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ADVANCED SUPERSONIC TECHNOLOGY CONCEPT STUDY - HYDROGEN FUELED CONFIGURATION

Summary Report

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16. Abstract

Conceptual designs of hydrogen fueled supersonic transport configurations for the 1990 time period were developed and compared with equivalent technology Jet A-1 fueled vehicles to determine the economic and performance potential of liquid hydrogen as an alternate fuel.

Parametric evaluations of supersonic cruise vehicles with varying design and transport mission characteristics established the basis for selecting a preferred configuration which was then studied in greater detail.

An assessment was made of the general viability of the selected concept including an evaluation of costs and environmental considerations, i.e., exhaust emissions and sonic boom characteristics. Technology development requirements and suggested implementation schedules are presented.

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FOREWORD

The Advanced Supersonic Technology Study - Hydrogen Fueled Configuration, summarized in this report was conducted under NASA Ames Research Center Contract NAS 2-7732 from July through December 1973. Details of the study findings are presented in the contract final report CR 114718 dated January 1974.

This report outlines the methodology and describes the work performed during the six month study, presents the results and conclusions, and makes recommendations for further investigation and development.

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

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

SUMMARY

This study has examined the feasibility of supersonic transport aircraft which use liquid hydrogen as the fuel. In Phase I a parametric analysis was carried out to determine a preferred configuration among the wide variety of possibilities that were examined. In Phase II, one vehicle of the selected configuration was studied to establish an acceptable basic design concept for the vehicle structure, the cryogenic fuel tanks, and the tank thermal protection system. The size, weight, and cost of this design of hydrogen fueled AST aircraft were then determined as required for the following mission capability:

Cruise speed	Mach 2.7
Range	7778 km. (4200 n.mi.)
Payload	22,226 kg. (49,000 lb.) (234 passengers)

Design tradeoffs, and performance and cost sensitivities were then evaluated.

An analysis was made of the environmental compatibility of the hydrogen fueled aircraft in terms of noise, sonic boom overpressure, and exhaust emissions. The design was then compared with that of a conventionally fueled AST airplane designed to the same criteria. The hydrogen fueled aircraft was found to provide advantages in nearly every category of comparison:

		Jet A-1	LH_2
Gross Weight	(lb.) kg.	(750,000) 340,194	(368,000) 166,922
Operating Empty Weight	(lb.) kg.	(309,700) 140,478	(223,100) 101,196
Fuel Weight	(lb.) kg.	(391,300) 177,491	(95,900) 43,500
Engine Thrust	(lb.) newtons	(89,500) 398,100	(46,000) 204,600
Cost			
RDT & E	$\$ \times 10^9$	4.28	3.32
Production Aircraft	$\$ \times 10^6$	67.33	47.97
Noise			
Sideline	EPNdB	108	106.1
Flyover	EPNdB	108	104.2
Sonic Boom Overpressure	(psf) newton/m^2	(1.86) 89.1	(1.32) 63.2
Energy per Seat Mile	(Btu/seat n.mi.) joule/seat m	(6102) 3479	(4274) 2437
Emissions		CO	None
		HC	None
		NO_x	Minimal
		H_2O	~Twice as much
		Noxious Odor	None

A comparison of direct operating cost and/or return on investment is strongly dependent on the cost of fuel. Analysis has shown the DOC of the two vehicles to be approximately equal when the cost of liquid hydrogen, in $/BTU, is not more than 1.75 times that of Jet A-1 fuel. At current prices being paid for petroleum based fuels, this ratio is well within the cost estimated by several authorities for making liquid hydrogen from coal and water.

A program for developing technologies required for designing, building, and operating liquid hydrogen-fueled supersonic transport aircraft is described and recommended for implementation. One of the urgent items is a recommendation to carry out a flight demonstration program using existing, subsonic transport aircraft converted to liquid hydrogen, to provide practical, operational experience in establishing design requirements and handling specifications for the new fuel.

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1.0 INTRODUCTION

This is the summary report of a study performed by Lockheed-California Company for NASA-Ames Research Center. The NASA Request for Proposal (RFP 2-19866, HK-94) dated March 29, 1973, sought promising new ideas for advanced technology supersonic transport concepts. The intent was to complement the existing AST studies and provide feasibility information for new, different concepts. Contemporary AST studies were all based on use of conventional (kerosene) type fuel (Jet A-1).

The approach proposed by Lockheed, reported herein, was to investigate the feasibility of using liquid hydrogen as the fuel in a supersonic transport of advanced design. This approach was suggested as a result of recognition of the impending energy crisis and the fact that the world's supply of petroleum will be significantly diminished by 1990, according to recent projections (References 1, 2, and 3). The prospects of having the demands of a fleet of SST's, with their prodigious appetite for fuel, imposed on the dwindling reserves of crude oil in that time period could very well be the cause for rejection of America's bid to build such aircraft.

On the other hand, preliminary conceptual analyses performed by Lockheed had indicated that use of hydrogen as the fuel in supersonic transport aircraft could conceivably lead to the following advantages:

- Significantly lower gross weight
- Reduced pollution
- Lower sonic boom overpressure
- Lower noise
- Decreased costs

The subject study was thus performed to investigate the potential of liquid hydrogen fueled AST aircraft and to discover if these significant advantages might be realized.

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2.0 DESCRIPTION OF THE STUDY

The objective of the study was to explore the economic and performance potential of liquid hydrogen (LH_2) fueled supersonic commercial transport aircraft. The study was conducted in two phases.

Phase I was an exploratory analysis, conducted to parametrically identify the potential of a large number of different configurations of LH_2 fueled AST aircraft, and to determine a preferred design concept, as well as a set of design requirements, for more specific analysis in Phase II. Table 1 is a list of the mission and vehicle configuration parameters and their values which were studied in every viable combination during Phase I. A list of the premises used to establish a basis for the study is shown in Table 2.

Phase I involved, first, a preliminary sizing investigation to establish approximate sizes for various example aircraft representing the scope of the study. Next, aerodynamic, propulsion, weight, and cost parameters were generated to appropriately represent the candidate vehicles in the ASSET (Advanced System Synthesis and Evaluation Technique) computer program. This work was heavily influenced by the experience of the participating personnel with existing transport aircraft, plus experience with both previous and on-going studies on supersonic as well as hypersonic transport aircraft. Engine decks were generated to represent the performance of hydrogen fueled versions of both turbojet and turbofan engines for Mach 2.7 cruise aircraft, and of a turbojet for a Mach 2.2 aircraft. ASSET runs were then made to determine performance capability, costs and significant design tradeoffs for each of the candidate vehicles. These results were analyzed to determine the four most attractive vehicle configurations for consideration leading to selection of the one preferred vehicle for more detailed study in Phase II. The final event of Phase I was the Mid Term Oral Review on 17 October 1973, following which NASA specified the design and performance requirements of the Phase II airplane.

TABLE 1

VEHICLE MISSION AND CONFIGURATION PARAMETERS

CRUISE MACH	M	= 2.2 AND 2.7
WING AREA m^2 (ft^2)	S_W	= 743.2 (8000), 1021.9 (11,000) AND 1300.6 (14,000)
WING THICKNESS RATIO (PERCENT)	t/c	= 3, 5 AND 7
FUSELAGE SECTION AREA m^2 (ft^2)	A_B	= 13.66 (147), 19.17 (206), 23.05 (248) AND 27.22 (293)
THRUST/WEIGHT N/kg	T/W	= 0.5, 0.6, 0.7 AND 0.8
ENGINE TYPES		TURBOFAN AND TURBOJET
GROSS WEIGHT kg (lb)	W_G	= 124,738 (275,000) to 317,515 (700,000)
RANGE km (n.mi.)	R	= 5926 (3200), 7408 (4000), 9260 (5000) AND 10,186 (5500)

TABLE 2

BASIC GUIDELINES

Fuel - liquid hydrogen
Planform - NASA Arrow - wing
Initial Operational Capability - 1990

Use of advanced materials and technology postulated to be developed by 1985. (Data available from Lockheed AST studies (References 1 and 4)).

Certification - FAR Part 25 and SST White Book

Noise - FAR Part 36 minus 5 EPNdB

Fuel Reserves - FAR Part 121.648

Runway Length Determination - FAR Part 25 for 305.6°K (90°F) day and 304.8 m (1000 ft) airport altitude.

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

Aircraft Design Life - 50,000 flying hours

Sonic Boom - no boom at ground level over populated areas

Stability - control configured aircraft

Cost - production up to 600 aircraft. Use modified ATA formulas for DOC evaluation at passenger load factor = 0.55. Use 1972 dollars for direct comparison with AST results. LH_2 available at airports.

Payload - 28,032 kg (61,800 pounds) (258 to 300 passengers, depending on class mix).

Phase II primarily was an analysis to provide design, performance, and cost information for the selected configuration of LH_2 fueled aircraft at a greater level of detail. The design basis and criteria were selected so as to provide a direct comparison of the cost, performance, and design characteristics of the LH_2 fueled supersonic transport with those of an equivalent design of JP fueled aircraft. The JP airplane selected to provide this comparison was one being evolved in a concurrent study by Lockheed for NASA-Langley Research Center under contract NAS 1-12288 titled, "Study of Structural Design Concepts for an Arrow-Wing Supersonic Transport Configuration." (Reference 5). Thus, the following mission requirements were established:

Cruise speed	Mach 2.7	
Range	7778 km.	(4200 n.mi.)
Payload (234 passengers)	22,226 kg.	(49,000 lb.)

To assure equivalency in design and evaluation between the JP and LH_2 aircraft being evolved in the two separate NASA studies, several changes from the basic premises used in Phase I of the subject study were made for Phase II. Table 3 lists the differences from those presented in Table 2.

TABLE 3

CHANGES IN BASIC GUIDELINES FOR PHASE II

(Refer to Table 2)

	Phase I	Phase II
Materials and Technology State-of-the-Art * technology level defined per agreement for contract NAS 1-12288	1985	1981 *
Noise	FAR 36 minus 5	FAR 36
Cost	1972 dollars	1973 dollars
Payload	28,032 kg (61,800 lb.) (300 passengers)	22,226 kg (49,000 lb.) (234 passengers)

Using these criteria, and the duct-burning turbofan engine shown to be preferred in Phase I, preliminary size and performance relationships for the Phase II point design aircraft were established. Aerodynamic data were rechecked and analyses of significant structural areas and of the cryogenic tank thermal protection system were made. Based on these results, new weight and cost relationships were established for input to ASSET. A series of ASSET runs were made to determine the most advantageous set of values for parameters such as thrust-to-weight ratio (T/W) and wing loading (W/S) which would produce an airplane that would perform the desired mission with the best combination of lowest gross weight, lowest fuel weight, and minimum cost. The result was definition of the point design LH_2 fueled AST airplane. Design tradeoffs and sensitivities to various parameters were established to provide information about the importance of each of the significant design and cost variables.

Finally, an assessment was made of the general viability of the concept, including an evaluation of environmental considerations such as exhaust emissions and sonic boom characteristics. Major technology development requirements were enumerated, along with suggested schedules for their implementation. Recommendations were made for follow-on development activity.

3.0 STUDY RESULTS

3.1 Phase I: Parametric Study

The purpose of Phase I was to "parametrically explore the potential of liquid hydrogen fueled AST concepts to determine a preferred configuration and set of design requirements for a specific design to be subsequently examined in greater detail in Phase II" (Reference 4).

3.1.1 Design Trends: The design trends and results of the Phase I Parametric Study are presented in this section. All vehicles represented in the curves meet all the constraints of takeoff and landing distances and the noise limitations, and were picked from ASSET computer results on the basis of minimum weight.

Figure 1 shows the effect of wing thickness (t/c) and wing area (and wing loading) on gross weight for a range of 8704 km (4700 n.mi.).

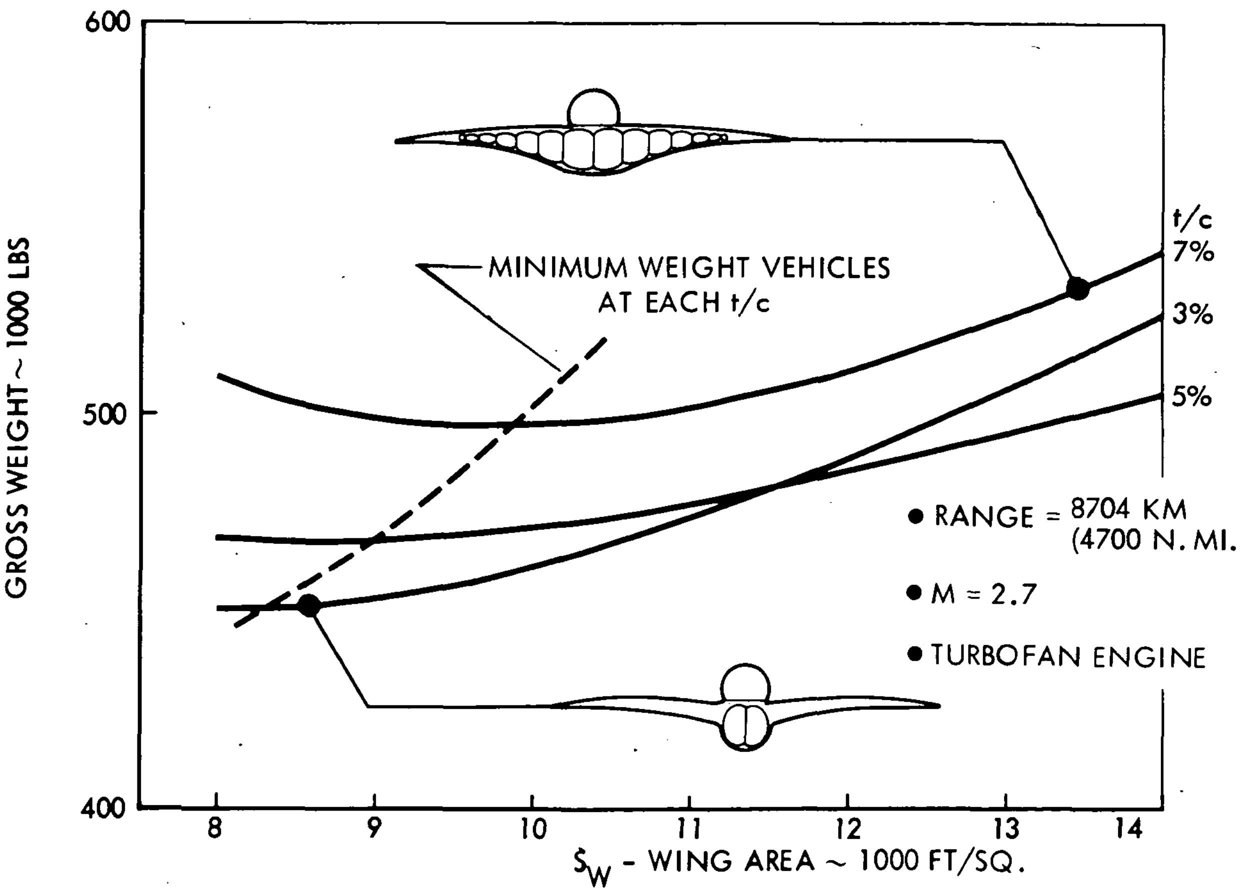

Figure 1. Wing Area and t/c vs. Gross Weight

The range of t/c's and wing areas considered produce vehicle cross sections which vary from a discrete wing-body (t/c = 3, Sw = 743.2 m^2 (8000 FT^2), to a blended wing-body (t/c = 7, Sw = 1300.9 m^2 (14,000 FT^2), as indicated by the sketches on the figure. The trend of the curves illustrates the tradeoff between drag and structural weight. Increasing wing area, relative to the fuselage size, causes the L/D to increase rapidly and the amount of fuel required to decrease. This is more than offset by the rapidly increasing wing weight. The net effect is an increase in the gross weight required to carry the fixed payload. A further effect of increasing wing size (low W/S) is that as the cruise wing loading is lowered, higher altitudes are required to achieve L/D maximum. These higher altitudes require more thurst and the net result is a compromise limiting the actual cruise L/D compared to the maximum attainable. Figure 2, which is derived from Figure 1, shows the effect of wing thickness on gross weight, indicating about the same gross weight for t/c's from 3 to 4% but a rapid rise due to the drag increase beyond 5%. Figure 2 illustrates that in general, due to the low quantity of fuel burn-off of hydrogen AST's the wing is sized by landing field length and not the takeoff condition.

