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TECHNOLOGY REQUIREMENTS FOR ADVANCED EARTH-ORBITAL TRANSPORTATION SYSTEMS

Final Report

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16. Abstract Areas of advanced technology that are either critical or offer significant benefits to the development of future Earth-orbit transportation systems were identified. Technology assessment was based on the application of these technologies to fully reusable, single-stage-to-orbit (SSTO) vehicle concepts with horizontal landing capability. Study guidelines included mission requirements similar to Space Shuttle, an operational capability beginning in 1995, and main propulsion to be advanced hydrogen-fueled rocket engines. Also evaluated was the technical and economic feasibility of this class of SSTO concepts and the comparative features of three operational take-off modes, which were vertical boost, horizontal sled launch, and horizontal take-off with subsequent inflight fueling.

Projections of both normal and accelerated technology growth were made. Figures of merit were derived to provide relative rankings of technology areas. The influence of selected accelerated areas on vehicle design and program costs was analyzed by developing near-optimum point designs.

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PREFACE

This study was performed by Martin Marietta Corporation, Denver Division, under NASA Contract NAS1-13916. Three reports describe the study and results, as follows:

"Technology Requirements for Advanced Earth-Orbital Transportation Systems"

- Summary Report

- Final Report

- Dual-Mode Propulsion

The authors wish to acknowledge the substantial contributions of engineering personnel at NASA Langley Research Center and Lewis Research Center as well as many persons in the Martin Marietta Corporation, Denver Division.

Certain commercial materials are identified in this paper in order to specify adequately which materials were investigated in the research effort. In no case does such identification imply recommendation or endorsement of the product by NASA, nor does it imply that the materials are necessarily the only ones or the best ones available for the purpose. In many cases equivalent materials are available and would probably produce equivalent results.

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SUMMARY

Areas of advanced technology that are either critical or offer significant benefits to the development of future Earth-orbit transportation systems were identified. Technology assessment was based on the application of these technologies to fully reusable, single-stage-to-orbit (SSTO) vehicle concepts with horizontal landing capability. Study guidelines included mission requirements similar to Space Shuttle, an operational capability beginning in 1995, and main propulsion to be advanced hydrogen-fueled rocket engines. Also evaluated was the technical and economic feasibility of this class of SSTO concepts and the comparative features of three operational take-off modes, which were vertical boost, horizontal sled launch, and horizontal take-off with subsequent inflight fueling.

The four basic tasks making up this study were (1) a projection of "normal" technological growth in pertinent vehicle system areas, (2) design of vehicle systems and definition of their performance potential based on these nominal growth projects, (3) a perturbation of selected technology areas to define the impact of R&T funding support for accelerated technology programs, and an assessment of various technology parameters in terms of cost/performance/benefit figure of merit, and (4) sensitivity and trade studies to define the impact of these focused program on vehicle characteristics and mission performance, and an identification of critical and high-yield technology.

INTRODUCTION

Various space vehicle systems that offer the potential for substantial improvements in our future space transportation capabilities relative to the Space Shuttle-based transportation system are being studied by NASA. Improved capabilities emphasize cost reductions but may also include different mission requirements from Shuttle. Although the Space Shuttle provides greatly improved capabilities over current expendable launch vehicles and is a cost-effective solution for the projected missions in the 1980-1990 decade, the evolution of launch vehicles is far from being mature. Traffic growth, new technology, and changing mission requirements will eventually make it cost effective to supplement or to replace the Shuttle. One class of potential future systems is the single-stage-to-orbit (SSTO) with horizontal landing capability. SSTO concepts that have been investigated in recent years at Langley Research Center and are considered in this present study have the potential for low recurring cost also present a considerable challenge to many of the associated technologies.

For the purposes of this study, an SSTO was postulated to be the Space Shuttle replacement system beginning flight operations in 1995. (The Shuttle operational lifetime would be about 15 years.) Allowing for an SSTO vehicle development lead time of about eight years, the required technology readiness date is 1987. The ten years between now and 1987 would be available for development of the required technology base. Many technology areas will advance during that time period without special SSTO funding because of ongoing technology programs and transfer from similar areas such as Space Shuttle and aeronautical technology programs; however, in selected areas, it would be necessary or desirable to accelerate the normal technology growth. The identification and prioritization of such areas has been the central issue of this study.

The primary goal of this study has been to identify areas of technology associated with SSTO systems that are either critical to their development of offer significant cost and performance benefits. This was accomplished by assessing the impact of technology perturbations on the vehicle program life-cycle costs (LCC) relative to the research program costs. Secondary goals had to do with the evaluation of SSTO system characteristics, including (1) the definition of performance potential in terms of vehicle design characteristics and life cycle costs, and (2) a comparison of three operational modes. These study goals were met by performing the four major tasks described below.

Government and industrial publications were reviewed in Task 1 to generate historical and future projections of "normal" technology growth primarily in the structures, materials, and propulsion disciplines with secondary emphasis on flight controls, trajectory optimization, and aerodynamics. Funding projections based on recent NASA and DOD actual expenditures and forecasts were made to be used as an aid to predicting "normal" technology growth.

During Task 2, preliminary design were developed for three hydrogen-fueled SSTO vehicles: VTO (vertical takeoff), HTO (horizontal takeoff sled launched), and IFF (inflight fueled). Each was designed for a payload capability of 29 500 kilograms (65 000 pounds), as easterly launch from KSC, and a horizontal landing. Both conventional bell nozzle rocket engines and linear rocket engines were considered. Various thermostructural and propulsion system concepts were evaluated for the three designs. A primary figure of merit (FOM) for vehicle design was minimum dry weight based on use of "normal" technological growth. An economic comparison was made of the total program costs for each concept.

Selected technology areas were perturbed during Task 3 beyond the "normal" growth level to identify the greatest potential payoffs for an accelerated technology vehicle design during Task 4. Technology parameters were assessed in terms of cost/performance/benefit figures of merit relative to the Task 1 and Task 2 base. The results of normal growth and normal funding from the Task 1 evaluation were considered in developing the costs and gains for an accelerated technology vehicle design. The Task 2 VTO vehicle design was used to derive the sensitivity information used in the figure-of-merit (FOM) assessment in performing the assessment of the figures of merit. Performance sensitivities were derived for those technology programs with a high-yield potential.

All technologies offering a clear payoff on a cost/performance/benefit figure of merit were then included in Task 4 designs of near-optimal vehicle configurations. The cost effectiveness of the total system, which used the accelerated technological forecasts, was then evaluated.

Based on these studies of normal and accelerated technological forecasts, funding, vehicle design requirements, and figures of merit, assessments of high-yield and critical areas of technology were made. These provided a basis for recommendations of areas of technology that should be vigorously pursued to support cost-effective, advanced earth-orbital transportation systems.

This summary report presents highlights of the study results. Future studies are anticipated to consider other vehicle alternatives such as use of dual-mode propulsion and control-configured vehicle concepts.

SYMBOLS

b'	Wing structural span
C_D	Drag coefficient
$C_{n\beta}$	Directional stability derivative
CER	Cost estimating relation
DDT&E	Design, development, test and evaluation
FOM	Figure of merit
F_{vac}	Engine vacuum thrust
F/W	Thrust/weight ratio
GLOW	Gross liftoff weight
g	Acceleration of gravity
h	Altitude
HTO	Horizontal takeoff
IFF	Inflight fueled
I_{SP}	Specific impulse
Ix, Iy, Iz	Moments of inertia about x, y, and z axes, respectively
L/D	Lift/drag ratio
LF	Load factor
ℓ_{ref}	Reference length
M	Mach number
MR	GLOW/WBO; O/F mixture ratio
NPSH	Net positive suction head
n_x	Force in x-direction/weight
n_z	Force in z-direction/weight
P_A	Atmospheric pressure

P_C	Thrust chamber pressure
P_{xy} P_{xz} P_{yz}	Products of inertia about xy, xz, and yz axes, respectively
q	Dynamic pressure
R.B.	Rudder bias
S_E	Elevon area
S_{Ref}	Reference area
SSTO	Single-stage-to-orbit
S_{VT}	Vertical tail area
$S_{VT_{EXP}}$	Vertical tail exposed area
S_W	Wing theoretical area
T	Temperature
t	Thickness; time
TPS	Thermal protection system
t_R	Thickness of wing root
VTO	Vertical takeoff
W	Weight (mass)
WBO	Burnout weight
WBS	Work breakdown structure
WLOSS	Ascent weight losses
WP	Ascent propellant weight
WPL	Payload weight
W_E	Elevon weight
W_L	Landing weight

W_{VT}	Vertical tail weight
x, y, z	Vehicle coordinate axes
$x/\ell_{c.g.}$	Longitudinal center of gravity
α	Angle of attack
α_{TRIM}	Trim angle of attack
ΔW_{DRY}	Dry weight increment
$\Delta \$LCC$	Undiscounted life cycle cost increment
$\Delta \$LCC_D$	Discounted life cycle cost increment
$\Delta \$R$	Undiscounted research cost increment
$\Delta \$R_D$	Discounted research cost increment
δ_e	Elevon deflection
ϵ	Nozzle expansion ratio
θx, θy, θz	Angles measured from x, y, and z axes, respectively
Λ_{LE}	Wing leading edge sweep angle
Λ_{TE}	Wing trailing edge sweep angle
λ	Propellant mass fraction; wing taper ratio
$\Sigma \$R_D$	Summation of discounted research costs

"NORMAL" TECHNOLOGY AND FUNDING PROJECTIONS

The primary objective of Task 1 was to define a base level of technology that would probably exist at the time needed to support the assumed SSTO program schedule without special technology development funding. Improvements in the base level of technology were assumed to occur between now and the needed date due to (1) transfer of technology developments from related programs such as existing space programs (especially the Space Shuttle) and commercial and military aircraft programs and (2) focus of technology programs on SSTO-related areas within a historically based "normal" funding level.

Our approach to Task 1 has been to use historical data for applicable technologies that are related to current space programs and commercial and military aircraft programs. Future technology capabilities and R&T funding were projected by trend curves based on data from Congressional records, Government technology and budgetary documents, and industrial reports. Mission objectives and the overall program plan have been used as a basis for our timing of these projections. This is reflected in Figure 66 shown in the Program Cost Analysis section.

Primary emphasis has been on technological developments that have a strong impact on the vehicle weight and c.g. locations; i.e., materials, structures, and propulsion. A secondary emphasis was given to technology related to other vehicle subsystems including aerothermodynamics, performance optimization, aerodynamics, computer technology, control systems, and auxiliary power.

The funding projections were based on NASA and DOD funding using both "top-down" and "bottom-up" estimating procedures. Funding was considered applicable only when it related to development of technologies that would be used on an SSTO vehicle. Some of the assumptions applicable to the technology and funding projections are (1) space programs to proceed as currently planned, (2) sources of transferable technology, such as commercial and military aircraft programs, to proceed at current expected levels, (3) existing levels, focus, and trends of technology programs to continue as expected, and (4) no major disasters or wars occur during this time period.

RATIONALE AND SCOPE

The main requirement for a technology to be evaluated was that it is applicable for use on an SSTO vehicle. The technology should be applicable to the vehicle and the program objectives. Advancements in technology were assumed to be continually funded and focused to achieve program goals. All technological options were retained unless a valid reason for elimination was uncovered.

The initial screening was used to select all known technology candidates within the scope of the study guidelines. The screening included identifying all critical characteristics of the technological advancements. Considerations included the applicable ranges of operating environment and the potential for minimizing vehicle dry weight. Options with little promise were rejected in favor of those with better performance, applicability, reliability, reusability, maintainability, and manufacturing possibilities.

The second stage of the screening process was to collect historical characteristic data on the options that passed initial screening. Correlation factors were then developed using the important characteristic parameters that represent the technological status of the options. The historical data were plotted against years using these correlation factors. Other correlation parameters were then selected for further projection activities.

In the final screening process, expert opinions were received and evaluated on the relative values of technology parameters for the 1995 time frame for initial operating capability. The historical data on NASA and DOD R&T funding were projected to 1990 along with specified nominal, maximum, and minimum yearly averages. The projections of historical data parameters were based on previous trends, the expert opinions of technological growth possibilities and knowledge of the "normal" funding anticipated. The total results were then used to select nominal, maximum, and minimum values of characteristic parameters based on engineering judgement of the validity of the projections.

TECHNOLOGY PROJECTIONS

Technology projections are discussed in three primary technological categories: (1) materials and structures, (2) propulsion, and (3) secondary technology areas. The potential improvement in the materials, structures, and propulsion technologies are presented in detail because they have large effects on vehicle dry weight. Data results for the secondary technologies are summarized in this chapter, and presented in Appendix A.

Materials and Structures

Rationale for materials and structures technology projection.- Structural and thermal protection system materials were initially screened to identify significant effects of materials on vehicle dry weight. Structural metals such as aluminum, titanium, high strength steels, superalloys, and beryllium alloys including Lockalloy have been improved in the last 20 years in the area of reliability, but with relatively little increase in strength/density or modulus/density. This trend is expected to continue and future projections of metallic materials will show minor improvements relative to vehicle dry weight. Advanced composite materials for primary and secondary structures have experienced significant advancement in strength and density and modulus and density properties as well as refined analysis and production methods. Projections for these materials show significant improvements based on historical performance and expected funding levels. Surface insulator materials have been dramatically improved in the last 15 years. Projections indicate a continued increase in upper limit temperature and weight efficiency. Figure 1 illustrates the rationale for the selection of materials for "normal" technology projections.

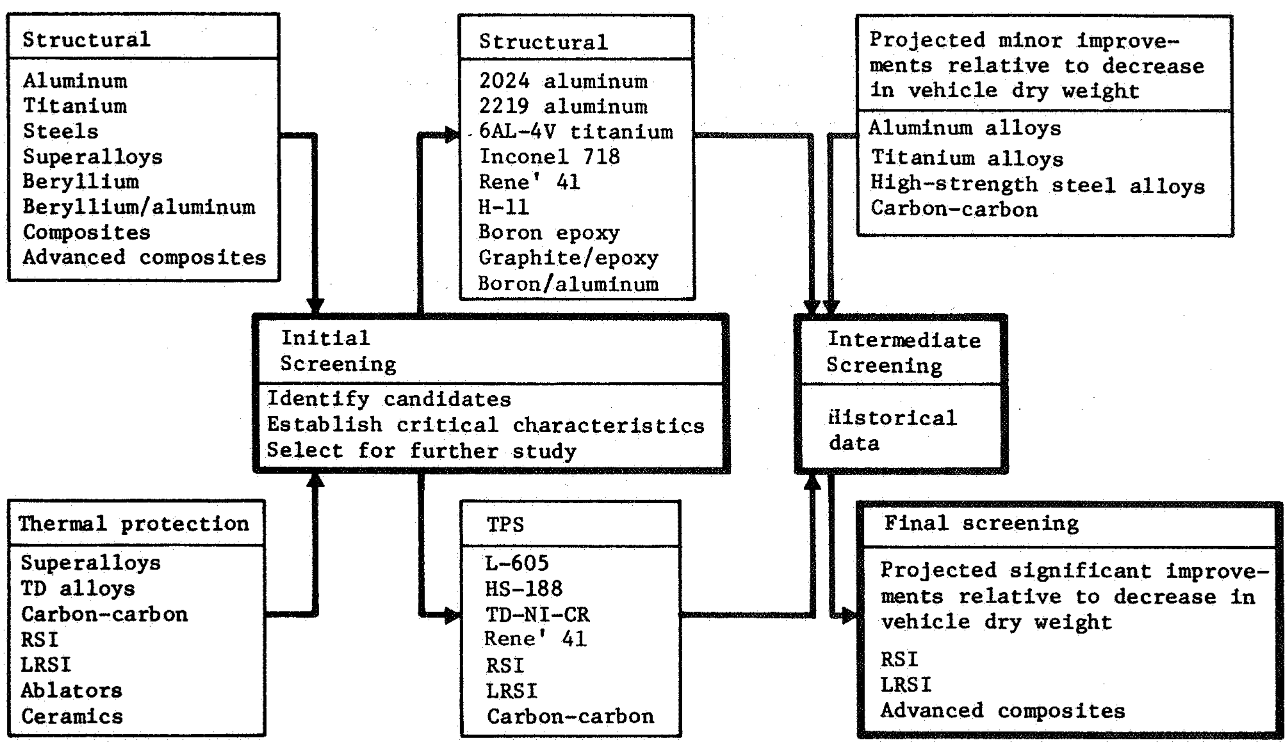


Figure 1.- Rationale for materials technology projection

The relative importance of the various structural and TPS components is shown on Figure 2. The combined weights of these subsystem components represent 60% of the SSTO vehicle dry weight. The components selected for the projections were the wing and elevon structure, the vertical tail structure, and the propellant tanks, the thrust structure, the landing gear, and the thermal protection system (TPS).

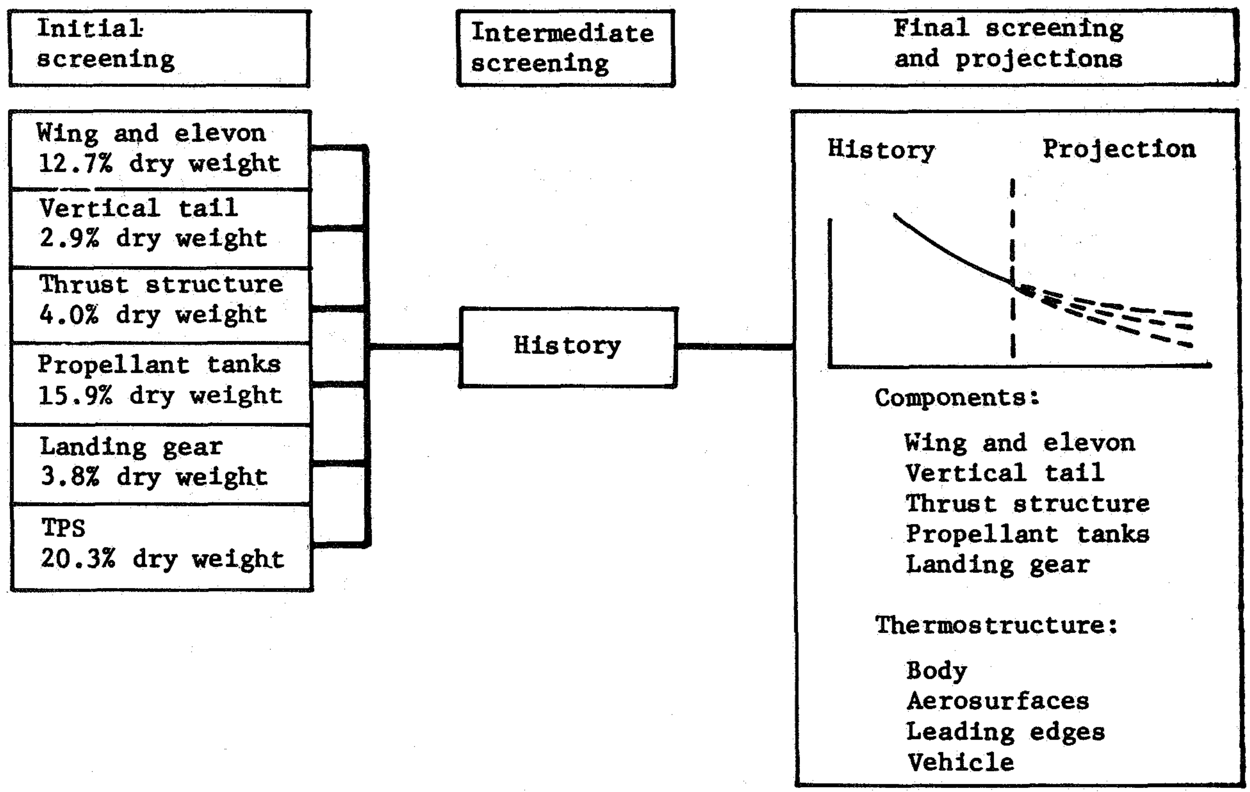


Figure 2.- Rationale for thermostructural technology projections

Normal technology advancement of structural and thermal protection system materials and structural components was based on the funding level of 1973 through 1975 R&T technology projected to the 1995 time period. The projected improvements in materials and structural components were based on consideration of "normal" goals to be achieved by research activities focused on SSTO applications. Historical data of materials and structural components, including the landing gear, were obtained from References 1 through 3 as well as unpublished industrial data. These included mass properties estimation methods (Martin Marietta), Space Shuttle external tank mass properties (Martin Marietta), C-5 airplane weights (Lockheed-Georgia), 747-airplane weights (Boeing), Phase B Space Shuttle reports (McDonnell Douglas Astronautics/Martin Marietta and Rockwell International), and Titan launch vehicle mass properties (Martin Marietta).

TPS materials.- Materials for external vehicle thermal protection systems have had dramatic improvements in the past 15 years, particularly in terms of lower density and thermal conductivity and increased reusability. Figure 3 illustrates insulation density history and the future projections for leading edge and surface areas. The leading edge density projections make use of higher temperature RSI materials for reuse in the 1645°K (2500°F) to 1867°K (2900°F) temperature range. This projection is based on RSI material, developed at NASA Ames Research Center, which has been tested to 1701°K (2600°F). The lower surface insulation is represented by families of ablators, glass phenolics, low density silicone ablators, and the RSI materials developed for the Space Shuttle. The future projections show a nominal density of 104 ± 8.0 kg/m^3 (6.5 ± 0.5 lb/ft^3). The upper surface RSI is the low temperature reusable insulators such as SLA-220 and Nomex felt. The projection for this material class is a nominal density of 72 ± 8.0 kg/m^3 (4.5 ± 0.5 lb/ft^3). The final selection of TPS densities versus temperature is listed in Table 1 where the lower surface insulation is indicated for two ranges of temperature.

TABLE 1.- TPS DENSITIES (NOMINAL PROJECTIONS TO 1987 TECHNOLOGY)

Temperature		Density	
°K	(°F)	kg/m^3	(lb/ft^3)
Up to 590	(Up to 600)	72	(4.5)
590 to 1367	(600 to 2000)	96	(6.0)
1367 to 1645	(2000 to 2500)	128	(8.0)
1645 to 1867	(2500 to 2900)	352	(22.0)

Structural materials.- Materials used for primary and secondary structures showing the greatest historical improvements and having the highest potential for future increases are the advanced composites. The historical data of advanced composites show dramatic step improvements in either strength or elastic modulus or in the case of the boron filaments both strength and modulus. Figures 4 and 5 show the data for filaments of glass, boron, graphite, and Kevlar. The maximum future projection of filament improvements is based on the "Outlook for Space" projections (ref. 4) and the minimum is based on engineering judgement of improved processing of present materials.

