	330592.	0.	4200.	0.0	200.7	0.	0.	0.	0.0	0.0	0.0
RESET	0.	0.0	431579.	0.	330592.	-4200.	0.	*****	0.0	0.	0.	0.	0.0	0.0	0.0
RESERVE	0.	0.0	431579.	23141.	353734.	0.	0.	0.0	0.0	0.	0.	0.	0.0	0.0	0.0
CLIMB	0.	0.200	408437.	2795.	356529.	2.	2.	0.4	0.4	0.	47201.	0.	9.62	1.444	0.0
CLIMB	1500.	0.505	405642.	13632.	370161.	23.	25.	3.1	3.5	0.	47201.	0.	9.91	1.506	0.0
CRUISE	43000.	0.900	392010.	10320.	380481.	165.	190.	18.9	22.4	0.	-47101.	0.	11.03	0.935	0.0
DESCENT	43000.	0.905	381690.	761.	381242.	58.	248.	8.0	30.4	0.	47501.	0.	4.92	-0.491	0.0
CRUISE	43000.	0.900	380929.	712.	381954.	12.	260.	1.3	31.7	0.	-47101.	0.	11.02	0.930	0.0
LOITER	15000.	0.403	380217.	13799.	395752.	0.	260.	30.0	61.7	0.	-47101.	0.	11.59	0.842	0.0

TOGW= 762171.3   FUEL A=395752.3   FUEL R=395752.3

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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)
	11096.2	133.53	0.0	68.583	72.500	166.16	110.78
INBOARD WING—	AREA(SQ.FT)	EXP. AREA	L.E. SWEEP	REF L(FT)	SFLE(SQ.FT)	AVG T/C	
	11096.2	9355.6	72.50	101.73	0.0	2.65	
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	101.73	0.0	2.65	
TOTAL WING—	AREA(SQ.FT)	EFF AR	AVG T/C	CR(FT)	CT(FT)	(B/2)/LW	P
	11096.2	1.61	2.65	166.16	0.0	0.315	0.392
WING TANK—	CBAR1(FT)	CBAR2(FT)	FTL(FT)	FVWING(CU FT)	FVBOX(CU FT)		
	152.06	0.0	61.10	4694.86	2863.63		
FUSELAGE—	LENGTH(FT)	S WET(SQ FT)	HW(FT)	EQUIV D(FT)	SFE(SQ FT)		
	297.00	7676.0	10.91	13.22	137.50		
	WW(FT)	HH(FT)	SWB(SQ FT)	FVB(CU FT)			
	11.33	11.50	9177.00	20000.00			
TAIL—	SHT(SQ.FT)	SHTX(SQ.FT)	HT REF L(FT)	SVT(SQ.FT)	SVTX(SQ.FT)	VT REF L(FT)	
	1112.01	938.69	23.93	473.30	473.30	26.06	
PROPULSION—	ENG L(FT)	ENG D(FT)	POD L(FT)	POD D(FT)	POD S WET	NO. PODS	INLET L(FT)
	22.29	7.09	39.19	8.03	3955.69	4.	0.0

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T/C	AR	Λ	T/W
2.65	1.61	68.7	0.456

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

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
TAKE-OFF WEIGHT	(762171.)		
FUEL AVAILABLE	395752.	FUEL	51.92
ZERO FUEL WEIGHT	(366419.)		
PAYLOAD	49060.	PAYLOAD	6.43
OPERATING WEIGHT	(317419.)		
OPERATING ITEMS	5466.	OPERATING ITEMS	1.42
STANDARD ITEMS	5334.		
EMPTY WEIGHT	(306619.)		
WING	104477.		
TAIL	9397.		
BODY	39181.	STRUCTURE	25.92
LANDING GEAR	31017.		
SURFACE CONTROLS	8471.		
NACELLE AND ENGINE SECTION	4989.		
PROPULSION	(75761.)	PROPULSION	9.94
WEIGHT OF LIFT ENGINES	0.		
VECTOR CONTROL SYSTEM	0.		
ENGINES	49651.		
THRUST REVERSAL	0.		
AIR INDUCTION SYSTEM	19123.		
FUEL SYSTEM	5375.		
ENGINE CONTROLS + STARTER	1613.		
INSTRUMENTS	1237.		
HYDRAULICS	5799.		
ELECTRICAL	4562.		
AVIONICS	1903.	EQUIPMENT	4.37
FURNISHINGS AND EQUIPMENT	11526.		
ENVIRONMENTAL CONTROL SYSTEM	8297.		
AUXILIARY GEAR	0.		
A.M.P.R.	(239509.)	TOTAL	(100.00)

~~EXCESS FUEL CAPACITY - BODY [illegible]~~
~~EXCESS FUEL CAPACITY - WING 0.~~
~~EXCESS BODY LENGTH - FT 0.0~~

~~STRUCTURE ALUMINUM~~

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WEIGHTS MATRIX

ELEMENT/ / MATERIAL	AL	TIT.	STEEL	COMP.	OTHER	TOTAL
WING	4806.	89432.	2090.	6477.	1672.	104477.
TAIL	0.	5638.	0.	3759.	0.	9347.
FUSEL	0.	19473.	0.	13439.	6269.	39181.
L. G.	0.	6203.	9739.	3722.	11352.	31017.
NACELLE	112.	1237.	0.	999.	147.	2495.
AIR INDUCT	0.	3525.	9562.	5737.	0.	18823.
S. CTLS	0.	1949.	1440.	847.	4236.	8471.
TOTALS	4918.	127757.	22830.	34979.	23675.	214161.

COST SUMMARY

RDT AND E

	TOTAL*
PROTOTYPE AIRCRAFT	963.91
DESIGN ENGINEERING	908.87
DEVELOPMENT TEST ARTICLES	446.60
FLIGHT TEST	118.06
ENGINE DEVELOPMENT CRUISE	1001.29
ENGINE DEVELOPMENT LIFT	0.0
AVIONICS DEVELOPMENT	0.0
MAINTENANCE TRAINER DEVEL	0.0
OPERATOR TRAINER DEVELOP	0.0
DEVELOPMENT TOOLING	720.99
SPECIAL SUPPORT EQUIPMENT	19.28
DEVELOPMENT SPARES	143.86
TECHNICAL DATA	21.61
TOTAL RDTE	4344.47

INVESTMENT

	TOTAL*	PER PROD A/C**
PRODUCTION AIRCRAFT	36844.92	61408.21
PRODUCTION ENGINEERING	0.0	0.0
MAINTENANCE TRAINERS	0.0	0.0
OPERATOR TRAINERS	0.0	0.0
PRODUCTION TOOLING	288.81	481.35
SPECIAL SUPPORT EQUIPMENT	1510.64	2517.74
PRODUCTION SPARES	5390.42	6984.04
TECHNICAL DATA	220.7	366.96
TOTAL INVESTMENT	44254.97	73758.25

DIRECT OPERATIONAL COST (DOC)

	C/SM***	PERCENT
FLIGHT CREW	0.09542	5.38013
FUEL AND OIL	0.58753	33.12669
INSURANCE	0.15386	8.67518
DEPRECIATION	0.49505	27.91241
MAINTENANCE	0.44172	24.90562
TOTAL DOC	1.77359	100.000

INDIRECT OPERATIONAL COST (IOC)

	C/SM***	PERCENT
SYSTEM	0.00400	0.44337
LOCAL	0.19341	21.46818
AIRCRAFT CONTROL	0.00512	0.56870
CABIN ATTENDANT	0.06867	7.62464
FOOD AND BEVERAGE	0.02373	2.63467
PASSENGER HANDLING	0.13656	15.17208
CARGO HANDLING	0.00849	0.94210
OTHER PASSENGER EXPENSE	0.33550	37.27577
OTHER CARGO EXPENSE	0.00278	0.30863
GENERAL + ADMINISTR.	0.12179	13.53126
TOTAL IOC	0.90005	100.000

MISC. DATA

RANGE (ST. MILES)	4033.18
BLOCK SPEED (MPH)	1344.21
FARE ($)	248.73
FLEET SIZE	14.02
PRODUCTION BASIS	600.00
REV.PASSENG.(MIL.PER YR)	1.81
AVER. CARGO PER FLIGHT	2000.00
FLIGHT PER A/C PER YEAR	1001.24

RETURN ON INVESTMENT (ROI)

TOTAL REVENUE PER YEAR *	469.74
TOTAL EXPENSE PER YEAR *	424.49
TOTAL INVESTMENT * INCL. FACILITIES	1135.69
ROI BEFORE TAXES	7.97
ROI AFTER TAXES	4.14

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

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RESEARCH DEVELOPMENT TEST EVALUATION (RDTE)

	DEVELOPMENT AND DESIGN		CONTRACTOR TEST AND EVALU		DEVELOPMENT AIRCRAFT		TOTAL RDT AND E	
AIRFRAME		1417.28		491.01		675.39		2583.67
ENGINEERING								
HOURS	45499.		10487.		3321.		5930[illegible].	
LABOR RATE	8.17		8.17		8.17		8.17	
OVERHEAD RATE	9.20		9.20		9.20		9.20	
TOTAL		790.32		182.76		57.69		1030.18
TOOLING								
HOURS	31069.		2768.		5536.		39371.	
LABOR RATE	6.09		6.09		6.09		6.09	
OVERHEAD RATE	12.36		12.36		12.36		12.36	
TOTAL		626.95		51.07		102.13		780.15
MANUFACTURING								
HOURS			11071.		22142.		33214.	
LABOR RATE			5.12		5.12		5.12	
OVERHEAD RATE			10.72		10.72		10.72	
TOTAL				175.37		350.74		526.10
QUALITY CONTROL								
HOURS			2214.		4428.		6643.	
LABOR RATE			6.29		6.29		6.29	
OVERHEAD RATE			10.72		10.72		10.72	
TOTAL				37.66		75.33		112.99
MATERIAL								
RAW AND PRCHSD			13.21		26.41		39.62	
PURCHASED EQUIP			24.53		49.06		73.5[illegible]	
TOTAL				37.73		75.47		113.20
MISCELLANEOUS								
HOURS			443.		886.		1329.	
LABOR RATE			5.12		5.12		5.12	
OVERHEAD RATE			10.72		10.72		10.72	
TOTAL				7.01		14.03		21.04
ENGINES		1001.29				83.91		1085.20
AVIONICS		0.0				2.00		2.00
PROFIT(AIRFRAME)		212.59		73.65		101.31		387.55
SPARES+TAXES						67.54		67.54
WARRANTY						33.77		33.77
SUBTOTAL		2631.16		564.66		963.91		4159.73
OTHER ITEMS								184.75
TOTAL (W. TE)								4344.47

PRODUCTION

	PRODUCTION YEARS 1	2	3	4	5	6	7	8	9	10	TOTAL
AIRFRAME	2174.55	2061.39	2290.79	2529.28	2750.83	2569.39	2434.37	2331.58	2249.47	2181.63	23583.31
ENGINEERING											
HOURS	7442.	6428.	6902.	7228.	7839.	6923.	6423.	6044.	5743.	5496.	66187.
LABOR RATE	9.17	9.17	9.17	9.17	9.17	8.17	9.17	9.17	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	129.62	111.65	118.14	125.55	132.69	120.26	111.56	104.93	99.76	95.46	1149.67
TOOLING											
HOURS	8955.	7713.	8162.	8673.	9167.	8309.	7707.	7253.	6892.	6595.	79424.
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	165.21	142.31	150.59	160.02	169.12	153.29	142.20	133.81	127.15	121.67	1465.38
MANUFACTURING											
HOURS	74622.	64277.	68016.	72278.	76389.	69235.	64227.	60440.	57430.	54955.	661369.
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	1182.02	1018.15	1077.39	1144.98	1210.00	1096.68	1017.36	957.36	909.69	870.49	10484.00
QUALITY CONTROL											
HOURS	14924.	12855.	13603.	14456.	15279.	13847.	12845.	12088.	11486.	10991.	132374.
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	253.86	218.67	231.39	245.89	259.87	235.54	218.50	205.62	195.38	186.96	2251.68
MATERIAL											
RAW AND PURCH	139.81	185.46	234.57	282.50	329.26	321.92	316.42	312.03	308.39	305.28	2734.63
PURCHASED EQUIP	257.79	344.42	435.63	524.64	611.49	597.85	587.63	579.48	572.72	566.96	5078.59
TOTAL	396.59	529.88	670.19	807.14	940.75	919.76	904.05	891.51	881.11	872.24	7813.22
MISCELLANEOUS											
HOURS	2435.	2571.	2721.	2891.	3056.	2769.	2569.	2416.	2297.	2198.	26475.
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	47.23	40.73	43.10	45.80	48.40	43.87	40.69	38.29	36.39	34.82	419.36
ENGINES	383.41	450.49	539.83	623.93	705.19	673.24	649.81	631.43	616.40	603.71	5876.43
AVIONICS	12.00	18.00	24.00	30.00	36.00	36.00	36.00	36.00	36.00	36.00	300.00
PROFIT	326.19	309.21	343.62	379.39	414.13	385.41	365.16	349.74	337.42	327.24	3537.50
INSUR.+TAXES	217.46	206.14	229.09	252.93	276.08	256.94	243.44	233.16	224.95	218.16	2358.33
WARRANTY	103.73	103.07	114.54	126.46	138.04	128.47	121.72	116.58	112.47	109.08	1179.17
TOTAL FLYAWAY	3222.37	3146.30	3540.86	3942.00	4330.27	4049.45	3850.49	3698.49	3576.71	3466.00	36844.92

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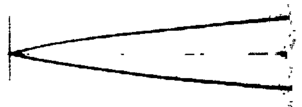
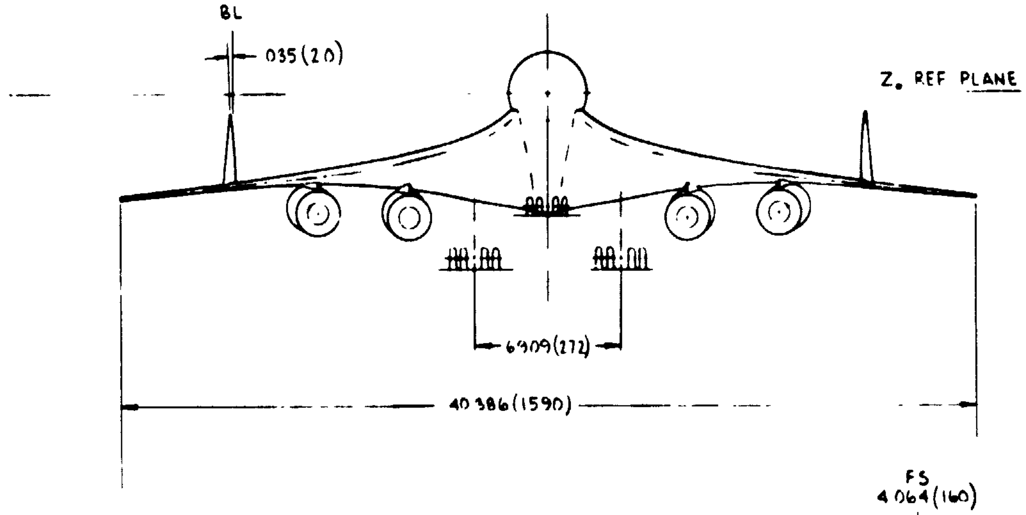
CHARACTERISTICS

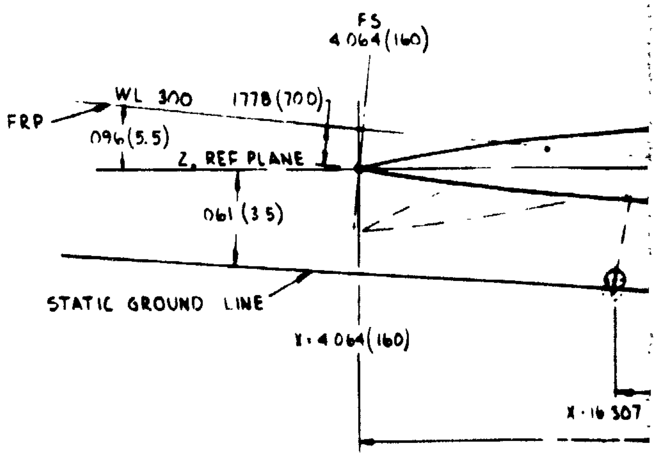
POWERPLANT - DUCT BURNING TURBO FAN
UNINSTD THRUST - 346 770 012 (77, 957) SLS

TAXI WEIGHT - 340,194.278 (150.000)