The primary consideration in the choice of engines is the ability to meet the Phase I ground rule of FAR 36 minus 5 EPNdB with regard to airport and community noise. The jet noise generated is based on the relative jet exhaust velocity and noise suppressor effectiveness for both turbojet and turbofan engines. The net effect for both types of engines is to limit the relative exhaust velocities to the range of 1800 - 1900 fps. With this velocity fixed the second consideration is to meet the 10500 ft. engine-out takeoff distance which requires a thrust/weight of approximately 0.3.. The relative jet velocity required to meet the noise constraint is achieved by power cut-back. In the case of the turbojet, this amounts to 61% cut back (about 39% of maximum thrust) while the turbofan can use 44% cut back, (about 56% power). The net result is that the uninstalled thrust/weight of the turbojet is 0.8 compared to only 0.58 for the turbofan to meet both the noise and takeoff distance constraints. The high installed thrust/weight required of the turbojet is partially offset by

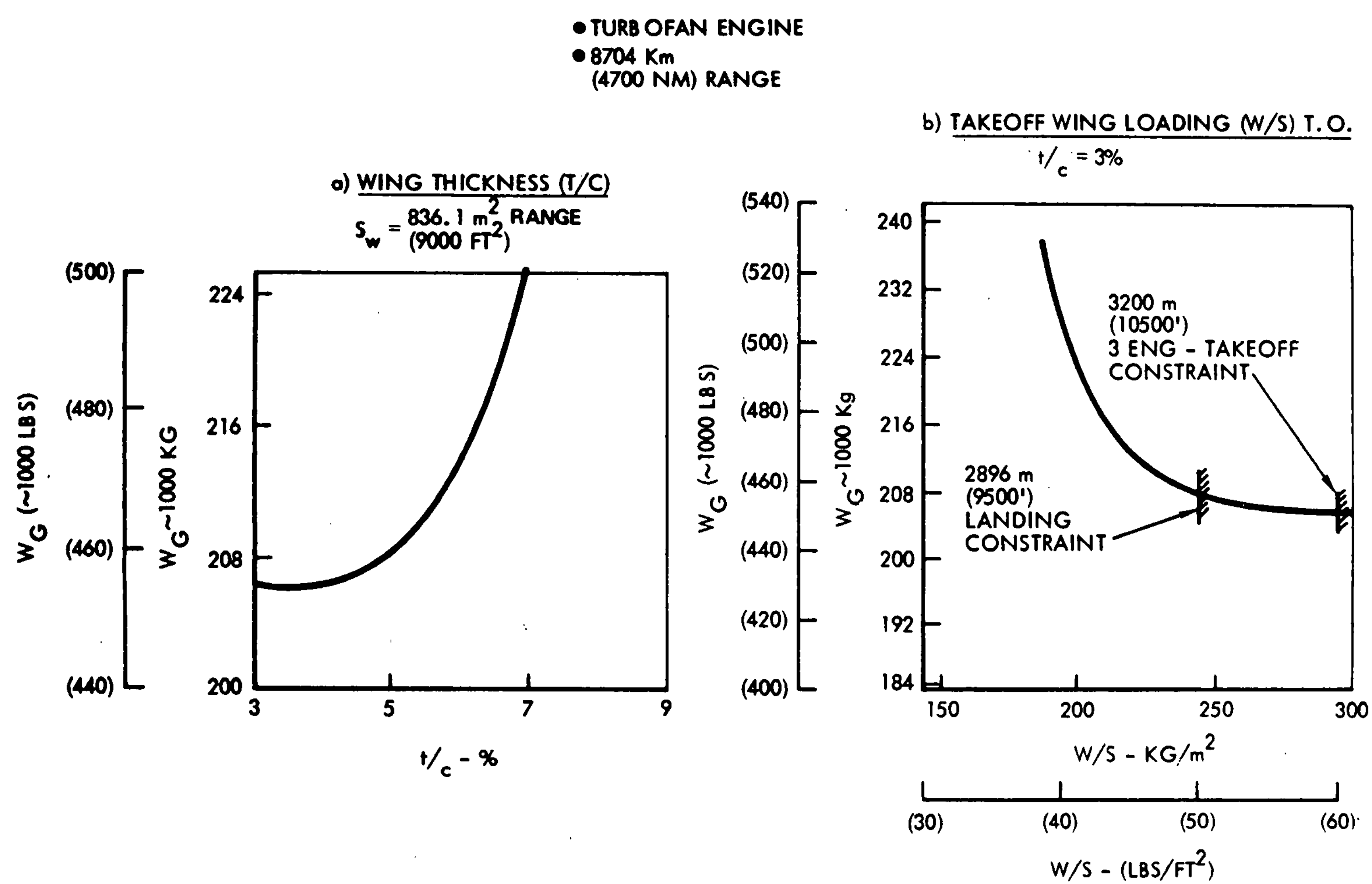

Figure 2. Effect of t/c and W/S on Gross Weight

a lower SFC during supersonic cruise. This is illustrated in Figure 3 which shows the turbojet superior to the turbofan at ranges exceeding 9260 km (5000 n.mi.).

Figure 3 is a summary curve showing a comparison of range vs. gross weight of a series of Mach 2.7 vehicles powered by turbojets and turbofans and Mach 2.2 vehicle powered by turbojets. While turbofan-powered Mach 2.2 vehicles were not examined, it is not expected that the results would be any different than the M 2.7 comparison. The Mach 2.2 vehicles exhibit slightly lower gross weights all the way up to the maximum range investigated (8890 km (4800 n.mi.)).

A further consideration, while not explicit in the study ground rules, is the ability to accomplish off-design missions in which either initial or final cruise legs are flown subsonically to avoid sonic booms in populated areas. Figure 4 shows a comparison of vehicles powered with turbofan and turbojet engines which reflects the lower SFC (0.29) of the turbofan compared to the turbojet (0.37) during subsonic cruise. It should be emphasized that the turbojet cycle characteristics were not optimized for this consideration and that were a subsonic leg actually a design requirement the range deterioration could possibly be reduced. In any event, the turbofan engine cycle is inherently more flexible with regard to choice of performance and noise characteristics than the turbojet.

3.1.2 <u>Cost Trends</u>: The direct operating costs (DOC) for the vehicles shown in Figure 3 are presented in Figure 5. It should be remembered that this is not a plot of a given airplane flying different ranges but rather, different point designs flying at various design ranges. Consequently, the longer the range, the larger the vehicle with an attendent increase in DOC. These DOC's are based on an arbitrary cost of 22¢/kg (10¢/lb.) for the liquid hydrogen fuel.

Table 4 is presented to illustrate a typical flyaway cost for an LH_2 fueled AST, compared with a Jet A-1 fueled AST. The $/kg ($/lb) cost factors indicated have been increased for the hydrogen vehicle by an estimated complexity factor, where appropriate. In both cases the range and payload are the same.

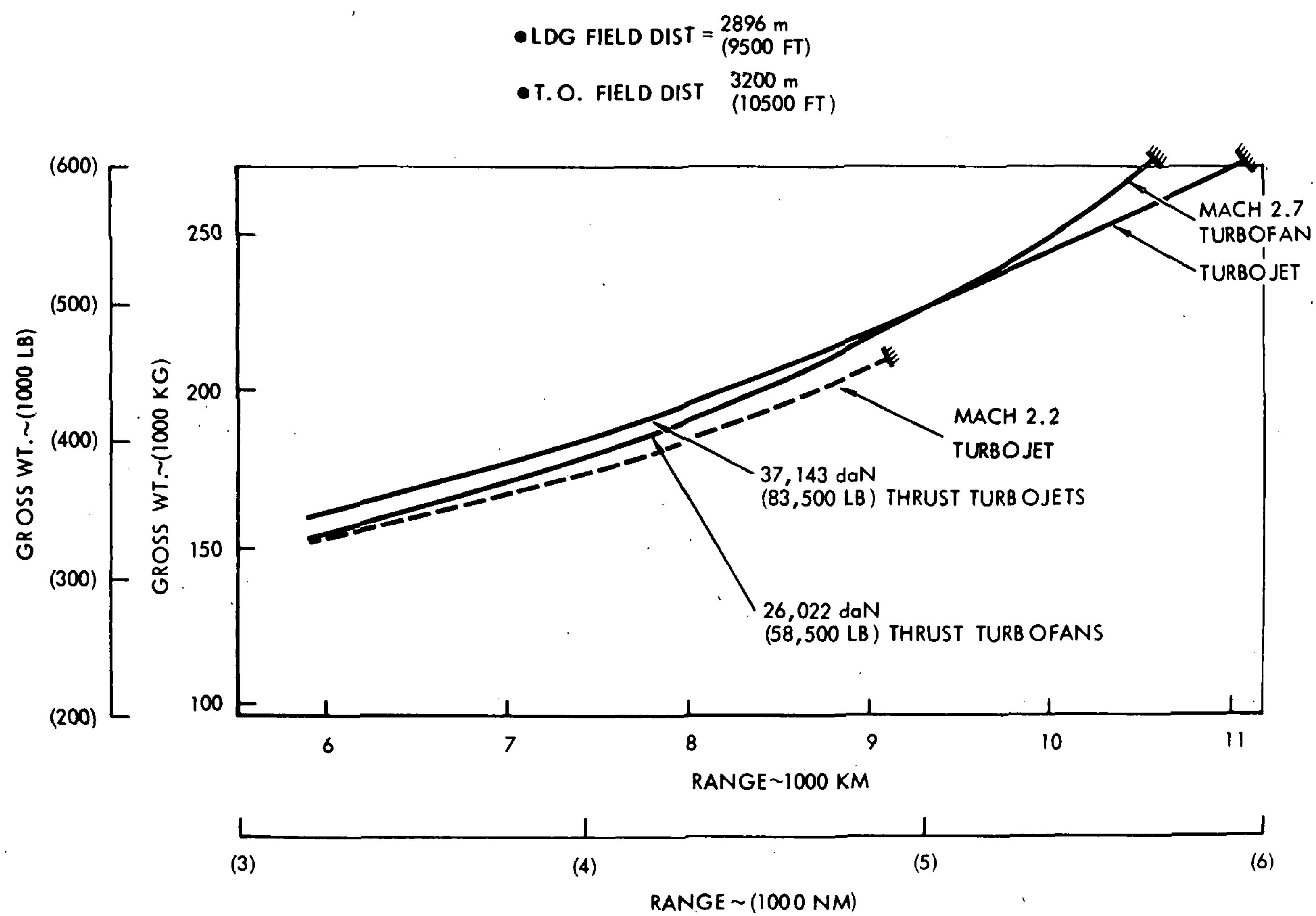

Figure 3. Vehicle Range - Weight and Engine Comparison

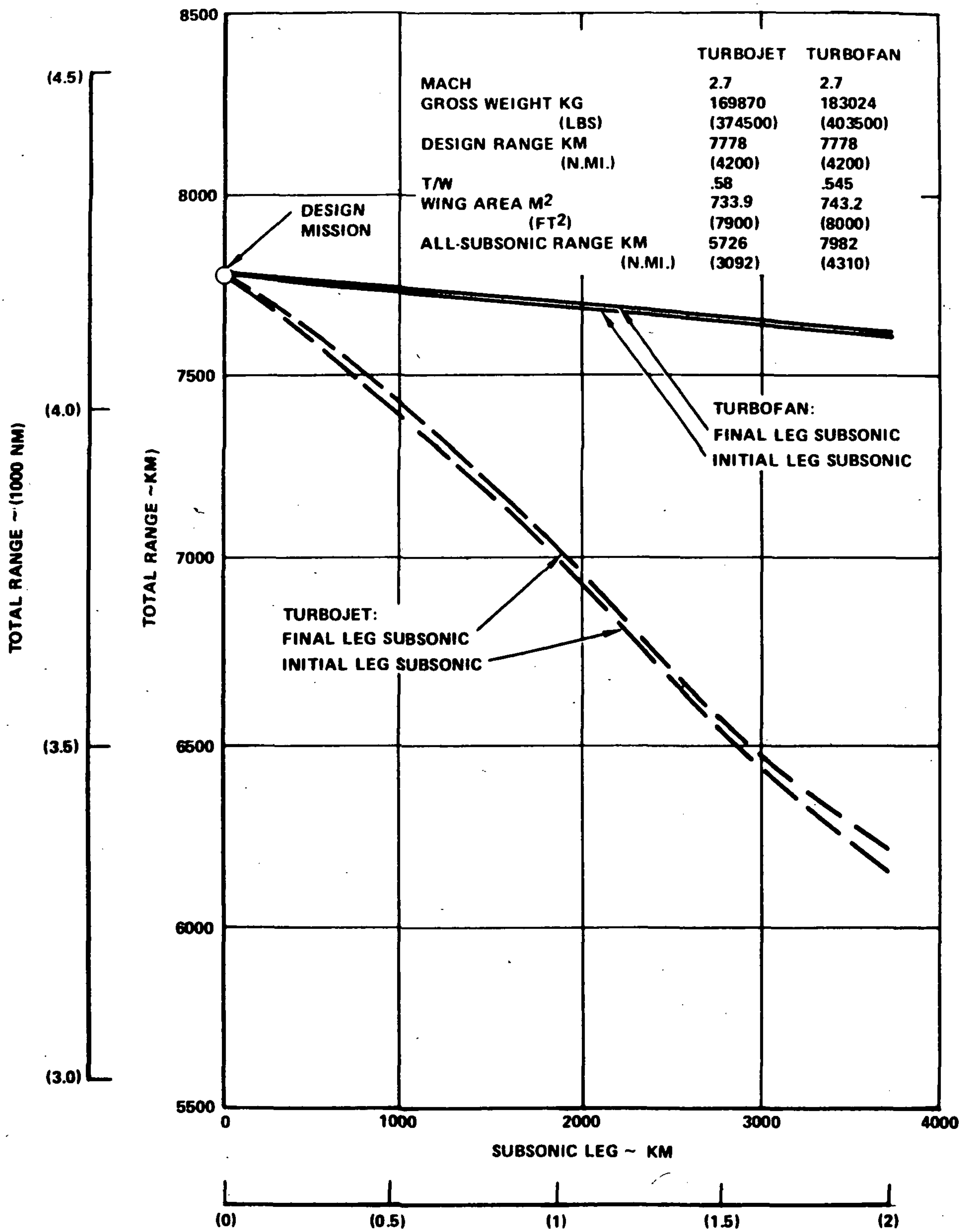

Figure 4. Comparison of Turbojet and Turbofan Engine in Performing Missions With Subsonic Cruise Legs