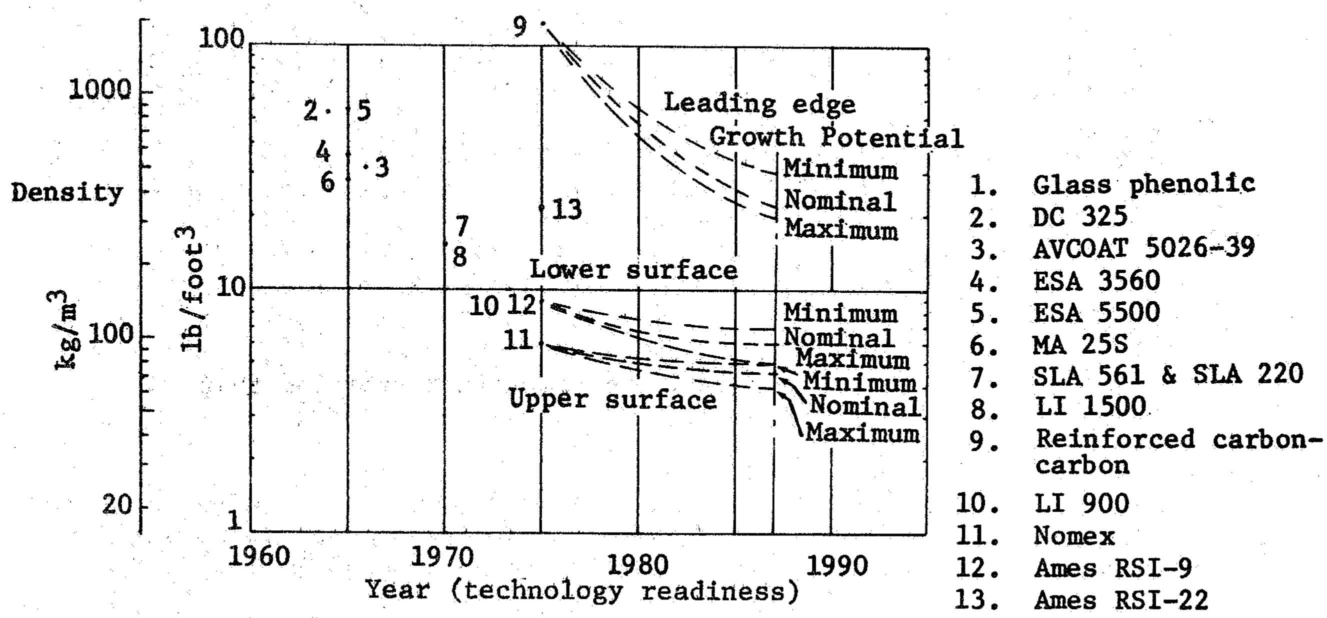


Figure 3.- Surface insulation history and projection

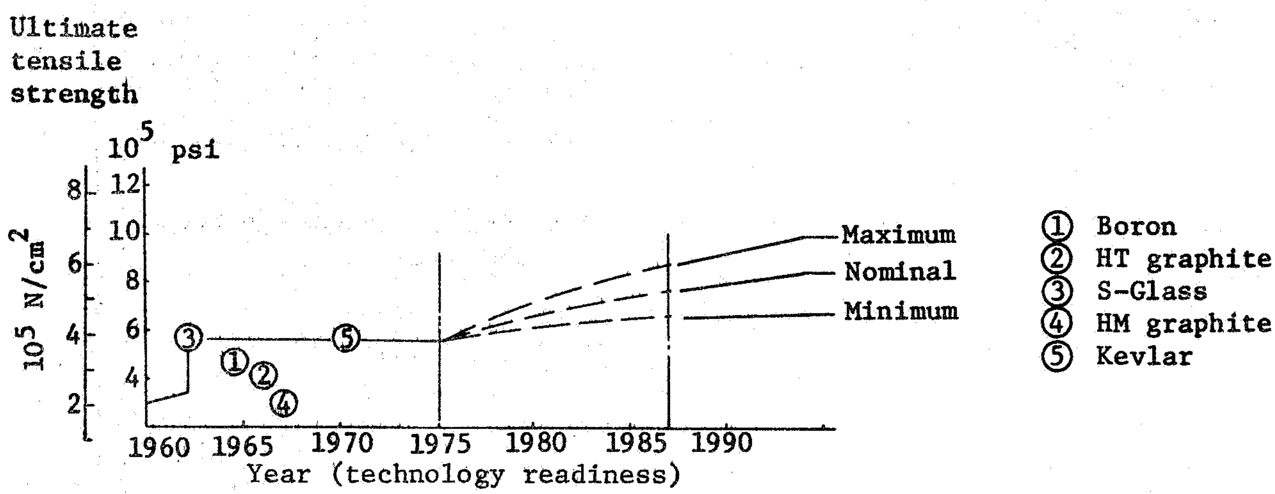


Figure 4.- Advanced composite fibers - ultimate strength history and projection

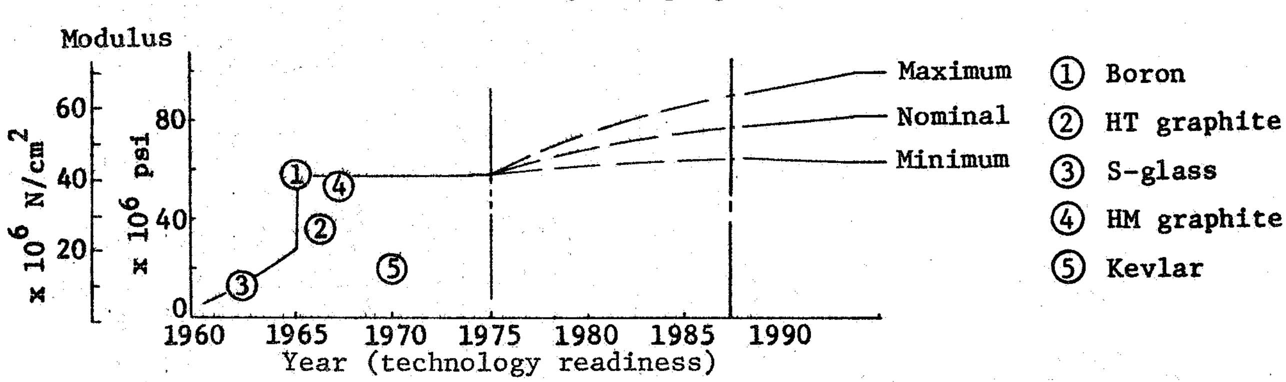


Figure 5.- Advanced composite fibers - modulus history and projection

Wing structure.- Wing geometry, loads, and weights were gathered to provide parametric weight data for estimating wing weights. Figure 6 shows wing data from 17 aircraft, the Space Shuttle orbiter, and two Shuttle Phase B booster vehicles. The wing weights are plotted as a function of a structural parameter α. The projection curves represent weight reductions that can be achieved by changing the present aluminum wing structure to one that uses advanced composite materials for both primary and secondary structures. The wing weight equation in Figure 6 was used for preliminary wing weights during subsequent vehicle sizing. Table 2 lists the aircraft and spacecraft vehicles that are used as data points in Figures 6 through 11.

TABLE 2.- AIRCRAFT AND SPACE VEHICLE HISTORICAL DATA POINTS

1.	B-36J	13.	F-106B	25.	Space Shuttle Phase B Booster, MDAC/MMC
2.	B-47B	14.	F-108	26.	Space Shuttle Phase B Booster, NAR/GDC
3.	B-52A	15.	F-101B	27.	Space Shuttle Phase B Orbiter, MDAC/MMC
4.	YB-60	16.	880	28.	Space Shuttle Phase B Orbiter, NAR
5.	C-135A	17.	990	29.	Space Shuttle Phase C&D Preposal, GAC/MMC
6.	B-58A	18.	C-141A	30.	Titan III Stage I
7.	F-105A	19.	F-111B	31.	Titan III Stage II
8.	F-104F	20.	C-5A	32.	Saturn SIVB
9.	C-133B	21.	747	33.	Saturn SII
10.	A3J-1	22.	F-4D	34.	Saturn S-IC
11.	XB-70A	23.	F-15	35.	Titan I
12.	F-102A	24.	Space Shuttle		

Elevon structure.- Elevon weight and geometry historical data are shown in Figure 7 for the B-58A, XB-70A, and the Space Shuttle orbiter. Studies of Space Shuttle Phase B and Phases C and D preproposal vehicle studies are included to give a better range of elevon area. The projections are based on use of advanced composite structure.

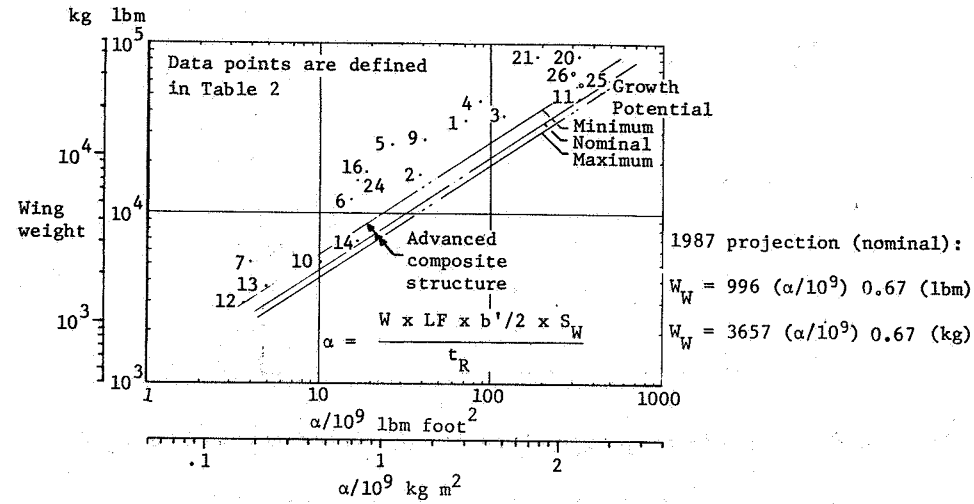


Figure 6.- Wing structure weight history and projection

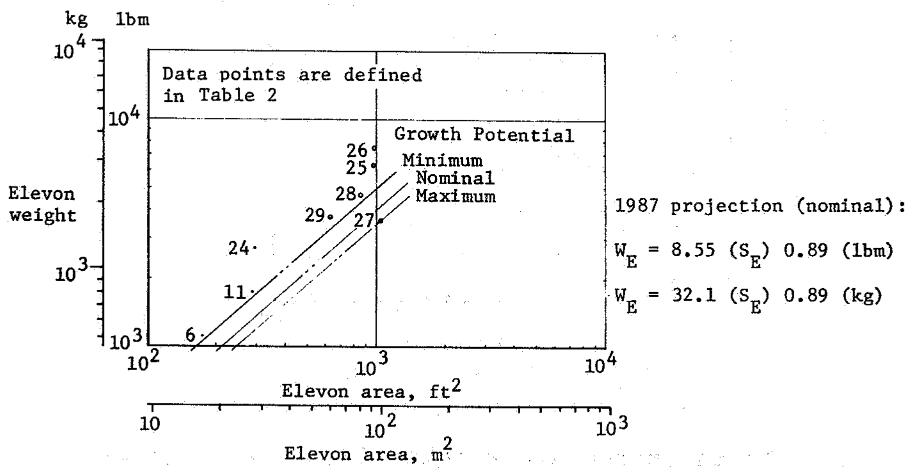


Figure 7.- Elevon structure weight history and projection

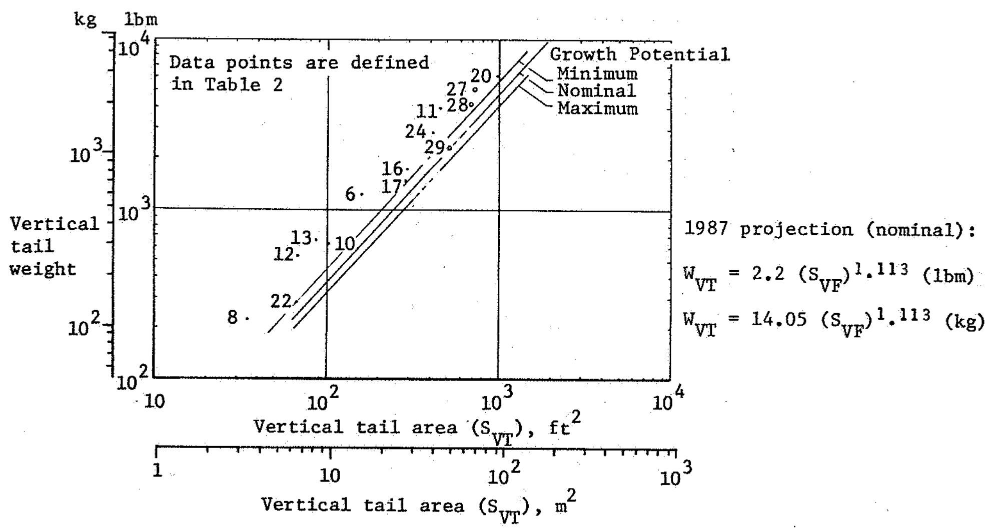


Figure 8.- Vertical tail structure history and projection

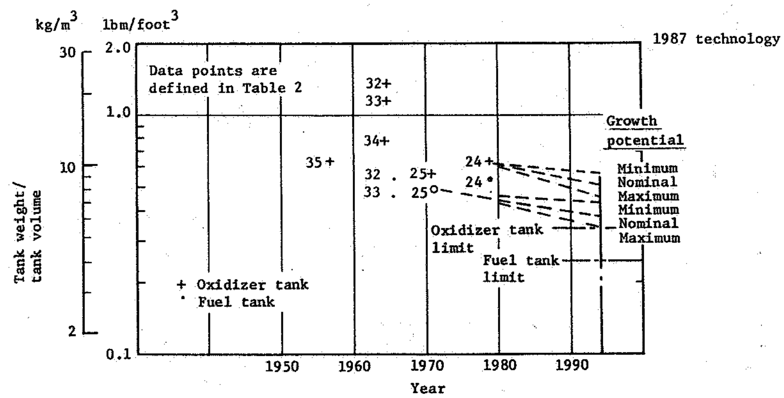


Figure 9.- Integral propellant tanks history and projection

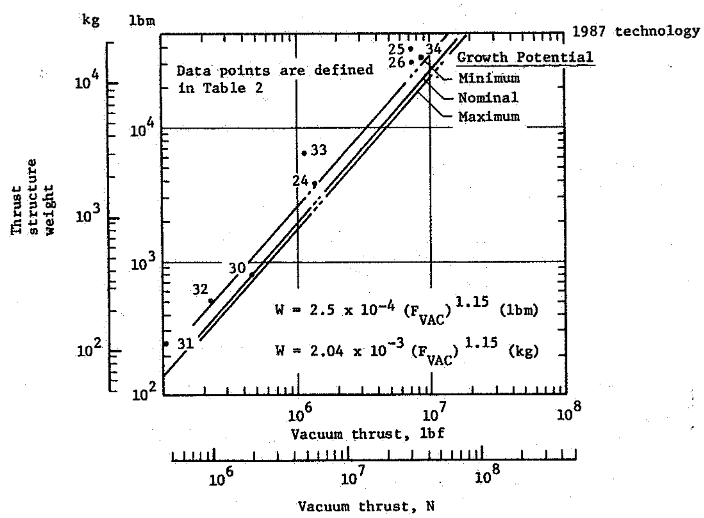


Figure 10.- Thrust structure weight history and projection

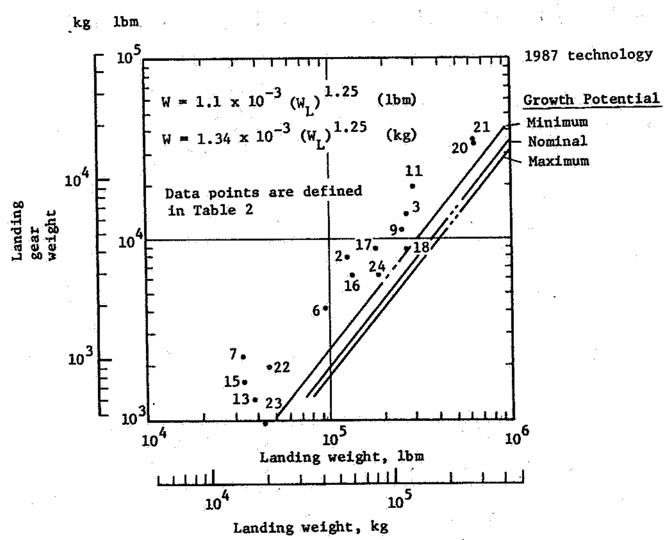


Figure 11.- Landing gear weight versus landing weight

Vertical tail structure.- Vertical tail geometry and weights were used to provide the data shown on Figure 8. The projections represent weight reductions from the present aluminum vertical tail structure by using advanced composite materials for both primary and secondary structures. The vertical tail weight equation shown on Figure 8 was used for later vehicle sizing analysis.

Propellant tanks.- Historical data for liquid hydrogen and liquid oxygen tanks of Stage I and Stage II rocket vehicles were used to identify historical trends of weight reduction. Figure 9 shows tank weight data for Saturn, Titan, and Space Shuttle external tank. Also included are the tanks designed on the Space Shuttle Phase B contract. The external hydrogen tank weights were modified to remove weight penalties due to the orbiter attachment design. The weight parameter shown is tank weight and tank volume. The oxidizer tank and hydrogen tank limits shown are for membrane tank designs and were used to aid in shaping the projections.

Thrust structure.- The thrust structure historical data are shown on Figure 10 for typical missiles and space vehicles as well as the Space Shuttle orbiter. The complex thrust structure of the Shuttle Phase B boosters is also included. The projections are based on use of advanced composites for the thrust structure. The thrust structure weight equation is shown in the figure.

Landing gear.- Landing gear weight data are plotted in Figure 11 as a function of landing weight. The landing gear weight equation was used for later vehicle sizing analysis.

Thermostructural subsystem concepts.- Figure 12 illustrates relative weights of body and propellant tank area thermostructural concepts. Using projections in propellant tankage and TPS weights, the three concepts shown have the indicated relative weights. Backup data for the relative unit weights are shown in Table 3. The unit weights for the radiative TPS concept are based on unpublished data derived during the Space Shuttle phase B study.

Aerosurface thermostructural concepts are compared on Figure 13. The lightest concept is the advanced composite structure wing with RSI/isolator bonded directly to the skin. The relative weights of concepts 1, 3, 4 and 5 are based on a wing trade study during the Phase B Shuttle study.

TABLE 3.- BODY THERMOSTRUCTURE CONCEPTS

Item	Unit Weight Comparison: Concept I kg/m^2	Concept I (lb/ft^2)	Concept II kg/m^2	Concept II (lb/ft^2)	Concept III kg/m^2	Concept III (lb/ft^2)
TPS (Nonmetallic)						
Surface insulation	6.80	(1.39)	6.80	(1.39)	----	----
Subpanels	1.95	(0.40)	----	----	----	----
Support structure	4.78	(0.98)	----	----	----	----
TPS (metallic)						
Radiative panels	----	----	----	----	5.13	(1.05)
Support structure	----	----	----	----	8.79	(1.80)
Insulation	----	----	----	----	4.83	(0.99)
Insulation packaging	----	----	----	----	1.86	(0.38)
Load bearing shell	----	----	13.03	(2.67)	----	----
Propellant tank	13.03	(2.67)	7.91	(1.62)	13.03	(2.67)
Tank insulation	1.41	(0.29)	1.41	(0.29)	1.41	(0.29)
Tank support	----	----	1.21	(0.25)	----	----
Total	27.97	(5.73)	30.36	(6.22)	35.05	(7.18)
w/w_I	1.0		1.09		1.25	

Figure 14 shows relative unit weights of leading edge concepts. The reinforced carbon-carbon is representative of the present Space Shuttle leading edge concept. The two active cooled leading edge designs are from Phase B Shuttle studies. The RSI leading edge concept is our projected technology design that assumes higher temperature reuse capability for the RSI materials.

Thermostructural concepts selection.- Material and component technology projections are integrated in three thermostructural vehicle concepts as shown in Figure 15.

In Concept I, the integral multiple lobe propellant tanks are covered with a standoff of advanced composite honeycomb subpanel with RSI bonded to the exterior surface. The aerosurfaces are advanced composite primary structure with RSI and strain isolator bonded to the surface.

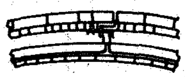
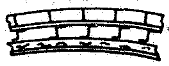
Concept			Comment
Concept I RSI bonded to advanced/composite subpanels integral aluminum tankage insulation		1.0	Recommended for SSTO baseline
Concept II RSI and strain isolator bonded to aluminum structure nonintegral tank with external insulation		1.09	
Concept III Standoff metallic radiative heat shield aluminum tankage with internal insulation		1.25	

Figure 12.- Body thermostructural concepts

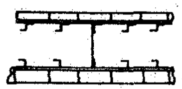
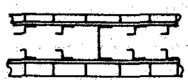
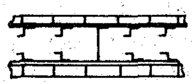
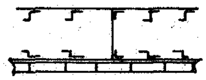
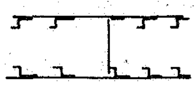
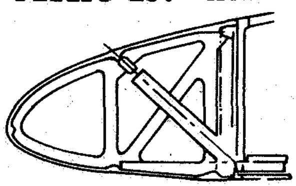
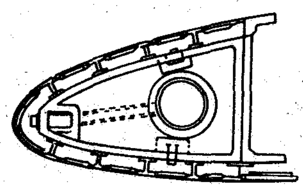
Comment		Relative Weight	Comments
RSI and strain isolator bonded to aluminum structure		1.0	
RSI and strain isolator bonded to advanced/composite structure		0.85	Recommended for SSTO baseline
RSI and strain isolator bonded to titanium structure		0.88	
Partial shielded (RSI) titanium structure		1.25	Problem areas: differential thermal strains
Hot structure		2.9 to 6.5 (Function of material used)	Problem areas: Oxidation coatings differential thermal strains

Figure 13.- Aerosurfaces thermostructural concepts



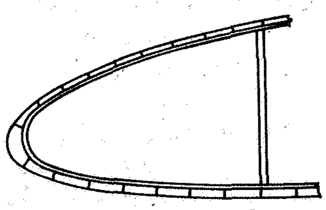
Reinforced carbon-carbon

W = 24.4 kg/m^2 (5 lbm/ft^2)



Transpiration cooling

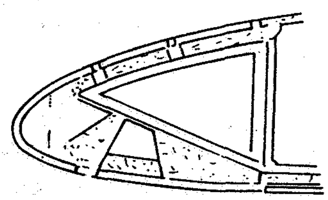
W = 43.9 kg/m^2 (9 lbm/ft^2)



RSI - Recommended for baseline

W = 14.6 kg/m^2 (3 lbm/ft^2)

Passive TPS



Heat pipe

W = 34.2 kg/m^2 (7 lbm/ft^2)

Active TPS

Figure 14.- Leading edge TPS - passive versus active

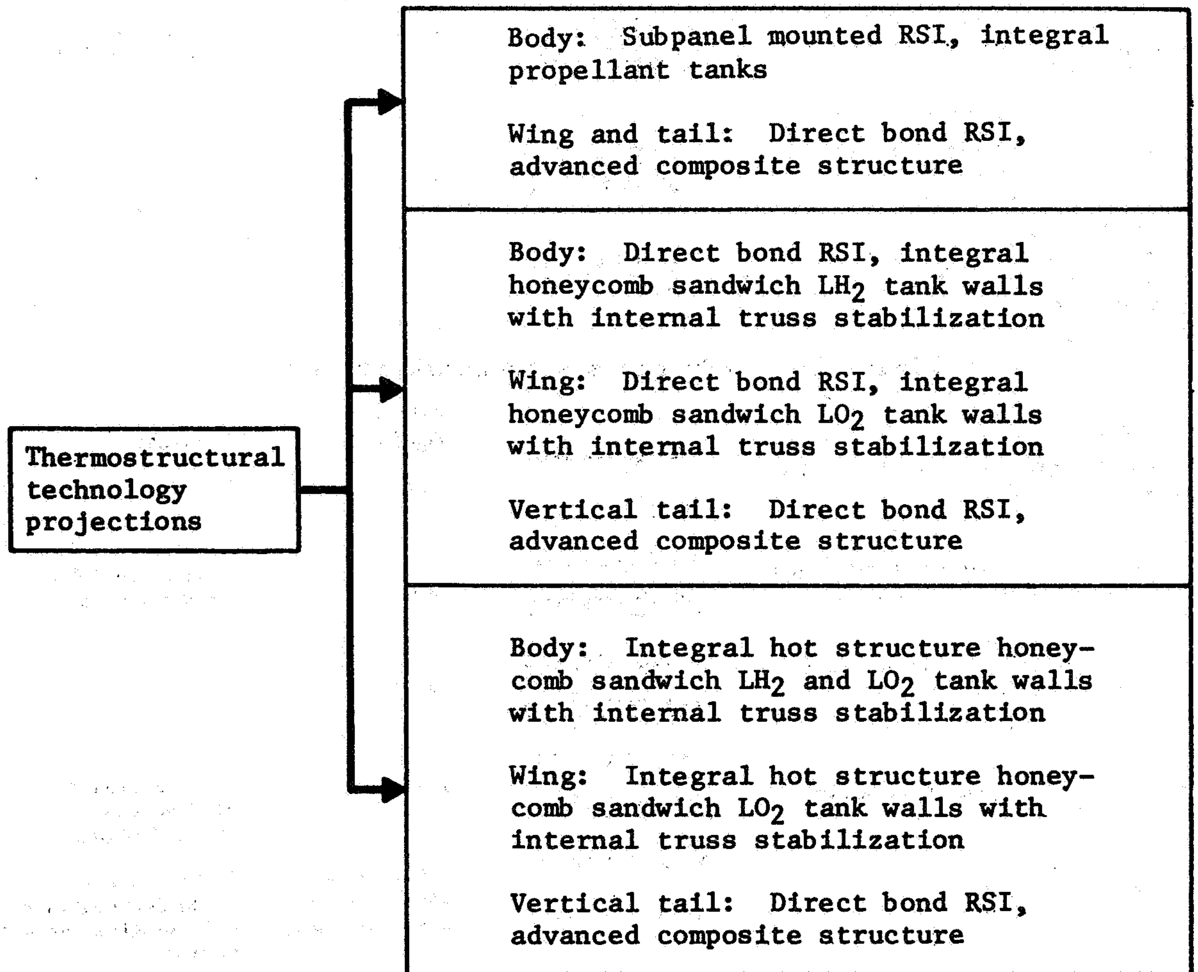


Figure 15.- Thermostructural vehicle concepts

Concept II is an integrated propellant tank wing and body configuration. The integral LH_2 tank is shaped to lifting body configuration and RSI tiles with a strain isolator are bonded directly to the sandwich tank walls. The walls are stabilized by internal truss structure. The LO_2 tanks form the wings of the vehicle and are constructed of honeycomb sandwich skins internally truss-stabilized with direct-bond RSI/strain isolator. The vertical tail and the area control surfaces are advanced composite structure with direct-bond RSI/strain isolator.