	WING	HORIZ TAIL	FUS VERT TAIL	WING VERT TAIL (EACH)
AREA	1005.397(10,822)	73 858(795)	26.942(290)	21.646(233)
ASPECT RATIO	1 62	1 707	0 517	0.495
TAPER RATIO	0 08	0 225	0.23	0 136
SPAN	40.386(1590)	11 217(36.8)	3.719 (12 2)	3.777 (1075)
ROOT CHORD	55.766(2195.5)	10.739(422 8)	11 737 (462 1)	11 643 (458.4)
TIP CHORD	4 460(175 6)	2.416(951)	2 700(106 3)	1 585 (62 4)
MAC	34.488 (1357 8)	7 455(293 5)	8 161 (321.3)	7.889 (310 6)
LE SWEEP	1 292 (74)	1.058 (60 64)	1.190 (68.20)	1 281 (73 42)
	1.236(70.84)			
	1128 (64 64)			







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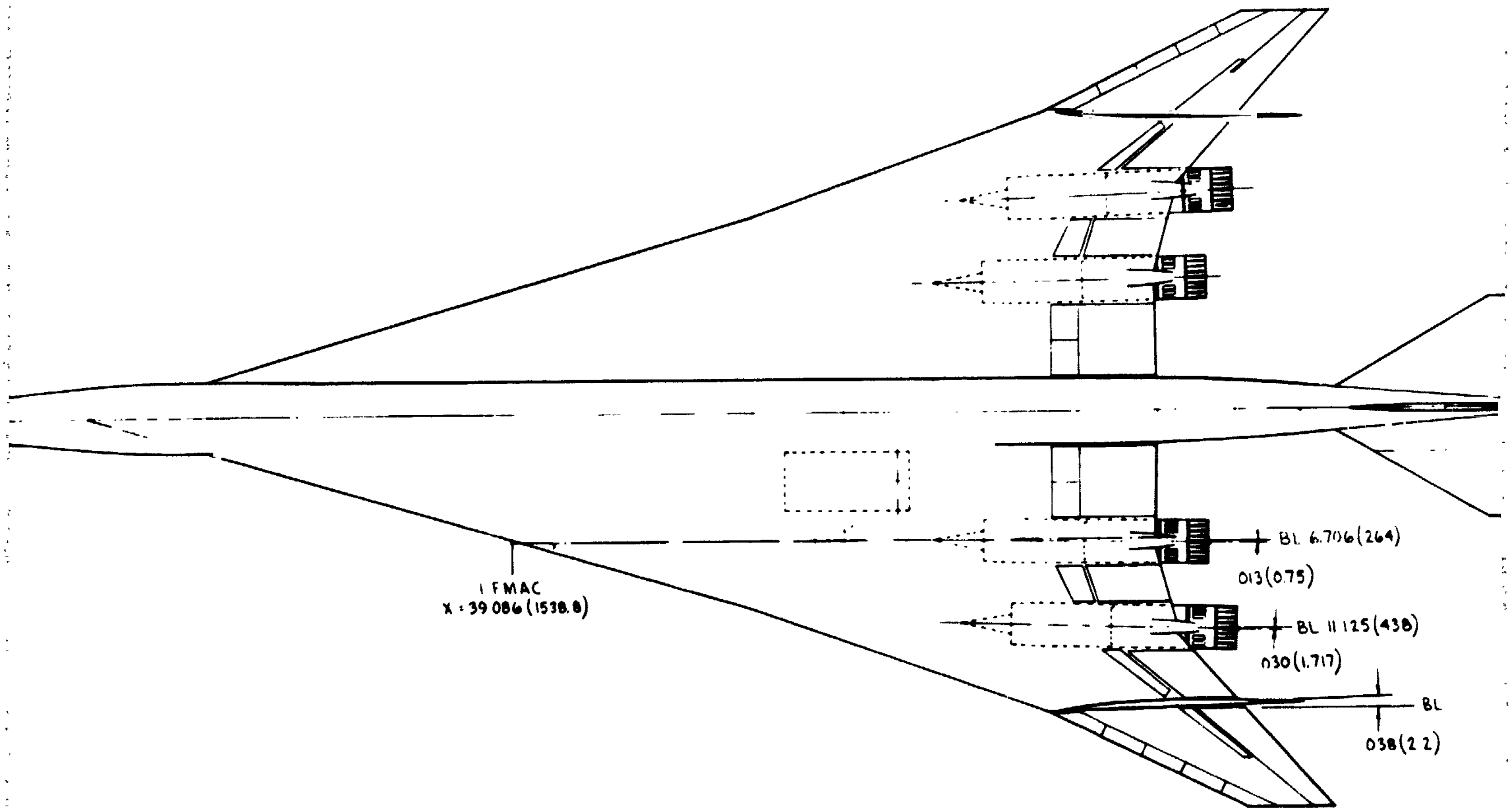
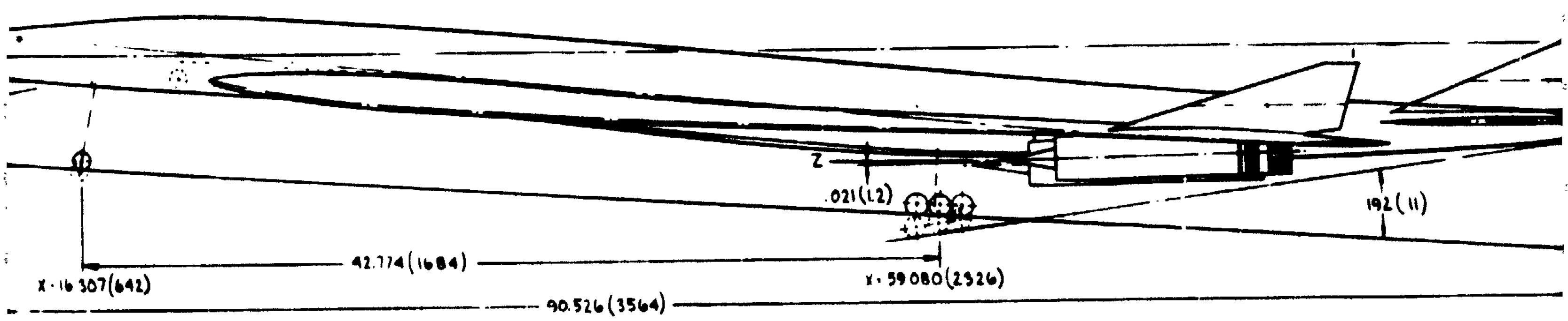
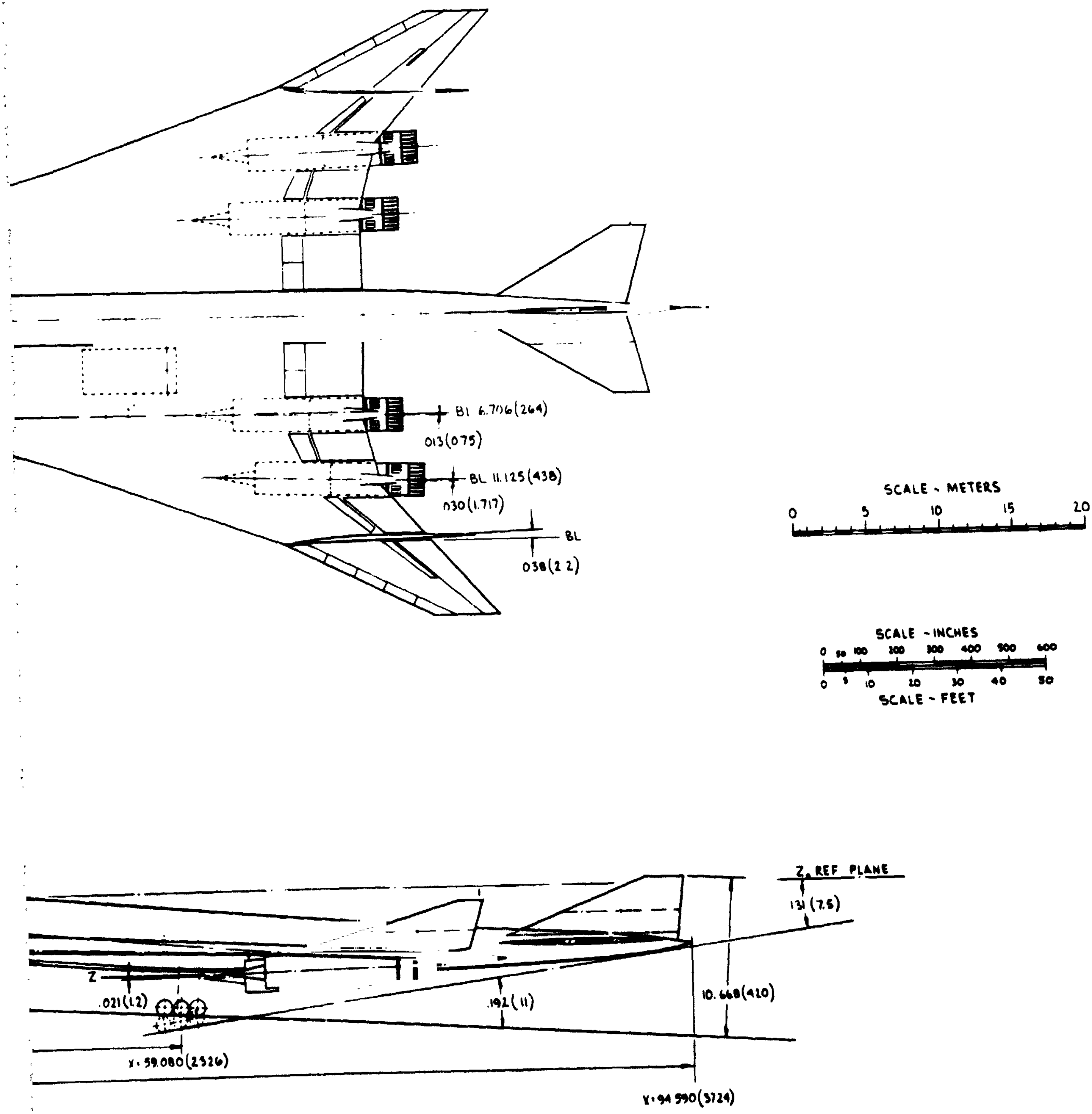
FOLDOUT FRAME /





NOTE:
1. DIMENSIONS IN SI UNITS (ENGLISH)

FOLDOUT FRAME 2



NOTE:
1. DIMENSIONS IN SI UNITS (ENGLISH)

Figure A-1. General Arrangement - Baseline Jet A M2.7 SCV

SUMMARY NO. 165 A S S E T P A R A M E T R I C A N A L Y S I S AUGUST 14 197

AIRCRAFT MODEL —CL 1607-9
I.O.C. DATE —1980
DESIGN SPEED —SUPERSONIC

ENGINE I.D. — 197000
SLS SCALE 1.0 = 79466
NUMBER OF ENGINES = 4.

WING QUARTER CHORD SWEEP = 63.75 DEG
WING TAPER RATIO = 0.0

1 W/S	87.4	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
2 T/W	0.500	0.0	0.0	0.0	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	2.06	0.0	0.0	0.0	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	2.65	0.0	0.0	0.0	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	673108	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
7 FUEL WEIGHT	355387	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
8 OP. WT. EMPTY	268720	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
9 ZERO FUEL WT.	317720	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
10 THRUST/ENGINE	94130	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
11 ENGINE SCALE	0.940	0.0	0.0	0.0	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	7701.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
13 WING SPAN	126.0	0.0	0.0	0.0	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	593.3	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	323.6	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	297.0	0.0	0.0	0.0	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—MILLION DOLLARS/AIRCRAFT																
17 FLYAWAY COST	3.297	0.0	0.0	0.0	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 AIRFRAME COST	65.553	0.0	0.0	0.0	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 ENGINE COST	1.099	0.0	0.0	0.0	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 AVIONICS COST	1.704	0.0	0.0	0.0	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—BILLION DOLLARS																
21 TOTAL INVEST.	0.859	0.0	0.0	0.0	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 TOTAL SYSTEM	5.495	0.0	0.0	0.0	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

APPENDIX A2

Reference Jet A Airplane (CL 1607-3)

Mach 2.2 SCV

Range = 4200 n. mi.

Payload = 234 Passengers (49,000 lb)

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

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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	673100.	3235.	3235.	0.	0.	10.0	10.0	0.	-197101.	0.	0.0	0.577	0.0
POWER 2	0.	0.300	669973.	4110.	7345.	0.	0.	0.5	10.5	0.	197204.	0.	4.87	0.964	0.0
CLIMB	0.	0.300	665763.	4754.	12099.	5.	5.	1.2	11.7	0.	197204.	0.	7.00	1.006	0.0
CRUISE	5000.	0.413	661009.	4114.	16213.	0.	5.	4.0	15.7	0.	-197101.	0.	7.82	0.732	0.0
ACCEL	5000.	0.413	656895.	1302.	17515.	1.	6.	0.2	15.9	0.	197201.	0.	9.76	1.248	0.0
CLIMB	5000.	0.537	655593.	16100.	33615.	29.	35.	3.8	19.7	0.	197201.	0.	11.75	1.291	0.0
CLIMB	34000.	0.964	639493.	26099.	59714.	137.	172.	8.7	28.4	0.	197201.	0.	8.56	1.374	2.61
CLIMB	54000.	2.116	613394.	0.	59714.	0.	172.	0.0	29.4	0.	197201.	0.	8.37	1.435	2.16
CRUISE	54000.	2.120	613394.	238410.	298124.	3868.	4040.	187.4	215.9	0.	-197201.	0.	8.18	1.288	2.16
DECEL	63000.	2.120	374984.	62.	298186.	15.	4055.	0.8	216.6	0.	197501.	0.	6.15	-0.337	1.51
DESCENT	63000.	1.892	374922.	922.	299108.	114.	4168.	11.6	228.2	0.	197501.	0.	10.12	-0.354	1.83
CRUISE	63000.	2.120	374000.	1529.	300637.	32.	4200.	1.5	229.8	0.	-197201.	0.	8.06	1.295	1.51
CRUISE	5000.	0.413	372470.	2317.	302954.	0.	4200.	5.0	234.8	0.	-197101.	0.	11.09	0.830	0.0
RESET	0.	0.0	370154.	0.	302954.	0.	4200.	0.0	234.8	0.	0.	0.	0.0	0.0	0.0
RESET	0.	0.0	370154.	0.	302954.	-4200.	0.	*****	0.0	0.	0.	0.	0.0	0.0	0.0
RESERVE	0.	0.0	370154.	21207.	324161.	0.	0.	0.0	0.0	0.	0.	0.	0.0	0.0	0.0
CLIMB	0.	0.200	348947.	1651.	325812.	2.	2.	0.4	0.4	0.	197204.	0.	9.60	1.002	0.0
CLIMB	1500.	0.500	347296.	8811.	334623.	16.	18.	2.3	2.7	0.	197201.	0.	10.90	1.254	0.0
CRUISE	41000.	0.900	338485.	7608.	342231.	162.	180.	18.5	21.2	0.	-197101.	0.	12.02	0.885	0.0
DESCENT	42000.	0.900	330876.	688.	342919.	63.	242.	8.6	29.8	0.	197501.	0.	11.13	-0.487	0.0
CRUISE	42000.	0.900	330188.	806.	343725.	17.	260.	2.0	31.8	0.	-197101.	0.	12.00	0.884	0.0
LOITER	15000.	0.530	329382.	11663.	355388.	0.	260.	30.0	61.8	0.	-197101.	0.	11.99	0.857	0.0