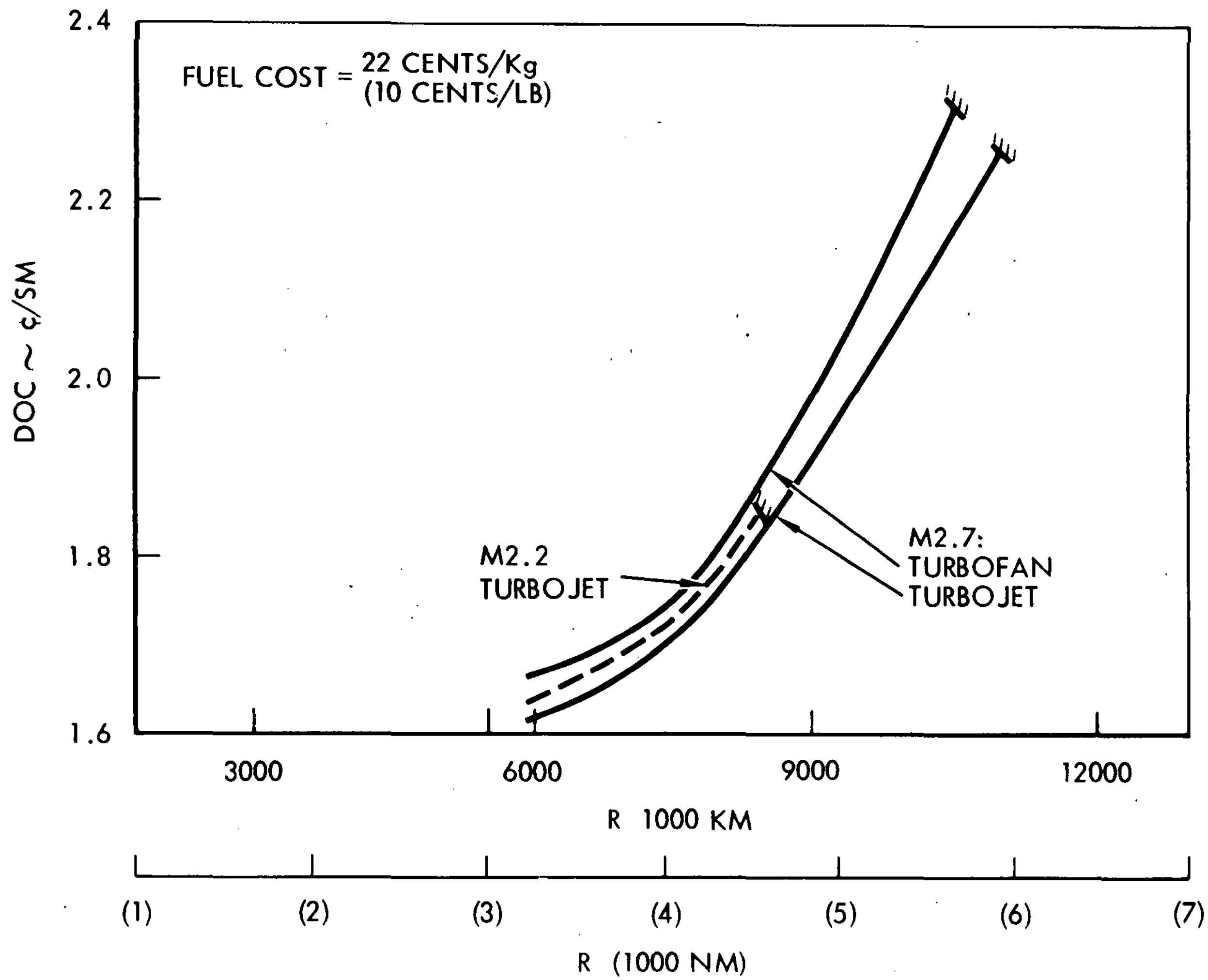

Figure 5. DOC Cost Comparison

TABLE 4

FLYAWAY COST COMPARISON JET A-1 vs. LH_2 FUELED AST

BASIS:

- No R&D Amortization
- 300 Aircraft
- 1972 Dollars Using 1980-85 Technology

	7,778 km (4,200 n.mi.)/2.7/TF			7,778 km (4,200 n.mi.)/2.7/TJ		
	Jet A-1 Fuel			LH_2 Fuel		
	Weight~kg (lbs)	Dollars	\$/kg (\$/lb)	Weight~kg (lbs)	Dollars	\$/kg (\$/lb)
Wing	43,551 (96,013)	13,670,000	313 (142)	23,479 (51,763)	8,998,000	384 (174)
Fuselage	17,056) (37,602)	5,247,000	309 (140)	19,121 (42,154)	6,906,000	362 (164)
Fuel System and Tank	2,400 (5,291)	832,000	346 (157)	12,100* (26,675*)	1,660,000	137 (62)
Other	47,106 (103,851)	38,320,000	814 (369)	30,220 (66,624)	27,088,000	897 (407)
Engine	20,615) (45,448)	7,684,000	373 (169)	11,733 (25,868)	7,604,000	648 (294)
Avionics	863 (1,903)	500,000	5,798 (2,630)	863 (1,903)	500,000	5,798 (2,630)
	131,590 (290,108)	66,253,000		97,516 (214,987)	52,756,000	

*Includes LH_2 Tankage

3.1.3 General Conclusions: As a result of the parametric design study of Phase I, the following general conclusions were reached concerning liquid hydrogen fueled supersonic transport aircraft:

- Configurations with relatively thin wings and large fuselages (to contain the fuel) provide superior performance.

- LH_2 Fueled AST aircraft are capable of ranges in excess of 11,112 km (6,000 n.mi.) with reasonable gross weights.

- Low mission fuel burn-off dictates wing loading (W/S) of approximately 195.3 kg/m^2 (40 Lbs/Ft^2) to meet landing field length of 2896 m (9500 ft.).

- Low take-off wing loading 220 - 224 kg/m^2 (45 - 50 lb/ft^2) means aircraft reach L/D max at low q's (high altitude) requiring high thrust/weight (.55 - .60).

- High thrust/weight results in satisfactory engine-out takeoff field length performance even with power cut back required by noise constraint.

- Turbofan engines are most promising for shorter ranges... Turbojet engines are most promising for long range, all-supersonic missions.

- Use of turbojets requires very large engines, deeply throttled at takeoff, to meet noise constraints.

3.1.4 Candidate Vehicle Selection

The completion of the Phase I parametric studies required the selection of the four most promising vehicles as candidates for the Phase II configuration refinement study. In cooperation with NASA, it was decided the four vehicles selected should reflect two range capabilities: 7778 km (4200 n.mi.), which represents transatlantic capability; and 10186 km (5500 n.mi.), representing transpacific capability, each with adequate subsonic cruise range either before or after supersonic cruise. It was also decided the selected vehicles should be designed for Mach 2.7 cruise, except that one vehicle should be designed for Mach 2.2 cruise and 7778 km (4200 n.mi.) range, to provide a specific comparison. Finally, although the turbojet engine was demonstrated to be more economical at long range cruise (Figure 3), it was felt that aircraft with both types of engines should be compared at the shorter range.

As a result the following requirements were established for the four aircraft designs which were to be compared leading to the ultimate selection of one design for detailed study in Phase II.

Cruise Speed	Range	Engine Type
M 2.7	7,778 km (4200 N.Mi.)	Turbofan
M 2.7	7,778 km (4200 N.Mi.)	Turbojet
M 2.7	10,186 km (5500 N.Mi.)	Turbojet
M 2.2	7,778 km (4200 N.Mi.)	Turbojet

Table 5 presents the characteristics of the four vehicles which were picked from the parametric data generated in Phase I to define the most attractive candidate aircraft to satisfy the stated requirements. There are several interesting items to note in the table. For example, columns 1 and 2 provide a comparison of turbofan vs. turbojet powered Mach 2.7 aircraft, each designed for a range of 7778 km (4200 n.mi.). The turbofan airplane has a significantly higher SFC in cruise and yet it requires only 3,039 kg (6,700 lbs) more fuel. Also, its gross and empty weights are appreciably less than those of the turbojet aircraft. Explanations for this involve several factors. First, the SFC of the duct-burning turbofan in both subsonic cruise and loiter is much lower than the counterpart turbojet, partially offsetting its higher supersonic cruise SFC; consequently, less fuel is needed to meet the low speed and reserve requirements. Second, the turbofan engines need not be throttled as deeply at takeoff to meet the noise limitation so smaller, lighter engines can be used. Thirdly, because the accumulation of such factors results in a lower landing weight, the wing area required to meet the landing distance requirements is smaller, leading to additional weight saving, finally resulting in the values of OEW, ZFW, and G. W. shown in the table.

A comparison of columns 2 and 4 offers some insight into the effect of cruise speed on vehicle performance and cost. Both the Mach 2.2 aircraft of column 4 and the Mach 2.7 aircraft of column 2 are powered by turbojet engines and both are designed for the same payload/range. The wing area of the Mach 2.2 vehicle is significantly smaller than that of the Mach 2.7 aircraft. The aircraft have wing loadings of 286.5 kg/m^2 (58.6 lb/ft^2) and 227.0 kg/m^2 (46.5 lb/ft^2), respectively. This is due to the higher aspect ratio (AR = 2) of the Mach 2.2 airplane compared to only 1.62 for the Mach 2.7. The wing is sized by airport performance requirements in both cases so the higher available lift coefficient (0.69) for the Mach 2.2 airplane, compared to 0.48 for the lower aspect ratio wing of the Mach 2.7 design, allows a reduction in wing size and correspondingly higher wing loading. The Mach 2.2 vehicle shows a slightly lower gross weight than the Mach 2.7 but almost equal fuel consumption. The DOC of the Mach 2.2 is slightly higher than the 2.7 in spite of its lower cost. This is due to its higher crew, insurance, and depreciation cost per seat mile which results from the lower productivity of the slower Mach 2.2 vehicle (888 flights per year vs 1039 for the Mach 2.7).

Column 3 lists the characteristics of a Mach 2.7 turbojet powered aircraft designed for trans-Pacific range capability. Comparison of those data with column 2 provides an appreciation of the differences in design and operating cost resulting from increasing design range from 7778 km (4200 n.mi.) to 10,186 km (5500 n.mi.). Gross weight increases over 32 percent, production cost of the aircraft increases nearly 28 percent, and Direct Operating Cost increases almost 19 percent.

3.2 Phase II: Vehicle Point Design Study

The purpose of Phase II was to "establish design, performance, and cost characteristics of a selected configuration of SST at a greater level of detail to provide confidence in the results and guidance for additional development" (Reference 4).

TABLE 5

CHARACTERISTICS OF SELECTED VEHICLES FROM PHASE I

	Vehicle Configuration No.	1	2	3	4
General	Mach	2.7	2.7	2.7	2.2
	Range ~ km (n.mi.)	7,778 (4,200)	7,778 (4,200)	10,186 (5,500)	7,778 (4,200)
	Engine Type	Turbofan	Turbojet	Turbojet	Turbojet
	T/W	.58	.80	.80	.80
	Wing Area - m^2 (ft^2)	743.2 (8,000)	836.1 (9,000)	1,035.8 (11,150)	630.3 (6,785)
	W/S - kg/m^2 (lbs/ft^2)	246 (50.4)	227 (46.5)	242 (49.6)	286 (58.6)
	t/c - %	3	3	3	3
	Aspect Ratio	1.62	1.62	1.62	2
Weights	Gross Wt. kg (lb)	183,024 (403,500)	189,737 (418,300)	250,835 (553,000)	180,392 (397,700)
	Fuel Wt. ~ kg (lb)	45,813 (101,000)	42,774 (94,300)	67,721 (149,300)	43,318 (95,500)
	Zero Fuel Wt.~kg (lb)	137,211 (302,500)	146,863 (324,000)	183,114 (403,700)	137,075 (302,200)
	Payload kg (lb)	28,032 (61,800	28,032 (61,800)	28,032 (61,800)	28,032 (61,800)
	OEW ~ kg (lb)	109,179 (240,700)	118,931 (262,200)	155,082 (341,900)	109,043 (240,400)
	PL/W_G	.153	.148	.112	.155
	WH_2/W_G	.25	.228	.27	.24
Performance	FAR T.O. Dist. ~ m (ft)	3,200 (10,500)	2,730 (8,970)	2,865 (9,400)	2,730 (8,950)
	FAR Ldg. Dist. ~ m (ft)	2,860 (9,382)	2,866 (9,404)	2,890 (9,483)	2,896 (9,500)
	Average Alt. - Cruise m (ft)	20,733 (68,000)	21,495 (70,500)	21,495 (70,500)	18,900 (62,000)
	Average L/D - Cruise	7.38	7.71	7.69	7.72
	Average SFC - Cruise kg/hr/daN (lbs/hr/lb)	.581 (.57)	.502 (.493)	.500 (.491)	.456 (.447)
	SFC - Subsonic Cruise kg/hr/daN (lbs/hr/lb)	.300 (.294)	.378 (.371)	.378 (.371)	.357 (.350)
	SFC - Subsonic Loiter kg/hr/daN (lbs/hr/lb)	.220 (.216)	.359 (.352)	.367 (.36)	.359 (.352)
Cost	DOC - ¢/SM	1.79	1.70	2.02	1.74
	*Basic Price $\$x10^6$	58.3	61.1	78	55.7

*300 Aircraft Production Base

3.2.1 Design Requirements

As described in Section 2, an overriding objective in Phase II was to provide a one-for-one basis for comparing a conventionally fueled (Jet A-1) AST with one fueled with liquid hydrogen. For that reason the payload requirement of the aircraft studied in Phase II was different from that used in Phase I. Otherwise, the mission corresponded to that of Candidate Vehicle No. 1 listed in Table 5. In addition, a more conservative definition of materials and technology state-of-the-art was assumed, again to correspond with that being used in the study being conducted by Lockheed for NASA-Langley Research Center, generally called the Arrow-Wing Structures Study (Reference 5). In that program, 1981 technology was defined as, basically, use of titanium skin and structure, reinforced with layup of boron-polyimide composite. By contrast, in Phase I of the subject study, 1985 technology was defined as use of almost 90 percent advanced composite materials in the wing, fuselage, and empennage structures. Because of these significant changes, plus consideration of the relaxed noise requirement (see Table 3), the first task of Phase II was to resize the candidate vehicle to establish a basis for study of the structural and thermal protection problems of the point design airplane.