Concept III is an integrated propellant tank wing and body configuration identical to Concept II except that it is constructed of high temperature alloys and has external TPS only on the vertical tail.

These three concepts were used in this study to determine which would yield the lightest vehicle dry weight when applied to single-stage-to-orbit vehicle designs. Both unit weight comparisons (Figures 12, 13 and 14) and vehicle weight comparisons (shown later herein) were made.

Propulsion

Approach.- Important propulsion parameters are projected for a 1995 operational date (IOC) extrapolation of historical data. The flow logic used to establish the projected performance values is shown on Figure 16. The critical parameters considered were specific impulse, engine thrust-to-weight ratio, thrust chamber pressure, and net positive suction head (NPSH). Historical data were collected from all types of rocket propulsion systems and used where applicable. As an example, even though the guidelines of the present study defined the main-engine propellant to be LO_2 and LH_2, any past or existing rocket system was investigated to provide a background for performance projections. The extrapolations were guided by recognition of possible hardware or design limitations and by advice from personnel at the Rocketdyne Division of Rockwell International and the Aerojet Liquid Rocket Company. It was assumed that a real need existed to improve each critical correlation parameter for the SSTO and that the available R&T funds would be directed correspondingly.

Considered all reasonable candidates.

Limited projections to LO_2/LH_2 propellants with bell and linear nozzle engines.

Assumed focused improvement effort with accompanying directed funding.

Establish Data Book.

Noncryogenic and cryogenic information for establishing trends

(Pump fed primarily applicable to main propulsion; pressure fed primarily applicable to OMS and RCS)

Select Critical Parameters.

Defined correlation parameters

- Specific impulse
- Chamber pressure
- Thrust and weight ratio
- Density
- NPSH

Established trend characteristics.

Make Final Projection.

Information and advice from Rocketdyne and Aerojet

Expert judgment

Design and hardware limitations

R&D funding definition

Projected correlation parameters

Figure 16.- Flow logic for propulsion technology projection

Extrapolations were made for the main propulsion system and the RCS and OMS auxiliary systems. In addition to the trend analysis of the critical parameters, a more general approach was taken to establish other pertinent performance parameters relating to linear and conventional bell-nozzle engines, nozzle configurations, mixture ratios, air-breathing engine concepts, and propellant bulk density.

Main engine propulsion system.- Projections for specific impulse, chamber pressure, engine thrust-to-weight ratio, and NPSH were made for the main propulsion system engines.

The specific impulse history and projection is shown in Figure 17 and the historical data bank used to perform the trend analysis and aid in the projection of 1995 vacuum specific impulse is shown in the insert. The noncryogenic data were used to determine improvement trend characteristics only. The projected nominal specific impulse value is 463.5 seconds. The rationale supporting the nominal projection was the time and funding that will exist to develop a LO_2 and LH_2 engine with a performance efficiency equal to 98% of theoretical with a chamber pressure of 31×10^6 N/m^2 (4500 psia), mixture ratio of 7.0, nozzle expansion ratio of 160, and probable use of both propellants for cooling. Presently the Space Shuttle main engine (SSME) has a 97% theoretical efficiency at a chamber pressure of 20.7×10^6 N/m^2 (3000 psia), expansion ratio of 77, and mixture ratio of 6.0.

The minimum projected value of 460 seconds was based on an expected SSME product improvement. The rationale supporting the maximum projection of 475 seconds consists of an engine with expansion ratio in excess of 300, mixture ratio of seven, 98% efficiency, and probable use of both propellants for cooling. The objective of engine development for high specific impulse was to minimize propellant load and gross liftoff weight giving consideration to the high mixture ratios required to increase propellant bulk densities.

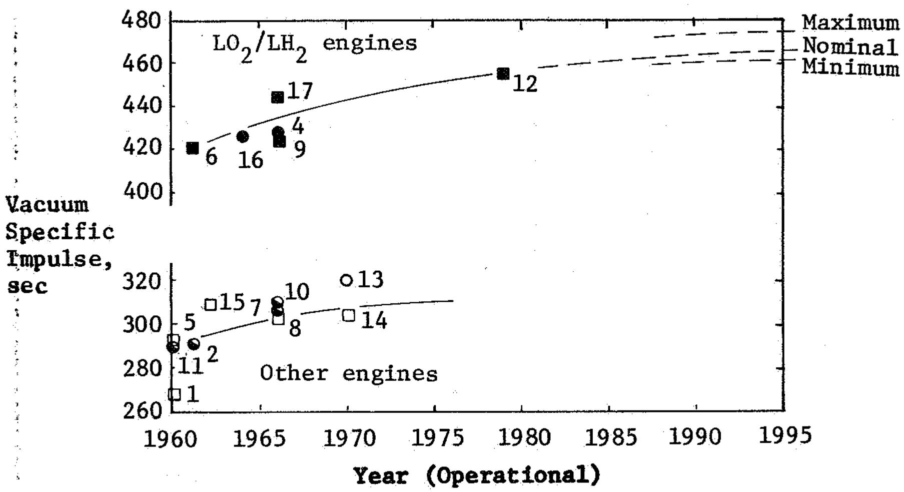
Engine envelope size was considered critical to optimize subsystem packaging in an SSTO vehicle. Because thrust level was dictated by the requirement of thrust to weight at liftoff, thrust chamber pressure was the only remaining variable available to reduce engine size. Chamber and nozzle diameters and lengths are inversely proportional to the square root of chamber pressure. Also, a significant sea level specific impulse improvement results from increased chamber pressure. As an example, the SSME sea level performance would increase from 363.2 seconds to 390.0 seconds if the chamber pressure were increased from 20.7×10^6 N/m^2 (3000 psia) to 31×10^6 N/m^2 (4500 psia).

The chamber pressure history and projection is shown on Figure 18 and shows a nominal projected value of 31×10^6 N/m^2 (4500 psia). The nominal value was based on an optimistic pump design limit for a staged combustion engine cycle and would require direct improvement efforts in such areas as materials, seals, and bearings. The minimum projected value was 26.2×10^6 N/m^2 (3800 psia) and is rationalized as an expected SSME improvement. The maximum value projected was 38.6×10^6 N/m^2 (5600 psia) and would require concentrated R&T effort in pump design, cooling, and material improvement.

The thrust-to-weight ratio projection is shown in Figure 19. There was no obvious trend in the historical data primarily because of the variations in engine configurations. The RL10A and SSME engines were used to make the trend projection. The nominal projected value of 82 was justified as a 10% reduction in SSME weights. The minimum value was representative of no improvement in SSME-accomplished thrust to weight. The maximum value of 90 was established as a 20% improvement and would require a concentrated weight reduction program.

The NPSH was considered critical to tank weight. Figure 20, produced by Rocketdyne, indicates a favorable trend to a NPSH of near zero. Obviously, improvements will be required in pump inducer designs.

<u>RCS/OMS</u>.- Propellant and system specific impulses were projected for RCS or OMS systems. For large total impulse auxiliary propulsion systems, the propellant specific impulse dominates the system specific impulse (total impulse/total system weight) levels; that is, the dry system weight becomes a much smaller percentage of total loaded weight. Therefore, system specific impulse approaches propellant specific impulse. Figure 21 presents historical and projected data for subsystem weight as a function of total impulse for monopropellant and bipropellant systems. The data bank is shown as an insert. The weights represent total penalty chargeable to the auxiliary propulsion system. The Shuttle OMS system specific impulse is 246 seconds compared to a propellant specific impulse of 314 seconds. The SSTO OMS total impulse requirement was projected to be approximately twice that of the Shuttle OMS and shows a definite need for improved propellant specific impulse.



Historical data

1. Delta, AJ10-118
2. Saturn, H-1
3. Thor, LR-79
4. M-1
5. Agena, LR-81
6. Centaur, RL-10
7. Saturn, F-1
8. Transtage, AJ10-138
9. Saturn, J-2S
10. Atlas, LR-105
11. Atlas, LR-89
12. Shuttle, SSME
13. Titan III, LR-91-11
14. Titan III, LR-87-11
15. Titan II, LR-91-5
16. Saturn, J-2
17. Centaur, RL-10

Figure 17.- Specific impulse history and projection

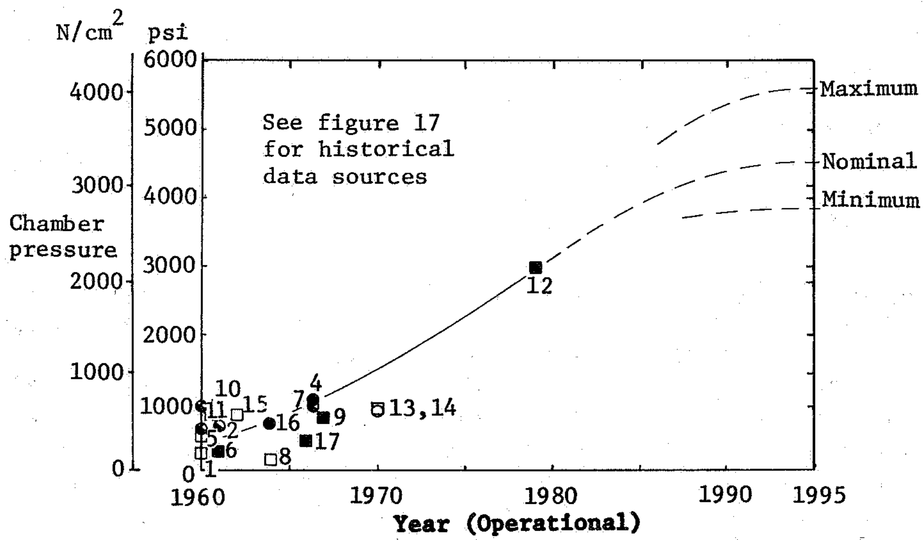


Figure 18.- Chamber pressure history and projection

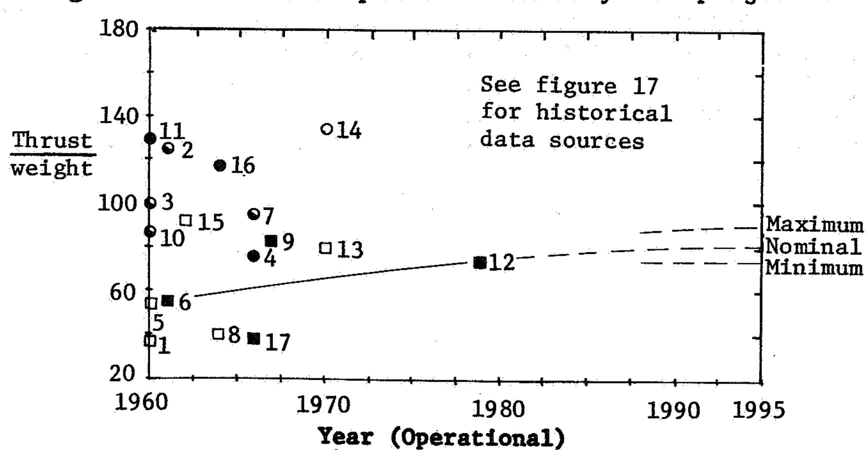


Figure 19.- Engine thrust and weight history and projection.
1987 technology

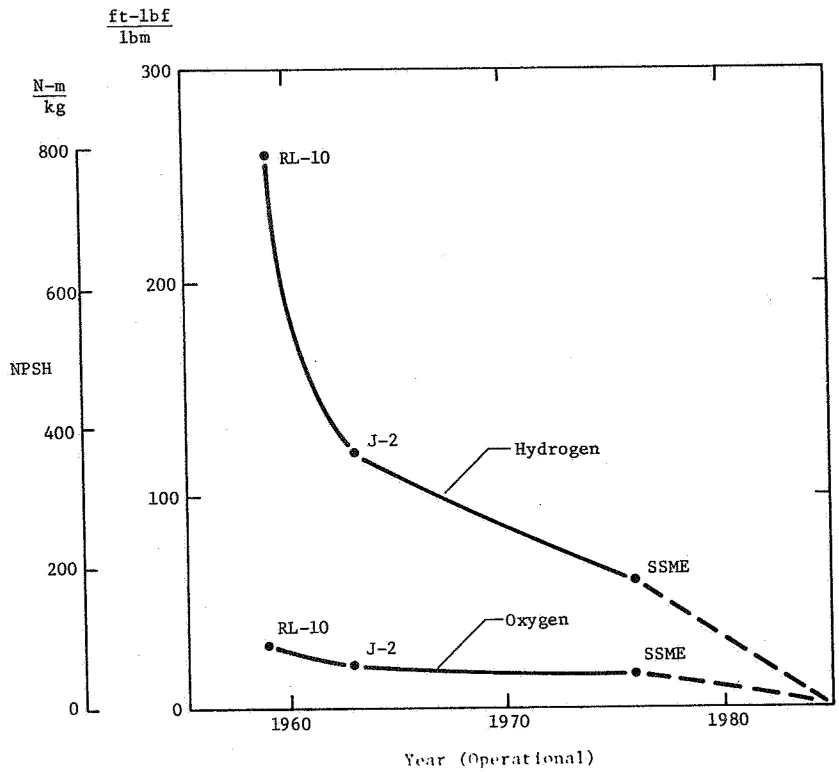


Figure 20.- Pump inducer NPSH history

Figure 22 presents projections of propellant specific impulse for the SSTO time frame. Significant gains can be realized by using oxygen and hydrogen bipropellant systems. The minimum projected value was based on the use of gaseous oxygen and hydrogen systems. The nominal value was associated with low chamber pressure liquid oxygen and hydrogen bipropellants, and the maximum value with high chamber pressure cryogenics.

The data from Figures 21 and 22 in conjunction with the historical data were used to predict system specific impulse for the SSTO. Rationale for the nominal value was based on the use of a bipropellant gaseous oxygen and hydrogen system with minimum component redundancy and a mixture ratio of 4 to 5. Propellants would be stored in a liquid state. The projected value represents the same system specific impulse-to-propellant specific impulse ratio as the present Space Shuttle OMS. The minimum projected value was for storable bipropellants supported by minimal improvement in current Space Shuttle OMS system specific impulse.

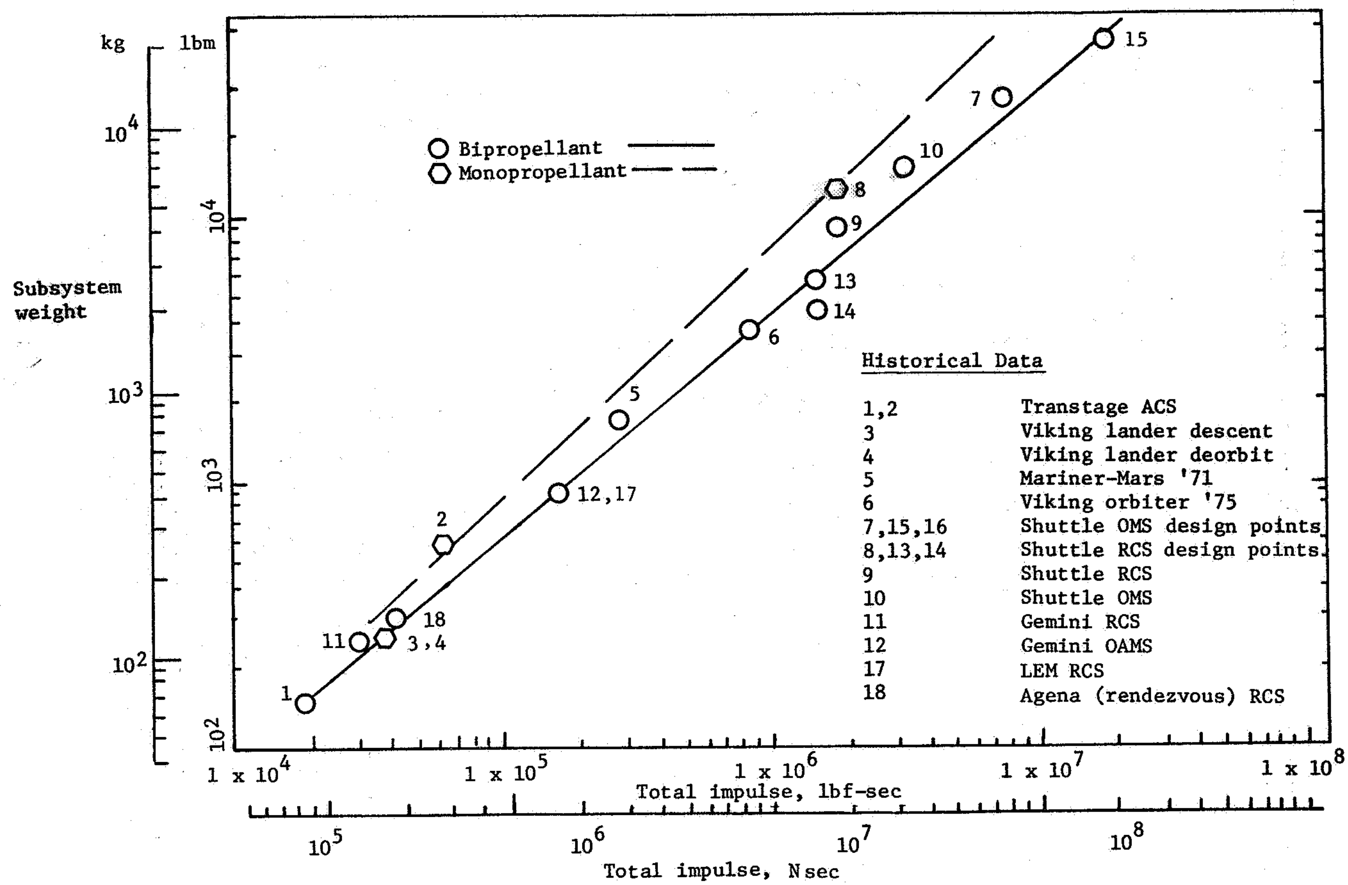


Figure 21.- Auxiliary propulsion system characteristics

The maximum system specific impulse would require significant component reliability improvements, use of liquid cryogenics, LH_2 storage at low pressure, and integrated tankage. The description and performance of this system was defined in the McDonnell Douglas Space Shuttle Auxiliary Propulsion System Design Study, Phase C Report, Report No. MDC E0523 under Contract NAS 9-12013.

General considerations.- Other propulsion parameters that affect the SSTO configuration and/or performances that were considered but were not analyzed by technology trend projections were propellant bulk density, engine configuration, and air-breathing engines.

1. Propellant bulk density.- An increase in propellant bulk density has a significant impact on decreasing vehicle dry weight and liftoff weight. With the restriction that the propellants are defined as LO_2 and LH_2, propellant bulk density can only be improved by using triple point or slush propellants. Densities

of the propellants as a function of state are presented in Table 4. These physical characteristics were obtained from the NBS, NASA, and Aerojet. Slush hydrogen has been produced, pumped, and handled at the National Bureau of Standards at Boulder, Colorado. It is now anticipated that the best usable slush propellants will have an average density equivalent to approximately 50% solid. Inasmuch as the present level of attention to this technology area has been small, triple point propellants were not selected for vehicle design using "normal" technology growth. However, with accelerated funding, these propellants could be available for SSTO applications.

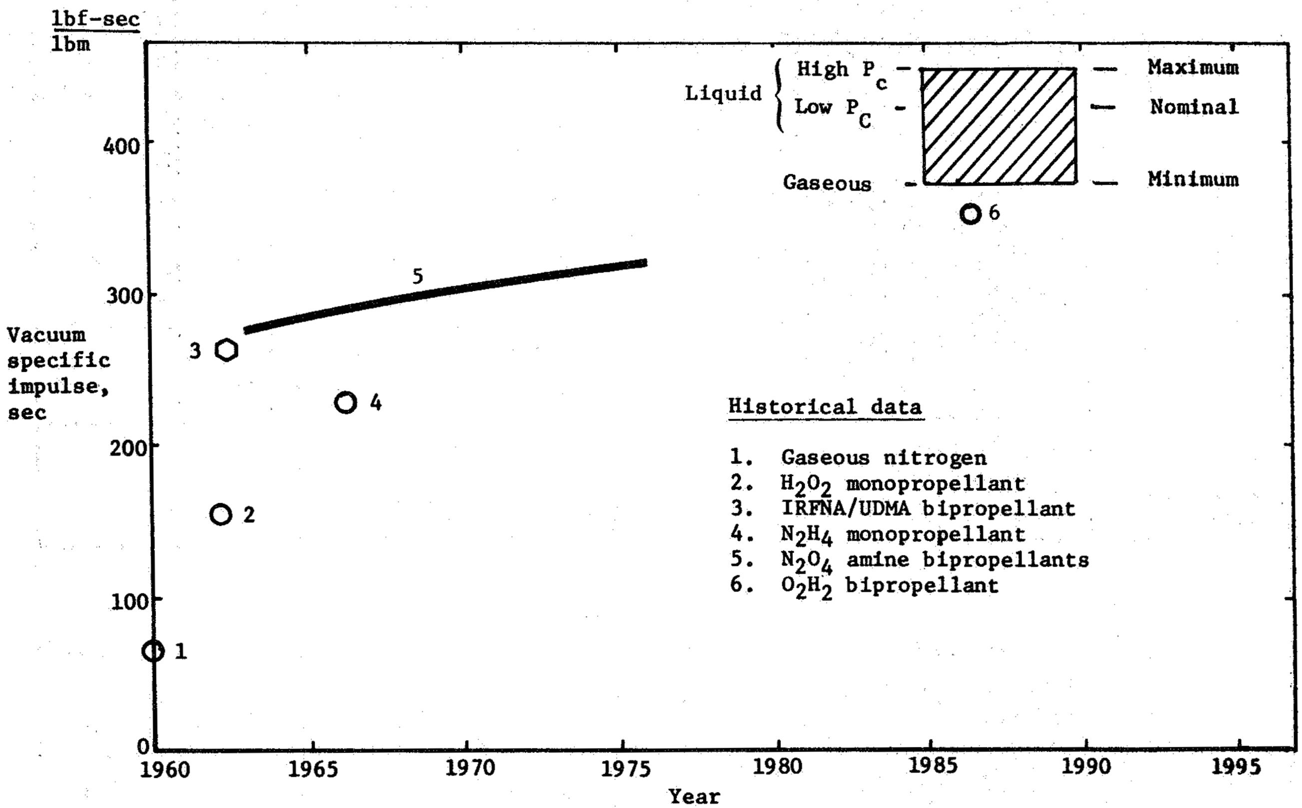


Figure 22.- RCS/OMS typical steady state specific impulse

TABLE 4.- CRYOGENIC PROPELLANT CHARACTERISTICS

Propellant	% Solid	Temperature,		Vapor pressure		Density	
		°K	°R	N/m^2	psia	kg/m^3	lb/ft^3
Oxygen	0*	90.8	163.5	137,900	20.0	1136	70.9
	0**	90.2	162.3	101,350	14.7	1141	71.23
	0***	54.3	97.8	1,379	0.2	1306	81.57
	50	54.3	97.8	1,379	0.2		
	100	54.3	97.8	0	0.0	1358	84.8
Hydrogen	0*	20.6	37.0	137,900	20.0	70.5	4.40
	0**	20.3	36.5	101,350	14.7	71.1	4.44
	0***	13.8	24.9	6,895	1.0	76.9	4.8
	50	13.8	24.9	6,895	1.0	81.4	5.08
	100	13.8	24.9	0	0.0	86.5	5.4

* Task 2 design
** Normal boiling point
*** Triple point

Initial estimates of potential dry weight improvements with the use of triple-point propellants are shown in Table 5. These values reflect no degradation in engine specific impulse because of lower propellant enthalpy. Vehicle weight reduction is directly attributable to tank volume reduction. Higher relative benefits resulted in the VTO concept compared to the HTO because of higher VTO volumetric efficiencies. The greatest impact is on the IFF configuration because of the large cruise propellant weights, which are proportional to gross weight. Because of the present technical status, implementation of increased density propellants was postponed until the Extended Performance Studies were completed.

TABLE 5.- POTENTIAL IMPROVEMENTS WITH TRIPLE-POINT PROPELLANTS

Oxygen	Hydrogen	% Dry weight change		
		VTO	HTO	IFF
Boiling point	Boiling point	Reference		
Triple point	Boiling point	-3.6	-2.7	-4.1
Boiling point	Triple point	-6.0	-5.0	-6.3
Boiling point	Triple point	-9.4	-8.2	-11.9

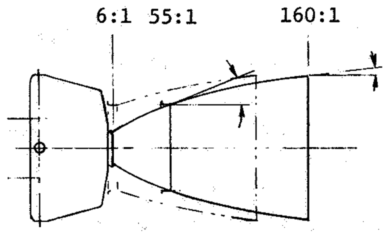
2. Engine configuration.- Engine configurations were studied to improve average flight specific impulse and vehicle packaging. Extendible multiposition conventional nozzles provide high performance at sea level (low expansion ratio) and also at altitude (extended high expansion ratio). The relatively large power head envelopes of the high chamber pressure engine limited the allowable forward retraction of the extendible nozzle designs. The total length of the engine in the extended position was dictated by contour considerations.