ILLRWT= 673100.0   FUEL A=355387.5   FUEL R=355387.6

CONFIGURATION GEOMETRY

BASIC WING—	AREA(SQ.FT)	SPAN(FT)	TAPER RATIO	C/4 SWEEP	L.E. SWEEP	CR(FT)	MAC(FT)
	7701.5	125.95	0.0	63.746	68.300	122.29	81.53
INBOARD WING—	AREA(SQ.FT)	EXP. AREA	L.E. SWEEP	REF L(FT)	SFLE(SQ.FT)	AVG T/C	
	7701.5	6385.7	68.30	74.24	0.0	2.65	
OUTBOARD WING—	AREA(SQ.FT)	Y BRK(FT)	L.E. SWEEP	REF L(FT)	SFLE(SQ.FT)	AVG T/C	
	0.0	0.0	68.30	74.24	0.0	2.65	
TOTAL WING—	AREA(SQ.FT)	EFF AR	AVG T/C	CR(FT)	CT(FT)	(B/2)/LW	P
	7701.5	2.06	2.65	122.29	0.0	0.398	0.386
WING TANK—	CBAR1(FT)	CBAR2(FT)	FTL(FT)	FVWING(CU FT)	FVBOX(CU FT)		
	111.29	0.0	57.31	2353.66	1533.84		
FUSELAGE—	LENGTH(FT)	S WET(SQ FT)	BWH(FT)	EQUIV D(FT)	SPI(SQ FT)		
	297.00	8463.5	11.26	13.22	137.30		
	BW(FT)	RH(FT)	SBW(SQ FT)	FVB(CU FT)			
	11.33	11.50	9177.00	20000.00			
TAIL—	SHT(SQ.FT)	SHTX(SQ.FT)	HT REF L(FT)	SVT(SQ.FT)	SVTX(SQ.FT)	VT REF L(FT)	
	593.29	497.19	18.44	328.56	325.87	20.47	
PROPULSION—	ENG L(FT)	ENG D(FT)	POD L(FT)	POD D(FT)	POD S WET	NO. PODS	INLET L(FT)
	20.52	6.20	31.07	7.33	2862.88	4.	0.0

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T/C	AR	W/S	T/W
2.65	2.06	87.4	0.500

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

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
TAKE-OFF WEIGHT	(673108.)		
FUEL AVAILABLE	355358.	FUEL	52.80
ZERO FUEL WEIGHT	(317771.)		
PAYLOAD	49000.	PAYLOAD	7.28
OPERATING WEIGHT	(268771.)	OPERATING ITEMS	1.55
OPERATING ITEMS	5459.		
STANDARD ITEMS	4953.		
EMPTY WEIGHT	(258309.)		
WING	77467.		
TAIL	5500.		
BODY	37002.	STRUCTURE	23.94
LANDING GEAR	28082.		
SURFACE CONTROLS	8567.		
NACELLE AND ENGINE SECTION	4370.		
PROPULSION	(66760.)	PROPULSION	9.92
WEIGHT OF LIFT ENGINES	0.		
VECTOR CONTROL SYSTEM	0.		
ENGINES	48080.		
THRUST REVERSAL	0.		
AIR INDUCTION SYSTEM	11949.		
FUEL SYSTEM	5138.		
ENGINE CONTROLS + STARTER	1593.		
INSTRUMENTS	1149.		
HYDRAULICS	4936.		
ELECTRICAL	4349.		
AVIONICS	1903.	EQUIPMENT	4.52
FURNISHINGS AND EQUIPMENT	11526.		
ENVIRONMENTAL CONTROL SYSTEM	6550.		
AUXILIARY GEAR	0.		
A.M.P.R.	(194374.)	TOTAL	(100.00)

~~ENGINE FUEL CAPACITY - BODY 319550.~~
~~ENGINE FUEL CAPACITY - WING 0.~~
~~ENGINE POD LENGTH - FT 0.0~~

~~STRUCTURAL ALUMINUM~~

WEIGHTS MATRIX

ELEMENT/MATERIAL	AL	TIT.	STEEL	COMP.	OTHER	TOTAL
WING	3573.	66483.	1553.	4815.	1243.	77667.
TAIL	0.	3300.	0.	2200.	0.	5500.
FUSEL	0.	18390.	0.	17692.	5920.	37002.
L. G.	0.	5616.	8818.	3370.	10276.	28012.
NACELLE	99.	1066.	0.	876.	129.	2190.
AIR INDUCT	0.	2390.	5975.	3585.	0.	11949.
S. CTLS	0.	1957.	1446.	851.	4253.	8507.
TOTALS	3671.	99221.	17792.	28388.	21823.	170995.

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RESEARCH DEVELOPMENT TEST AND EVALUATION (RDTE)

	DEVELOPMENT AND DESIGN		CONTRACTOR TEST AND EVALU		DEVELOPMENT AIRCRAFT		TOTAL RDT AND E	
AIRFRAME		913.63		374.50		545.47		1833.60
ENGINEERING								
HOURS	27610.		7194.		2670.		37473.	
LABOR RATE	8.17		8.17		8.17		8.17	
OVERHEAD RATE	9.20		9.20		9.20		9.20	
TOTAL		479.50		124.95		46.38		650.91
TOOLING								
HOURS	21504.		2225.		4450.		28184.	
LABOR RATE	6.09		6.09		6.09		6.06	
OVERHEAD RATE	12.36		12.36		12.36		12.36	
TOTAL		434.05		41.05		82.10		557.20
MANUFACTURING								
HOURS			8000.		17799.		26659.	
LABOR RATE			5.12		5.12		5.12	
OVERHEAD RATE			10.72		10.72		10.72	
TOTAL				146.97		281.94		422.91
QUALITY CONTROL								
HOURS			1700.		3560.		5340.	
LABOR RATE			6.29		6.29		6.29	
OVERHEAD RATE			10.72		10.72		10.72	
TOTAL				30.29		60.55		90.63
MATERIAL								
RAW AND PRCHSD			11.06		22.13		33.19	
PURCHASED EQUIP			20.55		41.10		61.66	
TOTAL				31.61		63.22		94.84
MISCELLANEOUS								
HOURS			356.		712.		1068.	
LABOR RATE			5.12		5.12		5.12	
OVERHEAD RATE			10.72		10.72		10.72	
TOTAL				5.64		11.20		16.92
ENGINES		866.44				82.30		948.74
AVIONICS		0.0				2.00		2.00
PROFIT(AIRFRAME)		137.04		56.17		81.62		274.84
DSN.-TRANS						54.55		54.55
WARRANTY						27.27		27.27
SUBTOTAL		1917.11		430.67		793.42		3141.20
OTHER ITEMS								155.86
TOTAL (RDTE)								3297.05

COST SUMMARY

RDT AND E

	TOTAL*
PROTOTYPE AIRCRAFT	793.42
DESIGN ENGINEERING	551.52
DEVELOPMENT TEST ARTICLES	360.69
FLIGHT TEST	69.98
ENGINE DEVELOPMENT CRUISE	266.44
ENGINE DEVELOPMENT LIFT	0.0
AVIONICS DEVELOPMENT	0.0
MAINTENANCE TRAINER DEVEL	0.0
OPERATOR TRAINER DEVELOP	0.0
DEVELOPMENT TOOLING	499.15
SPECIAL SUPPORT EQUIPMENT	15.87
DEVELOPMENT SPARES	123.57
TECHNICAL DATA	16.40
TOTAL RDTE	3297.05

MISC. DATA

RANGE (ST. MILES)	4833.16
BLOCK SPEED (MPH)	1161.01
FARE ($)	248.72
FLEET SIZE	16.23
PRODUCTION BASIS	600.00
REV.PASSENG.(MIL.PER YR)	1.81
AVER. CARGO PER FLIGHT	2000.00
FLIGHT PER A/C PER YEAR	864.78

INVESTMENT

	TOTAL*	PER PROD A/C**
PRODUCTION AIRCRAFT	31061.55	51769.25
PRODUCTION ENGINEERING	0.0	0.0
MAINTENANCE TRAINERS	0.0	0.0
OPERATOR TRAINERS	0.0	0.0
PRODUCTION TOOLING	85.16	141.94
SPECIAL SUPPORT EQUIPMENT	1273.52	2122.54
PRODUCTION SPARES	4702.60	7837.67
TECHNICAL DATA	185.61	309.36
TOTAL INVESTMENT	37308.45	62180.75

RETURN ON INVESTMENT (ROI)

TOTAL REVENUE PER YEAR *	469.74
TOTAL EXPENSE PER YEAR *	411.69
TOTAL INVESTMENT * INCL. FACILITIES	1096.62
ROI BEFORE TAXES	10.57
ROI AFTER TAXES	5.50

DIRECT OPERATIONAL COST (DOC)

	C/SM***	PERCENT
FLIGHT CREW	0.11049	6.48074
FUEL AND OIL	0.53846	31.58603
INSURANCE	0.14080	8.72847
DEPRECIATION	0.47875	29.06318
MAINTENANCE	0.42824	25.12016
TOTAL DOC	1.70473	100.000

INDIRECT OPERATIONAL COST (IOC)

	C/SM***	PERCENT
SYSTEM	0.00379	0.42638
LOCAL	0.17061	19.23012
AIRCRAFT CONTROL	0.00513	0.57736
CABIN ATTENDANT	0.07951	8.95057
FOOD AND BEVERAGE	0.02748	3.09327
PASSENGER HANDLING	0.13656	15.37358
CARGO HANDLING	0.00849	0.95562
OTHER PASSENGER EXPENSE	0.33550	37.77069
OTHER CARGO EXPENSE	0.00279	0.31274
GENERAL + ADMINISTR.	0.11822	13.30539
TOTAL IOC	0.88825	100.000

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

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PRODUCTION

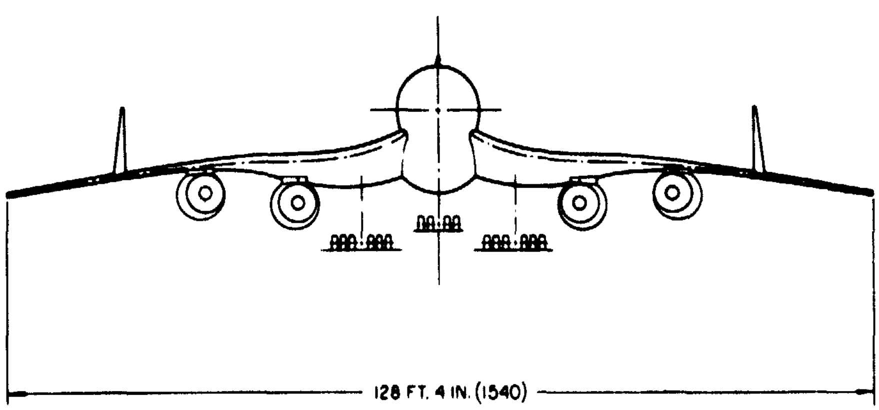
PRODUCTION YEARS

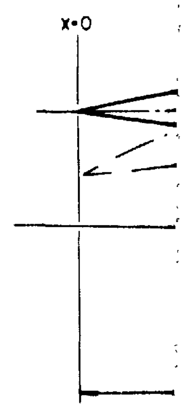
	1	2	3	4	5	6	7	8	9	10	TOTAL
AIRFRAME	1761.50	1675.02	1864.19	2060.54	2251.20	2096.60	1987.53	1904.48	1838.12	1783.28	19222.43
ENGINEERING											
HOURS	5999.	5167.	5448.	5810.	6141.	5565.	5163.	4858.	4617.	4418.	53205.
LABOR RATE	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17	
OVERHEAD RATE	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	9.20	
TOTAL	104.20	89.75	94.97	100.92	106.66	96.67	89.63	84.39	80.19	76.73	924.17
TOOLING											
HOURS	7192.	6200.	6561.	6972.	7369.	6679.	6196.	5830.	5540.	5301.	63846.
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	132.31	114.40	121.05	128.64	135.95	123.22	114.31	107.57	102.21	97.81	1177.96
MANUFACTURING											
HOURS	59956.	51670.	54674.	58191.	61496.	55655.	51630.	48595.	46166.	44176.	532046.
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	950.17	818.45	866.06	920.32	972.67	881.57	817.81	769.58	731.26	699.75	8427.65
QUALITY CONTROL											
HOURS	11997.	10334.	10935.	11620.	12281.	11131.	10326.	9717.	9233.	8835.	106410.
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	204.07	175.78	186.01	197.66	208.90	189.34	175.64	165.29	157.06	150.29	1810.03
MATERIAL											
RAW AND PURCH	116.29	155.37	196.51	236.66	275.34	269.69	265.08	261.40	258.35	255.75	2290.53
PURCHASED EQUIP	215.96	288.54	364.95	439.52	512.27	500.94	492.29	485.46	479.80	474.97	4254.99
TOTAL	332.24	443.91	561.45	676.19	788.11	770.63	757.37	746.86	738.15	730.72	6545.52
MISCELLANEOUS											
HOURS	2399.	2067.	2187.	2324.	2456.	2226.	2065.	1943.	1847.	1767.	21282.
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	38.31	32.96	34.84	36.81	38.91	35.26	32.71	30.79	29.25	27.99	337.11
ENGINES	376.09	441.99	528.54	612.01	691.71	660.38	637.39	619.37	604.62	592.17	5764.16
AVIONICS	12.00	18.00	24.00	30.00	36.00	36.00	36.00	36.00	36.00	36.00	300.00
PROFIT	264.22	251.25	279.63	309.08	337.68	314.49	298.13	285.67	275.72	267.44	2883.37
INSUR.+TAXES	176.15	167.50	186.42	206.05	225.12	209.66	198.75	190.45	183.81	178.33	1922.24
WARRANTY	88.07	83.75	93.21	103.03	112.56	104.83	99.38	95.22	91.91	89.16	961.12
TOTAL FLYAWAY	2679.03	2637.41	2975.98	3320.71	3654.28	3421.96	3257.18	3131.19	3030.18	2956.67	31062.55

CHARACTERISTICS

POWER PLANT - DUCT BURNING TURBOFAN
UNINSTALLED THRUST - 85,000 LB. SLS
TAXI WEIGHT - 680,000 LB.