3.2.2 Configuration Description

The final configuration of the point design airplane is shown in Figure 6 and 7. Figure 6 is the general arrangement showing the overall size and configuration of the design. The Arrow-wing planform prescribed by NASA-Langley for the AST Systems Studies and the Arrow-Wing Structures Studies (Reference 5) is used. Likewise, the wing section and high lift devices are the same. The wing has leading edge flaps and trailing edge flaps. Control surfaces are conventional. The familiar "droop snoot" is also incorporated. Four duct-burning turbofan engines are shown.

Figure 7 is an inboard profile which shows the location of significant items of equipment within the airplane. Of particular interest is the passenger cabin/hydrogen tank arrangement. This configuration was selected

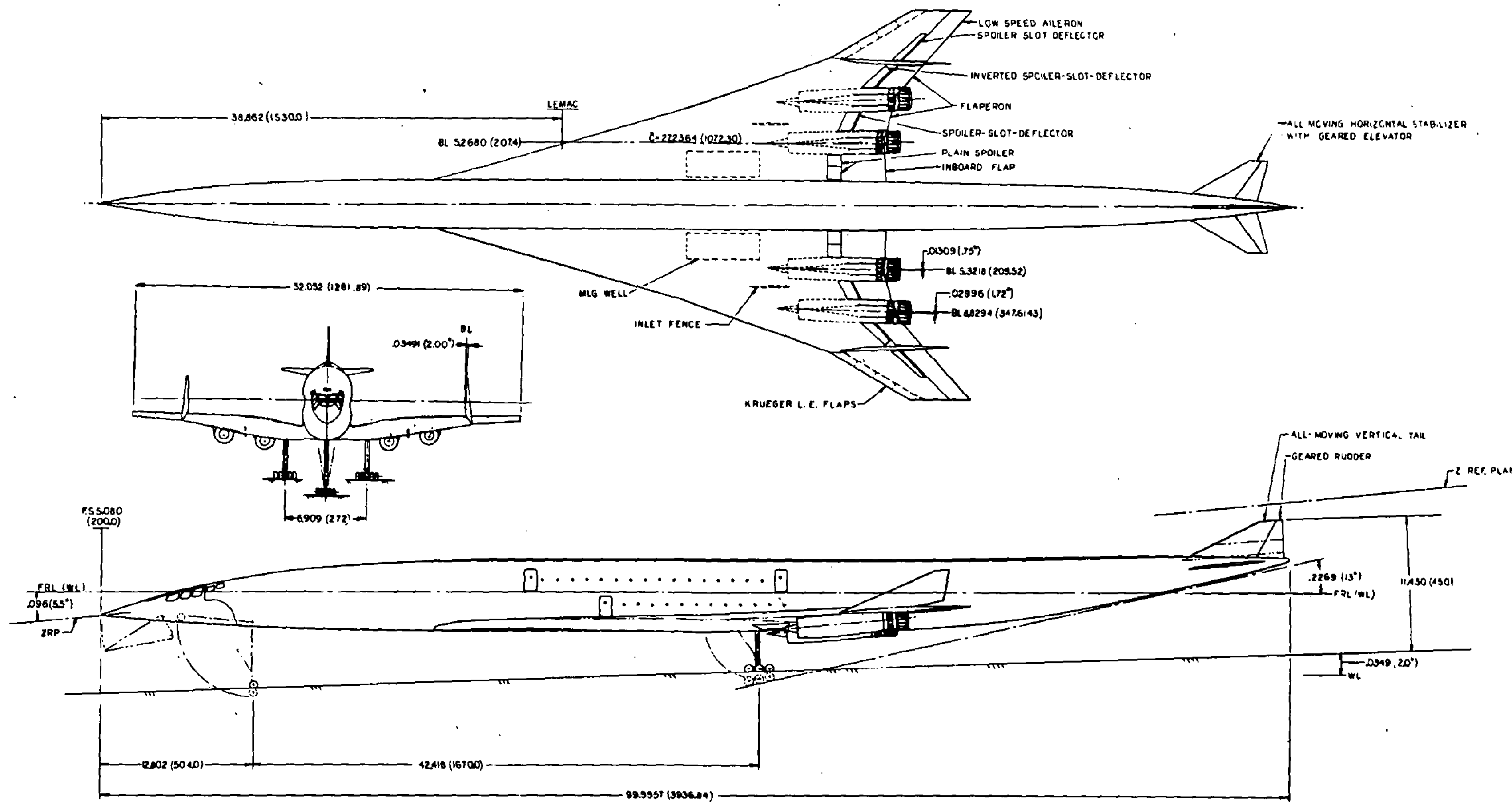

Figure 6. General Arrangement-Point Design LH_2 AST

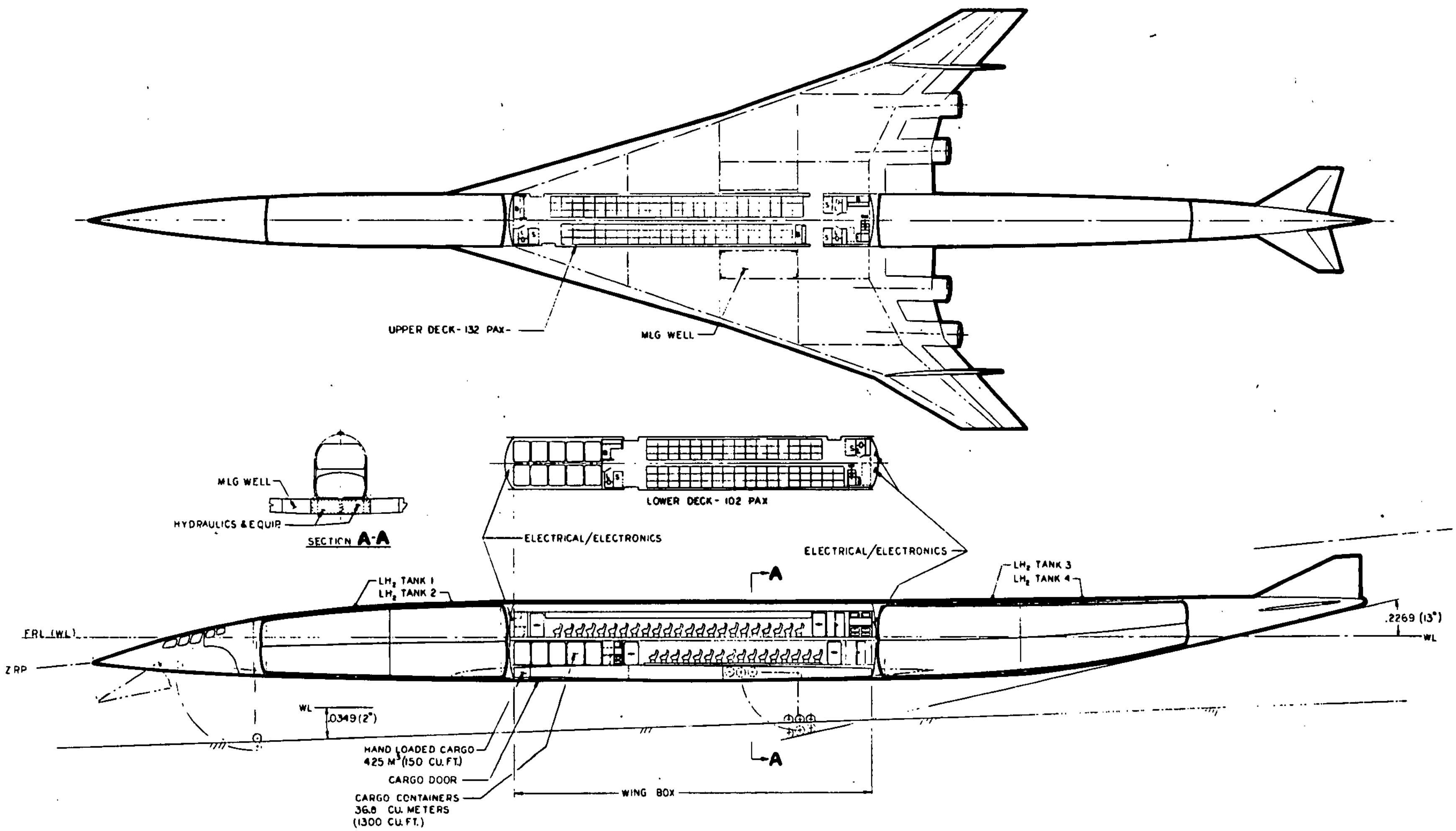

Figure 7. Interior Arrangement-Point Design LH_2 AST

after consideration of a number of other possibilities. The selection was based on the following criteria:

- o passenger/fuel separation distance
- o passenger/fuel relative locations
- o structural feasibility
- o airplane center-of-gravity control
- o passenger seating and accessibility
- o volume utilization
- o tank accessibility for maintenance and inspection

The double-lobe cross section was found to be more efficient for passenger seating than a simple circular shape, and the low-wing location offered structural advantages over a mid-wing arrangement. The wing structure is conceptually identical to that being evolved in the NASA-Langley Arrow-Wing Structures Study, modified in detail to account for:

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

The wing skin is titanium alloy and its structural framework is a series of spanwise beams located approximately 20 inches 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.

The fuselage, except for the hydrogen tanks, 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.

Both integral and non-integral tank design concepts were investigated for containing the liquid hydrogen fuel. An integral design was selected on the basis of the analysis which showed it offered advantages in both volumetric efficiency and structural weight fraction. The selected design has a welded aluminum tank skin, stiffened with integral longitudinal stringers, and stabilized with circumferential frames. Approximately every 5.08 m (200 inches) 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 walkway 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 consists of a layer of closed cell foam material bonded to the tank exterior surfaces for cryogenic insulation, a layer of high temperature insulation sealed to prevent air penetration, and a fiberglass/polyimide honeycomb core faced with graphite/polyimide surfaces to serve the combined functions of additional high temperature insulation and heat shield.

3.2.3 Vehicle Performance

Performance of the point design airplane was calculated on the basis of updated propulsion, aerodynamic, and weight inputs to ASSET. The appropriate input parameters were calculated to reflect the specific design shown on Figures 6 and 7. ASSET was then exercised to determine the combination of wing loading (W/S) and thrust/weight (T/W) at which the constraints of 2896 m (9500 ft.) landing field length, 3200 m (10,500 ft.) takeoff field length (with one engine out), and FAR Part 36 sideline and flyover noise limits would all be met most effectively, i.e., with the best combination of lowest gross weight, lowest fuel weight, and minimum cost airplane. The result of this investigation is shown in Table 6, a summary of the major items of interest to describe the characteristics of the point design airplane. Table 7 is a

TABLE 6

CHARACTERISTICS OF POINT DESIGN LH_2 AST

Payload		kg	(lb)	22,226	(49,000)
Range		km	(n.mi.)	7,778	(4,200)
Cruise Speed		Mach		2.7	
Takeoff Gross Weight		kg	(lb)	166,946	(368,054)
Zero Fuel Weight		kg	(lb)	123,419	(272,094)
Operating Empty Weight		kg	(lb)	101,193	(223,094)
Fuel Weight	Total	kg	(lb)	43,209	(95,960)
	Mission	kg	(lb)	36,940	(81,440)
Fuel Volume		m^3	(ft^3)		21,700
Wing Area		m^2	(ft^2)	639.2	(6,880)
Wing Loading (W/S)	Takeoff	kg/m^2	(lb/ft^2)	261.2	(53.5)
	Landing	kg/m^2	(lb/ft^2)	203.6	(41.7)
Span		m	(ft)	32.2	(105.6)
Overall Length		m	(ft)	100	(328)
Cruise Altitude		m	(ft)	20,574	(67,500)
Lift/Drag (cruise)				6.99	
Specific Fuel Consumption (cruise)		kg/hr/daN	(lb/hr/lb)	0.572	(0.561)
Thrust/Weight (SLS)				0.50	
Thrust per engine		Newton	(lb)	204,618	(46,000)

Weight Fractions	Percent
Fuel	26.1
Payload	13.3
Structure	33.5
Propulsion	15.8
Other	11.3

summary of mission characteristics for a 7778 km (4200 n.mi.) flight, carrying a full payload of 22,226 ks (49,000 lbs), cruising at Mach 2.7. The standard reserve requirement, flown at the end of the maximum range mission, is included. Most of the column headings in Table 7 are self-explanatory; the route segments are defined as follows:

Segment	Explanation
Takeoff	
Power 1	taxi power setting
Power 2	takeoff power setting
Climb	climb to 1524 m (5000 ft.) at takeoff power
Cruise	cruise in holding pattern for traffic clearance - no distance credit
Accel	accelerate at constant altitude
Climb	climb and accelerate
Climb	climb and accelerate to cruise speed
Climb	climb to start-of-cruise altitude
Cruise	cruise at Mach 2.7
Decel	decelerate at constant altitude
Descent	decelerate and descend to 1524 m (5000 ft.)
Cruise	a simulation of fuel expenditure to account for approach and landing

The point design airplane was studied to determine the effect perturbations of some of the design variables would have on its performance. The results of these design tradeoffs and sensitivity studies are shown in Figures 8 through 14, plus following pages.