These considerations are illustrated in Figure 23(a). A nozzle with a near-optimum contour in the extended position is shown in example A. This nozzle is split for retraction near the area ratio (55) for full seal level expansion. When retracted, the flow exit angle (at 55) causes sea level performance losses. Furthermore, the overhang of the rest of the nozzle (at 160) is so great that the flow emanating from the inner nozzle can impinge on the outer section and reduce performance adding heating problems. Splitting the nozzle farther aft (beyond 55) reduces this flow impingement problem but further degrades low-altitude performance. A modified contour (example B) is a preferred alternative, using a nonoptimum contour to reduce the exit angle (at 55), thereby reducing the overhand when this nozzle is retracted for low-altitude operation. A conical nozzle (example C) also exhibits minimal flow impingement when retracted, but has unacceptably severe losses because of the large exit angles. A rolled diaphragm nozzle skirt (example D) shows promise for improving extendible-nozzle performance, but needs much more development to be compatible with repetitive reuseability.

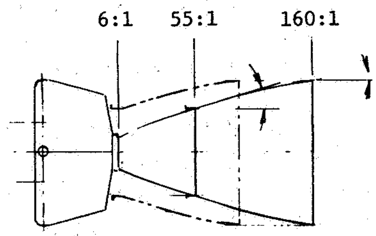
A linear engine configuration was evaluated as a possible means of improving vehicle packaging and providing higher average specific impulse. The engine configuration is shown on Figure 23. It is a multiple segment, split-combustor design that operates at a chamber pressure of 20.7×10^6 N/m^2 (3000 psia). A total of ten sets of the SSME turbopump assemblies, mounted internally between the upper and lower nozzle surfaces, supply propellants to ten grups of combustor segments. Thrust vector control is accomplished by differential throttling of combustors. Throttling or combuster shutdown is used to limit vehicle acceleration.

The graph in Figure 23 illustrates that this engine has less performance at low altitudes than the bell-nozzle engine, resulting in a lower average specific impulse. The engine configuration could be modified to improve its overall performance applied to an SSTO vehicle, but the parametric engine data required to do this have not been available.

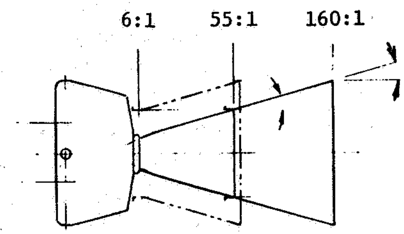
3. Airbreathing engines.- Airbreathing engine trends and requirements were reviewed as applicable to the IFF concept. Based on previous studies by Pratt and Whitney, turbojet engines can readily be adapted to use hydrogen fuel with an appreciable reduction in fuel consumption. Engines developed specifically for hydrogen would also result in reduced engine size and weight. Because of the limited operating range required for the SSTO, additional cost and weight benefits can be projected through engine simplification. Sophisticated fuel controls, variable geometry compressors, and variable exhaust nozzles now incorporated on military and commercial engines would not be necessary. Only auxiliary power for the engines themselves would be supplied and, combined with strictly ground-supplied start systems, such as compressed air turbine impingement, would reduce gear box requirements to an absolute minimum. If needed for landing, restart could be accomplished by windmilling.



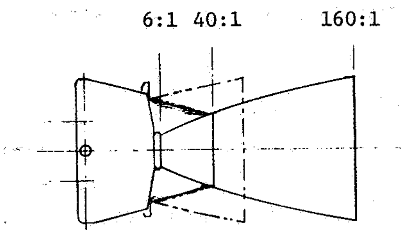
A. Optimum contour



B. Modified contour

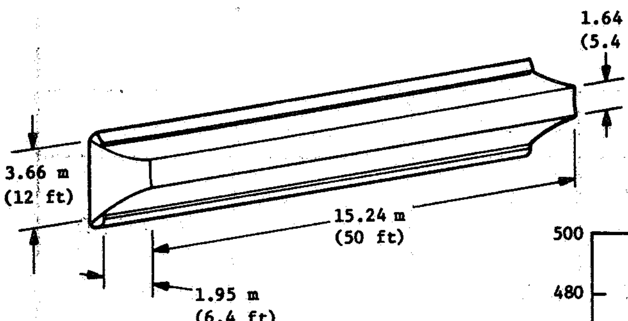


C. Conical



D. Rolled diaphragm

(a) Extendible nozzle considerations



P_C = 20.7 (10^6) $\frac{N}{m^2}$ (3000 psia)

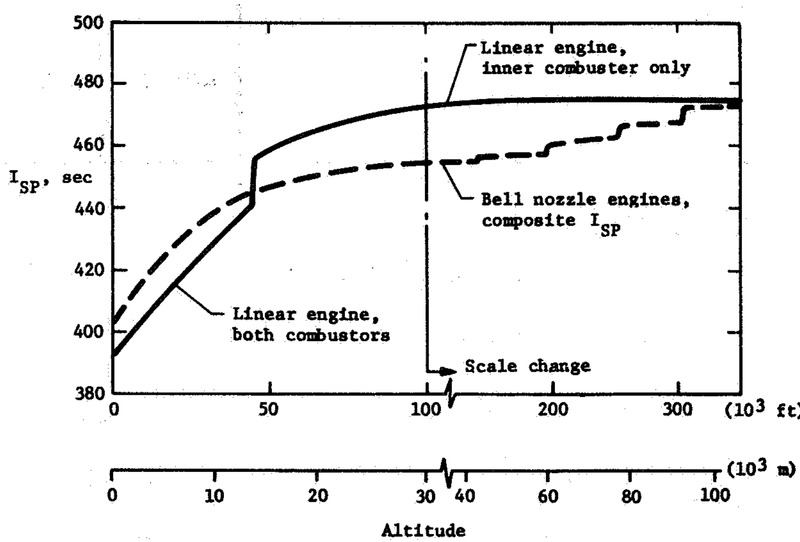
F/W = 684 N/kg (69.8 lbf/lbm

ϵ overall = 91

$I_{SP_{VAC}}$ = 475.0 sec

$I_{SP_{SL}}$ = 391.5 sec

O/F = 6.0



(b) Linear engine preliminary concepts

Figure 23.- Bell nozzle and linear engine concepts

The large engine installation weights and low fuel requirements of airbreathers, were traded against the low engine weight and high fuel consumption of rockets in later system definition analyses.

Summary of projections.- Table 6 presents a summary of projections used for later configuration definition representing normal propulsion technology growth. Propellant densities were taken at a vapor pressure of 137 900 N/m^2 (20 psia). The projected nominal values of specific impulse for the OMS and RCS systems are 440 seconds, and 420 seconds, respectively. Slush propellant considerations were projected for Extended Performance Studies. Normal growth configuration sizing is to be based on nominal values of performance parameters.

TABLE 6.- PROPULSION SYSTEM CONCEPTS SELECTION

	Fixed nozzle configuration		Extendable nozzle configuration	
Bell nozzle engines				
Chamber pressure MN/m^2 (psia)	27.6 (4000)		27.6 (4000)	
Area ratio	35		55/160	
Thrust/weight, vacuum	81.2		58.9	
	MR = 6	MR = 7	MR = 6	MR = 7
$I_{SP_{Vac}}$ (Sec)	441.4	436.1	466.4	463.5 ε = 160
$I_{SP_{SL}}$ (Sec)	408.2	404.0	399.5	395.5 ε = 55
Linear Engine				
Chamber pressure MN/m^2 (psia)	20.7 (3000)			
Area ratio (overall)	91			
Thrust/weight, vacuum	69.8			
$I_{SP_{VAC}}$ (Sec)	475.0			
$I_{SP_{SL}}$ (Sec)	391.5			

Secondary Technology Areas

A number of secondary technology areas were investigated, although to a lesser degree than the materials, structures, and propulsion areas. These secondary disciplines included aerothermodynamics, performance optimization, aerodynamics, computer technology, control systems, and auxiliary power. The general approach in studying these areas consisted of first identifying the current activities and their associated level of technology and then identifying the projected 1990 technology status and its impact on SSTO vehicle design. Table 7 summarizes the results of these studies and a more detailed analysis is presented in the secondary technology section of the Appendices. This investigation has shown that significant vehicle improvements leading to weight and cost reductions can be realized with future focused development in these secondary disciplines.

TABLE 7.- SUMMARY RESULT OF SECONDARY TECHNOLOGY STUDY

Areas of technology	Projections for improvements
Aerothermodynamics	Better knowledge of catalytic wall, lee surface heating, and B.L. transition effects
Performance optimization	Optimal trajectory guidance, reduced margins
Aerodynamics	Development of optimal configuration parameters (wing-body shape)
Computer technology	Advanced techniques for vehicle design and onboard flight operations
Control systems	Integrated digital systems, relaxed static stability, and improved load relief
Auxiliary power	Improved fuel cells and APU, higher pressure hydraulics, hot gas actuation

R&T FUNDING PROJECTIONS

Two approaches were taken to identifying and projecting NASA and DOD funding for "normal" technology growth. The first was a "top-down" method of selecting those portions of the total NASA budget that were considered to be applicable to the single-stage-to-orbit vehicle. The second method was a "bottom-up" approach whereby RTOPS documents, industry news services, and marketing reports were researched to identify the applicable NASA and DOD technology efforts and the efforts were then projected into the future. In each case, the historical data were organized, judgement was used to make linear projections, and polynomial regression curve fitting techniques were employed.

NASA Funding

Top-down.- There are many technology areas being funded by the OAST and OMSF offices of NASA that offer potential technology growth for SSTO designs (Refer to Table 8.). The total NASA obligations are the summation of budgets comprising OMSF, OSS, OA, OAST, Tracking and D/A, plus facilities and Research and Program Management. Actual dollar outlays for fiscal years 1973 and 1974 and estimates for 1975 through 1980 are listed in Table 9 and plotted in Figure 24. Shuttle funding is included in the OMSF category and all data are based on current 1975 dollars. The information sources used were (1) Budget Estimates, Office of Management and Budget, 1975, Vol. 1, NASA Summary Data, Research and Development; (2) NASA Planning Wage Guidelines, February 1975; and (3) NASA Fiscal Year 1976 Estimates and Budget Summary.

The portion of the total NASA funding that was judged to be related to SSTO technology has been separated and shown in Table 10 and plotted in Figure 25. Fluid dynamics and high and low speed flight dynamics were combined in one category. The 1975 and 1976 data are current fiscal year estimates and 1977 through 1990 data are linear projections based on judgement. The information sources used were (1) Budget Estimates, OMB, 1975, Vol. 1, NASA Summary Data, Research and Development; and (2) Aviation Week and Space Technology, 17 March 1975, pp 59-68.

Bottom-up.- The RTOPS documents for 1973, 1974, and 1975 were reviewed for purposes of identifying SSTO-related technology and funding on recent NASA research activities. Each RTOP was designated to be in one of five major categories (Refer to Table 11.). Individual items were summed in each of the five categories and linear projections to 1990 were made based on judgement. Polynomial regression curve fitting was then employed to derive the curves shown in Figure 26. The boundaries, which include a 95% probability range, are shown.

TABLE 8.- NASA RESEARCH AND TECHNOLOGY SUMMARY

Related NASA/DOD cooperative efforts
National Facilities Program
YF-12 (supersonic flight research)
X-24 (hypersonic flight research)
Entry technology configuration program
C-130E composite wing box
Support of military developments (F-14, F-15, F-16, B-1)
Aeronautical R&D Study
YF-12 flight experiments
Propulsion, air induction systems
Structures, flight loads predictions/correlations
Materials, flight evaluations of composites
Avionics and controls
Aerothermodynamics
AST related research (advanced supersonic technology)
Materials, composites
Advanced propulsion technology
LH_2 engines
Space technology (includes Shuttle, IUS)
Propulsion, LO_2/LH_2 engines, dual mode, lifetime
Materials, TPS
Analysis, ODIN/EDIN, NASTRAN, IPAD
Basic research
Aerofluid mechanics, flight mechanics, power
Materials, composites
Structures
Propulsion (air breathers)
Avionics
Mission systems and integration
Advanced development, composites, fabrication, propulsion, payloads
General purpose mission equipment
Advanced missions
Uses of space transportation system
Improvement of space systems
Cost/performance forecast methods
Development, test, and mission operations
Research and test operations (JSC and MSFC)
Life sciences (selection criteria for crew and passengers)
Launch systems operations
Space life sciences
Life support and protective equipment
Man-machine technology
Apollo-Soyuz test project
Rendezvous and docking systems
Space processing of materials
Space Shuttle
Systems and subsystems development and integration
Propulsion technology
Thermostructural technology

TABLE 9.- NASA FIVE-YEAR PLAN BASED ON CURRENT (1975) PROGRAM FUNDING

Dollars in millions†

FY	1973	1974	1975	1976	1977	1978	1979	1980
Shuttle*	377	475	798	1206	1276	1199	821	347
Total OMSF	1154	1000	1110	1414	1494	1419	1042	570
Total OSS	680	580	540	547	455	312	238	225
Total OA	189	161	178	167	140	116	85	80
Total OAST	233	234	245	236	221	203	187	180
Tracking & D/A	248	244	250	250	252	255	296	300
Other	4	8	12	17	17	17	17	17
Research & development	2508	2227	2335	2631	2579	2322	1875	1372
Construction of facility	79	101	158	130	120	70	50	50
Research & program management	722	727	727	721	721	721	721	721
Total	3309	3055	3220	3482	3420	3113	2646	2143

*Included in OMSF funding †Expressed in equivalent 1975 dollars

TABLE 10.- RELATED SSTO NASA FUNDING

Dollars in millions*

FY	1973		1974		1975	
Materials	6.0	9.2%	6.6	9.3%	6.9	9.2%
Structures	6.1	9.4%	6.4	9.1%	7.0	9.3%
Avionics	3.2	4.9%	3.2	4.5%	3.8	5.1%
Propulsion	8.2	12.6%	9.7	13.7%	10.4	13.9%
Airbreathing engines	8.0	12.3%	8.0	11.3%	8.0	10.7%
Fluid dynamics, high- and low-speed flight dynamics	28.6	43.8%	29.5	41.8%	30.1	40.2%
Other	5.1	7.8%	7.3	10.3%	8.7	11.6%
Total	65.2	100%	70.7	100%	74.9	100%

TABLE 11.- SELECTED NASA RTOPS TOTALS

Dollars in millions*

FY	1973		1974		1975	
Structures	3.94	32.9%	1.78	18.4%	4.75	39.0%
Materials	4.11	34.3%	3.11	32.2%	2.38	19.5%
Subtotal	8.05	67.2%	4.89	50.6%	7.13	58.5%
Propulsion-main engine plus auxiliary	2.65	22.1%	3.19	33.0%	3.05	25.0%
Airbreathing engine	0.55	4.6%	1.05	10.9%	1.26	10.3%
Hypersonic technology	0.73	6.1%	0.53	5.5%	0.76	6.2%
Total	11.98	100.0%	9.66	100.0%	12.20	100.0%

*Actual

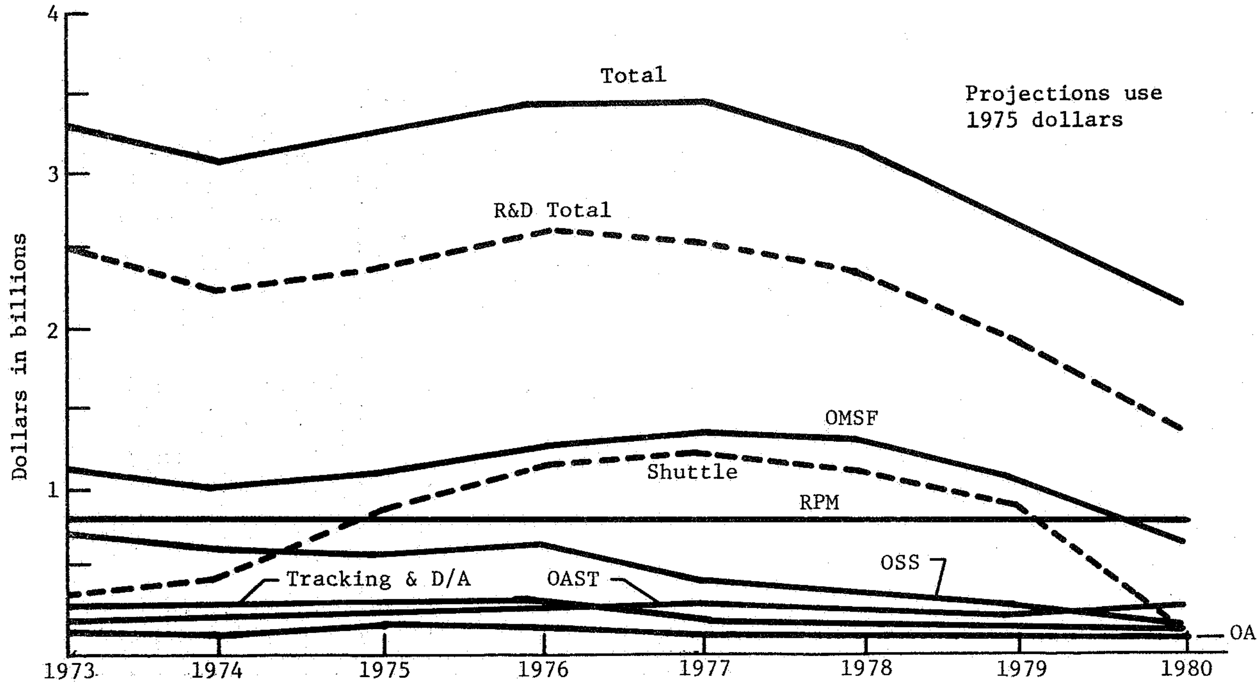


Figure 24.- NASA five-year funding plan

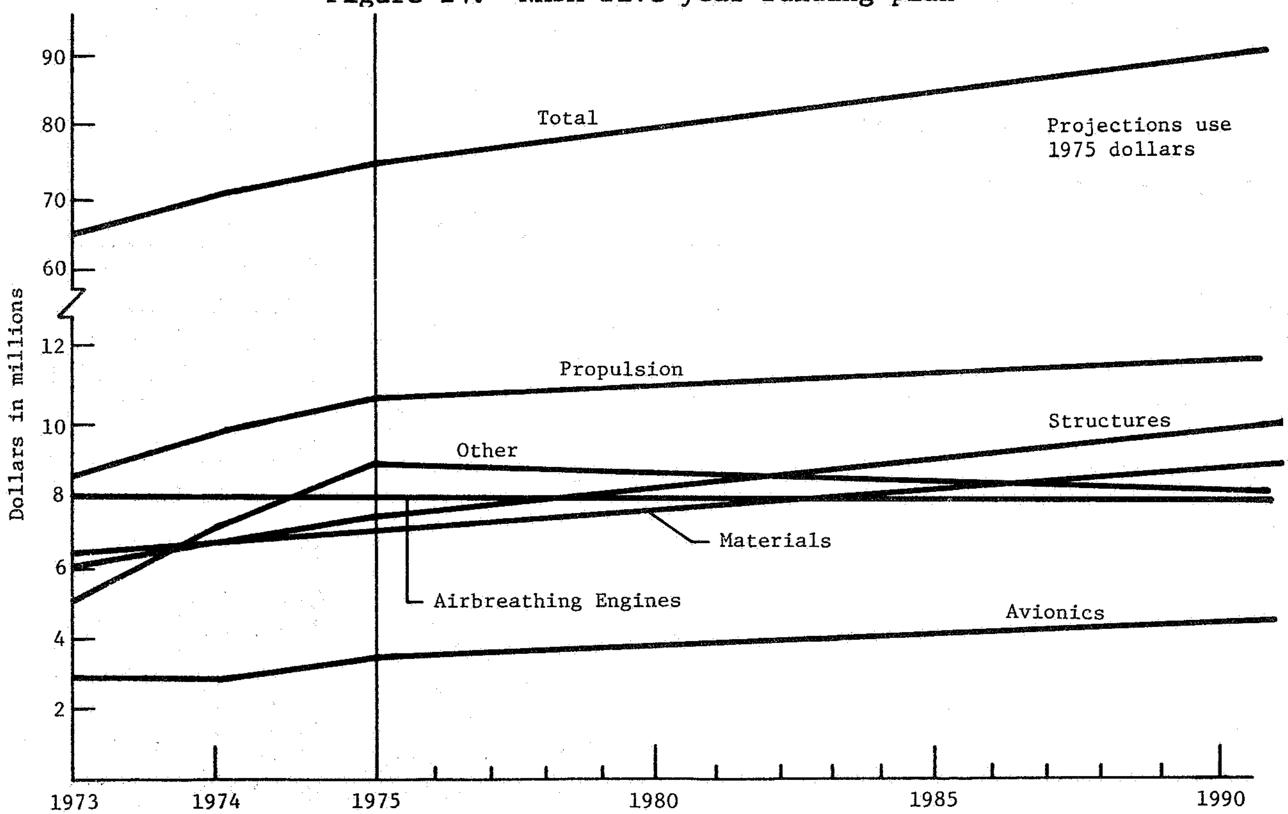


Figure 25.- SSTO-related NASA funding

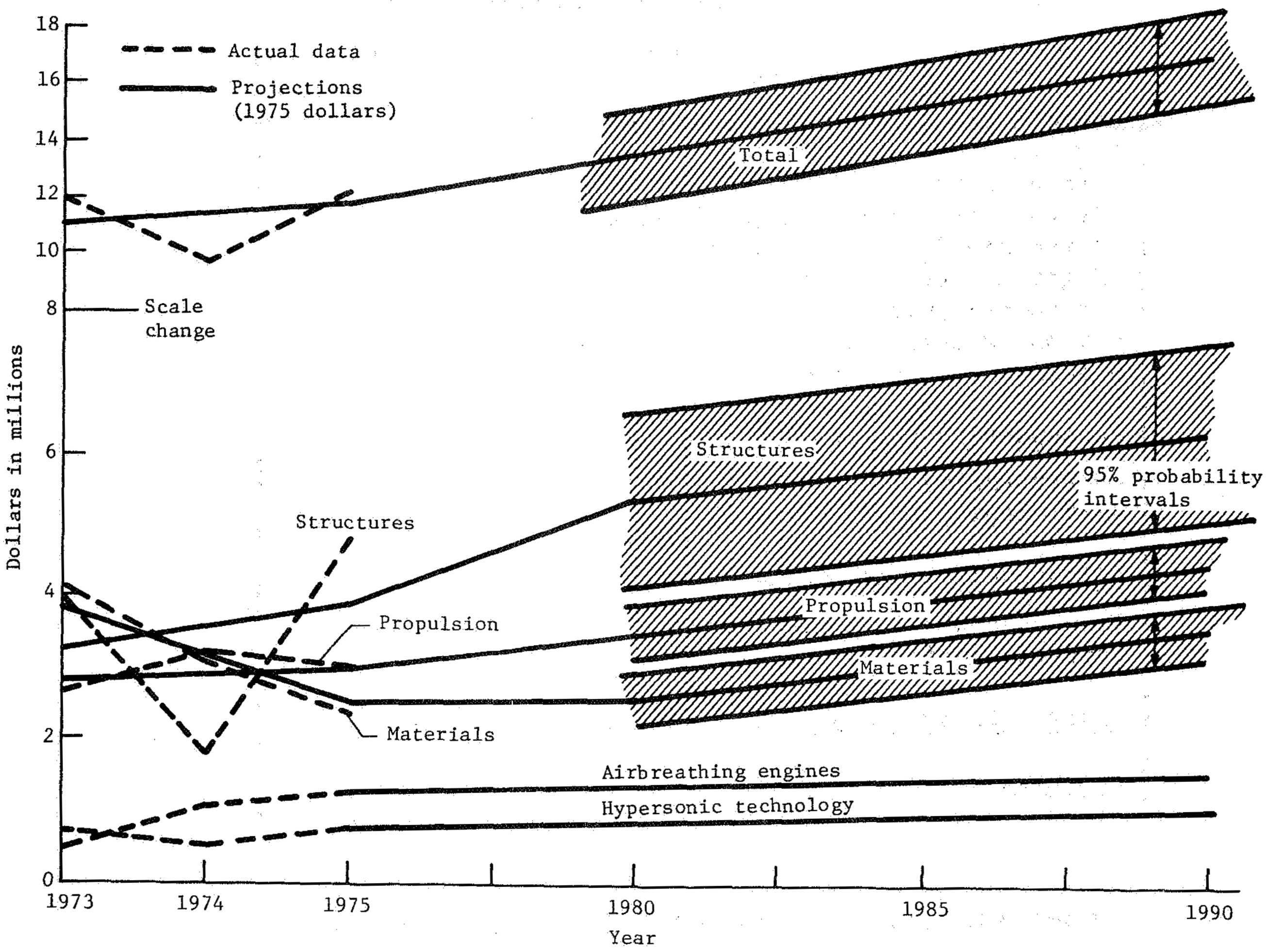


Figure 26.- Selected NASA RTOPS funding

DOD Funding

Bottom-up.- Table 12 summarizes R&T activities in the DOD that offer potential growth for SSTO designs. The selected applicable technology items are tabulated in Table 13. Selection was made based on research titles and consultation with experts working in the fields of interest. Basic airframe research was excluded from structures and materials; propulsion includes some subcategories in aircraft technology. Figure 27 shows the polynomial regression curves that were derived to fit the linear projections out to 1990 that were based on judgement. The boundaries encompassing the 95% probability range are shown. The information sources were (1) DMS Contract Quarterly, March 1975; (2) Industry News Service; (3) DMS Marketing Reports; and (4) committee on Armed Services, U.S. House of Representatives, 24 February 1975.