		WING	HORIZ. TAIL	FUS VERT TAIL	WING VERT. TAIL (EACH)
AREA	SQ. FT.	8,000	443.6	267	176.6
ASPECT RATIO		2.058	1.707	0.517	0.517
TAPER RATIO		0.1167	0.225	0.23	0.20
SPAN	IN.	1540.3	330.2	141	114.7
ROOT CHORD	IN	1672	315.8	443.4	369.5
TIP CHORD	IN.	195.1	71.1	102	73.9
MAC	IN.	1023.1	219.2	308.3	254.5
L.E. SWEEP	DEG.	70.2 66.11 52.15	56.64	68.2	73.42





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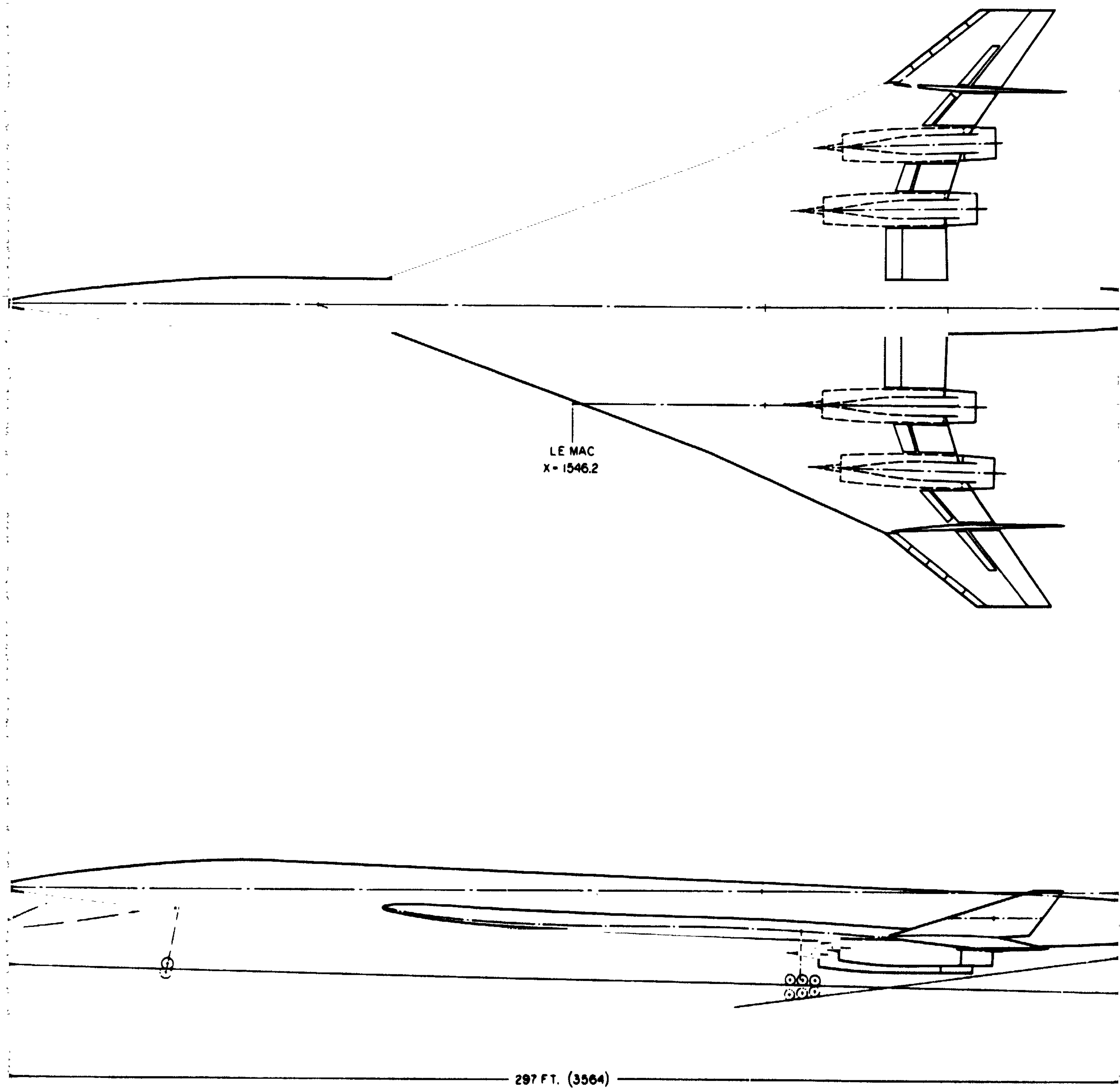

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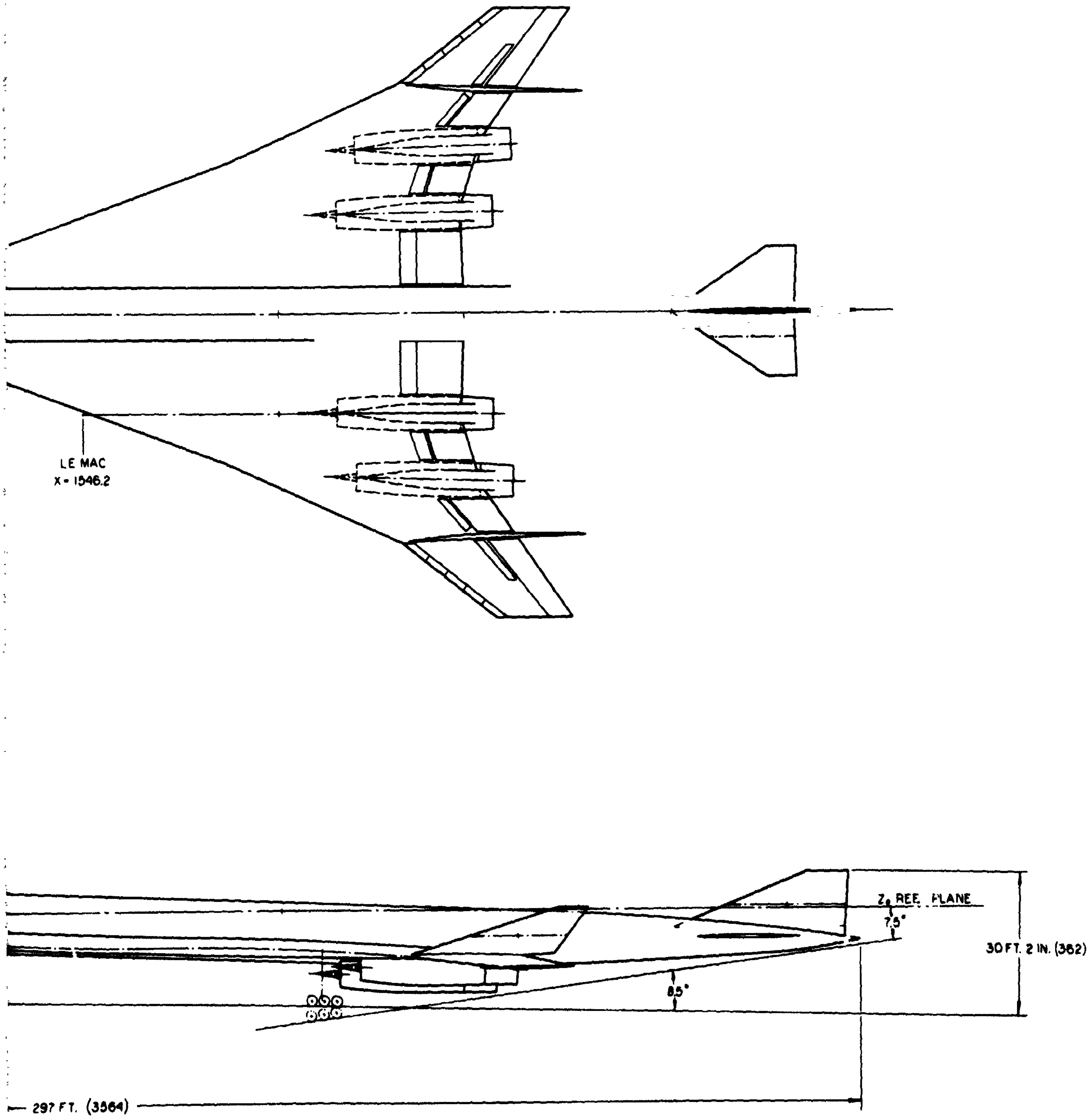

Figure A-2. General Arrangement - Baseline Jet A M2.2 SCV

APPENDIX B

WING AND FUSELAGE CROSS SECTIONS OF LH_2 FUELED SCV's

B1 - Mach 2.7 LH_2 SCV

B2 - Mach 2.2 LH_2 SCV

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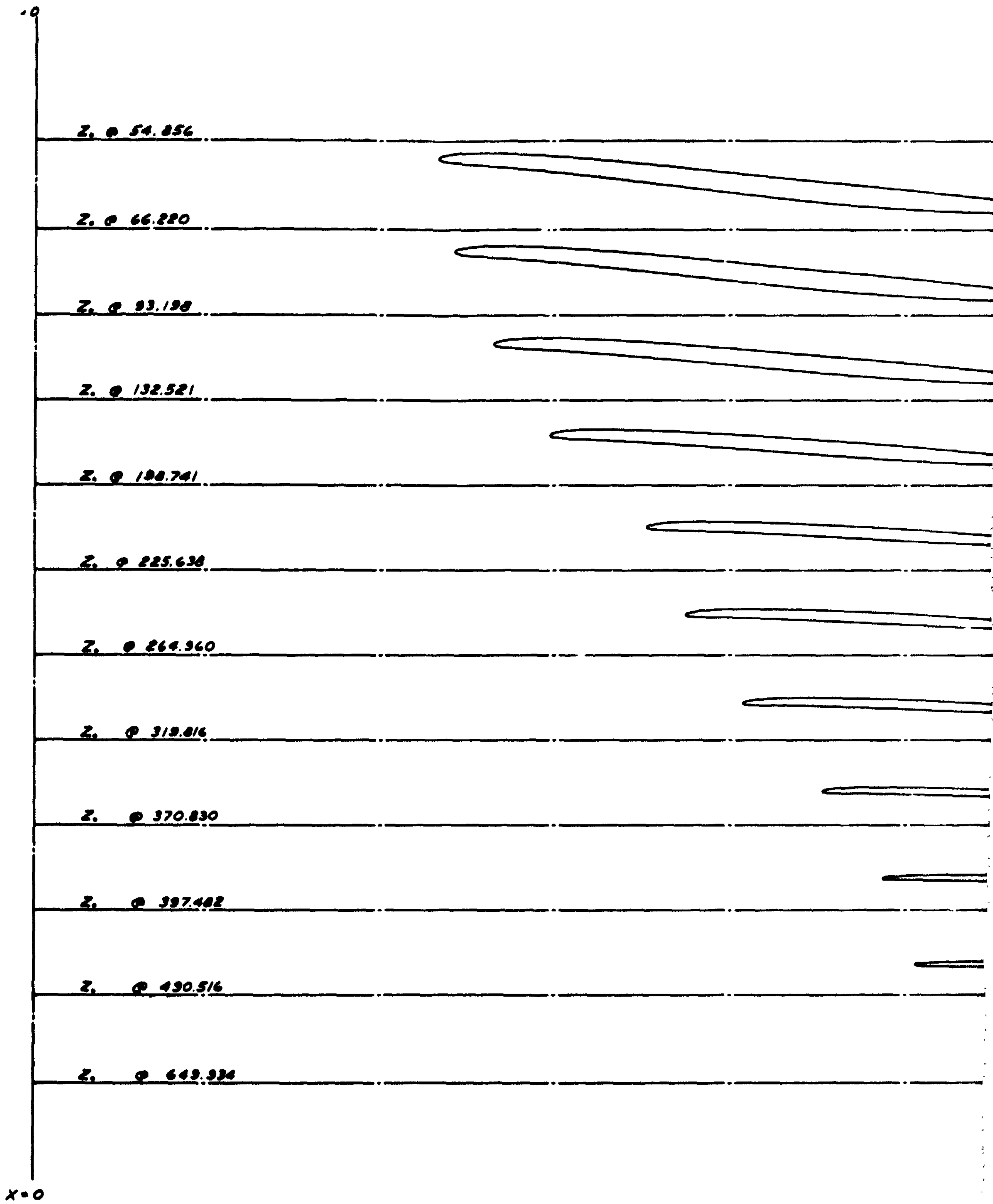

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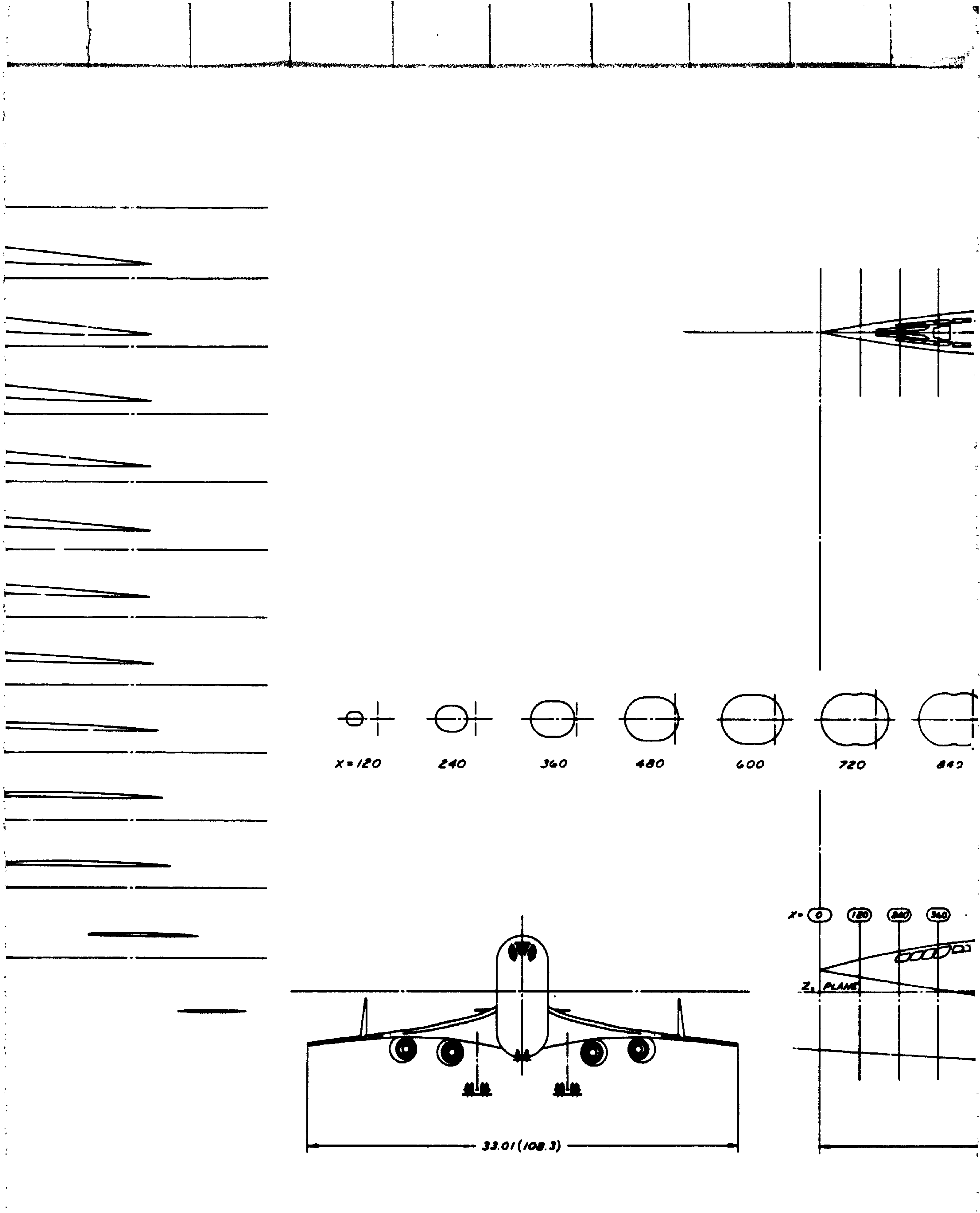

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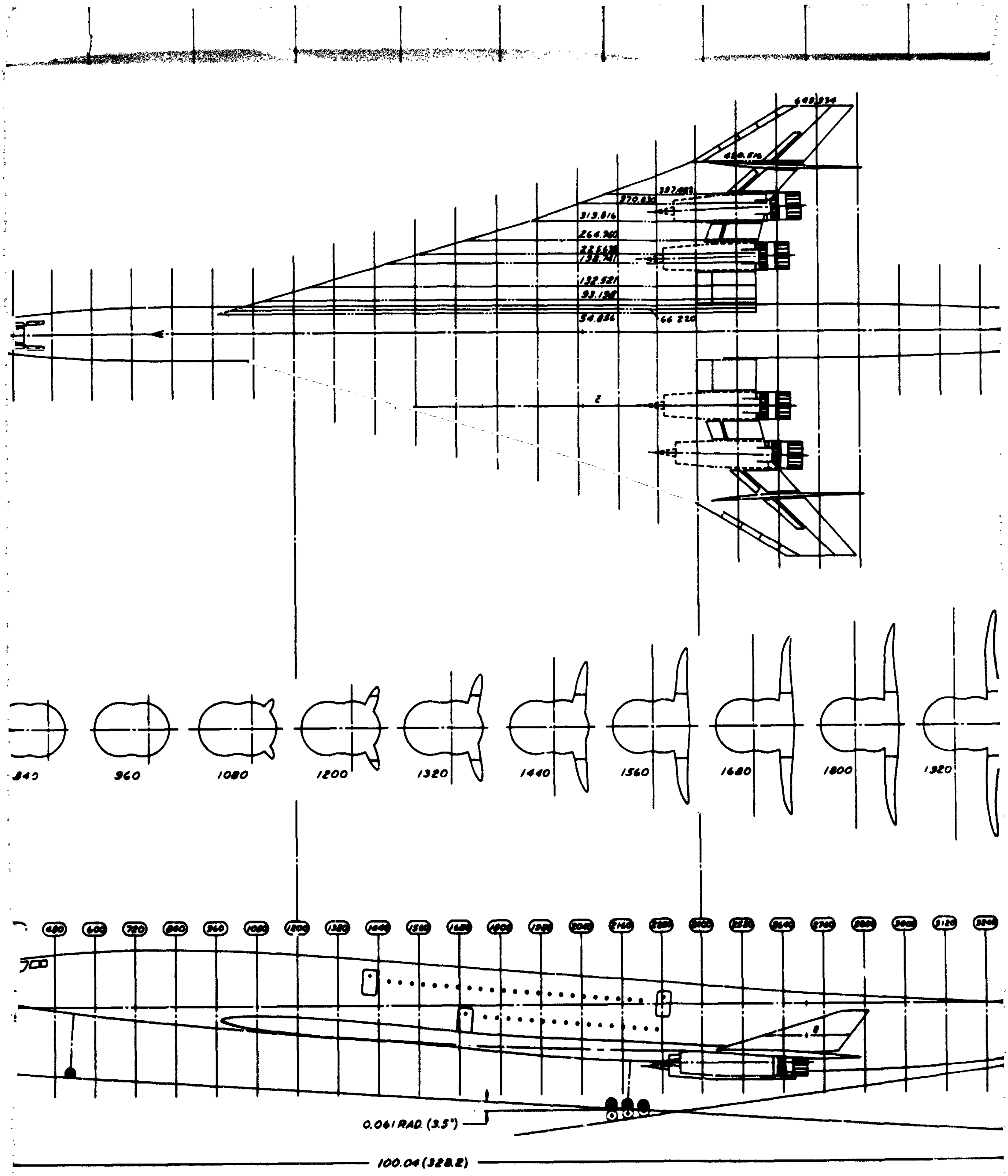