TABLE 7a

MISSION SUMMARY: POINT DESIGN LH_2 AST

(SI Units)

SEGMENT	INIT ALTITUDE (m)	INIT MACH NO	INIT WEIGHT (kg)	SEGMT FUEL (kg)	TOTAL FUEL (kg)	SEGMT DIST (km)	TOTAL DIST (km)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	AVG L/D RATIO	AVG SFC (kg/hr/daN)
TAKEOFF											
POWER 1	0.	0.0	166946.	207.3	207.3	0.	0.	10.0	10.0	0.0	.152
POWER 2	0.	0.300	166738.	315.7	523.0	0.	0.	0.5	10.5	6.35	.366
CLIMB	0.	0.300	166423.	426.8	949.8	9.3	9.3	1.2	11.7	8.38	.384
CRUISE	1524.	0.414	165996.	264.4	1214.2	0.	9.3	4.0	15.7	9.01	.219
ACCEL	1524.	0.414	165732.	91.6	1305.8	7.4	16.7	0.7	16.4	9.89	.237
CLIMB	1524.	0.539	165640.	2065.	3370.8	190.8	207.4	13.7	30.1	9.84	.342
CLIMB	10363.	0.989	163575.	7462.	10832.8	827.8	1035.	24.0	54.2	6.20	.569
CLIMB	19202.	2.700	156113.	201.4	11034.2	38.9	1074.	0.8	55.0	6.92	.585
CRUISE	20117.	2.700	155912.	25569.	36603.2	6401.	7475.	133.7	188.7	6.99	.572
DECEL	21031.	2.700	130343.	9.5	36612.7	59.3	7532.	1.3	190.0	6.99	-.213
DESCENT	21031.	2.282	130333.	87.5	36700.2	244.5	7778.	11.9	201.9	7.97	-.126
CRUISE	1524.	0.414	130245.	240.8	36941.	0.	7778.	5.0	206.9	9.80	.222
RESERVE	0.	0.0	130005.	2586.	39527.	0.	0.	0.0	0.0	0.0	.0
CLIMB	0.	0.200	127419.	262.2	39789.2	5.6	5.6	0.7	0.7	8.41	.382
CLIMB	457.	0.505	127156.	151.6	41305.2	188.9	194.5	13.4	14.2	9.10	.310
CRUISE	10973.	0.900	125640.	773.4	42078.6	194.5	388.9	12.3	26.4	9.67	.299
DESCENT	10973.	0.900	124866.	54.0	42132.6	92.6	381.5	7.1	33.5	9.07	-.172
CRUISE	457.	0.503	124812.	139.3	43525.6	0.	381.5	30.0	63.5	9.96	.228

NOTE: Negative values of SFC indicate drag is greater than thrust in those flight segments.

TABLE 7b

MISSION SUMMARY: POINT DESIGN LH_2 AST

(Customary U. S. Units)

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)	AVG L/D RATIO	AVG SFC (lb/hr/lb)
TAKEOFF											
POWER 1	0.	0.0	368054.	457.	457.	0.	0.	10.0	10.0	0.0	0.149
POWER 2	0.	0.300	367597.	696.	1153.	0.	0.	0.5	10.5	6.35	0.359
CLIMB	0.	0.300	366902.	941.	2094.	5.	5.	1.2	11.7	8.38	0.377
CRUISE	5000.	0.414	365960.	583.	2676.	0.	5.	4.0	15.7	9.01	0.215
ACCEL	5000.	0.414	365378.	202.	2879.	4.	9.	0.7	16.4	9.89	0.233
CLIMB	5000.	0.539	365175.	4552.	7431.	103.	112.	13.7	30.1	9.84	0.336
CLIMB	34000.	0.989	360623.	16451.	23882.	447.	559.	24.0	54.2	6.20	0.558
CLIMB	63000.	2.700	344172.	444.	23426.	21.	580.	0.8	55.0	6.92	0.574
CRUISE	66000.	2.700	343728.	56370.	80696.	3456.	4036.	133.7	188.7	6.99	0.561
DECEL	69000.	2.700	287358.	21.	80717.	32.	4067.	1.3	190.0	6.99	-0.209
DESCENT	69000.	2.282	287337.	193.	80910.	132.	4200.	11.9	201.9	7.97	-0.124
CRUISE	5000.	0.414	287143.	531.	81441.	0.	4200.	5.0	206.9	9.80	0.218
RESERVE	0.	0.0	286613.	5701.	87142.	0.	0.	0.0	0.0	0.0	0.0
CLIMB	0.	0.200	280912.	578.	87720.	3.	3.	0.7	0.7	8.41	0.375
CLIMB	1500.	0.505	280333.	3343.	91064.	102.	105.	13.4	14.2	9.10	0.304
CRUISE	36000.	0.900	276990.	1705.	92769.	105.	210.	12.3	26.4	9.67	0.293
DESCENT	36000.	0.900	275284	119.	92888.	50.	260	7.1	33.5	9.07	-0.169
CRUISE	15000.	0.503	275165.	3072.	95960.	0.	260.	30.0	63.5	9.96	0.224

NOTE: Negative values of SFC indicate drag is greater than thrust in those flight segments.

FIGURE 8 SHOWS THAT AT THE DESIGN POINT, I.E., A RANGE OF 4200 NM INDICATED BY THE SQUARE DOT, APPROXIMATELY 70 LB OF GROSS WEIGHT IS REQUIRED TO AFFECT A CHANGE OF DESIGN RANGE OF 1 NM. FOR SIGNIFICANTLY LARGE INCREASES IN RANGE, FOR EXAMPLE TO PROVIDE 5400 NM, THE GROSS WEIGHT OF THE LIQUID HYDROGEN-FUELED AST AIRCRAFT WOULD NEED TO GROW TO ABOUT 460,000 LB.

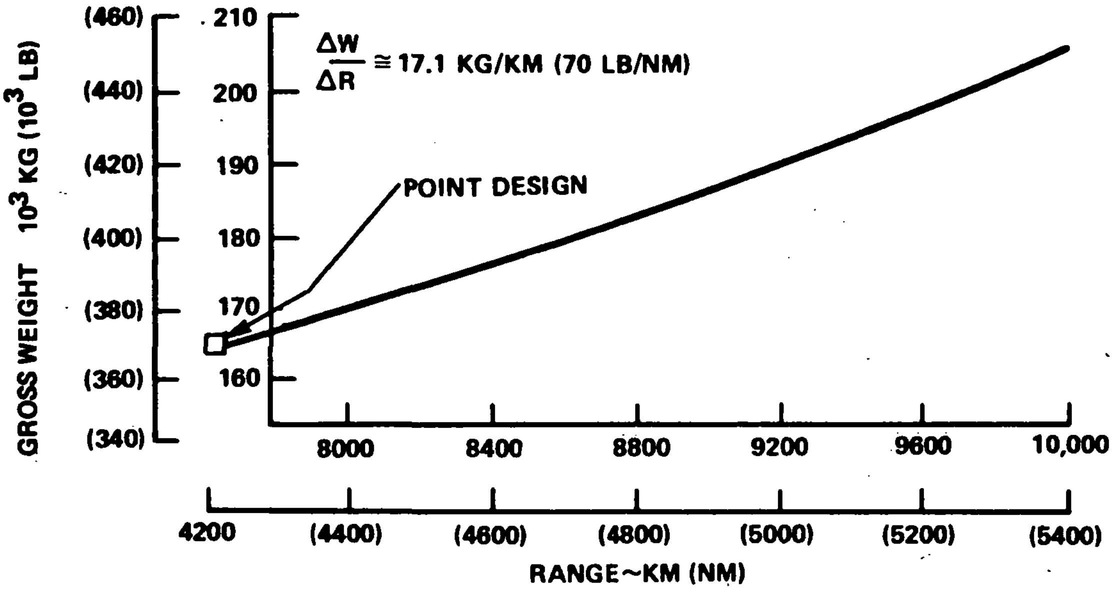

Figure 8. Gross Weight vs. Design Range

THE EFFECT ON RANGE OF A CHANGE IN NOMINAL SPECIFIC FUEL CONSUMPTION IS SHOWN IN FIGURE 9. IT AMOUNTS TO APPROXIMATELY 54.3 NM PER PERCENT CHANGE IN SFC AROUND THE DESIGN POINT. THIS IS A SENSITIVE PARAMETER. THE AVERAGE CRUISE SPECIFIC FUEL CONSUMPTION IS 0.572 $\left(\frac{kg}{hr}\right)$/daN $\left[0.561 \left(\frac{lb}{hr}\right)/lb\right]$. A ONE PERCENT CHANGE IN SFC THEREFORE REPRESENTS ONLY 0.00612 $\left(\frac{kg}{hr}\right)$/daN $\left[0.006 \left(\frac{LB}{HR}\right)/LB\right]$ DIFFERENCE.

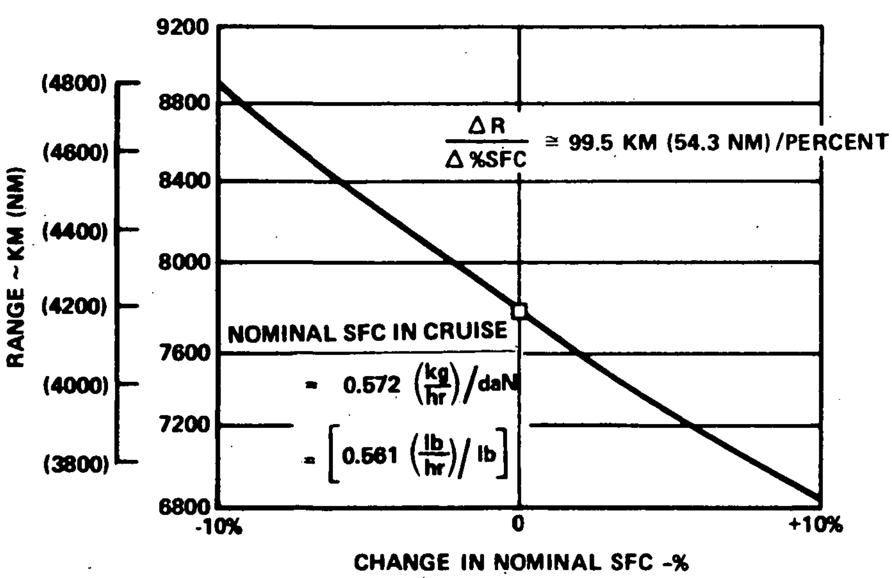

Figure 9. Range vs. Change in Specific Fuel Consumption

FIGURES 10a) AND 10b) ILLUSTRATE THE EFFECT INCREMENTAL EMPTY WEIGHT CHANGE WOULD HAVE ON VEHICLE CHARACTERISTICS. FIGURE 10a) SHOWS THE EFFECT OF EMPTY WEIGHT ON GROSS WEIGHT IF THE TREND TOWARD HIGHER EMPTY WEIGHTS IS OBSERVED BEFORE DESIGN FREEZE. AT THIS STAGE OF THE PROGRAM A FIRM TREND TOWARD APPRECIABLY HIGHER EMPTY WEIGHT CAN BE COMPENSATED FOR IN THE DESIGN BY INCREASING WING SIZE, ENGINE SIZE, PROVIDING FOR MORE FUEL WEIGHT, INCREASING LANDING GEAR WEIGHT, ETC. THESE CHANGES MIGHT ALL BE REQUIRED TO HOLD DESIGN PARAMETERS SUCH AS T/W AND W/S TO THEIR SPECIFIED VALUES. UNDER THESE CONDITIONS, GROSS WEIGHT MUST INCREASE 2.65 LB FOR EVERY POUND EMPTY WEIGHT INCREASES. WITHIN THE LIMITS ILLUSTRATED THE TREND IS A STRAIGHT LINE.

IF EMPTY WEIGHT CHANGES AFTER DESIGN FREEZE AND PAYLOAD MUST BE HELD CONSTANT SO THAT CHANGE IN RANGE IS THE ONLY TRADEOFF POSSIBLE, RANGE WILL BE AFFECTED AS SHOWN IN FIGURE 10b). AGAIN THE VARIATION IS A STRAIGHT LINE AROUND THE DESIGN POINT BUT IN THIS CASE THE TRADEOFF IS 4.14 NM PER 100 LB OF EMPTY WEIGHT CHANGE.

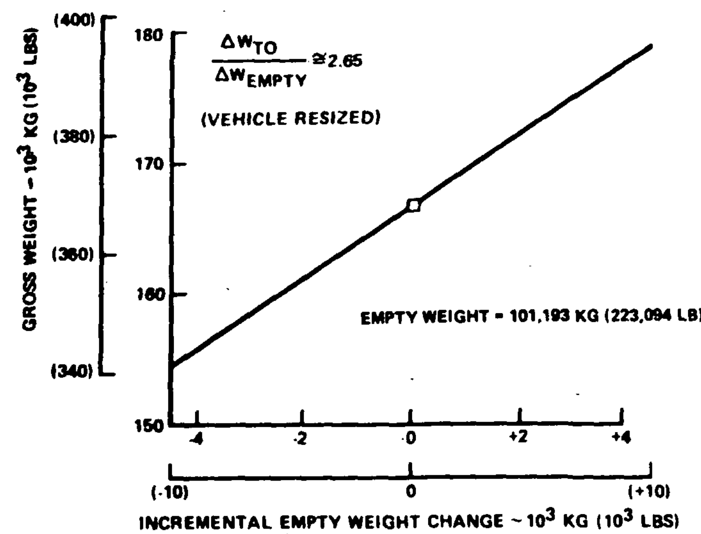

Figure 10a. Growth Factor: Gross Weight vs. Change in Empty Weight (Constant Range)

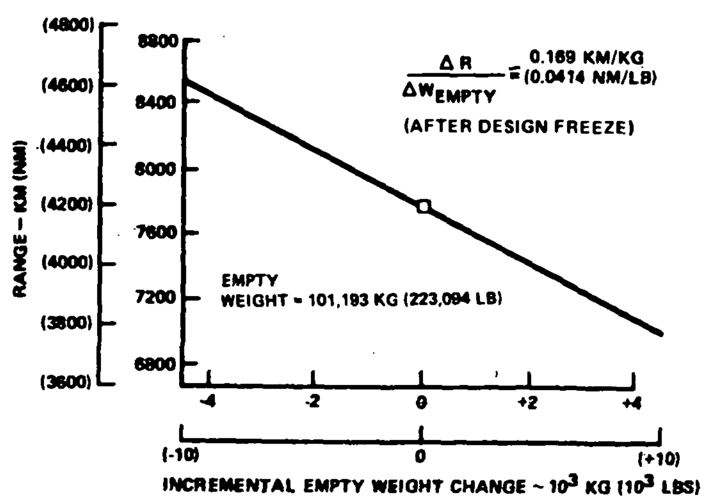

Figure 10b. Tradeoff: Range vs. Change in Empty Weight (Constant Gross Weight)

THE PENALTY IN TOTAL RANGE THAT RESULTS FROM HAVING TO FLY INITIAL OR FINAL LEGS AT SUBSONIC SPEEDS OVER POPULATED AREAS IS SHOWN IN FIGURE 11. THE DECAY IN TOTAL RANGE AMOUNTS TO ONLY ABOUT 100 N.MI. PER 1000 N.MI. OF SUBSONIC LEG. THIS LOW PENALTY MEANS THAT INABILITY TO CONTINUE A MISSION AT SUPERSONIC SPEEDS, E.G., AS A RESULT OF LOSS OF AN ENGINE, WOULD NOT PROHIBIT FLYING TO THE ORIGINAL DESTINATION EXCEPT IN A SMALL PERCENTAGE OF THE MAXIMUM RANGE MISSIONS WHERE THE DISTANCE TO THE DESIGNATED ALTERNATE FIELD IS TOO GREAT TO BE WITHIN THE TOTAL FUEL CAPACITY LIMIT INCLUDING LEGAL RESERVES AT THE ALTERNATE. IT IS OF INTEREST THAT IF THE WHOLE MISSION WERE FLOWN SUBSONICALLY THE RANGE WOULD ONLY DECAY 40 N.MI. ALTHOUGH IT WOULD BE A LONG, EXPENSIVE TRIP.