TABLE 12.- DOD RESEARCH AND TECHNOLOGY SUMMARY

Aerospace flight dynamics
Structural testing, design criteria, concepts, analysis
Dynamics
Aero-acoustics
Airframe propulsion compatibility
System simulation and analysis
Flight control systems
Aerothermodynamics
Composite structures
Stability and control
Aerospace propulsion
Rocket engines
Airbreathing engines
Flight vehicle technology
Transonic aircraft technology
Control configured vehicles
Space vehicle subsystems
Space Shuttle

TABLE 13.- SELECTED DOD (AIR FORCE) FUNDING TOTALS

Dollars in millions

	1973		1974		1975	
Structures	5.81	38.6%	6.18	39.3%	7.19	41.2%
Materials	2.67	17.8%	3.40	21.6%	3.78	21.7%
Subtotal	8.48	56.4%	9.58	60.9%	10.97	62.9%
Propulsion	6.13	40.8%	5.82	37.1%	6.04	34.6%
Other	0.42	2.8%	0.32	2.0%	0.44	2.5%
Total	15.03	100.0%	15.72	100.0%	17.45	100.0%

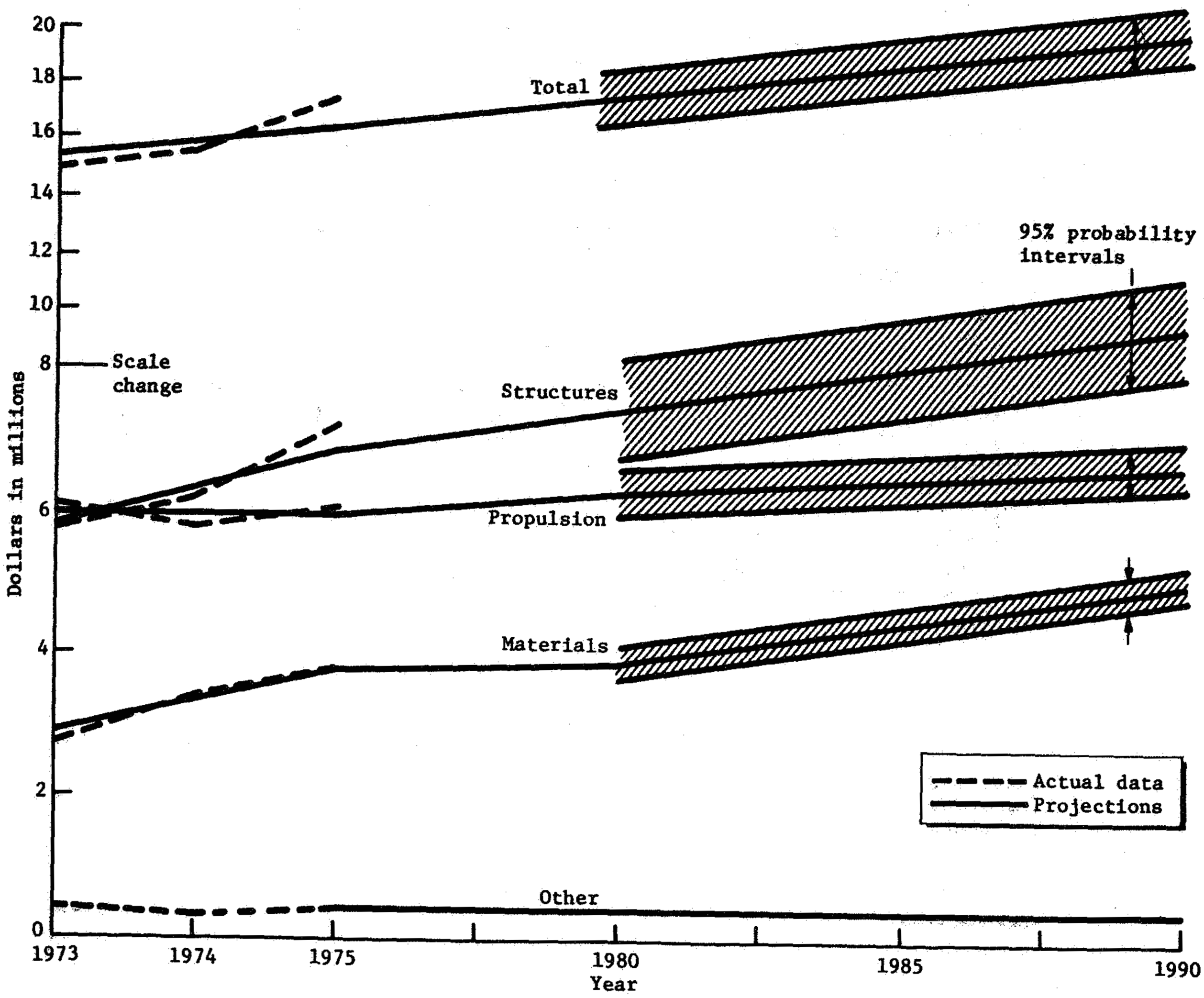


Figure 27.- Selected DOD (Air Force) funding

Summary Results

The 1975 funding levels for structures, materials, and propulsion for both the "top-down" and the "bottom-up" estimates are tabulated in Table 14. These funding levels are for R&T activities applicable to an SSTO vehicle concept. The projected annual average spending is based on the data from Tables 10, 11, and 13.

TABLE 14.- TASK 1 FUNDING PROJECTIONS

Dollars in millions

	Materials and structures	Propulsion
Top-down (1975)		
NASA-related SSTO	13.9	10.4
Bottom-up (1975)		
NASA - selected RTOPS	7.1	3.0
DOD - selected R&T	11.0	6.0
	18.1	9.0
Projected annual average spending for R&T	19.5	10.2

PERFORMANCE POTENTIAL OF VEHICLE SYSTEMS

RATIONALE AND SCOPE

Study guidelines and "normal" technology projections were used to configure three basic vehicles: VTO, HTO, and IFF. Thermostructural and configuration concepts were selected for the vehicles based on parametric studies that considered three thermostructural concepts and two propellant tankage concepts. The significant technologies are discussed and final mass properties tabulated for each vehicle concept.

Vehicle ascent was optimized by determining initial thrust and weight, the best combinations of dual position and fixed nozzle engines, and engine shutdown versus throttling efficiencies. Aerodynamic, aerothermodynamic, and flight performance analyses were performed. Critical airloads that were generated for the VTO vehicle were input to a finite element model of the fuselage tank-wing assembly to provide internal vehicle loads to use for substantiation of structural sizing results. Analysis was focused on vehicle concepts with the purpose of identifying key technology requirements.

The Statement of Work identified numerous design requirements and objectives that influenced the vehicle designs. Table 15 presents a summary of these items.

PARAMETRIC STUDIES AND CONCEPT COMPARISONS

Configuration Modifications

Initial vehicle sizing studies included parametric analyses of configuration arrangements to obtain the most forward center of gravity location and to minimize vehicle dry weight. Trends of various studies are given in Table 16 relative to an initial representative vehicle concept. The first two modifications were incorporated in the final vehicle configuration.

The wing carrythrough structure was located in the body, aft of the LO_2 tanks because of the following structural and configurational considerations:

(1) The propulsion feed system requires at least a 1.83 m (6 ft) straight run including prevalve; therefore, no more than a 3.35 m (11 ft) length could be saved in the aft compartment by reducing the wing box length.

TABLE 15.- GUIDELINE DESCRIPTION

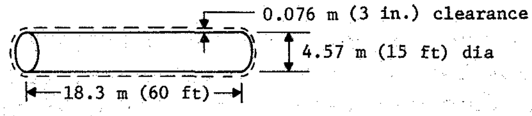
Design vertical takeoff, horizontal landing vehicles for minimum dry weight using dual-mode propulsion.
Use dual-mode engine performance and weights from advanced high-pressure engine study (ref. 2).
Use accelerated performance, accelerated technology projections (ref. 1).
n_x = 3-g ascent, n_z = 3-g entry, n_z = 2.5 g subsonic maneuver.
Safety factors: Prelaunch, liftoff, ascent, in-orbit: 1.4 Entry, subsonic maneuver, landing: 1.5
Design to low-cost refurbishment and maintenance. Life: 500 missions.
Payload cylinder 
Mission: Due east from KSC, 28.5-deg inclination, 29 500 kg (65 000 lbm) payload, 198 m/sec (650 ft/sec) OMS ΔV, 30.5 m/sec (100 ft/sec) RCS ΔV, Reference energy orbit, 93 x 186 km (50 x 100 n. mi.)
TPS design mission: Entry from a due east, 28.5-deg inclination, 370 km. (200 n. mi.)-altitude orbit, 29 500 kg (65 000 lbm) payload, and 2 050 km (1100 n. mi.) crossrange capability.
Vehicle loads with and without 29 500 kg (65 000 lbm) payload.
Maximum landed payload = 29 500 kg (65 000 lbm)
Landing requirements: Minimum speed = 306 ± 9 km/hr (165 ± 5 knots) α = 15 deg (sea-level conditions and maximum landed weight)
Aerodynamic requirements: Subsonic - 2% $\bar{c}$ minimum static longitudinal stability margin, 0.0015 minimum static directional stability margin, Hypersonic Trimmable α range (with/without payload) - 25 deg or less to 40 deg or greater, Landing sink speed - 3.05 m/sec (10 ft/sec) maximum Reentry - Trimmable with control surfaces longitudinally and laterally with RCS (non-CCV designs).
4-man crew cabin arrangement.
10% weight margin on all vehicle subsystems except engines.
Provide for stable dynamic properties by using RCS during periods of low dynamic pressure and aerodynamic control surfaces when dynamic pressures are sufficient.
Provide TPS for protecting the primary airframe, the crew, the payload, and vehicle subsystems from aerodynamic heating during ascent and entry and from engine exhaust convective and radiative heating.
Provide a positive docking mechanism (interception, engagement, and release of vehicle with other orbital elements).
OMS requirements: OMS tankage for ΔV capability of 381 m/sec (1250 ft/sec) OMS burn in either single long burn or a series of multiple burns, spread randomly over the mission duration.

TABLE 16 VEHICLE PARAMETRIC STUDIES - CENTER OF GRAVITY VARIATION

Configuration modification	C.G. shift (%) of body length (landing without payload)
Increase body length 4.3 m (14 feet) to allow for wing carrythrough structure.	1.5 forward shift
Move cargo module forward 9.7 m (32 feet).	1.2 forward shift
50% body length increase	1.0 aft shift
Move OMS system forward of crew compartment.	Concept considered impractical.

(2) The wing carrythrough torque box requires a length compatible with the vehicle loads. If the present 6.4 m (21 ft) wing carrythrough torque box were reduced to only 3.048 m (10 ft), it would introduce a load concentration problem.

(3) The wing carrythrough could be external below the LO_2 tanks with a penalty in cross section and a long standoff ramp for body fairing; however, this would introduce a large amount of unusable volume.

(4) Another concept would be to design smaller diameter LO_2 tanks allowing the wing carrythrough torque box to pass through the body under the revised tanks. This would, however, require longer LO_2 tanks and negate the desired shortening of the vehicle. It would also disrupt the direct load path of the propellant tank walls causing an increase in weight, creating additional unusable volume between LO_2 tanks, and moving the vehicle c.g. aft approximately 1%.

Bell Nozzle and Linear Engines

Figure 28 shows a VTO vehicle with bell-nozzle engines for comparison with a vehicle using linear-nozzle engines, shown in Figure 29. The bell-nozzle vehicle uses four dual position (ε = 55/160) and six fixed position nozzles (ε = 35) with engine sea level thrust values of 2 224 000 N (500 000 lb) and 2 447 000 N (550 000 lb). It is sized to meet a mass ratio requirement of 7.48, based on trajectory optimizations, whereas the linear-engine vehicle is sized to meet its mass ratio requirement of 7.89. The lower performance of the linear engine vehicle is attributed to nonoptimized expansion ratios for the initial, low altitude flight phase. Parametric engine data have not been available to pursue the optimization.

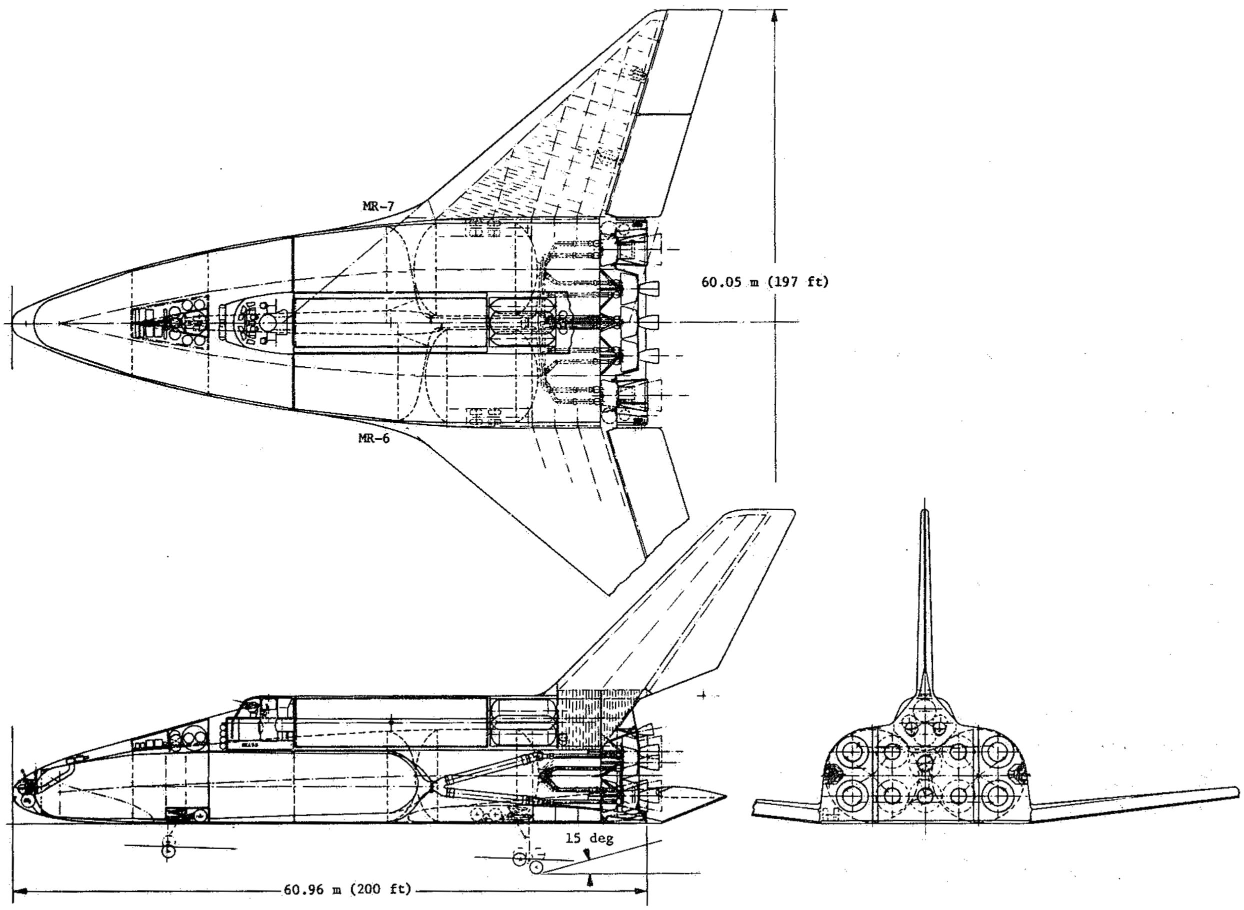


Figure 28.- Bell nozzle inboard profile

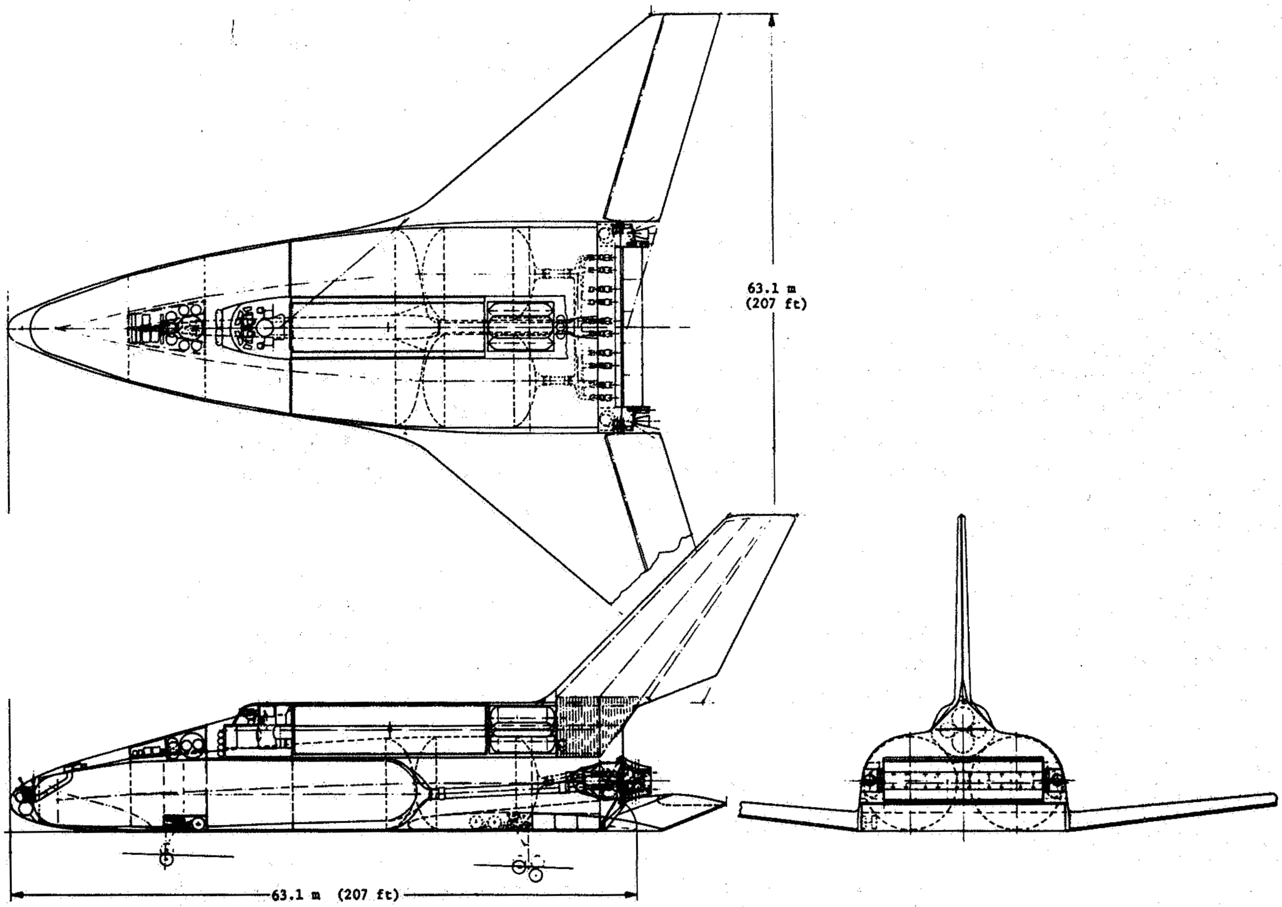


Figure 29.– VTO linear engine vehicle, inboard profile

The configuration and envelope dimensions of the linear engine adopted for this study, as well as the resulting I_{sp} versus altitude characteristics, are shown in Figure 23. This is a multiple segment split combustor engine operating at a nominal chamber pressure of 20.7 (10^6) N/m^2 (3 000 psia). A total of ten sets of SSME-type turbopump assemblies, mounted between the upper and lower nozzle surfaces, supply propellants to ten groups of combustor segments. Thrust vector control is accomplished by differential throttling of combustor segment groups, and thrust level is controlled by a combination of throttling, outer combustor shutdown, and shutdown of combustor groups. The nozzle expansion ratio is 91 with both inner and outer combustor segments operating, and is 320 with an inner segment only. The propellant feedlines and the engine mount structure are modified to accommodate the linear engine requirements.

Table 17 shows the vehicle weights using the two engine concepts. The dry weight of the vehicle with bell-nozzle engines is 10% lighter. It was concluded that this study would be continued using bell-nozzle engines, with the recommendation that studies by engine manufacturers should be initiated to develop linear-nozzle engine parameters.

TABLE 17.- BELL NOZZLE VERSUS LINEAR NOZZLE ENGINE VTO VEHICLE MASS PROPERTIES

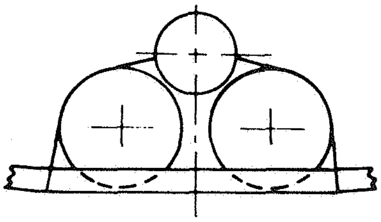
	Bell-nozzle engine vehicle		Linear-nozzle engine vehicle	
	Mass, kg	Weight, lb	Mass, kg	Weight, lb
Dry weight	201 249	443 678	223 466	492 658
Ascent propellant	1 660 998	3 661 873	1 976 129	4 356 618
GLOW	1 921 972	4 237 223	2 262 798	4 988 616

Propellant Mixture Ratio

Assessments of propellant mixture ratio effects led to the selection of O/F = 7.0 on the basis that the VTO bell-nozzle vehicle landing weight was 9 000 kg (20 000 pounds) less than with O/F = 6.0.

Thermostructural Concepts

Three thermostructural concepts (Figure 15) were identified in the technology assessment as candidates for SSTO application. Refer to "Normal" Technology and Funding Projections in which the three concepts are defined. Figure 30 illustrates the three concepts and lists the selected thermostructural criteria. Vehicle designs using these concepts were compared using the same propellant weight for each.



Baseline (Concept I)

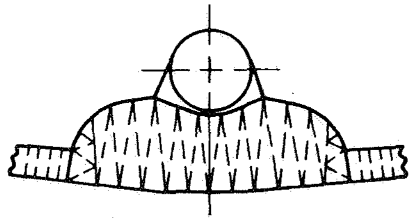
Thermostructure:

Body - aluminum clustered tanks, RSI/subpanel TPS

Aerosurfaces - Borsic/aluminum composite structure, RSI/strain isolator direct bond

Tank pressure (ultimate); 207 000 N/m^2 (30 psi)

Maximum structural temperature: 450 K (350°F)



Concept II

Thermostructure: Rene' 41 sandwich

Tank pressure (ultimate): 207 000 N/m^2 (30 psi)

Maximum structural temperature: 1144 K (1600°F)

Concept III

Thermostructure: Titanium sandwich, RSI/strain isolator direct bond

Tank pressure (ultimate): 207 000 N/m^2 (30 psi)

Maximum Structural temperature: 533 K (500°F)

Figure 30.- Design concept comparison approach

The vehicle in Figure 31 uses an integral tank structure of aluminum alloy with all the propellants in the fuselage (Concept I). The aerosurfaces and nontank skirts are advanced-composite structure. The TPS consists of external RSI directly bonded to the aerosurfaces by means of a strain isolator and RSI bonded to advanced composite sandwich subpanels on the fuselage-tank area. The vehicle shown in Figure 32 uses a truss-supported flattened tank (Concept II). This concept is a hot structure vehicle using Rene´ 41 sandwich tank panels with no external TPS. Concept III is a hybrid vehicle using titanium sandwich tank panels with an external bond-on insulation of RSI. The results of the lowest dryweight and represents an advantage in thermostructural technology, design development, manufacturing, and operations requirements. Technology advantages include the current active developments of RSI-protected aluminum structure for the Space Shuttle and avoidance of hot-structures with their associated thermal expansion, aerosmoothness, and temperature limit concerns. The selection of RSI for the thermal protection provides the lightest weight and also permits a wide entry flight corridor because it can sustain higher heating rates than metals. The integral membrane tankage concept was therefore selected for continued studies.