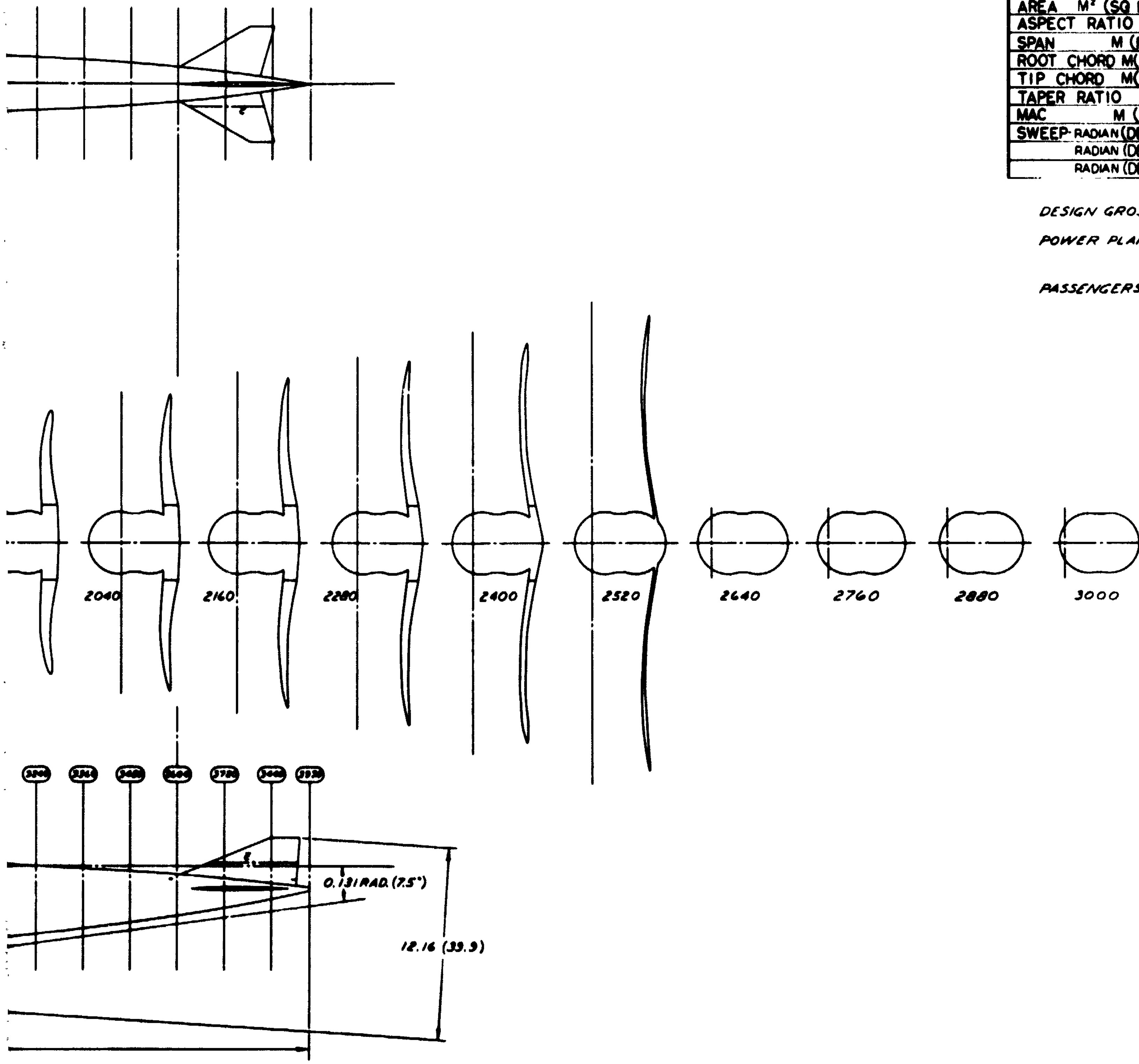
FOLDOUT FRAME 3

CHARACTERISTICS
AREA M² (SQ FT
ASPECT RATIO
SPAN M (FT
ROOT CHORD M(IN)
TIP CHORD M(IN)
TAPER RATIO
MAC M (IN)
SWEEP- RADIAN (DEG)
RADIAN (DEG)
RADIAN (DEG)

DESIGN GROSS

POWER PLANT

PASSENGERS -



FOLDOUT FRAME 4

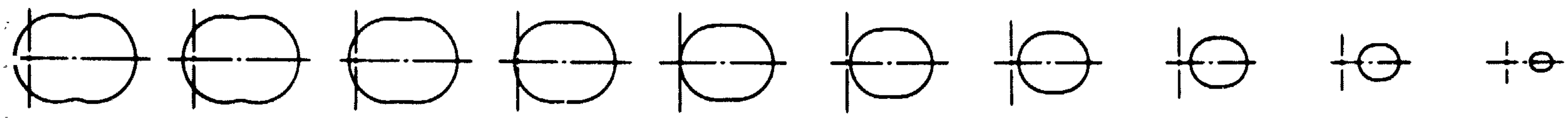
1

CHARACTERISTICS	WING	HORIZ. TAIL	FUS VERT TAIL	WING VERT. TAIL
AREA M^2 (SQ FT)	678.17 (7300)	31 22 (336.1)	13.40 (144.2)	17 00 (183.0)
ASPECT RATIO	1.607	1.707	0.517	0.517
SPAN M (FT)	33 01 (108.3)	7.32 (24.0)	2.62 (8.6)	2.96 (9.7)
ROOT CHORD M(IN)	45.63 (1796 6)	6.98 (274 9)	8.28 (326.0)	9.56 (376.2)
TIP CHORD M(IN)	5.26 (2070)	1.57 (61.8)	1.91 (75.0)	1.91 (75.2)
TAPER RATIO	0.1135	0.225	0.23	0.20
MAC M (IN)	28.07 (1105.0)	4.85 (190.8)	5.76 (226.7)	6.58 (259.1)
SWEEP-RADIAN (DEG)	1.292 (74)	1.058 (60.64)	1.190 (68.2)	1.281 (73.42)
RADIAN (DEG)	1.236 (70.84)	—	—	—
RADIAN (DEG)	1.047 (60)	—	—	—

DESIG' GROSS WEIGHT - 164,657 KG. (363,000 LBS.)

POWER PLANT - SCV LH_2 M2.7 DUCT BURNING TURBOFAN
UNINSTALLED THRUST - 217,952 NEWTONS (49,000 LBS.)

PASSENGERS - 234



2. DIM. IN METERS (FEET), OR NOTED
DWG NO CL 1701-9,3V1,3V2,3V3,7A,& 7B
1. THIS DESIGN DEVELOPED ON COMPUTER GRAPHICS,

Figure B-1. Cross Sections - M2.7 LH2 SCV Wing and Fuselage

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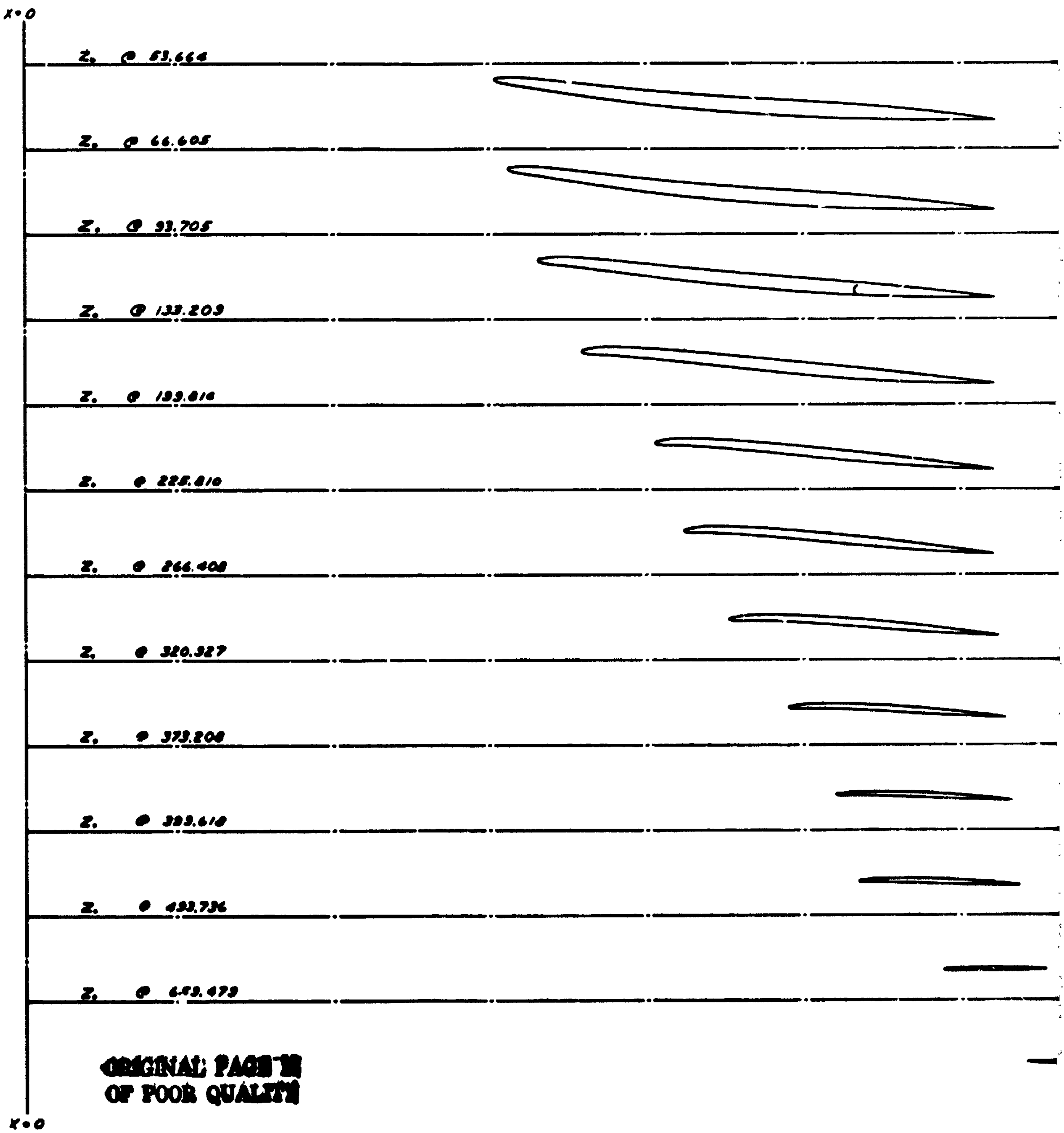