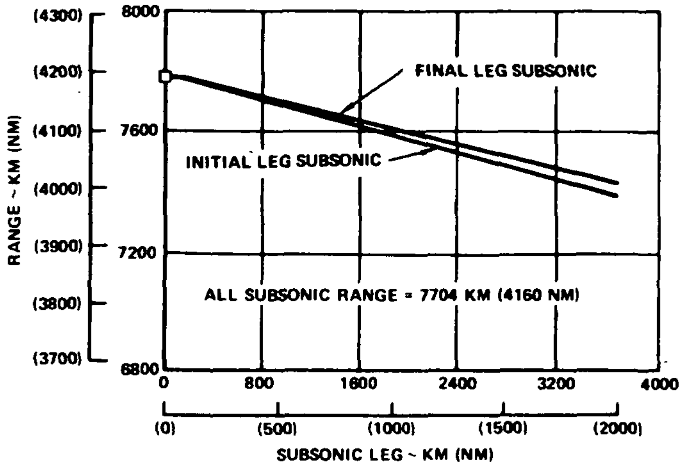

Figure 11. Effect of Subsonic Cruise Leg on Total Range (Point Design Aircraft)

FIGURE 12 REPRESENTS THE CHANGE IN RANGE AS PAYLOAD IS OFF-LOADED. THE INCREASE IS ABOUT 0.073 KM/KG (1.8 N.MI. PER 100 LB) OF PAYLOAD. OF INTEREST HERE IS THAT AS DESIGNED, THE POINT DESIGN VEHICLE IS FUEL VOLUME LIMITED AND NO ADDITIONAL FUEL CAN BE ADDED AS THE PAYLOAD IS REDUCED, IN CONTRAST TO THE CASE FOR MOST 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. OFF LOADING THE ENTIRE 22,226 KG (49,000 LB) OF PAYLOAD WOULD RESULT IN A FERRY RANGE FOR THE SUBJECT LH_2 AST AIRPLANE OF APPROXIMATELY 9450 KM (5100 N.MI.).

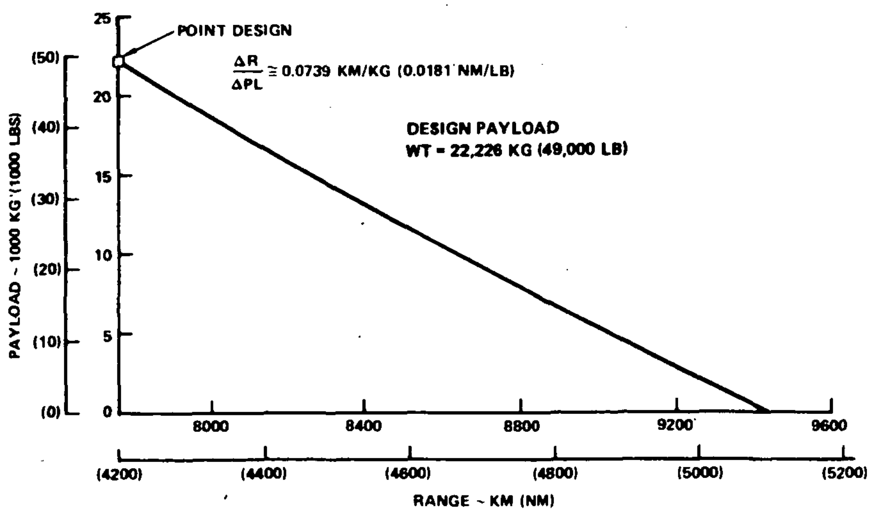

Figure 12. Range vs. Change in Payload

IN FIGURE 13, THE EFFECT OF A REDUCTION FROM THE SELECTED NOISE LIMITED TAKE-OFF DUCT BURNING TEMPERATURE OF 2160°R IS SHOWN FOR THE POINT DESIGN AIRPLANE. AS THE POWER IS REDUCED, THE TAKE-OFF DISTANCE INCREASES, MAXIMUM SIDELINE NOISE DECREASES AND FLYOVER NOISE INCREASES. THE INCREASE OF FLYOVER NOISE IS DUE TO THE LOWER FLIGHTPATH ANGLE WITH A SUBSEQUENT REDUCTION IN ALTITUDE AT THE 3.5 N.MI. MEASURING POINT.

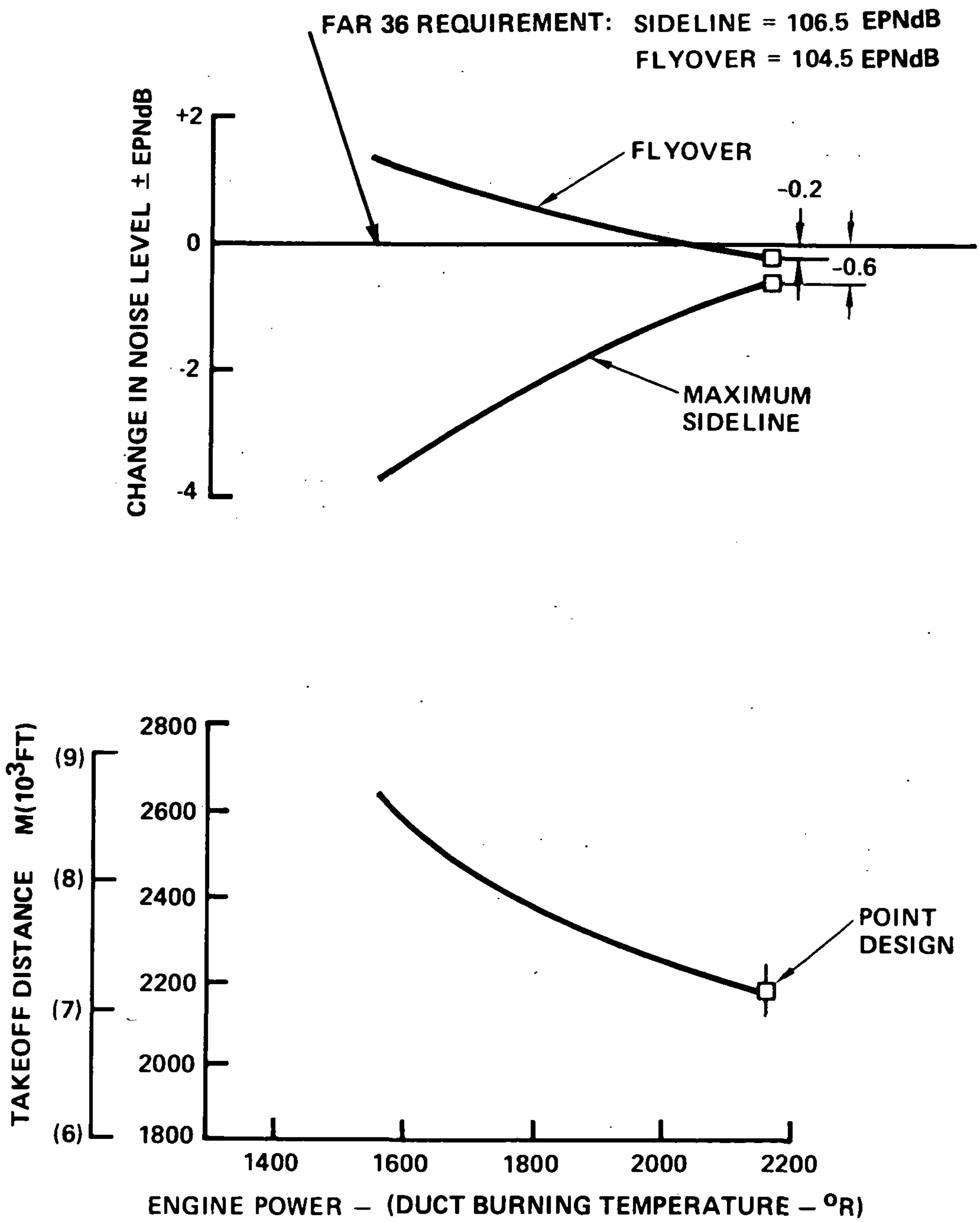

Figure 13. Takeoff Distance and Noise vs. Engine Power Setting

OTHER TRADEOFFS EVALUATED

TAKEOFF GROSS WEIGHT VS FAR 36 NOISE REQUIREMENT:

$$\frac{\Delta W_{TO}}{\Delta EPNdB} \cong 3000 \text{ LB/EPNdB}$$

THE TRADEOFF IN GROSS WEIGHT, AS A FUNCTION OF FAR PART 36 NOISE SPECIFICATION, IS EFFECTIVE FOR ONLY A VERY SMALL RANGE, I.E., FOR ONLY 1 OR 2 dB CHANGE AROUND THE DESIGN POINT. THE VALUE SHOWN ILLUSTRATES THAT APPROXIMATELY 3,000 LB OF VEHICLE GROSS WEIGHT NEEDS TO BE ADDED IN ORDER TO DESIGN THE AIRCRAFT TO MEET A NOISE SPECIFICATION WHICH IS MORE STRINGENT BY ONLY 1 dB. THE CHANGE WOULD RESULT FROM A NEED TO DECREASE WING LOADING AND INCREASE THRUST LOADING IN ORDER TO TAKEOFF WITH THE ENGINES MORE DEEPLY THROTTLED.

RANGE VS DRAG:

$$\frac{\Delta R}{\Delta \text{ DRAG COUNT}} \cong 54.1 \text{ M MI/COUNT}$$

THE EFFECT OF INCREASED DRAG ON THE VEHICLE RANGE IS ILLUSTRATED TO COST 54.1 NM IN RANGE PER ADDITIONAL DRAG COUNT, THEREBY MAKING IT A TRADEOFF EQUALLY IMPORTANT TO THAT PREVIOUSLY SHOWN FOR THE CHANGE IN SPECIFIC FUEL CONSUMPTION. THE CHANGE IN DRAG COUNT MAY, FOR EXAMPLE, RESULT FROM ADDITION OF AN EXTERNAL ANTENNA. FOR REFERENCE, THE NOMINAL DRAG OF THE SUBJECT POINT DESIGN AIRCRAFT IN CRUISE IS 122.5 COUNTS.

TAKEOFF GROSS WEIGHT VS FAR LANDING DISTANCE:

$$\frac{\Delta TOGW}{\Delta \text{ LDG DIST.}} \cong 1900 \text{ LB/100 FT}$$

TAKEOFF GROSS WEIGHT IS QUITE SENSITIVE TO FAR LANDING DISTANCE REQUIREMENTS, AS ILLUSTRATED BY THE 1900 LB PER 100 FT CHANGE IN LANDING DISTANCE. THIS CHANGE ALSO WOULD REQUIRE ADJUSTMENT IN VEHICLE WING LOADING TO MEET A CHANGE IN THE LANDING DISTANCE SPECIFICATION.

3.2.4 Cost

Cost of the point design LH_2 AST was calculated on the following bases:

- Number of aircraft = 300 and 600
- Fare = \$9 + .0496 x Range in Statute miles
- Passenger load factor = 0.55
- Aircraft utilization = 3600 hrs/year
- Fuel Cost: LH_2 = 22¢/kg (10¢/lb)

Table 8 is a summary of the significant factors. The effects of various sensitivities and tradeoffs on costs of the aircraft are shown in Figures 14 through 19.

TABLE 8

COST SUMMARY: POINT DESIGN LH_2 AST
(Refer to Table 6 for vehicle data)

COST ELEMENTS		Number of Aircraft 300	600
RDT&E	$\$10^6$		
Engine		659	659
Airframe		2,661	2,661
Total		3,320	3,320
Production Aircraft, each	$\$10^6$	47.96	40.89
Return on Investment (ROI) (after taxes)	Percent	6.04	10.93
Direct Operating Cost (DOC)	¢/seat km (¢/SM)		
Flight Crew		0.061 (0.098)	0.061 (0.098)
Fuel and Oil		0.457 (0.735)	0.457 (0.735)
Insurance		0.085 (0.137)	0.067 (0.108)
Depreciation		0.274 (0.441)	0.216 (0.348)
Maintenance		0.234 (0.376)	0.211 (0.340)
Total		1.111 (1.787)	1.012 (1.629)
Indirect Operating Cost (IOC)	¢/seat km (¢/SM)	0.498 (0.801)	0.497 (0.799)

FIGURES 14 AND 15 SHOW THAT ROI IS EXTREMELY SENSITIVE TO BOTH STAGE LENGTH AND FARE LEVEL. AS PREVIOUSLY NOTED, FARE WAS CALCULATED ON THE BASIS OF $9 + 0.0496R, WHERE RANGE WAS MEASURED IN STATUTE MILES. AS STAGE LENGTH IS DECREASED AND FARE IS DECREASED, ROI IS SQUEEZED ACCORDINGLY. FOR EXAMPLE, VARYING STAGE LENGTH BETWEEN THE NOMINAL DESIGN DISTANCE OF 7778 KM (4200 N MILE) DOWN TO ABOUT 4074 KM (2200 N MILE), THE DISTANCE FROM L.A. TO HONOLULU, INCREASES DOC FROM 2¢ PER SEAT N MILE TO APPROXIMATELY 2.5¢ PER SEAT N MILE. ROI ON THE OTHER HAND TAKES A PLUNGE FROM THE NOMINAL DESIGN POINT OF 6% DOWN TO A NEGATIVE 2-1/4% AS STAGE LENGTH IS REDUCED OVER THE SAME LIMITS. FIGURE 15 SHOWS THAT THIS CONDITION CAN BE CORRECTED BY FAIRLY SMALL MODIFICATION OF FARE LEVEL.