Note: The legend appears in Figure 35.

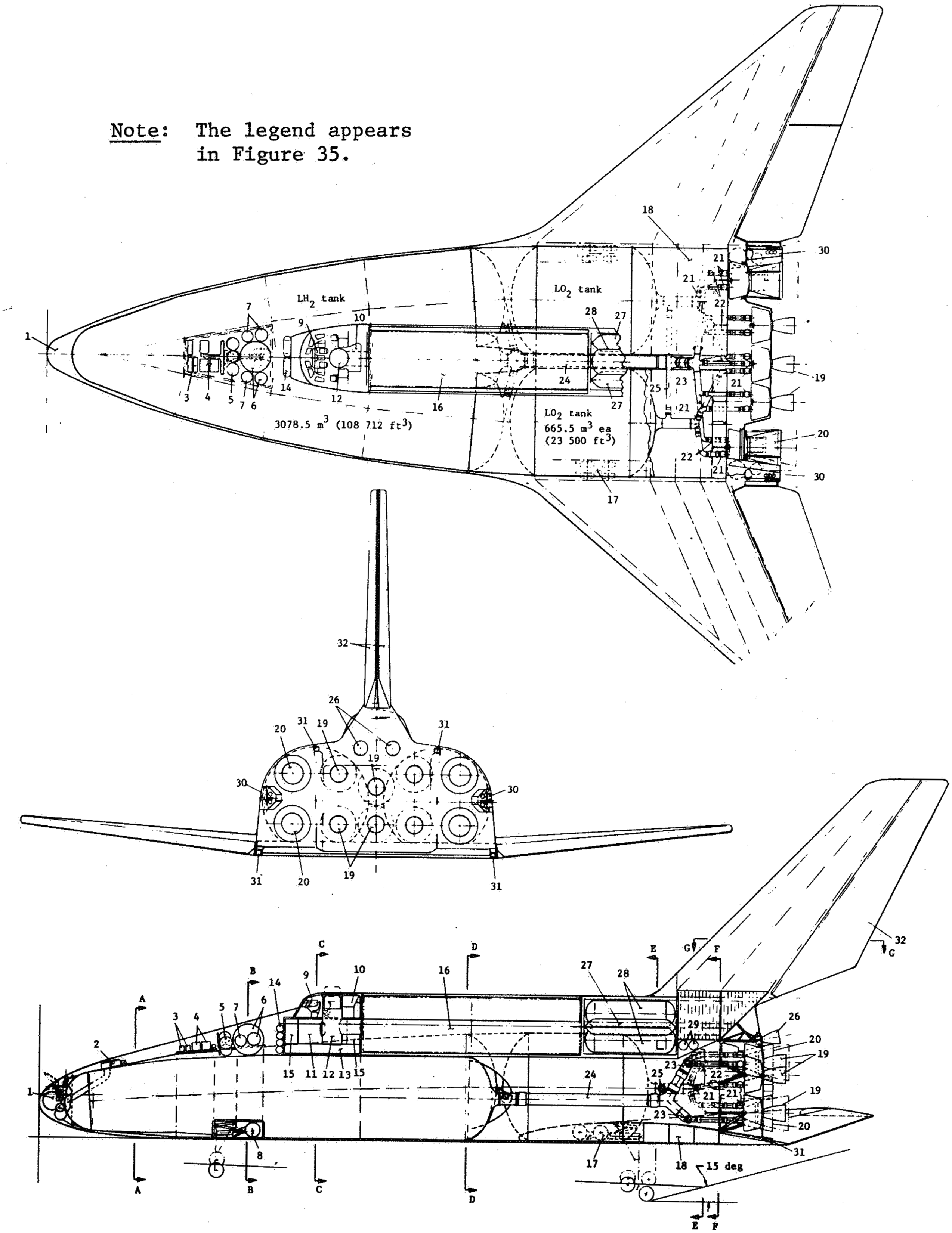


Figure 31.- VTO integral membrane tankage, Concept I

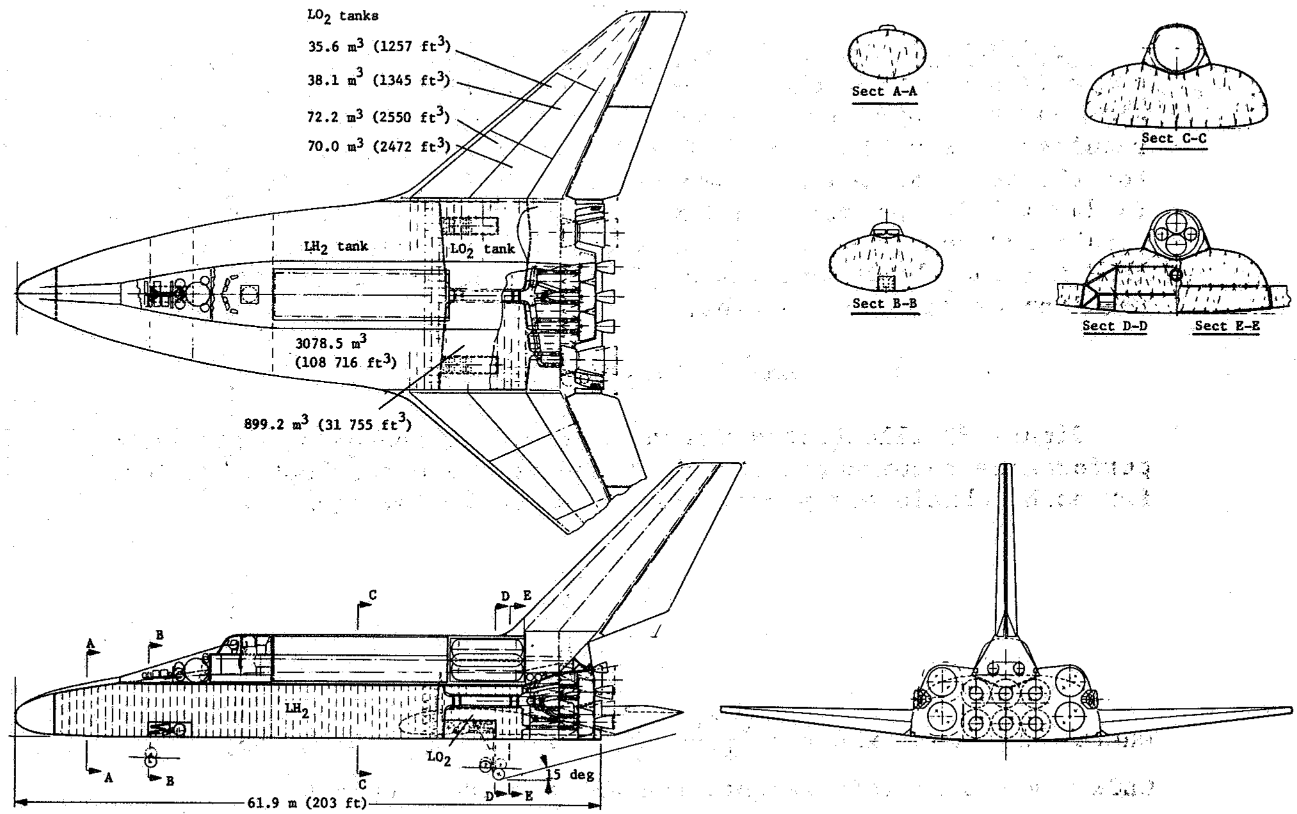


Figure 32.- VTO truss-supported, flattened tank, Concept II

TABLE 18 WEIGHT COMPARISON OF CONCEPTS

Function Description	Mass, kg (weight, lb)					
	Concept I		Concept II		Concept III	
Wing	23 502	(51 813)	43 081	(94 977)	28 383	(62 573)
Vertical tail	5 265	(11 607)	5 265	(11 607)	5 265	(11 607)
Body	35 750	(78 816)	88 723	(195 601)	52 238	(115 166)
Induced environmental protection	39 568	(87 232)	3 402	(7 500)	21 864	(48 201)
Propellant system	4 818	(10 621)	5 380	(11 860)	5 380	(11 860)
Fixed weight	78 715	(173 537)	78 715	(173 537)	78 715	(173 537)
Margin	15 272	(33 669)	18 966	(41 813)	15 694	(34 600)
Dry weight	202 890	(447 295)	243 532	(536 895)	207 539	(457 545)

Vehicle comparison VTO dry wing versus wet wing.- A separate comparison study of the VTO vehicle using LO_2 propellant in the wing cavity (approximately 30% of the total vehicle LO_2 propellant) resulted in a vehicle GLOW of 2.04 million kg (4.5 million pounds) for the wet wing vehicle compared to a GLOW of 1.92 million kg (4.243 million pounds) for the dry wing vehicle. This comparison result led to the selection of a dry wing vehicle concept and was used for the three vehicles of Task 2 based on the commonality requirement for the vehicles.

VEHICLE SIZING APPROACH

Figure 33 illustrates the vehicle sizing approach. The ascent performance requirement curves, based on trajectory optimizations, for each vehicle are plotted using the following equation:

$$\lambda = \frac{1 - \frac{1}{MR}}{1 - \frac{WPL}{GLOW}} \tag{1}$$

where MR = mass ratio = $\frac{GLOW}{WBO}$; WPL = 29 480 kg (65 000 lb); GLOW = gross liftoff weight; and WBO = burnout weight.

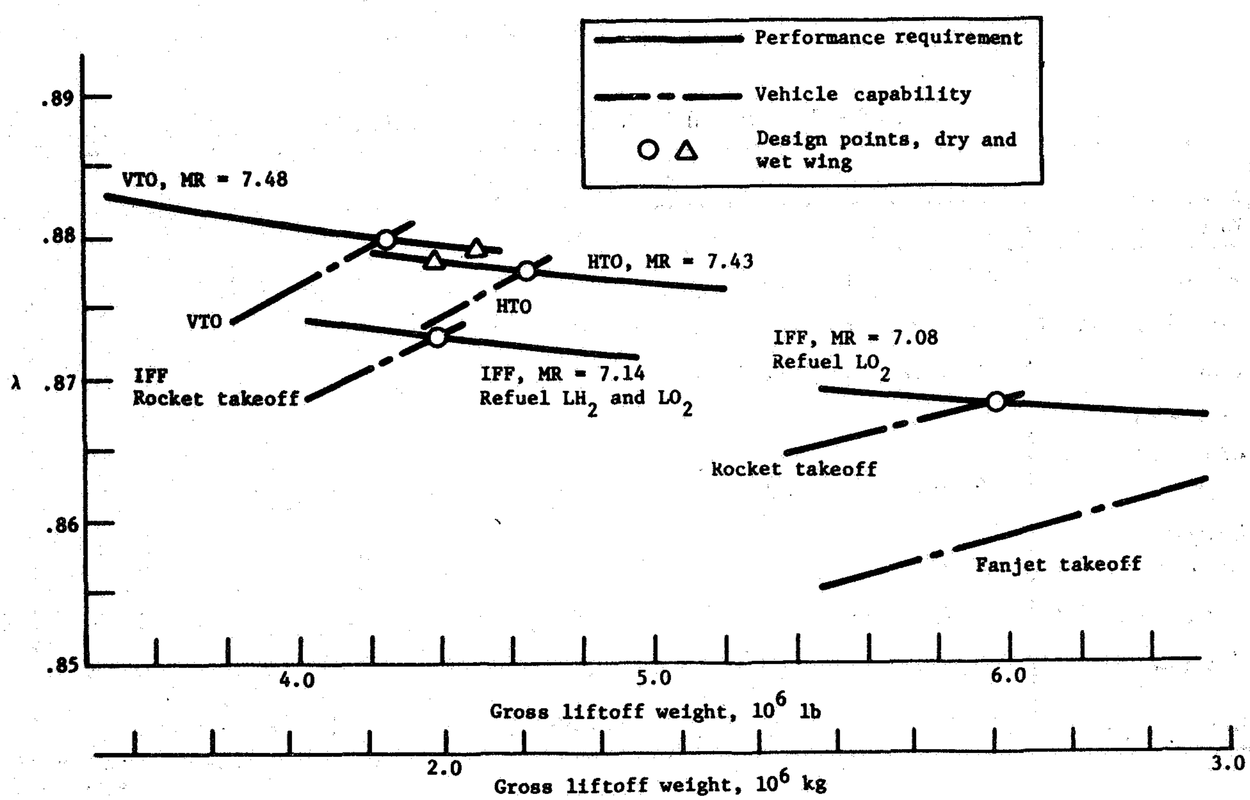


Figure 33.- Approach to sizing vehicles

The vehicle capability curves, based on parametric vehicle weights analyses, are plotted using the equation:

$$\lambda = \frac{WP}{GLOW - WPL - \frac{WLOSS}{2}} \tag{2}$$

where WP = ascent propellant weight; and WLOSS = ascent weight losses. The design points for the vehicles are at the intersection of the performance requirement curves with the vehicle capability curves. The VTO and HTO vehicles were sized with and without propellant in the wings. The IFF vehicles were sized using both rocket and turbofan takeoff propulsion concepts, and for refueling either LO_2 only or both LO_2 and LH_2.

The vehicles to be described in this chapter were designed to carry a payload of 29 480 kg (65 klb). The mass ratio requirements were calculated using ascent performance, employing estimates of lift and drag derived early in the study. Later, aerodynamics for these vehicle configurations were derived and applied to performance calculations of mass ratio requirements for the VTO and HTO vehicles.

The results of using the revised aerodynamics, which exhibited smaller drag coefficients than the initial aerodynamics, showed that the vehicles were capable of lifting payloads heavier than the guideline payload of 29 480 kg (65 klb). Alternately, the vehicle designs could be modified to a smaller size to meet the guideline payload capability. The HTO vehicle size was found to be considerably improved by drag reductions.

Estimates of the VTO and HTO vehicle mass properties based on the revised aerodynamics were made using sensitivity relations. These estimations, as well as the detailed design characteristics of the vehicles, are presented in subsequent sections of this chapter.

VTO VEHICLE DESIGN

The variables studied during initial VTO vehicle sizing were initial thrust to weight, propellant mixture ratio, number of dual-position nozzle engines, number of fixed-nozzle engines, engine shutdown sequence, engine throttling, ascent lift, and duration of constant lift. The POST ascent trajectory program was used to optimize the ascent trajectory. Configuration arrangement was varied and studies of the effect on vehicle c.g. with resulting wing and vertical tail area requirements were compared. An objective leading to minimum dry weight was to arrange the vehicle design for a c.g. as forward as possible, as this leads to smaller wing and vertical tail areas and significant reductions in vehicle size.

General Arrangement

The VTO vehicle shown on Figure 34 is a nearly optimum vehicle within the study groundrules and practical considerations of design. The vehicle is 61.9 meters (203 ft) long and has a wing span of 60.2 meters (197.4 ft). Ten rocket engines in the fuselage base are arranged with four dual-position (ε = 55/160) nozzles outboard and six fixed-nozzle (ε = 35) engines inboard. The wing has leading edge and trailing edge sweeps of 50 deg and 20 deg respectively; the vertical tail, 45 deg and 28 deg. The vertical tail is a 10 deg wedge configuration with the capability of forming a double wedge configuration by actuating the split rudders and speed brakes inward as shown on Figure 35, Section G-G.

Inboard Profile

Figure 35 shows the inboard profile of the VTO vehicle. The major components are the fuselage tank module, the crew and payload module, and the exposed wing assemblies.

Fuselage tank module.- The fuselage tank module consists of the liquid hydrogen and liquid oxygen tanks connnected by inner tank skirts and an aft skirt compartment made up of the wing carrythrough structure, the engine mount beams, and the aft heat shield structure. The hydrogen tank is a three lobe tank configured to conform to the desired fuselage shape and to be compatible with good structural load paths. The outlet of the fuel tank is centrally located to pass between the two oxidizer tanks. The main outlet from the center lobe of the fuel tank is also connected to the two outer cells for complete drainage of the tank. The two oxidizer tanks are structurally connected to the outer lobes of the fuel tank. Each oxidizer tank has a main drain with a connecting line between the two. The single fuel feedline splits aft of the oxidizer tank outlets and each of the two lines feeds five engines as shown in Figure 35, Section E-E. The straight portion of the fuel and oxidizer feedlines have both upper and lower valves to drain the propellant lines as each pair of engines are shut down on ascent, thus minimizing residual propellant weight.

The four dual position nozzle engines are set forward of the six fixed nozzle engines to minimize plume interference after the nozzles are extended. The engine mount beams connnect the oxidizer tank skirts and the wing carrythrough.

Weight			C.G. % Ref Length
Payload	29 483 kg	(65 000 lb)	58.9
Dry weight	202 753 kg	(446 993 lb)	
Landing without payload	207 643 kg	(457 774 lb)	72.7
Landing with payload	237 126 kg	(522 774 lb)	71.0
Ascent propellant	1 660 998 kg	(3 661 873 lb)	
Gross liftoff weight	1 924 654 kg	(4 243 136 lb)	70.1

Volume		
LH_2 tank	3078.5 m^3	(108 712 ft^3)
LO_2 tank	1331.0 m^3	(47 000 ft^3)
Payload		
Diameter	4.572 m	(15 ft)
Length	18.288 m	(60 ft)
Payload Bay Clear Opening		
Diameter	4.725 m	(15.5 ft)
Length	18.517 m	(60.75 ft)

Area		
Body plan area	984.2 m^2	(10 594 ft^2)
Wing, theoretical	1126.0 m^2	(12 120 ft^2)
Wing, exposed	573.0 m^2	(6 168 ft^2)
elevon	181.1 m^2	(1 950 ft^2)
Vertical tail	205.3 m^2	(2 210 ft^2)
rudder	74.3 m^2	(800 ft^2)
Body wetted area	2635.6 m^2	(28 370 ft^2)

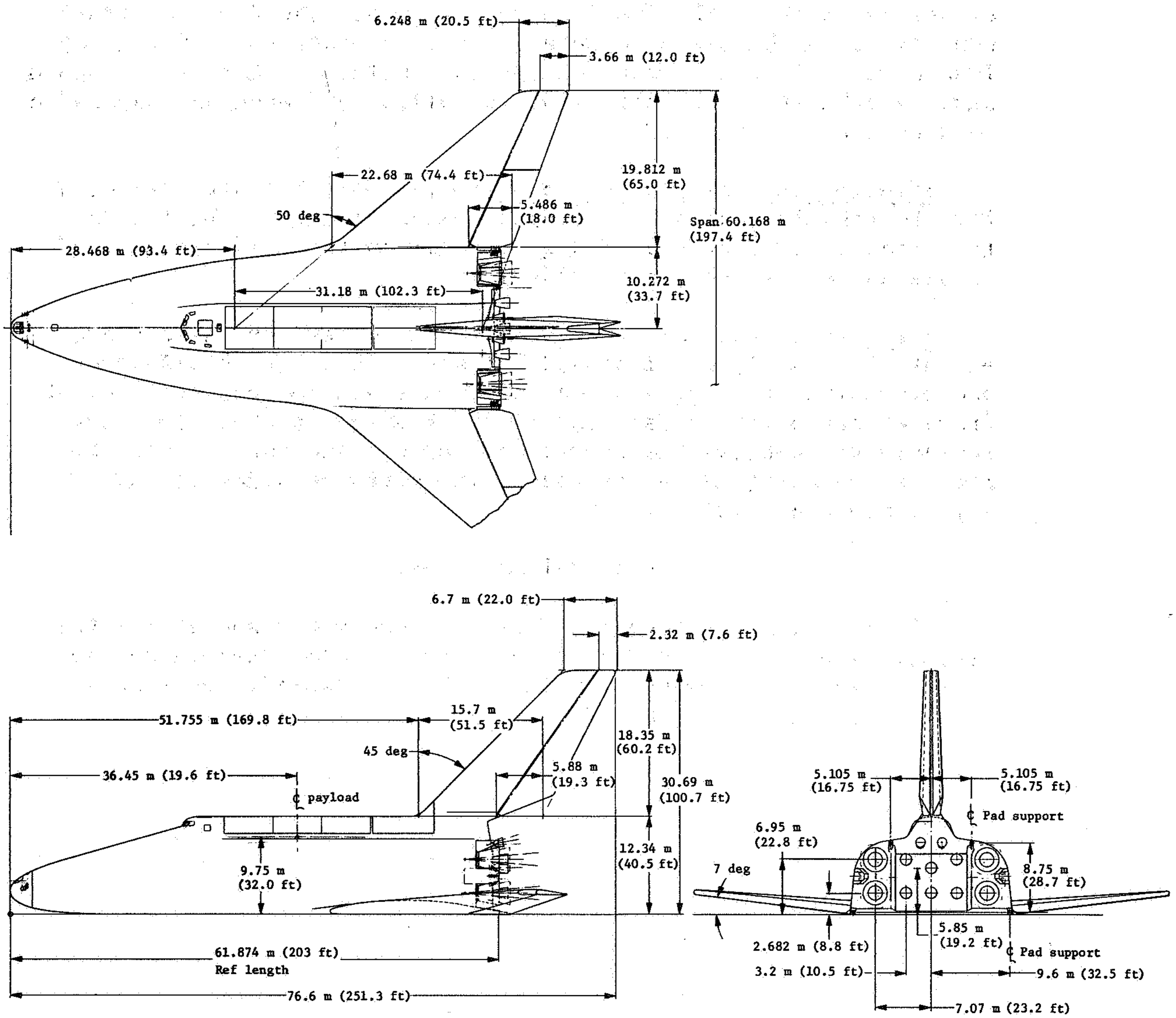


Figure 34.- VTO general arrangement

The main landing gears are nested between the oxidizer tank and the wing closing rib. The nose landing gear is retracted into a cavity in the hydrogen tank center lobe as shown in Figure 35, Section B-B.

Crew and payload module.- The assembly containing the crew compartment, payload bay, OMS propellant tankage, and vertical tail, is a separate module attached primarily at three points to the fuselage/tank module. The crew compartment is similar to the Space Shuttle orbiter crew compartment except for the integral docking facility between the flight deck and the operations deck. The payload bay is adjacent to the operations deck as in the Space Shuttle. The OMS propellant tankage consists of four cylindrical vessels located aft of the payload bay. The support structure for the vertical tail is aft of the OMS tanks and includes the aft structural ties to the fuselage tank module. The forward attachment is at the bulkhead between the crew compartment and the payload bay. This attachment concept is similar to that of the Space Shuttle orbiter to external tank and allows differential expansion between the two modules.

External thermal protection system.- The TPS system selected for the vehicle consists of subpanel-mounted RSI on the fuselage tank module and direct bond RSI isolator on the crew and payload module as well as the aerosurfaces.

Equipment.- Much of the equipment is located at the forward end of the vehicle for improved balance (e.g., electrical power and hydraulic power generation components are located on a pallet frame on the upper forward end of the hydrogen tank). The nose compartment contains the forward RCS module and the two aft RCS modules are attached to the respective outboard sides of the engine mounted bulkhead.

Structural Arrangement

The structural arrangement showing load paths and structural members is presented in Figure 36. The crew and payload module is shown removed from the final assembly to clarify the structural continuity of each module.