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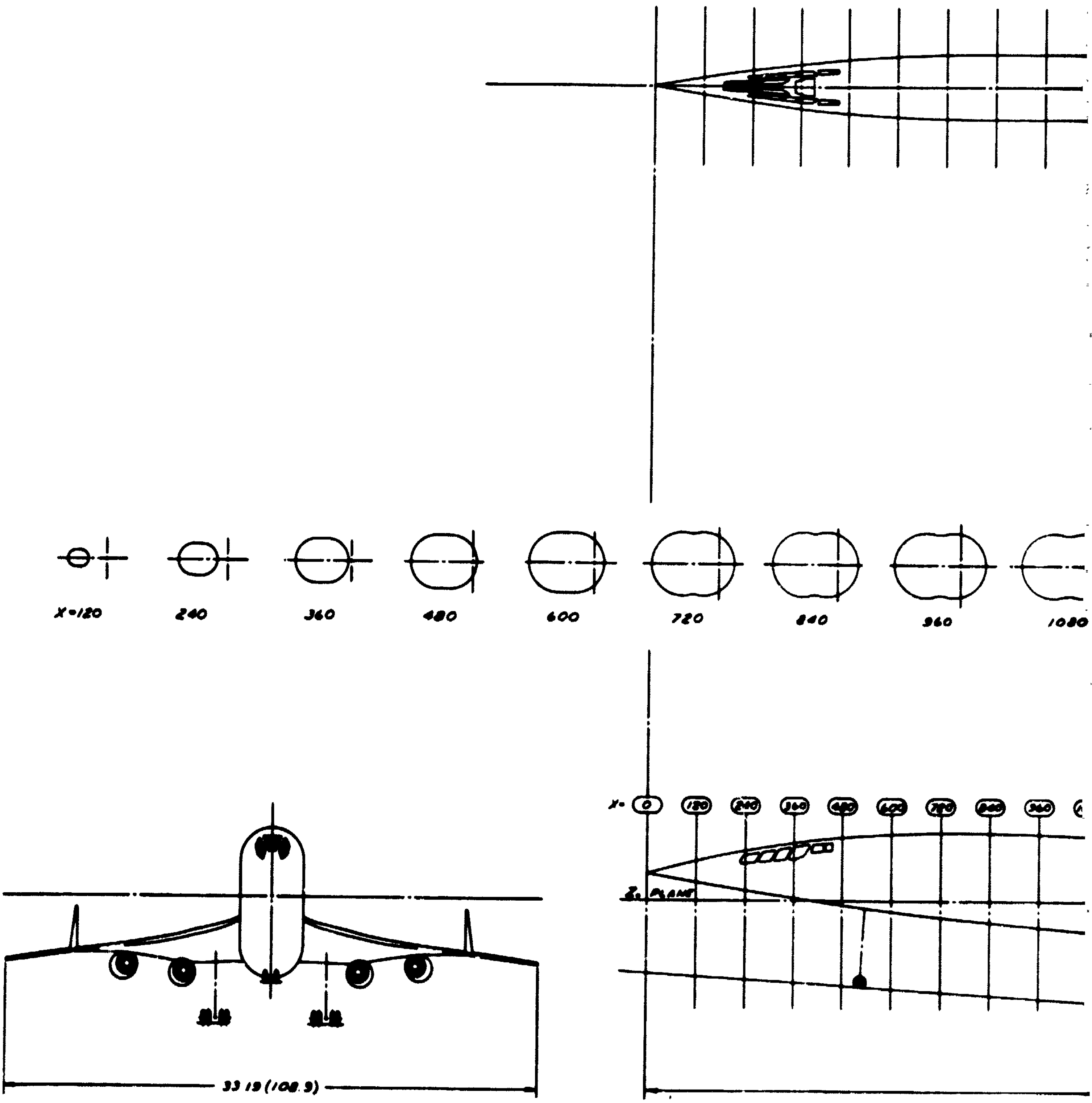
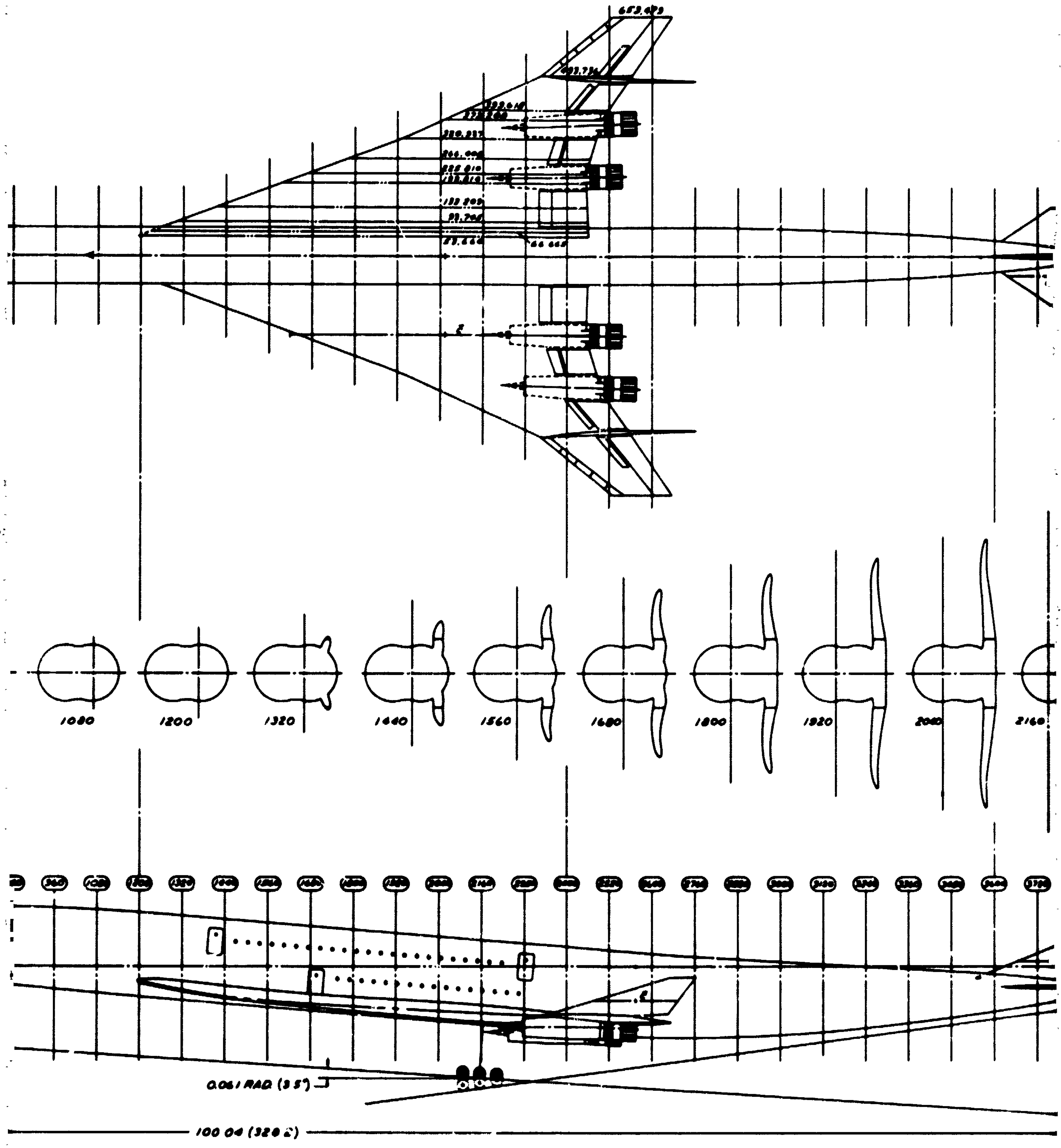
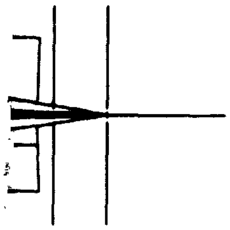


FOLDOUT FRAME 2



FOLDOUT FRAME 3

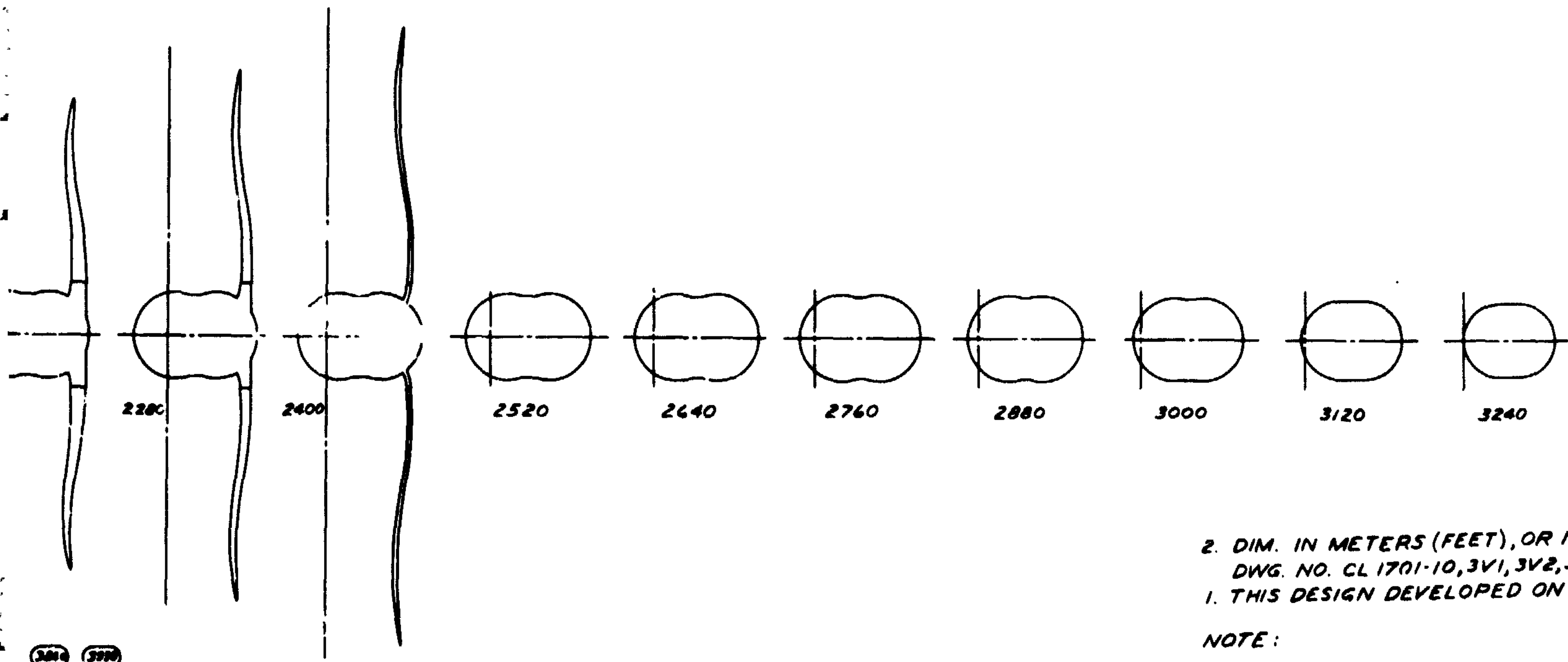


CHARACTERISTICS	WING	HORIZ. TAIL
AREA M² (SQ FT)	535.10 (5760)	26 68 (287.2)
ASPECT RATIO	2.058	1.707
SPAN M (FT)	33 19 (108.9)	6.74 (22.1)
ROOT CHORD M(IN)	36 03 (1418.7)	6 45 (254.1)
TIP CHORD M(IN)	4.21 (165.6)	1.45 (57.2)
TAPER RATIO	0.1167	0.225
MAC M (IN)	22.05 (868.1)	4.48 (176 4)
SWEEP-RADIAN (DEG)	1.225 (70.20)	0.988 (56.64)
RADIAN (DEG)	1.154 (66.11)	—
RADIAN (DEG)	0 910 (52.15)	—

DESIGN GROSS WEIGHT - 161,935 KG. (357,

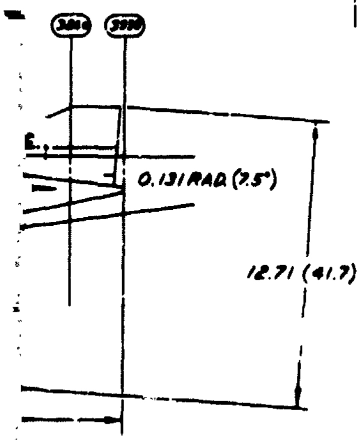
POWER PLANT - SCV LH_2 M2.2 DUCT BU
UNINSTALLED THRUST - 2

PASSENGERS - 234



2. DIM. IN METERS (FEET), OR NO
DWG. NO. CL 1701-10, 3V1, 3V2, 3V
1. THIS DESIGN DEVELOPED ON C

NOTE:



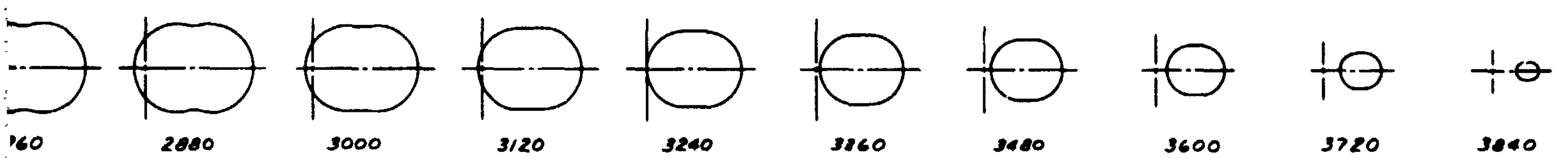
FOLDOUT FRAME 4

CHARACTERISTICS	WING	HORIZ. TAIL	FUS. VERT. TAIL	WING VERT. TAIL
AREA M^2 (SQ FT)	535.10 (5760)	26.68 (287.2)	17.79 (191.5)	14.15 (152.3)
ASPECT RATIO	2.058	1.707	0.517	0.517
SPAN M (FT)	33.19 (108.9)	6.74 (22.1)	3.05 (10.0)	2.71 (8.9)
ROOT CHORD M(IN)	36.03 (1418.7)	6.45 (254.1)	9.54 (375.5)	8.72 (343.4)
TIP CHORD M(IN)	4.21 (165.6)	1.45 (57.2)	2.19 (86.4)	1.74 (68.7)
TAPER RATIO	0.1167	0.225	0.23	0.20
MAC M (IN)	22.05 (868.1)	4.48 (176.4)	6.63 (261.1)	6.01 (236.6)
SWEEP-RADIAN(DEG)	1.225 (70.20)	0.988 (56.64)	1.190 (68.2)	1.281 (73.42)
RADIAN(DEG)	1.154 (66.11)	—	—	—
RADIAN(DEG)	0.910 (52.15)	—	—	—

DESIGN GROSS WEIGHT - 161,935 KG. (357,000 LBS.)

POWER PLANT - SCV LH_2 M2.2 DUCT BURNING TURBOFAN
UNINSTALLED THRUST - 232,234 NEWTONS (52,211 LBS.)

PASSENGERS - 234



2. DIM. IN METERS (FEET), OR NOTED
DWG. NO. CL 1701-10, 3V1, 3V2, 3V3, 7A, & 7B
1. THIS DESIGN DEVELOPED ON COMPUTER GRAPHICS,

NOTE:

Figure B-2. Cross Sections - M2.2 LH_2 SCV Wing and Fuselage

FOLDOUT FRAME 5

APPENDIX C

SELECTED ASSET COMPUTER PRINTOUT PAGES OF BASELINE LH_2 FUELED SCV'S

C1 - Mach 2.7 Min W_G LH_2 SCV

C2 - Mach 2.2 Min W_G LH_2 SCV

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SUMMARY II NO. 1 — A S S E T P A R A M E T R I C A N A L Y S I S — AUGUST OF 1975

AIRCRAFT MODEL --CL 1701-9
I.O.C. DATE --1990
DESIGN SPEED --SUPERSONIC

ENGINE I.D. -- 126000
SLS SCALE 1.0 = 84500
NUMBER OF ENGINES = 4.

WING QUARTER CHORD SWEEP = 67.25 DEG
WING TAPER RATIO = 0.115

1 W/S	49.7	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
2 T/W	0.534	0.0	0.0	0.0	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	1.61	0.0	0.0	0.0	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	394913	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
7 FUEL WEIGHT	100678	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
8 OP. WT. EMPTY	245235	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
9 ZERO FUEL WT.	294235	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
10 THRUST/ENGINE	52519	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
11 ENGINE SCALE	0.626	0.0	0.0	0.0	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	7952.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
13 WING SPAN	113.1	0.0	0.0	0.0	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	367.0	0.0	0.0	0.0	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	527.1	0.0	0.0	0.0	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	340.2	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
COST DATA																
17 RDTE - BIL.	3.778	0.0	0.0	0.0	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.	59.61	0.0	0.0	0.0	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-MIL.	6.892	0.0	0.0	0.0	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	2.185	0.0	0.0	0.0	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	0.840	0.0	0.0	0.0	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	-1.24	0.0	0.0	0.0	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)	7803	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
24 AP SPEED-KT(1)	158.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0

APPENDIX C1

Mach 2.7 LH_2 SCV

Range = 4200 n. mi.

Payload = 234 Passengers (49,000 lb)

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

INTERNATIONAL MISSION STD DAY + 14.4 F

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	394914.	539.	539.	0.	0.	10.0	10.0	0.	-126101.	0.	0.0	0.164	0.0
POWER 2	0.	0.300	394375.	916.	1455.	0.	0.	0.3	10.3	0.	126201.	0.	6.70	0.524	0.0
CLIMB	0.	0.300	393459.	860.	2315.	10.	10.	2.3	12.6	0.	126101.	0.	9.14	0.250	0.0
CRUISE	5000.	0.414	392599.	578.	2893.	0.	10.	4.0	16.6	0.	-126101.	0.	9.81	0.217	0.0
ACCEL	5000.	0.414	392022.	221.	3114.	4.	13.	0.8	17.3	0.	126101.	0.	10.49	0.235	0.0
CLIMB	5000.	0.539	391800.	4847.	7960.	110.	123.	14.4	31.7	0.	126101.	0.	10.24	0.331	0.0
CLIMB	34000.	0.980	386953.	16032.	23993.	384.	507.	20.8	52.6	0.	126201.	0.	6.21	0.567	0.0
CLIMB	61000.	2.606	370921.	1630.	25623.	74.	581.	2.9	55.5	0.	126201.	0.	7.18	0.577	0.0
CRUISE	68000.	2.620	369291.	58545.	84168.	3419.	4000.	133.7	189.1	0.	-126201.	0.	7.42	0.575	0.0
DECEL	72000.	2.620	310746.	11.	84179.	15.	4015.	0.6	189.7	0.	1501.	0.	7.47	-0.252	0.0
DESCENT	72000.	2.448	310735.	235.	84414.	157.	4172.	13.1	202.9	0.	1501.	0.	8.16	-0.130	0.0
CRUISE	72000.	2.620	310500.	442.	84855.	28.	4200.	1.1	203.9	0.	-126201.	0.	7.38	0.578	0.0
CRUISE	5000.	0.414	310058.	539.	85394.	0.	4200.	5.0	208.9	0.	-126101.	0.	10.48	0.219	0.0
RESET	0.	0.0	309519.	0.	85394.	0.	4200.	0.0	208.9	0.	0.	0.	0.0	0.0	0.0
RESET	0.	0.0	309519.	0.	85394.	-4200.	0.	*****	0.0	0.	0.	0.	0.0	0.0	0.0
RESERVE	0.	0.0	309519.	5978.	91372.	0.	0.	0.0	0.0	0.	0.	0.	0.0	0.0	0.0
CLIMB	0.	0.200	303541.	647.	92019.	3.	3.	0.8	0.8	0.	126101.	0.	8.94	0.386	0.0
CLIMB	1500.	0.505	302895.	3646.	95664.	106.	109.	13.9	14.6	0.	126101.	0.	9.20	0.301	0.0
CRUISE	36000.	0.900	299249.	1323.	96987.	76.	185.	8.7	23.3	0.	-126201.	0.	9.97	0.306	0.0
DESCENT	36000.	0.900	297926.	134.	97122.	52.	237.	7.2	30.6	0.	1501.	0.	9.17	-0.169	0.0
CRUISE	36000.	0.900	297791.	403.	97525.	23.	260.	2.6	33.2	0.	-126201.	0.	9.96	0.306	0.0
LOITER	15000.	0.470	297388.	3152.	100677.	0.	260.	30.0	63.2	0.	-126101.	0.	10.33	0.220	0.0