THE DIAGONAL CURVE ON THE RIGHT HAND SIDE OF FIGURE 15 SHOWS THE RETURN ON INVESTMENT USING A FARE ($249) CALCULATED FOR THE DESIGN STAGE LENGTH OF 7778 KM (4200 N MI). ROI IS SHOWN TO BE 6 PERCENT. THE DIAGONAL LINE ON THE LEFT REPRESENTS THE 4074 KM (2200 N MI) STAGE LENGTH. THE EXTRAPOLATION FROM FIGURE 14 SHOWED THAT A NEGATIVE ROI OF APPROXIMATELY 2% WOULD BE REALIZED IF THE FARE ($135) CALCULATED BY THE ABOVE FORMULA WERE CHARGED FOR THE REDUCED STAGE LENGTH. THE PRESENT FIGURE SHOWS THAT IF THE FARE WAS INCREASED TO $166, THE ROI WOULD BE RETURNED TO THE NOMINAL 6% VALUE. THIS $166 FARE FOR THE TRIP FROM LAX TO HONOLULU WOULD BE 27% ABOVE THE CURRENT (SEPT 1973) COACH FARE AND 13% BELOW THE CURRENT FIRST CLASS FARE.

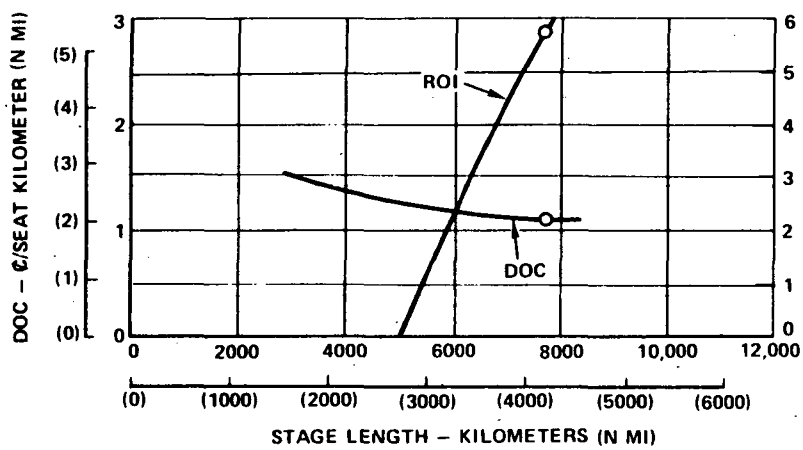

Figure 14. DOC/ROI vs. Stage Length

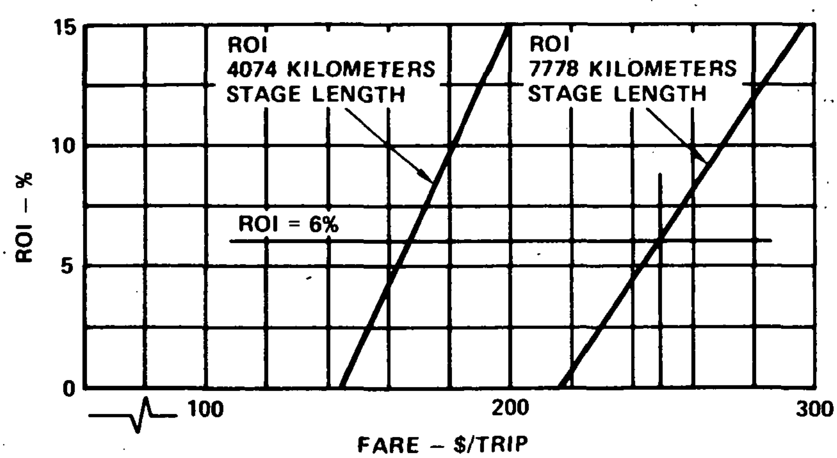

Figure 15. ROI vs. Fare Level

FIGURE 16 SHOWS THE RELATIONSHIP BETWEEN THE PRODUCTION QUANTITY AND THE PRODUCTION COST OF THE VEHICLE. THE RESULTS ARE DEPENDENT UPON THE SLOPES OF THE LEARNING CURVES CHOSEN FOR THE LABOR, MATERIAL, ENGINE AND AVIONICS. FOR EXAMPLE, A SLOPE OF 0.8 WAS USED FOR LABOR. THE DOC AND ROI ARE SENSITIVE TO THE COST OF THE VEHICLE AS THE VEHICLE COST CONTRIBUTES APPROXIMATELY 32 PERCENT TO THE DOC IN TERMS OF INSURANCE AND DEPRECIATION.

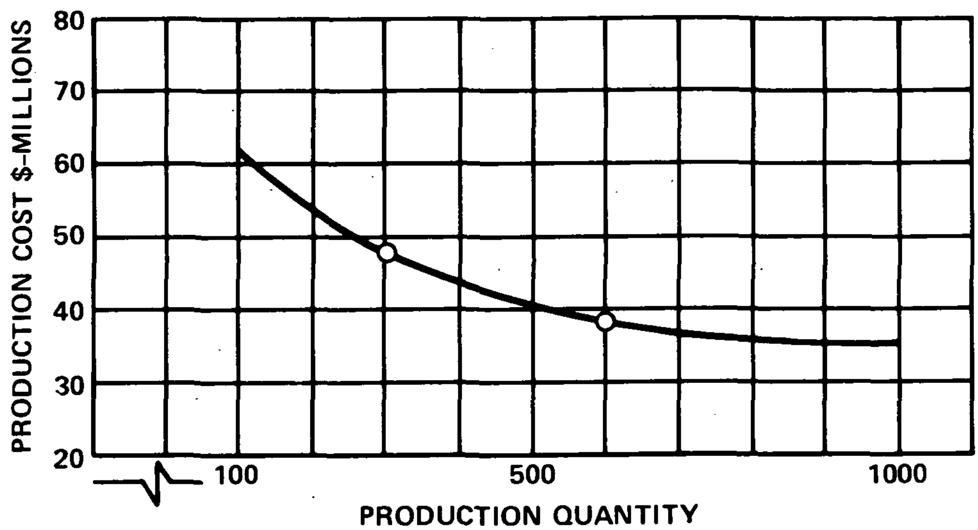

Figure 16. Production Cost vs. Production Quantity

AS SHOWN ON FIGURE 17, THE DOC AND ROI ARE VERY SENSITIVE TO FUEL COST. THIS IS DUE TO THE FACT THAT FOR THE AST THE FUEL IS A LARGE PERCENTAGE (41 PERCENT) OF THE TOTAL DOC. A 50 PERCENT INCREASE IN THE FUEL COST CAUSES A DROP IN ROI FROM 6 PERCENT TO ZERO IF NO ADJUSTMENT IS MADE IN THE FARE.

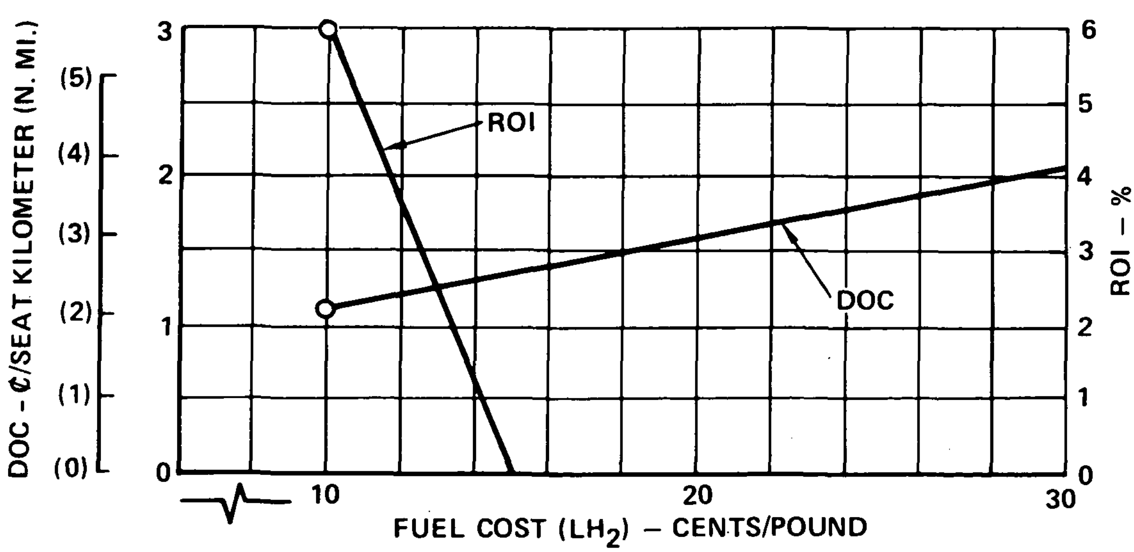

Figure 17. DOC/ROI vs. Fuel Cost

THE SENSITIVITY OF DOC AND ROI TO UTILIZATION IS SHOWN IN FIGURE 18. THE UTILIZATION IS VARIED FROM THE BASE POINT OF 3,600 BLOCK HOURS TO A LOW OF 3,000 HOURS AND A HIGH OF 4,000 HOURS. IN ALL CASES THE PASSENGER DEMAND REMAINS CONSTANT. THE CHANGE IN DOC IS NOT DRAMATIC BUT THE CASCADING EFFECT ON ROI IS SIGNIFICANT.

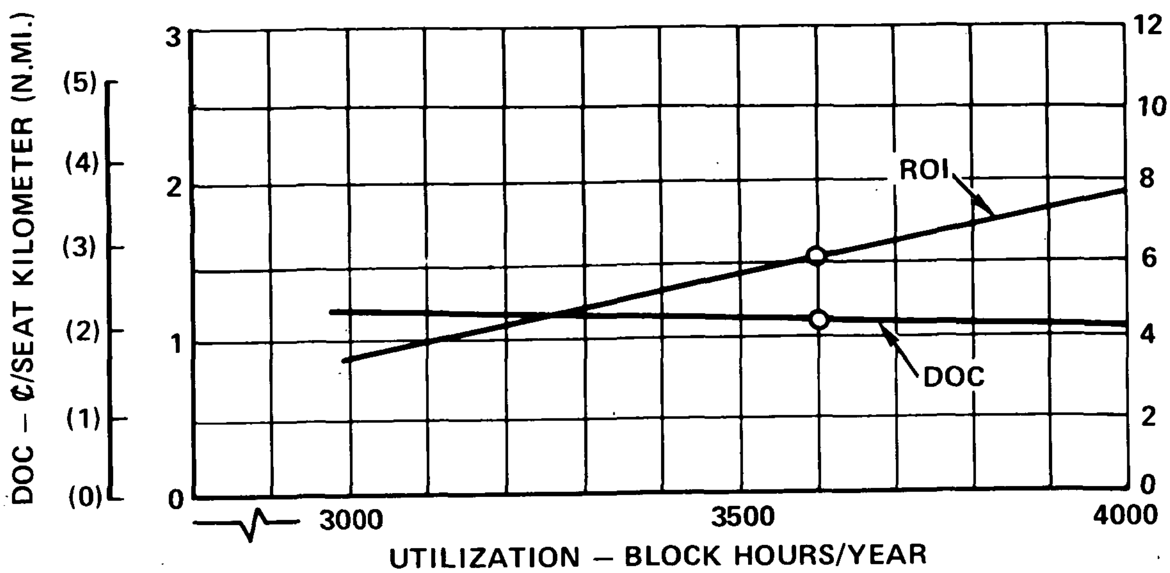

Figure 18. DOC/ROI vs. Utilization

THE LOAD FACTOR VARIATION HAS NO EFFECT ON DOC BUT DOES ALTER THE IOC AND THE NUMBER OF VEHICLES IN THE FLEET. THE CHANGE IN THE IOC IS CAUSED BY THE CHANGE IN THE NUMBER OF PASSENGERS HANDLED AT EACH FLIGHT AND THE NUMBER OF VEHICLES REQUIRED IS CHANGED BECAUSE THE PRODUCTIVITY OF THE AIRPLANE IS CHANGED WHILE THE TOTAL PASSENGER DEMAND REMAINS CONSTANT. THE CHANGE IN ROI WITH A CHANGE IN LOAD FACTOR RANGING FROM 0.50 TO 0.60 IS SHOWN IN FIGURE 19.

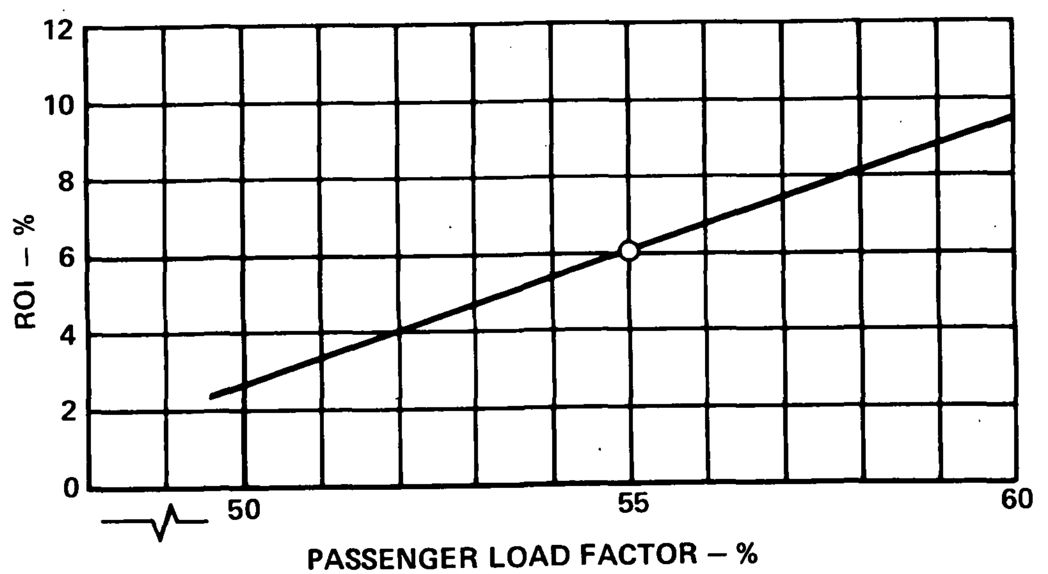

Figure 19. ROI vs. Load Factor

3.2.5 Environmental Summary

Engine Noise The duct-burning turbofan engines incorporate design features and suppression materials to quiet their operation at takeoff in accordance with latest design trends, anticipating some margin of improvement over current capability. In addition, the engines are throttled during takeoff to further quiet their operation in order to meet FAR Part 36 requirements. Fan duct temperature is restricted to 2160°R during takeoff and climb to 5000 ft. As a result, both sideline and flyover noise specifications are met. Maximum noise levels calculated are as follows:

	LH_2 AST	FAR Part 36 Spec.
Sideline 0.648 km (0.35 n.mi.)	105.9 EPNdB	106.6 EPNdB
Flyover 6.48 km (3.5 n.mi.)	104.3 EPNdB	104.5 EPNdB

Sonic Boom The maximum sonic boom overpressure for the LH_2 AST point design aircraft was found to occur during climbout. At M1.4 and an altitude of 12,802 m (42,000 ft), an overpressure of 99.6 n/m^2 (2.08 psf) was calculated. At the start of the cruise leg the value was calculated to be 1.31 psf. By the end of cruise the aircraft was lighter and higher and the sonic boom overpressure was reduced to 1.17 psf.