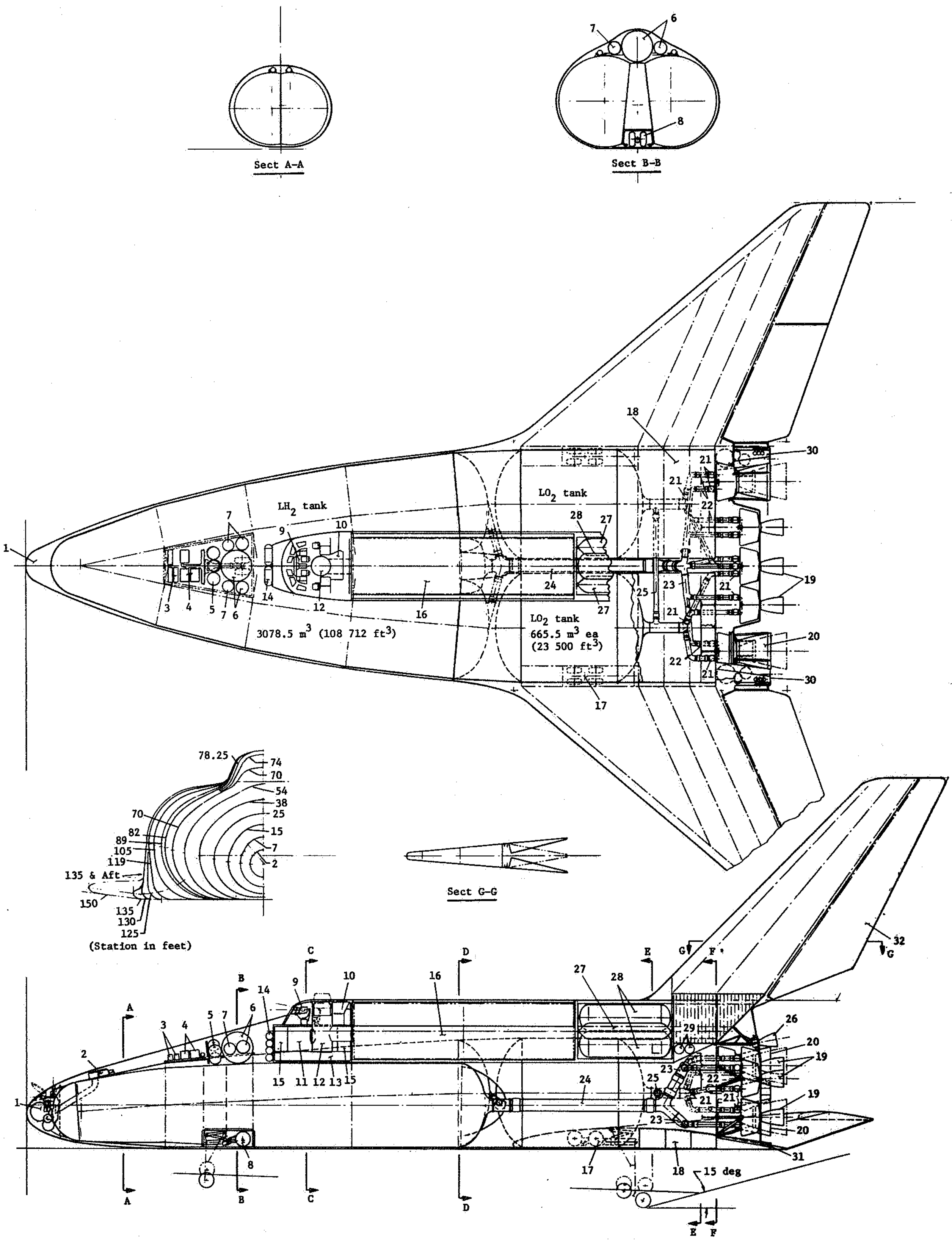
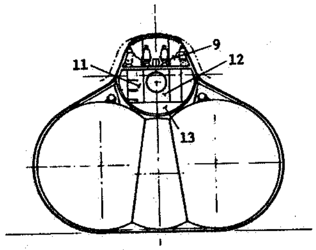
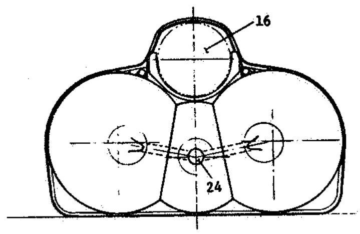


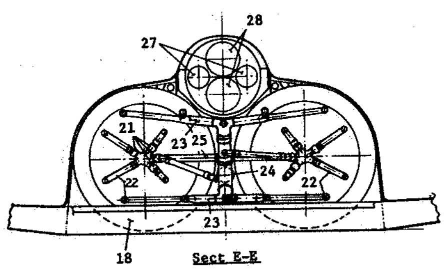
Figure 35.- VTO inboard profile



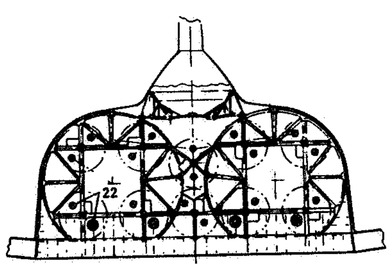
Sect C-C



Sect D-D



Sect E-E



Sect F-F

Legend:

1. Forward RCS module
2. LH_2 tank vent and pressurization valves
3. Electrical power system, fuel cells
4. Power system, APUs
5. Fuel cell propellants (LO_2-LH_2)
6. APU propellant (LO_2-LH_2)
7. Pressurants (He)
8. Nose landing gear
9. Flight deck
10. Operations deck
11. Rest and passenger area
12. Airlock and docking module
13. ECLSS - system
14. ECLSS supply and purge gas tanks
15. Avionics
16. Payload bay
17. Main landing gear
18. Wing carrythrough structure
19. Main propulsion engine, $\varepsilon = 35$, fixed nozzle, not gimballed
20. Main propulsion engine, $\varepsilon = 55/160$, extendable nozzle, gimballed
21. Propellant prevalve
22. Propellant feedlines
23. LH_2, upper and lower feedline manifolds
24. LH_2 main feedline
25. LO_2 tank interconnect line
26. OMS engine, LO_2-LH_2
27. OMS propellant tank, LO_2
28. OMS propellant tank, LH_2
29. OMS pressurant tanks (He)
30. Aft RCS modules
31. Pad support hard points
32. Split rudder

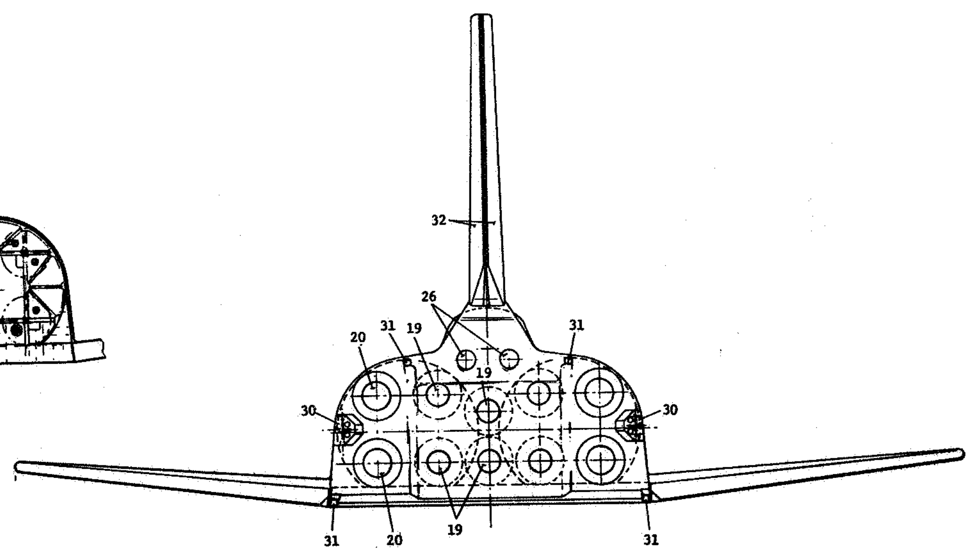


Figure 35.- Concluded

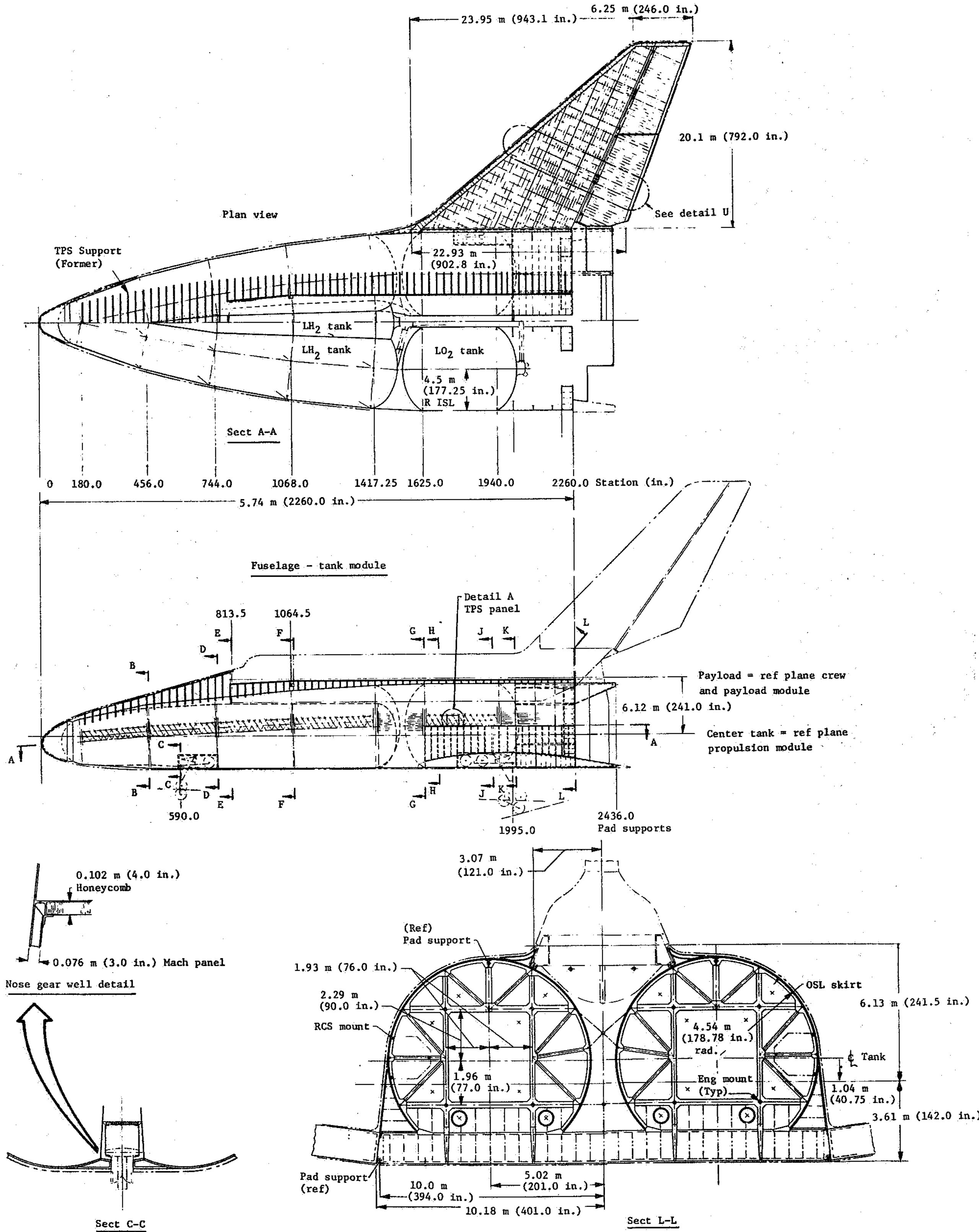


Figure 36.- VTO structural arrangement and details

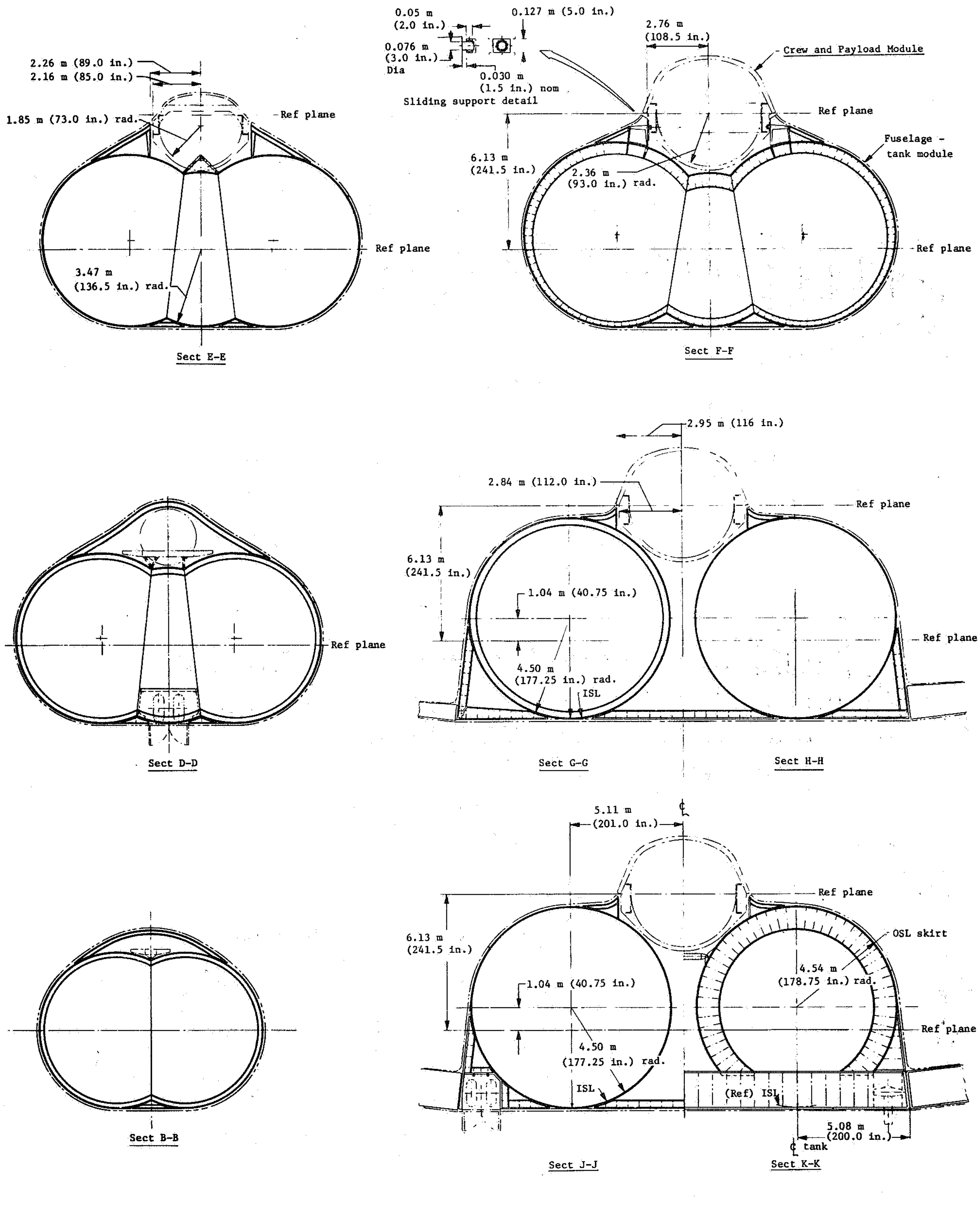
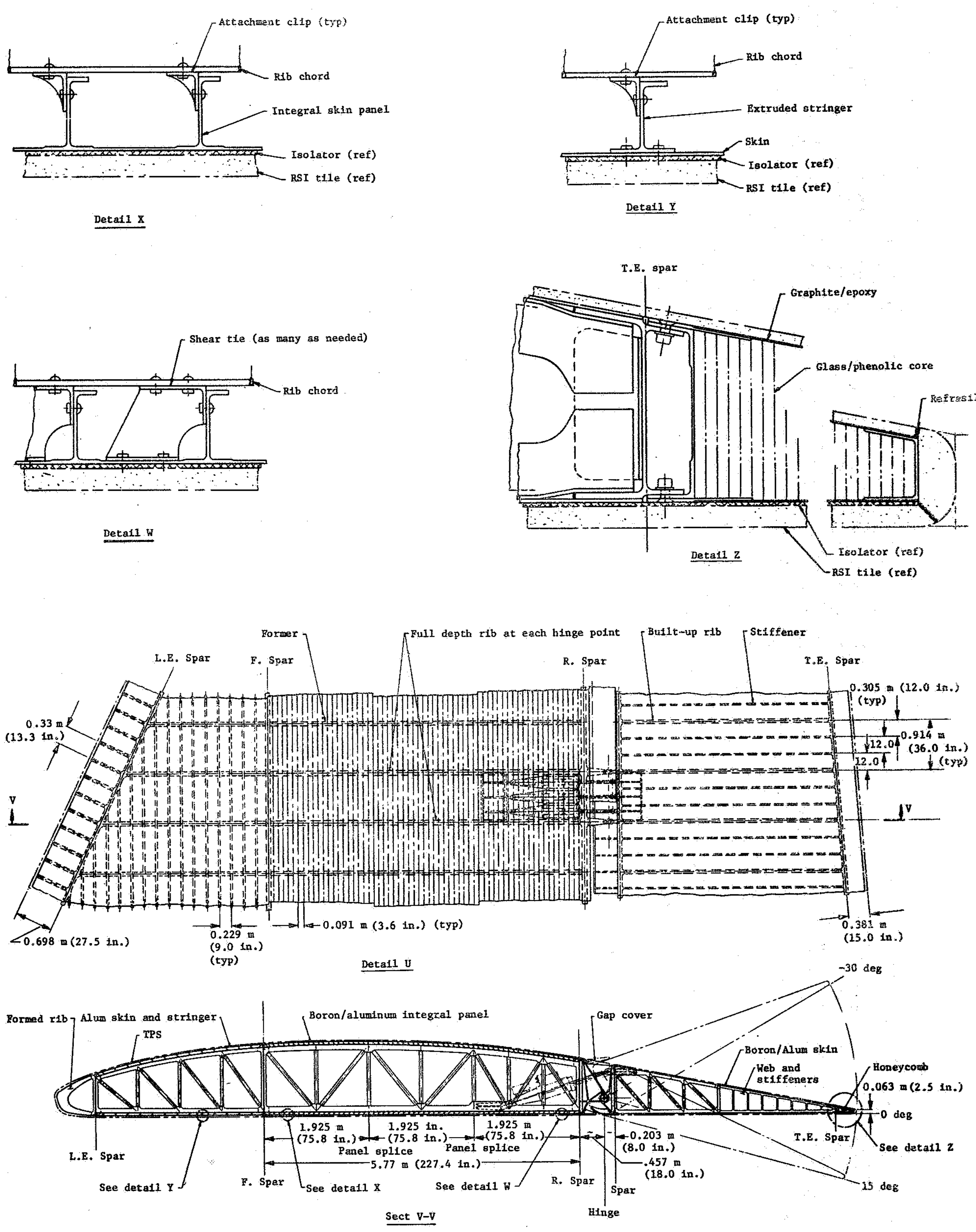


Figure 36.- Continued



WING STRUCTURE DETAILS

Figure 36.- Continued

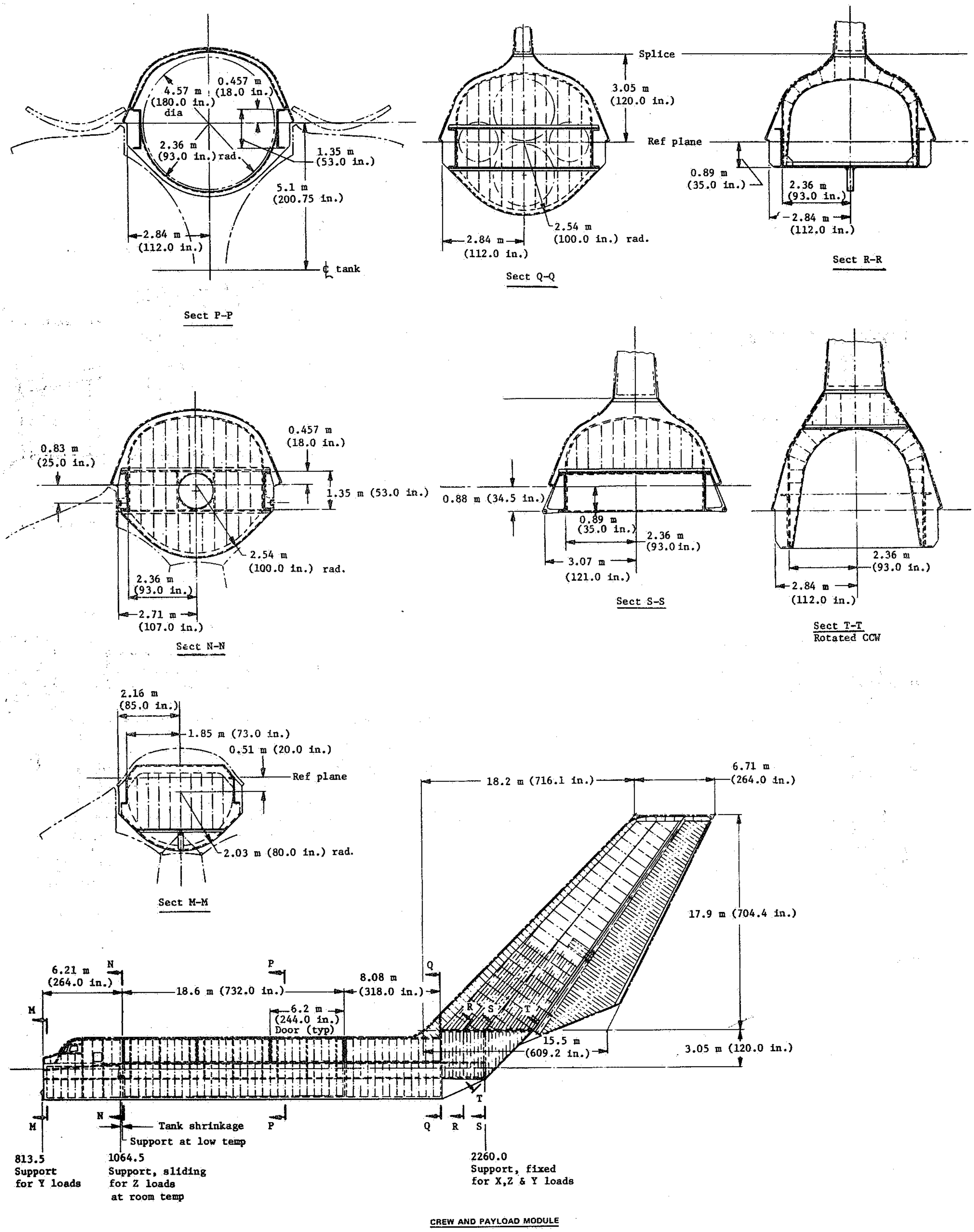


Figure 36.- Continued

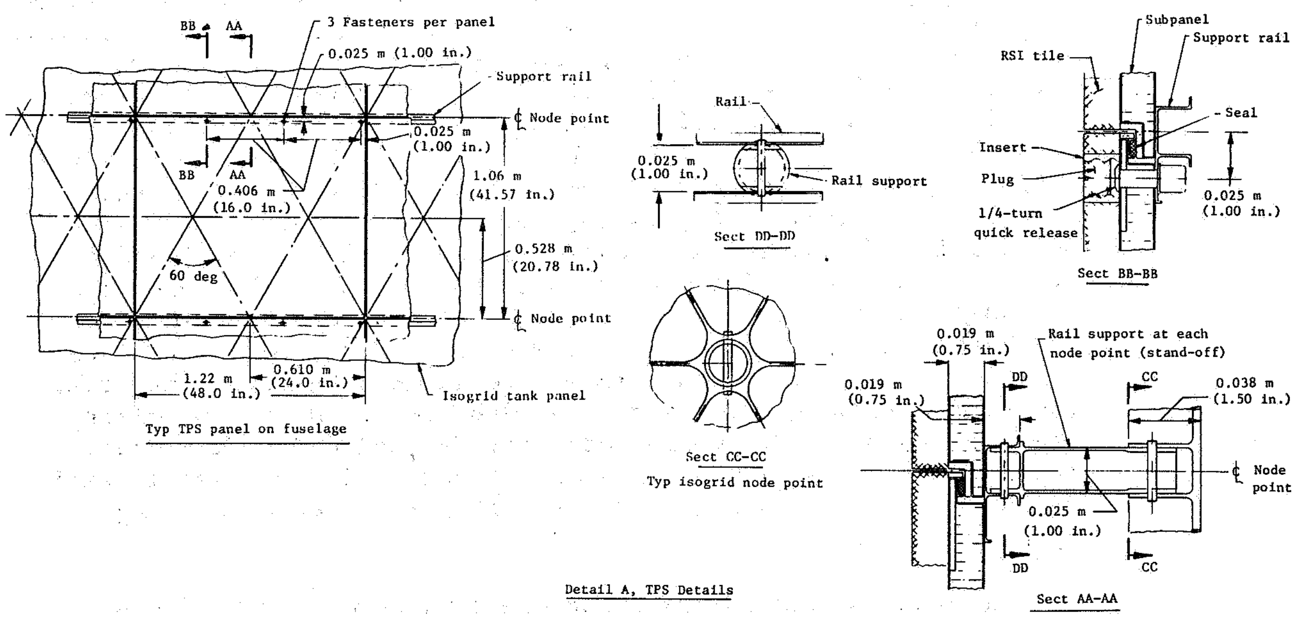


Figure 36.- Concluded

Fuselage tank module.- The LH_2 fuel tank is the forward three-lobed tank. The outer two lobes are connected to the two separate LO_2 oxidizer tanks by two intertank cyclindrical shells. The intertank shells are locally cut out to clear the fuel tank outlet lines. The two oxidizer tanks are separate and provide the load paths between the fuel tank and the aft skirt-engine mount-wing carrythrough structural component. The engine mount beam structure is shown on Figure 36, Section L-L. The horizontal and vertical beams transmit engine loads to the two cylindrical aft skirts and the wing carrythrough torque box. The nose gear is housed in the center lobe of the fuel tank as shown in Figure 36, Section D-D. The gear loads are reacted by the internal tank frames and the two internal lobe intersection beams of the tank. The main landing gears are housed outboard of the two oxidizer tanks as shown in Figure 36, Sections J-J and K-K. Main gear loads are reacted by the frame aft of the oxidizer tanks as well as the beams tied to the oxidizer tank forward bulkhead wing spar connection.