TOGWT= 394913.8   FUEL A=100678.5   FUEL R=100676.8

Appendix C

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INTERNATIONAL MISSION STD DAY + 14.4 F

T/C	AR	W/S	T/W
3.00	1.61	49.7	0.535

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	MAC(FT)	
	7952.4	113.06	0.115	67.248	70.840	85.10	
WING PANELS--	AREA(SQ FT)	EXP. AREA	AVG T/C	L.E. SWEEP	SFLE(SQ FT)	REF L(FT)	
	4784.7	2569.7	2.73	74.000	0.0	109.64	
	1217.0	1217.0	2.52	74.000	0.0	75.08	
	490.4	490.4	2.58	70.840	0.0	55.46	
	817.7	817.7	2.71	70.640	0.0	40.16	
	652.7	652.7	2.92	60.000	0.0	23.97	
TOTAL WING--	AREA(SQ FT)	EFF AR	AVG T/C	CR(FT)	CT(FT)	MAC(FT)	L(FT)
	7962.6	1.61	2.75	156.27	16.00	94.18	151.76
FUSELAGE--	LENGTH(FT)	S WET(SQ FT)	HW(FT)	EQUIV D(FT)	SPI(SQ FT)		
	340.24	15537.6	15.55	17.32	236.00		
	BW(FT)	BH(FT)	SBW(SQ FT)				
	12.90	15.43	15538.64				
HORZ. TAILS--	SHT1(SQ FT)	SHX1(SQ FT)	REF L1(FT)	SHT2(SQ FT)	SHX2(SQ FT)	REF L2(FT)	
	367.00	250.60	14.09	0.0	0.0	0.0	
VERT. TAILS--	SVT1(SQ FT)	SVX1(SQ FT)	REF L1(FT)	SVT2(SQ FT)	SVX2(SQ FT)	REF L2(FT)	
	190.95	190.95	21.73	231.38	331.38	20.55	
PROPULSION--	ENG L(FT)	ENG D(FT)	POD L(FT)	POD D(FT)	POD S WET	NO. PODS	INLET L(FT)
	18.88	5.22	32.24	6.47	2820.29	4.	0.0
FUEL TANKS--	WING(CU FT)	BOX(CU FT)	FUS(CU FT)				
	0.0	0.0	24448.38				

INTERNATIONAL MISSION STD DAY + 14.4 F

T/C	AR	W/S	T/W
3.00	1.61	49.7	0.535

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

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
TAKE-OFF WEIGHT	(394914.)		
FUEL AVAILABLE	100679.	FUEL	25.49
ZERO FUEL WEIGHT	(294235.)		
PAYLOAD	49000.	PAYLOAD	12.41
OPERATING WEIGHT	(245235.)		
OPERATING ITEMS	5376.	OPERATING ITEMS	2.86
STANDARD ITEMS	5927.		
EMPTY WEIGHT	(233932.)		
WING	57858.		
TAIL	5170.		
BODY	55895.	STRUCTURE	36.53
LANDING GEAR	17812.		
SURFACE CONTROLS	4600.		
NACELLE AND ENGINE SECTION	2920.		
PROPULSION	(59503.)	PROPULSION	15.07
WEIGHT OF LIFT ENGINES	0.		
VECTOR CONTROL SYSTEM	0.		
ENGINES	29053.		
THRUST REVERSAL	0.		
AIR INDUCTION SYSTEM	10951.		
FUEL SYSTEM	18127.		
ENGINE CONTROLS + STARTER	1372.		
INSTRUMENTS	1102.		
HYDRAULICS	3005.		
ELECTRICAL	4761.		
AVIONICS	1903.	EQUIPMENT	7.64
FURNISHINGS AND EQUIPMENT	11526.		
ENVIRONMENTAL CONTROL SYSTEM	7877.		
AUXILIARY GEAR	0.		
A.M.P.R.	(192042.)	TOTAL	(100.00)
EXCESS FUEL CAPACITY - BODY	-0.		
EXCESS FUEL CAPACITY - WING	0.		
EXCESS BODY LENGTH - FT	0.0		

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OF POOR QUALITY

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

ELEMENT/ / MATERIAL	AL	TIT.	STEEL	COMP.	OTHER	TOTAL
WING	2661.	49527.	1157.	3587.	926.	57858.
TAIL	0.	3102.	0.	2068.	0.	5170.
FUSEL	18222.	10955.	1006.	19172.	6540.	55895.
L. G.	0.	3562.	5593.	2137.	6519.	17812.
NACELLE	66.	724.	0.	534.	36.	1460.
AIR INDUCT	0.	2190.	5476.	3285.	0.	10951.
S. CTLS	0.	1058.	782.	460.	2300.	4600.
TOTALS	20949.	71119.	14014.	[illegible]	16371.	152746.

COST SUMMARY

RDT AND E	TOTAL*	INVESTMENT	TOTAL*	PER PROD A/C**	DIRECT OPERATIONAL COST (DOC)	C/SM***	PERCENT
PROTOTYPE AIRCRAFT	722.60	PRODUCTION AIRCRAFT	27299.93	45499.87	FLIGHT CREW	0.09916	4.52551
DESIGN ENGINEERING	859.55	PRODUCTION ENGINEERING	0.0	0.0	FUEL AND OIL	1.23291	56.32735
DEVELOPMENT TEST ARTICLES	337.46				INSURANCE	0.12014	5.48657
FLIGHT TEST	99.44				DEPRECIATION	0.36639	17.65306
ENGINE DEVELOPMENT CRUISE	875.69				MAINTENANCE	0.35038	16.00757
ENGINE DEVELOPMENT LIFT	0.0				TOTAL DOC	2.18912	100.000
AVIONICS DEVELOPMENT	0.0						
MAINTENANCE TRAINER DEVEL	0.0	MAINTENANCE TRAINERS	0.0	0.0	INDIRECT OPERATIONAL COST (IOC)	C/SM***	PERCENT
OPERATOR TRAINER DEVELOP	0.0	OPERATOR TRAINERS	0.0	0.0			
DEVELOPMENT TOOLING	744.48	PRODUCTION TOOLING	219.32	365.54	SYSTEM	0.00340	0.40520
SPECIAL SUPPORT EQUIPMENT	14.45	SPECIAL SUPPORT EQUIPMENT	1365.00	2274.99	LOCAL	0.10022	11.93495
DEVELOPMENT SPARES	105.73	PRODUCTION SPARES	3943.54	6572.56	AIRCRAFT CONTROL	0.00513	0.61075
TECHNICAL DATA	18.80	TECHNICAL DATA	164.14	273.56	CABIN ATTENDANT	0.07129	8.48945
					FOOD AND BEVERAGE	0.02464	2.93381
TOTAL RDTE	3778.20	TOTAL INVESTMENT	32991.91	54986.51	PASSENGER HANDLING	0.13656	16.26285
MISC. DATA		RETURN ON INVESTMENT (ROI)			CARGO HANDLING	0.00845	1.01090
RANGE (ST. MILES)	933.11	TOTAL REVENUE PER YEAR *		464.73	OTHER PASSENGER EXPENSE	0.33550	39.95506
BLOCK SPEED (MPH)	1294.90	TOTAL EXPENSE PER YEAR *		480.83	OTHER CARGO EXPENSE	0.00278	0.33081
FARE ($)	248.72	TOTAL INVESTMENT * INCL. FACILITIES		891.97	GENERAL + ADMINISTR.	0.15170	18.06621
FLEET SIZE	14.55	ROI BEFORE TAXES		-2.49	TOTAL IOC	0.83994	100.000
PRODUCTION BASIS	600.00	ROI AFTER TAXES		-1.29			
REV.PASSENG.(MIL.PER YR)	1.81						
AVER. CARGO PER FLIGHT	2000.00						
FLIGHT PER A/C PER YEAR	964.53						

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

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RESEARCH DEVELOPMENT TEST AND EVALUATION (RDTE)

	DEVELOPMENT AND DESIGN	CONTRACTOR TEST AND EVALU	DEVELOPMENT AIRCRAFT	TOTAL FLT AND E
AIRFRAME	1294.81	379.91	610.34	2285.06
ENGINEERING				
HOURS	42031.	8441.	2519.	53001.
LABOR RATE	8.17	8.17	8.17	8.17
OVERHEAD RATE	9.20	9.20	9.20	9.20
TOTAL	747.43	146.63	43.76	937.12
TOOLING				
HOURS	22090.	2100.	4194.	28379.
LABOR RATE	6.09	6.09	6.09	6.09
OVERHEAD RATE	12.36	12.36	12.36	12.36
TOTAL	647.38	38.74	77.47	763.59
MANUFACTURING				
HOURS		8398.	16796.	25194.
LABOR RATE		9.12	9.12	9.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		133.03	266.0[illegible]	399.1[illegible]
QUALITY CONTROL				
HOURS		1680.	3359.	5039.
LABOR RATE		6.29	6.29	6.29
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		28.57	57.14	85.71
MATERIAL				
RAW AND PURCHASED		9.67	19.34	29.01
PURCHASED EQUIP		17.96	35.92	53.88
TOTAL		27.63	55.26	82.89
MISCELLANEOUS				
HOURS		336.	672.	1008.
LABOR RATE		5.12	5.12	5.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		5.32	10.64	15.96
ENGINES	875.69		57.16	932.85
AVIONICS	0.0		2.00	2.00
PROFIT(AIRFRAME)	209.22	56.99	76.55	342.76
INSUR.+TAXES			51.03	51.03
WARRANTY			25.52	25.52
SUBTOTAL	2479.73	436.90	722.60	3639.22
OTHER ITEMS				138.98
TOTAL (RDTE)				3778.20

PRODUCTION

	PRODUCTION YEARS 1	2	3	4	5	6	7	8	9	10	TOTAL
AIRFRAME	1639.12	1549.74	1720.08	1897.39	2069.52	1924.65	1822.64	1745.20	1683.19	1631.96	17683.90
ENGINEERING											
HOURS	5661.	4876.	5159.	5483.	5795.	5252.	4872.	4585.	4356.	4165.	50207.
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	98.32	84.69	89.62	95.23	100.65	91.22	84.63	79.64	75.67	72.41	872.09
TOOLING											
HOURS	6753.	5851.	6191.	6579.	6953.	6302.	5846.	5502.	5228.	5002.	60248.
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	125.32	107.95	114.23	121.[illegible]	128.29	116.28	107.87	101.51	96.45	92.29	1111.58
MANUFACTURING											
HOURS	56615.	48756.	51594.	54627.	57965.	52514.	48720.	45847.	43564.	41687.	502066.
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	896.63	772.33	817.26	868.46	917.65	831.90	771.73	726.22	690.06	660.32	7952.73
QUALITY CONTROL											
HOURS	11321.	9752.	10319.	10965.	11589.	10504.	9744.	9169.	8713.	8337.	100413.
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	192.57	165.87	175.52	186.52	197.13	178.67	165.75	155.97	148.21	141.82	1706.03
MATERIAL											
RAW AND PURCH	101.64	135.81	171.77	206.86	241.11	235.73	231.70	226.49	225.82	221.55	2002.48
PURCHASED EQUIP	188.76	252.21	318.99	384.18	447.77	437.78	430.30	424.34	419.38	415.16	3718.89
TOTAL	290.41	388.01	490.76	591.04	688.88	673.51	662.00	652.83	645.21	636.71	5721.36
MISCELLANEOUS											
HOURS	2264.	1950.	2064.	2193.	2318.	2101.	1949.	1834.	1743.	1667.	21083.
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	35.27	30.89	32.69	34.74	36.71	33.28	30.87	29.05	27.60	26.41	318.11
ENGINES	261.19	306.89	367.07	425.04	480.39	458.63	442.67	430.15	419.91	411.26	4003.18
AVIONICS	12.00	18.00	24.00	30.00	36.00	36.00	36.00	36.00	36.00	36.00	300.00
PROFIT	245.87	232.46	258.01	284.61	310.43	288.73	273.43	261.78	252.48	244.79	2652.59
INSUR.+TAXES	163.91	154.98	172.01	189.74	206.95	192.49	182.28	174.52	168.32	163.20	1768.39
WARRANTY	81.96	77.49	86.00	94.87	103.48	96.24	91.14	87.26	84.16	81.60	884.19
TOTAL FLYAWAY	2464.05	2339.57	2627.17	2921.64	3202.77	2996.94	2848.36	2734.91	2644.05	2576.49	27299.93

ORIGINAL PAGE IS
OF POOR QUALITY

SUMMARY ID NO. 1 A S S E T P A R A M E T R I C A N A L Y S I S SEPTEMBER 25 1975

AIRCRAFT MODEL —CL 1701-10
I.O.C. DATE —1990
DESIGN SPEED —SUPERSONIC

ENGINE I.D. — 130000
SLS SCALE 1.0 = 78900
NUMBER OF ENGINES = 4.

WING QUARTER CHORD SWEEP = 63.93 DEG
WING TAPER RATIO = 0.117

1 W/S	62.6	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
2 T/W	0.567	0.0	0.0	0.0	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	2.06	0.0	0.0	0.0	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	3.60	0.0	0.0	0.0	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	376916	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
7 FUEL WEIGHT	101659	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
8 OP. WT. EMPTY	226256	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
9 ZERO FUEL WT.	275256	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
10 THRUST/ENGINE	53427	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
11 ENGINE SCALE	0.677	0.0	0.0	0.0	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	6020.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
13 WING SPAN	111.3	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	293.8	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
15 V. TAIL AREA	514.6	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	341.7	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.	9.094	0.0	0.0	0.0	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.	53.69	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 INVESTMENT-BIL.	6.929	0.0	0.0	0.0	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	2.259	0.0	0.0	0.0	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	0.853	0.0	0.0	0.0	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	-2.71	0.0	0.0	0.0	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)	7906	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
24 AP SPEED-KT(1)	150.6	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

SUMMARY ID NO. 1 ASSET PARAMETRIC ANALYSIS SEPTEMBER 25 1975

AIRCRAFT MODEL —CL 1701-10
I.O.C. DATE —1990
DESIGN SPEED —SUPERSONIC

ENGINE I.D. — 130000
SLS SCALE 1.0 = 78500
NUMBER OF ENGINES = 4.

WING QUARTER CHORD SWEEP = 61.93 DEG
WING TAPER RATIO = 0.117

1 W/S	62.6	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
2 T/W	0.567	0.0	0.0	0.0	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	2.06	0.0	0.0	0.0	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	3.60	0.0	0.0	0.0	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	376916	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
7 FUEL WEIGHT	101659	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
8 OP. WT. EMPTY	226256	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
9 ZERO FUEL WT.	275256	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
10 THRUST/ENGINE	59477	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
11 ENGINE SCALE	0.677	0.0	0.0	0.0	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	6020.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
13 WING SPAN	111.3	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	293.8	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
15 V. TAIL AREA	514.6	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	341.7	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.	9.094	0.0	0.0	0.0	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.	53.69	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-MIL.	6.929	0.0	0.0	0.0	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	2.259	0.0	0.0	0.0	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	0.853	0.0	0.0	0.0	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	-2.71	0.0	0.0	0.0	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)	7906	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
24 AP SPEED-KT(1)	150.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0

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APPENDIX C2

Mach 2.2 LH_2 SCV

Range = 4200 n. mi.