Emissions Exhaust emissions from the hydrogen fueled duct-burning turbofan engines will consist of oxides of nitrogen (NO_x) and water vapor H_2O). The amount of NO_x emitted will be a function of the burner design. There is theoretical feasibility for virtually NO_x-free operation if full advantage of the following characteristics of gaseous hydrogen is taken in design of the engine combustor:

- o very wide flammability limits
- o very high diffusivity

NO_x does not form in significant quantities if combustion temperatures are limited to less than 3600°R and if residence times are short. It has been demonstrated that both of these conditions can be met as a result of the characteristic smooth, rapid mixing and combustion of gaseous hydrogen in air.

Water vapor is the principal product of combustion of hydrogen. It is expected a hydrogen fueled AST will produce not quite twice the quantity of H_2O emitted by a conventionally fueled (Jet A-1) supersonic transport. During cruise the subject point design airplane uses 3.19 kilograms (7.03 lbs) of LH_2 per second. Assuming 100 percent combustion efficiency, this generates 28.5 kilograms (62.8 lbs) of H_2O per second. By contrast, an equivalent design of Jet A-1 fueled AST will use 11.6 kilograms (25.6 lbs) of fuel per second and, again assuming 100 percent combustion efficiency, will generate 15.06 kilograms (33.2 lbs) of H_2O per second.

4.0 CONCLUSIONS

4.1 Comparison With Equivalent JP Vehicle

One of the overriding objectives of the Phase II effort was to provide a design of LH_2 fueled AST which could be compared directly with a JP fueled version. The payload and original ground rules of the subject study were accordingly modified to provide a comparable basis for design with the JP fueled AST being developed under contract NAS1-12288 (Reference 5). Table 9 presents a comparison of a number of relevant factors for aircraft designed to use each of the fuels. Both aircraft are 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 5 as that which is presumed to be available for start of hardware development in 1981.

Table 10 lists some pertinent cost data for comparison of the two types of aircraft. Direct operating cost (DOC) is strongly influenced by the cost of the fuel. Figure 20 presents a plot of DOC for each type of aircraft as a function of the cost of its fuel.

It is significant to note that in September 1973, Jet A-1 sold for approximately \$31.70 \$/m^3 (12¢/gal., 1.78¢/lb., or 97¢ per 10^6 BTU). By early January 1974, the price had risen to 60.76\$/m^3 (23¢/gal., 3.42¢/lb., or \$1.86 per 10^6 BTU). The cost of LH_2 produced in large quantities from coal, is variously quoted at prices from \$2368 to \$4735 per 10^6 J (\$2.50 to \$5.00 per 10^6 BTU) (12.9 to 25.8¢/lb) delivered to the airport. The data of Figure 20 shows a hydrogen fueled AST can be competitive on the basis of DOC when LH_2 costs approximately \$1.50 per million BTU more than Jet A-1. In other words, when Jet A-1 costs \$1894 per 10^6 J (\$2.00 per 10^6 BTU) (3.68¢/lb.), airline operators could afford to pay \$3315 per 10^6 J (\$3.50 per 10^6 BTU) (18.5¢/lb.) for LH_2. It is significant that this comparison, favorable as it is to the hydrogen aircraft, does not include consideration of the lower maintenance requirements and the longer life anticipated for components of engines fueled with liquid hydrogen. These additional benefits should be evaluated and included in subsequent analyses.

TABLE 9

COMPARISON OF JET A-1 AND LH_2 FUELED SUPERSONIC TRANSPORTS OF ADVANCED DESIGN

Fuel			JET A-1		LH_2	
Payload	(lb)	kg.	(49,000)	22,226	(49,000)	22,226
Range	(n.mi.)	km.	(4,200)	7,778	(4,200)	7,778
Cruise Speed	Mach		2.7		2.7	
Takeoff Gross Weight	(lb)	kg.	(750,000)	340,194	(368,000)	166,922
Operating Empty Weight	(lb)	kg.	(309,700)	140,478	(223,100)	101,196
Fuel Weight, Mission	(lb)	kg.	(326,000)	147,871	(81,440)	36,941
Total	(lb)	kg.	(391,300)	177,491	(95,900)	43,500
Fuel Volume	(ft^3)	m^3	(8,290)	234.7	(21,700)	614.5
Wing Area	(ft^2)	m^2	(10,822)	1005.4	(6,880)	639.2
Wing Loading (W/S) Takeoff	(lb/ft^2)	kg/m^2	(69.3)	338.4	(53.5)	261.2
Landing	(lb/ft^2)	kg/m^2	(39.1)	190.9	(41.7)	203.6
Span	(ft)	m	(132.5)	40.39	(105.6)	32.19
Overall Length	(ft)	m	(297)	90.5	(328)	100.0
Lift/Drag (cruise)			8.5		6.99	
Specific Fuel Consumption (cruise)	((lb/hr)/lb)	kg/hr/daN	(1.51)	1.54	(0.561)	.572
Thrust/Weight (SLS)			0.477		0.50	
Thrust Per Engine	(lb)	kg.	(89,500)	40,597	(46,000)	20,865
Weight Fractions	Percent					
Fuel			52.2		26.1	
Payload			6.5		13.3	
Structure			25.3		33.5	
Propulsion			10.0		15.8	
Equipment and Operating Items			6.0		11.3	
Energy/Seat. Mi.	(BTU/ seat n.mi)	joule/ seat m	(6102)	3,479	(4274)	2,437

TABLE 10

COST COMPARISON: JET A-1 VS. LH_2 AST's
(Refer to Table 9 for vehicle data)

Cost Elements*		Aircraft	
		Jet A-1	LH_2
RDT&E	$\$10^6$		
Engine		950	659
Airframe		3,327	2,661
Total		4,277	3,320
Production Aircraft, each	$	67,328,000	47,967,000
Return On Investment (ROI) (After taxes)	Percent	2.24	6.04
Direct Operating Cost (DOC)	¢/SM		
Flight Crew		0.088	0.098
Fuel and Oil		0.568	0.735
Insurance		0.181	0.137
Depreciation		0.583	0.441
Maintenance		0.468	0.376
Total		1.888	1.787
Indirect Operating Cost (IOC)	¢/SM	0.888	0.801

* Basis for Costs:

- production of 300 aircraft
- Fare = \$9 + .0496 x Range (statute miles)
- passenger load factor = 0.55
- aircraft utilization = 3600 hrs/year
- fuel cost: Jet A-1 = 1.97¢/lb
  LH_2 = 10¢/lb

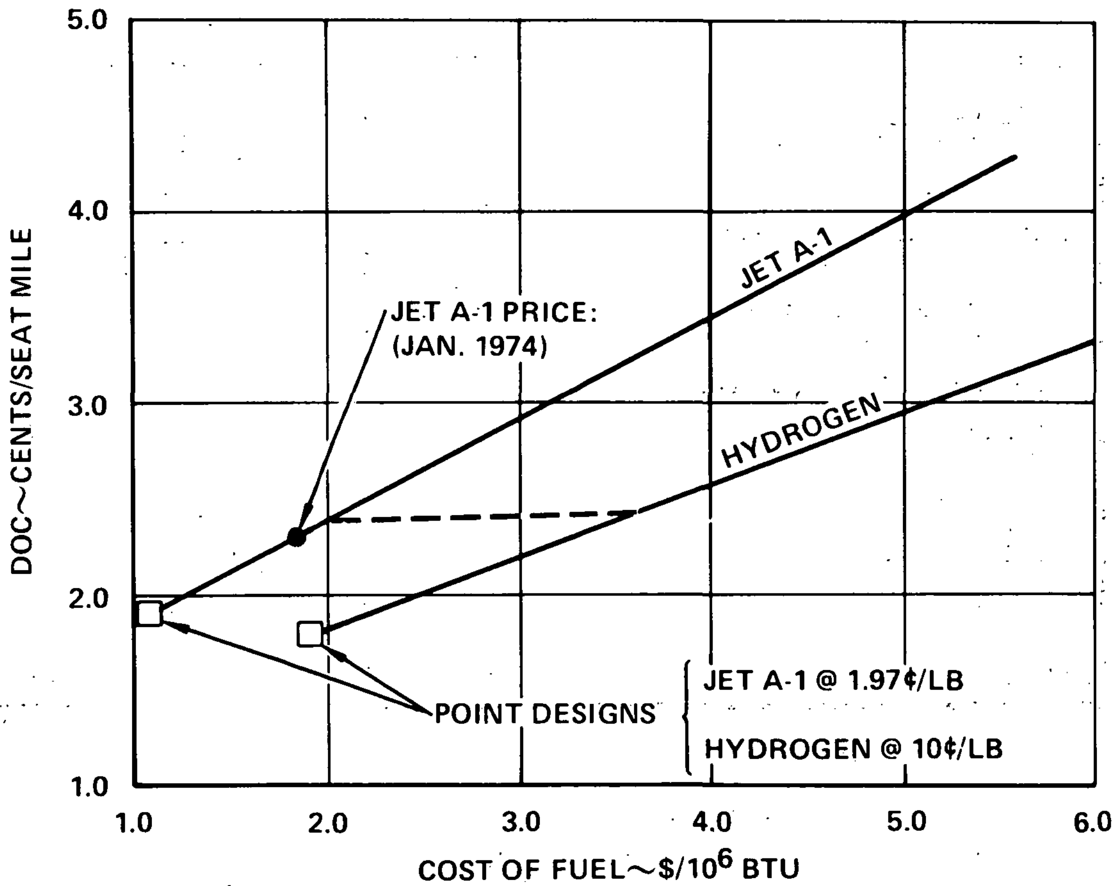

Figure 20. DOC vs Cost of Fuel

Another factor of particular interest to compare the relative desirability of the two aircraft is energy expended per available seat mile. The Jet A-1 AST uses 43 percent more BTU/available seat mile than does the LH_2 AST; 6102 BTU vs. 4274 BTU/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.

4.2 Major Technology Development Required

Technology development required to permit start of development of LH_2 fueled AST aircraft can be considered in two categories: minimum and desirable. The minimum category consists of those items which are necessary to accommodate the requirements of designing, fabricating, operating, handling, and maintaining aircraft of the subject design with its cryogenic fuel in a safe, economical manner; the desirable category includes additional items which can be seen will lead to further significant improvement in the operation or cost of LH_2 fueled AST aircraft. Table 11 presents the items of technology development required for both categories.

Table 11
Major Technology Development Required
For LH_2 Fueled AST Aircraft

Minimum (necessary for the point design aircraft)

- Duct-burning turbofan engines designed to operate efficiently on hydrogen fuel.
- Lightweight cryogenic insulation, e.g., PVC or polyurethane foam, which is impervious to air, which can be bonded to an aluminum tank and can demonstrate an acceptable effective useful life.
- Lightweight high temperature insulation, impervious to air, satisfactory for exposure to temperatures from 0°F to +400°F.
- Lightweight heat shield structural material having low thermal conductivity, e.g., fiberglass core, graphite/polyimide faced honeycomb sandwich, which is satisfactory for airline service.

- Continued -

Table 11 (Continued)

- Lightweight aluminum tankage, capable of withstanding airline service, plus exposure to cryogenic temperatures and attendant thermal stresses.
- A satisfactory vent system for the LH_2 fueled aircraft.
- An aircraft fuel feed system including pumps, valves, quantity sensors, heat exchanger, pressurization system and control, and vacuum-jacketed lines acceptable for airline service.
- A ground supply and fuel handling system for use at airline terminals.
- An acceptable specification and set of standards for handling liquid hydrogen in routine airline operation.
- A flight demonstration program involving conversion of existing aircraft to LH_2 fuel and use in simulated airline operations.

<u>Desirable</u> (improvements for additional advantage)

- Cryogenic insulation material which is impervious to gaseous or liquid hydrogen and can be used inside the aircraft fuel tanks. Alternatively, development of a barrier film which can be applied over a cryogenic insulation to prevent permeation by gaseous hydrogen into the insulation.
- Composite materials satisfactory for use as structure for an integral cryogenic tank.
- Heat shield material and design which serves efficiently as a high temperature insulation for application over the integral tank structure.

5.0 RECOMMENDATIONS

Use of liquid hydrogen for fuel in a supersonic transport of advanced design looks very attractive. Particularly in view of the dramatic developments of the past two months relative to the cost and availability of Jet A-1 fuel, the advantages to the air transport industry of using a synthetic fuel which is completely independent of the supply of petroleum are apparent.

Aside from the economic and availability considerations, the LH_2 fueled AST offers advantages in other areas. Energy expended during the flight, per available seat mile, is significantly lower than for a Jet A-1 aircraft of comparable design. Environmental pollution is drastically reduced. Noise is lower and sonic boom overpressures are lower.

It is recommended that development of technology for LH_2 fueled supersonic transport aircraft be pursued. The following actions are recommended to further explore the potential of such aircraft and to establish technology feasibility:

- study alternate configuration concepts which appear to offer advantage, e.g., the wide-body version discussed at the Mid Term Oral Review.
- perform additional studies of the point design aircraft to establish better definition of the design.
- build and test insulated model tanks to determine their capability for withstanding thermal cycling under simulated structural loading conditions.
- investigate thermal protection system concepts.
- study aircraft ground handling and refueling operations to establish specifications for equipment and procedures to assure safe, economical practices.

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