Payload = 234 Passengers (49,000 lb)

MISSION SUMMARY

INTERNATIONAL MISSION STD DAY + 14.4 F

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	376917.	639.	639.	0.	0.	10.0	10.0	0.	-130101.	0.	0.0	0.204	0.0
POWER 2	0.	0.300	376277.	736.	1376.	0.	0.	0.3	10.3	0.	130202.	0.	6.62	0.444	0.0
CLIMB	0.	0.300	375539.	762.	2138.	8.	8.	2.0	12.3	0.	130101.	0.	8.75	0.248	0.0
CRUISE	5000.	0.414	374777.	617.	2757.	0.	8.	4.0	16.3	0.	-143101.	0.	9.53	0.276	0.0
ACCEL	5000.	0.414	374160.	194.	2951.	3.	11.	0.5	16.8	0.	140101.	0.	10.73	0.254	0.0
CLIMB	5000.	0.539	373966.	3655.	6606.	96.	107.	12.3	29.1	0.	150101.	0.	10.84	0.365	0.0
CLIMB	34000.	0.900	370220.	9199.	15805.	237.	344.	14.6	43.7	0.	130202.	0.	6.71	0.569	0.0
CLIMB	52000.	2.103	361022.	1902.	17707.	81.	425.	3.9	47.7	0.	[illegible]	0.	7.37	0.524	0.0
CRUISE	57000.	2.120	359119.	69211.	86609.	3625.	4050.	175.7	223.3	0.	-130201.	0.	7.25	0.522	0.0
DECEL	61000.	2.120	290909.	13.	86621.	13.	4063.	0.7	224.0	0.	1501.	0.	7.98	-0.104	0.0
DESCENT	61000.	1.885	290896.	211.	86832.	106.	4170.	10.8	234.8	0.	1501.	0.	8.91	-0.119	0.0
CRUISE	61000.	2.120	290685.	514.	86765.	30.	4200.	1.5	236.3	0.	-130201.	0.	7.15	0.521	0.0
CRUISE	5000.	0.414	290171.	553.	87298.	0.	4200.	5.0	241.3	0.	-130101.	0.	10.71	0.245	0.0
RESET	0.	0.0	289618.	0.	87298.	0.	4200.	0.0	241.3	0.	0.	0.	0.0	0.0	0.0
RESET	0.	0.0	289618.	0.	87298.	-4200.	0.	*****	0.0	0.	0.	0.	0.0	0.0	0.0
RESERVE	0.	0.0	289618.	6111.	93409.	0.	0.	0.0	0.0	0.	0.	0.	0.0	0.0	0.0
CLIMB	0.	0.200	283508.	503.	93912.	3.	3.	0.8	0.8	0.	130101.	0.	8.92	0.321	0.0
CLIMB	1500.	0.505	283004.	2914.	96826.	41.	44.	11.6	12.4	0.	137101.	0.	9.59	0.291	0.0
CRUISE	36000.	0.900	280088.	1302.	98128.	91.	135.	10.4	22.7	0.	-140101.	0.	12.82	0.293	0.0
DESCENT	36000.	0.900	278788.	156.	98284.	56.	191.	7.8	30.5	0.	1501.	0.	9.88	-0.168	0.0
CRUISE	36000.	0.900	278632.	274.	98558.	19.	200.	2.2	32.6	0.	-140101.	0.	10.96	0.295	0.0
LOITER	15000.	0.501	278358.	3102.	101660.	0.	200.	30.0	62.6	0.	-130101.	0.	10.69	0.244	0.0

TOGWT= 376916.6 FUEL A=101659.9 FUEL R=101659.8

INTERNATIONAL MISSION STD DAY + 14.4 F

T/C	AR	W/S	T/W
3.00	2.06	62.6	0.567

CONFIGURATION GEOMETRY

BASIC WING--	AREA(SQ FT)	SPAN(FT)	TAPER RATIO	C/4 SWEEP	L.E. SWEEP	MAC(FT)	
	6020.1	111.36	0.117	61.910	66.110	65.35	
WING PANELS--	AREA(SQ FT)	EXP. AREA	AVG T/C	L.E. SWEEP	SFLE(SQ FT)	REF L(FT)	
	3610.9	1923.8	3.00	70.200	0.0	84.14	
	920.5	920.5	3.00	70.200	0.0	57.39	
	370.1	378.1	3.00	66.110	0.0	42.60	
	618.2	618.2	3.00	66.110	0.0	30.81	
	492.7	492.7	3.00	52.150	0.0	18.40	
TOTAL WING--	AREA(SQ FT)	EFF AR	AVG T/C	CR(FT)	CT(FT)	MAC(FT)	L(FT)
	6019.9	2.06	3.00	120.86	14.10	73.96	140.80
*FUSELAGE--	LENGTH(FT)	S WET(SQ FT)	BWW(FT)	EQUIV D(FT)	SPI(SQ FT)		
	361.22	15517.6	15.32	17.36	236.80		
	BW(FT)	BH(FT)	SBW(SQ FT)				
	12.90	19.43	15517.58				
HORZ. TAILS--	SHT1(SQ FT)	SHX1(SQ FT)	REF L1(FT)	SHT2(SQ FT)	SHX2(SQ FT)	REF L2(FT)	
	293.77	193.54	12.43	0.0	0.0	0.0	
VERT. TAILS--	SVT1(SQ FT)	SVX1(SQ FT)	REF L1(FT)	SVT2(SQ FT)	SVX2(SQ FT)	REF L2(FT)	
	196.21	196.21	22.62	318.35	318.35	20.15	
PROPULSION--	ENG L(FT)	ENG D(FT)	POD L(FT)	POD D(FT)	POD S WET	NO. PODS	INLET L(FT)
	15.52	4.19	26.71	5.13	1723.28	4.	0.0
FUEL TANKS--	WING(CU FT)	HOX(CU FT)	FUS(CU FT)				
	0.0	0.0	24631.62				

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C.3

INTERNATIONAL MISSION STD DAY + 14.4 F

T/C	AR	W/S	T/W
3.00	2.06	62.6	0.567

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

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
TAKE-OFF WEIGHT	(376917.)		
FUEL AVAILABLE	101660.	FUEL	26.97
ZERO FUEL WEIGHT	(275257.)		
PAYLOAD	49000.	PAYLOAD	13.00
OPERATING WEIGHT	(276257.)		
OPERATING ITEMS	5376.	OPERATING ITEMS	2.94
STANDARD ITEMS	5713.		
EMPTY WEIGHT	(215166.)		
WING	46803.		
TAIL	4771.		
BODY	54824.	STRUCTURE	34.65
LANDING GEAR	17159.		
SURFACE CONTROLS	4420.		
NACELLE AND ENGINE SECTION	7627.		
PROPULSION	(56444.)	PROPULSION	14.98
WEIGHT OF LIFT ENGINES	0.		
VECTOR CONTROL SYSTEM	0.		
ENGINES	26847.		
THRUST REVERSAL	0.		
AIR INDUCTION SYSTEM	7185.		
FUEL SYSTEM	19035.		
ENGINE CONTROLS + STARTER	1377.		
INSTRUMENTS	1096.		
HYDRAULICS	2764.		
ELECTRICAL	4609.		
AVIONICS	1903.	EQUIPMENT	7.46
FURNISHINGS AND EQUIPMENT	11526.		
ENVIRONMENTAL CONTROL SYSTEM	6229.		
AUXILIARY GEAR	0.		
A.M.P.R.	(174240.)	TOTAL	(100.00)
EXCESS FUEL CAPACITY - BODY	-0.		
EXCESS FUEL CAPACITY - WING	0.		
EXCESS BODY LENGTH - FT	0.0		

WEIGHTS MATRIX

ELEMENT/ / MATERIAL	AL	TIT.	STEEL	COMP.	OTHER	TOTAL
WING	2153.	40064.	936.	2902.	749.	46803.
TAIL	0.	2862.	0.	1908.	0.	4771.
FUSEL	17872.	10745.	987.	18804.	6414.	54824.
L. G.	0.	3432.	5388.	2059.	6280.	17159.
NACELLE	59.	452.	0.	525.	78.	1314.
AIR INDUCT	0.	1437.	3593.	2156.	0.	7185.
S. CTLS	0.	1017.	751.	442.	2210.	4420.
TOTALS	20085.	60209.	11655.	28797.	15731.	136476.

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COST SUMMARY

RDT AND E	TOTAL*
PROTOTYPE AIRCRAFT	657.33
DESIGN ENGINEERING	571.28
DEVELOPMENT TEST ARTICLES	302.91
FLIGHT TEST	67.08
ENGINE DEVELOPMENT CRUISE	804.57
ENGINE DEVELOPMENT LIFT	0.0
AVIONICS DEVELOPMENT	0.0
MAINTENANCE TRAINER DEVEL	0.0
OPERATOR TRAINER DEVELOP	0.0
DEVELOPMENT TOOLING	562.68
SPECIAL SUPPORT EQUIPMENT	13.15
DEVELOPMENT SPARES	99.22
TECHNICAL DATA	15.39
TOTAL RDTE	3093.86

INVESTMENT	TOTAL*	PER PROD A/C**
PRODUCTION AIRCRAFT	25205.45	42009.07
PRODUCTION ENGINEERING	0.0	0.0
MAINTENANCE TRAINERS	0.0	0.0
OPERATOR TRAINERS	0.0	0.0
PRODUCTION TOOLING	30.19	50.32
SPECIAL SUPPORT EQUIPMENT	1260.27	2100.45
PRODUCTION SPARES	3736.33	6227.22
TECHNICAL DATA	151.16	251.94
TOTAL INVESTMENT	30383.40	50638.99

DIRECT OPERATIONAL COST (DOC)	C/SM***	PERCENT
FLIGHT CREW	0.11336	5.01934
FUEL AND OIL	1.26041	55.80737
INSURANCE	0.12506	5.53730
DEPRECIATION	0.40238	17.81628
MAINTENANCE	0.35729	15.81971
TOTAL DOC	2.25850	100.000

INDIRECT OPERATIONAL COST (IOC)	C/SM***	PERCENT
SYSTEM	0.00337	0.39543
LOCAL	0.09565	11.21785
AIRCRAFT CONTROL	0.00513	0.60146
CABIN ATTENDANT	0.08156	9.56783
FOOD AND BEVERAGE	0.02819	3.30648
PASSENGER HANDLING	0.13656	16.01555
CARGO HANDLING	0.00849	0.99553
OTHER PASSENGER EXPENSE	0.33550	39.34741
OTHER CARGO EXPENSE	0.00278	0.32578
GENERAL + ADMINISTR.	0.15541	18.22667
TOTAL IOC	0.85266	100.000

MISC. DATA	
RANGE (ST. MILES)	4833.10
BLOCK SPEED (MPH)	1131.48
FARE ($)	248.72
FLEET SIZE	16.66
PRODUCTION BASIS	600.00
REV.PASSENG.(MIL.PER YR)	1.81
AVER. CARGO PER FLIGHT	2000.00
FLIGHT PER A/C PER YEAR	842.80

RETURN ON INVESTMENT (ROI)	
TOTAL REVENUE PER YEAR *	469.73
TOTAL EXPENSE PER YEAR *	493.95
TOTAL INVESTMENT * INCL. FACILITIES	929.38
ROI BEFORE TAXES	-5.21
ROI AFTER TAXES	-2.71

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

RESEARCH DEVELOPMENT TEST AND EVALUATION (RDTE)

	DEVELOPMENT AND DESIGN	CONTRACTOR TEST AND EVALU	DEVELOPMENT AIRCRAFT	TOTAL RDT AND E
AIRFRAME	906.22	321.73	458.08	1766.04
ENGINEERING				
HOURS	28599.	6465.	2257.	37321.
LABOR RATE	8.17	8.17	8.17	8.17
OVERHEAD RATE	9.20	9.20	9.20	9.20
TOTAL	496.76	112.30	39.21	648.27
TOOLING				
HOURS	24255.	1881.	3762.	29898.
LABOR RATE	6.09	6.09	6.09	6.09
OVERHEAD RATE	12.36	12.36	12.36	12.36
TOTAL	409.46	34.71	69.42	593.58
MANUFACTURING				
HOURS		7525.	15049.	22574.
LABOR RATE		5.12	5.12	5.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		119.19	238.38	357.57
QUALITY CONTROL				
HOURS		1505.	3010.	4515.
LABOR RATE		6.29	6.29	6.29
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		25.60	51.20	76.80
MATERIAL				
RAW AND PRCHSD		8.81	17.62	26.43
PURCHASED EQUIP		16.36	32.72	49.08
TOTAL		25.17	50.34	75.51
MISCELLANEOUS				
HOURS		301.	602.	903.
LABOR RATE		5.12	5.12	5.12
OVERHEAD RATE		10.72	10.72	10.72
TOTAL		4.77	9.54	14.30
ENGINES	804.57		59.82	864.39
AVIONICS	0.0		2.00	2.00
PROFIT(AIRFRAME)	147.93	48.26	68.71	264.91
INSUR.+TAXES			45.81	45.81
WARRANTY			22.90	22.90
SUBTOTAL	1938.72	369.99	657.33	2966.04
OTHER ITEMS				127.76
TOTAL (RDTE)				3093.80

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PRODUCTION

	PRODUCTION YEARS 1	2	3	4	5	6	7	8	9	10	TOTAL
AIRFRAME	1472.98	1394.36	1548.51	1708.88	1864.57	1734.72	1643.15	1573.44	1517.77	1471.77	15930.14
ENGINEERING											
HOURS	5072.	4369.	4623.	4912.	5192.	4706.	4365.	4108.	3903.	3735.	44985.
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	88.10	75.88	80.30	85.33	90.18	81.74	75.83	71.35	67.80	64.88	781.38
TOOLING											
HOURS	6086.	5242.	5547.	5895.	6230.	5647.	5238.	4929.	4684.	4482.	53982.
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	112.29	96.72	102.35	108.76	114.95	104.18	96.65	90.95	86.42	82.69	995.96
MANUFACTURING											
HOURS	50716.	43687.	46228.	49125.	51918.	47056.	43653.	41078.	39033.	37351.	449847.
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	803.37	692.00	732.25	778.13	822.39	745.37	691.46	650.68	618.28	591.64	7125.56
QUALITY CONTROL											
HOURS	10144.	8737.	9246.	9825.	10384.	9411.	8731.	8216.	7807.	7470.	89969.
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	172.54	148.62	157.27	167.12	176.63	160.09	148.51	139.75	132.79	127.07	1530.38
MATERIAL											
RAW AND PURCH	92.59	123.71	156.47	188.44	219.64	214.74	211.07	208.14	205.71	203.64	1824.14
PURCHASED EQUIP	171.95	229.75	290.58	349.96	407.89	398.79	391.98	386.55	382.03	378.19	3387.69
TOTAL	264.54	353.46	447.05	538.41	627.53	613.53	603.05	594.69	587.75	581.83	5211.82
MISCELLANEOUS											
HOURS	2029.	1747.	1849.	1965.	2077.	1882.	1746.	1643.	1561.	1494.	17994.
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	32.13	27.68	29.29	31.13	32.90	29.81	27.66	26.03	24.73	23.67	285.02
ENGINES	273.34	321.16	384.14	444.61	502.74	479.96	463.26	456.16	439.44	430.39	4189.38
AVIONICS	12.00	18.00	24.00	30.00	36.00	36.00	36.00	36.00	36.00	36.00	300.00
PROFIT	220.95	209.15	232.28	256.33	279.69	260.21	246.47	236.02	227.67	220.77	2389.52
INSUR.+TAXES	147.30	139.44	154.85	170.89	186.46	173.47	164.31	157.34	151.78	147.18	1593.01
WARRANTY	73.65	69.72	77.43	85.44	93.23	86.74	82.16	78.67	75.89	73.59	796.51
TOTAL FLYAWAY	2200.21	2151.83	2421.20	2676.35	2962.68	2771.09	2635.34	2531.63	2448.54	2386.59	25205.45

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