The wing torque box carrythrough structure is aft of the oxidizer tank domes and provides moment and torque continuity between the exposed wing structures. This torque box is shown in Figure 36, Section L-L. The oxidizer tank aft skirt attaches to the upper surface of the torque box as shown.

The four dual-position-nozzle engines are mounted on the engine mount frames and the six fixed-nozzle engines are mounted on trusses that attach to the engine mount beams because of their offset mounting.

The crew and payload module forward attachment to the fuselage tank module is shown in Figure 36, Sections M-M and N-N. The A-frame attachment at Section M-M provides a Y-direction load capability with swivel design to prevent X-direction reaction loads. Section N-N shows a sliding lug design that will transmit Z-direction loads, but not Y or X. The aft attachment is illustrated in Section L-L showing a two-point attachment capable of transmitting X-, Y- and Z-direction loads.

Crew and payload module.- The crew and payload module structure is integrated structurally and consists of the crew compartment, the payload bay with six door sections, the OMS propellant tankage compartment, the vertical tail support structure, and the vertical tail. Figure 36, Sections M-M through S-S shows details of the shell structure, the payload door area, vertical tail-to-support structure continuity, and attachment points between modules. The two OMS engines mount to the aft end of this module.

Wing structure.- Figure 36, Detail U shows a typical area adjacent to an elevon actuator. The elevon design deflections are 15 degrees down to 30 degrees up. Details of the elevon structure and the hinge area are shown in Section V-V. The lower elevon cove gap seal is a flexible curtain and the upper gap is closed by a gap cover flap.

The basic wing structure consists of a torque box 5.78 m (18.9 ft) wide with spar webs at each end as shown in Detail U and Section V-V. The torque box upper and lower cover is an integral stiffened skin. The vertical tail structural concept is similar to the wing concept and is not shown in detail.

Thermal protection system.- The external thermal protection system consists of two basic concepts. The aerosurfaces and the crew and payload module use direct bond RSI tiles with strain isolators as shown in Figure 36, Details W through Z. In areas where the entry temperature is 570 K (600°F) or lower, felt insulation is used; e.g., upper aft wing surface, upper payload shell, and OMS tankage shell.

The fuselage tank module uses the second concept, which is a standoff subpanel-mounted RSI tile design. A typical subpanel-mounted RSI design is shown in Figure 36. The support rails, which run longitudinally, are attached to node points of the integrally stiffened isogrid structure of the propellant tanks as shown in Figure 36, Section AA. The sandwich subpanels are supported on only two sides by the rails with three quick-release fasteners per panel. Details of the fastener access are shown in Section BB. The thermal protection system is designed to limit the primary structure to 450 K (350°F) and the secondary structure (subpanels) to 533 K (500°F).

Structural and thermal protection system materials.- Figure 37 shows the structural and TPS materials of the vehicle. In each case where a specific material or alloy is called out, it is intended to indicate a material family. In some cases future material designations may be changed but they are expected to have family characteristics.

The main fuel and oxidizer tanks are integrally machined welded skins of 2219 aluminum alloy. The payload bay, OMS tankage bay, intertank shells, aft skirt shells, and vertical tail support structure are semimonocoque graphite/epoxy composite construction. The wing and vertical tail are advanced composite construction with borsic/aluminum skins. The selection of borsic/aluminum over graphite/epoxy was based on the advantage of the aluminum heat sink when determining the insulation, requirements.

The subpanels used for the fuselage tank module TPS mounting are sandwich panels of high-modulus graphite faces with glass phenolic cores. The TPS insulation materials are Nomex felt for the upper surfaces and RSI tiles for the lower surfaces and leading edges.

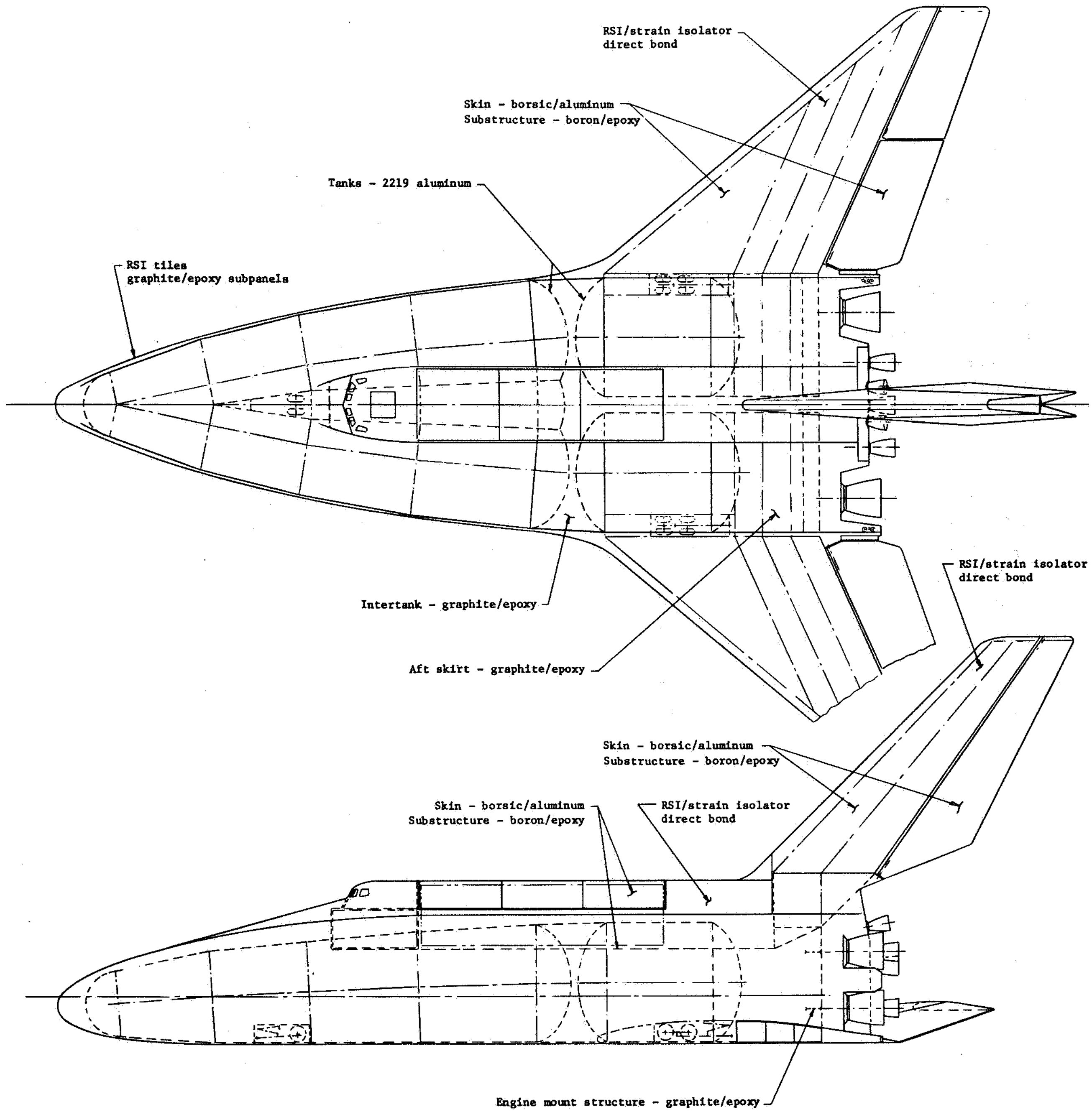


Figure 37.- Materials designation drawing

Propulsion

There are three separate and independent propulsion systems in the VTO vehicle concept: The main propulsion system, the OMS, and the RCS. Each of these is discussed and then the results of analysis of three alternative configurations are presented.

Main propulsion system.- The main propulsion system uses ten engines, each of $2.67(10^6)$N (600 000 lbf) nominal thrust, operating at an O/F ratio of 7.0 and a chamber pressure of $31(10^6)N/m^2$ (4500 psia). The propellants are supplied from an LH_2 tank of 3078 m^3 (108 700 ft^3), and two separate but interconnected LO_2 tanks of 665 m^3 (23 500 ft^3) each. The engines are assumed to be operable at zero NPSH at the engine and feed system interface.

Six of the ten engines are of a nongimbaled fixed-nozzle design. The remaining four outboard engines incorporate a movable nozzle extension to increase the nozzle area ratio for high altitude operation and are gimbal-mounted to provide thrust vector control. Except for these differences, the engines are essentially identical. The engines are sized to provide a 1.3 thrust/weight ratio at takeoff. The maximum acceleration is limited to 3 g by sequentially shutting down engines in symmetrical pairs, starting with shutdown of the fixed-nozzle engines. This procedure obviates any need for engine throttling and permits optimization of the area ratios of the three different nozzle configurations to provide a high average I_{sp}. The operating conditions and performance data for the three engine configurations are given in Table 19.

The propellant tanks are sized for normal boiling point propellants; i.e., bulk densities of 1 134 kg/m^3 (70.9 lbm/ft^3) and 70 kg/m^3 (4.4 lbm/ft^3) for LO_2 and LH_2, respectively, with initial ullage volumes of 3%. A key weight saving feature in the tank design is the minimization of the maximum operating pressure by using a zero-NPSH requirement for the engine inlets. With a zero-NPSH, the design criteria for the pressurization/feed system are suppression of cavitation in the feedlines and maintenance of a positive gage pressure at all times. A design value of 138 000 N/m^2-gage (20 psig) was used as the maximum working pressure for the tanks. Assuming propellants are saturated at near atmospheric pressure at launch, gives an allowance of 34 500 N/m^2 (5 psi) for inflight propellant temperature stratification, pressure regulator tolerance, and net feed system friction loss minus hydrostatic pressure gain. Because of the vehicle size and arrangement, the hydrostatic pressure component will make a significant contribution toward overcoming friction loss.

TABLE 19.- VTO ENGINE PERFORMANCE DATA

Nozzle type	Fixed	Dual	
Number per vehicle	6	4	
Engine weight - kg (lbm)	3070 (6769)	4120 (9084)	
Propellant flow rate - kg/sec (lbm/sec)	625 (1377)	625 (1377)	
LO_2 flow rate - kg/sec (lbm/sec)	547 (1205)	547 (1205)	
LH_2 flow rate - kg/sec (lbm/sec)	78 (172)	78 (172)	
Chamber pressure - 10^6 N/m^2 (psia)	31 (4500)	31 (4500)	
Throat Area - m^2 (in.^{2})	0.0424 (65.8)	0.0424 (65.8)	
Throat diameter - m (in.)	0.232 (9.15)	0.232 (9.15)	
Expansion ratio	35	55	160
Exit area - m^2 (in.^{2})	1.49 (2300)	2.33 (3620)	6.79 (10 530)
Exit diameter - m (in.)	1.38 (54)	1.72 (68)	2.94 (116)
Thrust, S.L. - 10^3N (10^3 lbf)	2470 (556)	2420 (544)	----
Thrust, vacuum - 10^3N (10^3 lbf)	2670 (600)	----	2840 (638)
I_{sp}, S.L. - sec	404.1	395.5	----
I_{sp}, vacuum - sec	436.1	----	463.5

The LH_2 feed system (Figure 35) consists of three outlets - one in each of the three tank lower dome segments - all feeding to a single main feedline. This main line then goes through a series of bifurcations to individual feedlines for each engine. The LO_2 system is similar except that it has two main feedlines, one for each tank, with a crossover between them. The LH_2 system is vacuum-jacketed to eliminate air condensation and to minimize pre-start propellant conditioning requirements. The LO_2 system is foam insulated. Isolation valves are located at the upstream end of each individual engine feedline to permit draining those lines through the engine after each engine is shut down. This, together with the sequence of engine shutdowns (i.e., the outboard engines being the last), ensures a minimum of trapped propellant in the feed system.

For purposes of this study, the feedlines were designed for maximum flow velocities of 4.9 m/sec (16 ft/sec) for LO_2 and 11.3 m/sec (37 ft/sec) for LH_2. These velocities result in equal diameters for both the LO_2 and LH_2 systems.

The pressurization system is autogenous, using hot propellant vapors bled from the engines. Except for the lower tank pressures, the pressurization system is assumed to be similar to that for the Space Shuttle external tank. Correcting for the pressure differences, this leads to pressurant densities at burnout of 2.0 kg/m^3 (0.125 lbm/ft^3) for the LO_2 tank and 0.176 kg/m^3 (0.011 lbm/ft^3) for the LH_2 tank. The corresponding average temperatures are 264°K (475°R) and 190°K (342°R), and the total pressurant weights are 2 665 kg (5 875 lbm) and 542 kg (1 196 lbm), respectively.

Although the vehicle designs use a 31 (10^6) N/m^2 (4 5000 psia) chamber pressure for the main engines (refer to page 20), later discussions with consulting rocket engine firms indicated that the projected pressure is optimistic. Subsequent studies were made using engines with 27.6 (10^6) N/m^2 (4 000 psia) champber pressure. These studies showed that the somewhat larger envelop dimensions could be accommodated and that by modifying the nozzle expansion ratios the vehicle performance could be maintained equal to that for the higher pressure engine. The modified expansion ratios are 55 for the fixed nozzle and 40/160 for the dual nozzle. The ratios with the 27.6 (10^6) N/m^2 (4 000 psia) pressure were used in the remainder of the study using all LO_2/LH_2 engines.

OMS.- The OMS consists of two LO_2/LH_2 pump-fed engines of 66 700 N (15 000 lbf) thrust each, operating at a 6 O/F ratio with a steady-state I_{sp} of 440 seconds. These engines are also assumed to be operable at zero NPSH, but in all other respects they are the same as the existing RL-10. Propellants are supplied from the LO_2 and LH_2 tanks which are sized for a ΔV of 381 m/sec (1250 ft/sec) using normal boiling point densities. The resultant tank data are tabulated as follows:

OMS tanks	Propellant wt		Tank volume	
	kg	lbm	m^3	ft^3
LO_2 (each tank)	9 200	20 300	8.5	300
LO_2 (total)	18 400	40 600	17.0	600
LH_2 (each tank)	1 540	3 400	22.6	800
LH_2 (total)	3 080	6 800	45.2	1 600
Total propellant*	21 480	47 400	62.2	2 200
*Sized for ΔV = 381 mps (1250 fps)				

The OMS propellant tanks are pressurized with He stored at ambient temperature and a pressure of $27.6(10^6)$ N/m^2 (4000 psia). The size of such a pressurization system depends essentially on the difference between tank total pressure and liquid vapor pressure, which for a zero-NPSH engine requirement, need only be sufficient to overcome the friction and transient start losses of the feed system. For this study, these pressure losses were taken to be 10 300 N/m^2 (1.5 psi) for the LO_2 tank, and 6900 N/m^2 (1.0 psi) for the LH_2 tank. This leads to a usable He requirement of 7.7 kg (17 lbm). Using the system weight to usable He weight ratio of 20 from the Space Shuttle OMS results in a pressurization system weight of 154 kg (340 lbm). The propellant tanks are designed for a maximum operating pressure of 138 000 N/m^2 (20 psia) so the allowable rise in propellant vapor pressure over the duration of the mission is 24 100 N/m^2 (3.5 psi) for LO_2 and 27 600 N/m^2 (4.0 psi) for LH_2. The OMS net system I_{sp} is 419 sec when loaded for a ΔV of 381 m/sec (1250 ft/sec). (Net system I_{sp} is the ratio of total impulse to total weight of the system). When off-loaded for a ΔV of 198 m/sec (650 ft/sec), the net system I_{sp} drops to 401 seconds.

RCS. The rCS consists of three similar units - one mounted in the vehicle nose and the other two located outboard of the main engine cluster (Figure 35). Each of these provides one-third of the total RCS ΔV capability of 30.5 m/sec (100 ft/sec). The propellants are O_2 and H_2 that are supplied to the thrusters as gases at an O/F ratio of 4.5 from accumulators at a pressure of 1.38 (10^6) N/m^2 (200 psia). The accumulators are replenished through a pump and evaporator-heater from low-pressure cryogenic storage tanks. The RCS propellant tank data are as follows:

RCS tanks	Propellant wt		Tank volume	
	kg	lbm	m^3	ft^3
LO_2 (each tank)	540	1190	0.50	17.5
LO_2 (total)	1620	3570	1.50	52.5
LH_2 (each tank)	118	260	1.74	61.5
LH_2 (total)	354	780	5.22	184.5
Total Propellant	1974	4350	6.72	236.5

These tankage requirements are based on an average thruster I_{sp} of 390 sec, normal boiling point propellants, and 4% initial ullage. The pumps are assumed to operate submerged at zero NPSH, so no pressurization of the liquid propellant tanks is required. Detailed analyses of the accumulator, pump, and heat exchanger requirements will depend on the system duty cycle. For this study, the system dry weight is estimated to be 480 kg (1060 lbm) per module, or 1440 kg (3180 lbm) total. The net system I_{sp} is 220 seconds.

Internal Loads Analysis

The VTO fuselage tank module was mathematically modeled using the Martin Marietta-Denver Space Frame Program (MDSFP). This finite element program uses the stiffness method to compute deflections and rotations of each node point for the applied loading condition and then calculate the compatible internal loads and stresses in each structural element. The model contains 239 node points with 1356 degress of freedom (See Figure 38.). Taking advantage of symmetry, only one-half of the structure was modeled to minimize computer costs. The 25 node points located on the plane of symmetry were fixed in the Y, θ_X, and θ_Z directions because these values must be zero for symmetrical structure with a symmetrical loading. Two X deflections and one Z deflection were fixed to provide X, Z, and θ_Y overall stability.

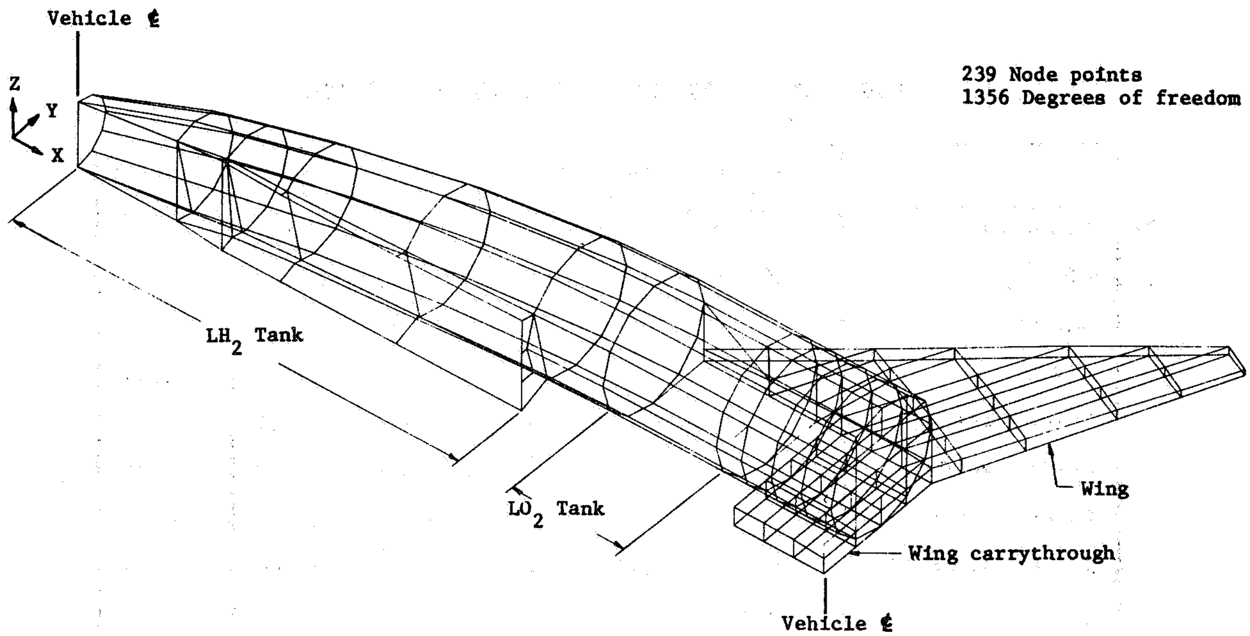


Figure 38.- Finite element model

Twelve bulkhead stations were used with node points typically spaced at 30 degree intervals circumferentially. The aft bulkhead was modeled in sufficient detail so engine loads could be applied at the proper loactions. Node points were also provided for the crew and payload module attachment loads and landing gear loads. Forty-two of the node points were in the wing outboard of the root rib. A total of 524 beam elements (axial, two bending, two shear, and torsional stiffness), 30 triangular plates and 269 quadrilateral plates (axial, inplane shear, and inplane bending stiffness) were used. The number of triangular panels was minimized because quadrilateral panels provide considerably more accurate results for a given number of node points. Because the quadrilateral plates must lie in a plane, it was necessary to use modified wing depths to define a set of upper node points to eliminate the panel curvature. The main beam depths and torque box areas were closely approximated.

Table 20 lists parametric data for the five major loading conditions that were considered. For each loading condition, the applied engine, gear, and/or airloads were distributed to the node points to give the proper moment about the VTO center of gravity. The weights were also distributed to the node points and then the balancing inertial reactions were calculated. The resultants of the applied and inertial reactions were input to the MDSFP for each overall loading condition. The resulting internal stresses were reviewed to ensure that no large inaccuracies existed in the structural element mechanical properties that were input to the program.

TABLE 20.- EXTERNAL LOADING CONDITIONS

	Maximum $q\alpha$ headwind	Maximum n_x ascent	Entry	2.5-g maneuver	Two-wheel landing
Mach number	1.53	5.93	16.0	0.6	
q,N/m^2	30 595	5 003	5 861	13 885	
(q,psf)	(639)	(104.5)	(122.4)	(290)	
α, deg	3.57	7.4	30	6.3	15.0
Vehicle mass, kg	1 424 500	919 200	237 200	237 200	237 200
(Vehicle weight, lb)	(3 140 500)	(2 026 400)	(523 000)	(523 000)	(523 000)
Normal airload, kg	823 900	62 500	521 900	593 100	245 600
(Normal airload, lb)	(1 816 507)	(137 813)	(1 150 600)	(1 307 500)	(541 450)
n_x	1.567	3.0			
n_z	0.578	0.068	2.2	2.5	1.794

Some of the more critical internal loads are given in Table 21. The results of this analysis were used to confirm that the sizing of the vehicle structural elements was correct and that their weights were properly represented in the mass properties analysis. The only structural modification that was indicated was a potential small reduction in wing weight because wing loads at maximum $q\alpha$ were less than the load capability of the wing.

TABLE 21.- INTERNAL LOADS SUMMARY

Vehicle location	Ultimate load kN/m (lb/in.)			
	Loading condition			
	Maximum $q\alpha$	Maximum longitudinal acceleration	2.5 g maneuver	Two-wheel landing
Inner tank	439 (2 505)	181 (1 031)	94 (535)	66 (377)
Aft skirt	809 (4 620)	712 (4 067)	167 (953)	132 (754)
Exposed wing root	2353 (13 440)	454 (2 590)	1201 (6 860)	199 (1 135)
Exposed wing mid-span	824 (4 703)	106 (603)	412 (2 353)	68 (390)

Mass Properties

Vehicle mass properties are based primarily on the Task 1 nominal projections for weight estimating relationships; however, where the internal loads generated by the finite element analysis indicated necessary changes, the Task 1 projections were modified.

The vehicle sized in this study is based on the initial aerodynamics estimate. The effects of revised aerodynamic characteristics are reported in the vehicle comparison study. The mass properties summary table indicates a payload of 29 484 kg (65 000 lb), with the increased performance capability, based on revised aerodynamics, shown as increased payload.

TABLE 22.- VTO MASS PROPERTIES

Code	System	Mass, kg	Weight, lb
1.0	Wing group	23,502	51 813
2.0	Tail group	5 265	11 607
3.0	Body group	52 873	116 565
4.0	Induced environmental protection	39 432	86 933
5.0	Landing & auxiliary systems	7 304	16 103
6.0	Propulsion-ascent	41 896	92 364
	6.1 Engine accessories	2 175	4 796
	6.2 Propellant system	4 816	10 618
	6.3 Engines (10)	34 904	76 950
7.0	Propulsion - RCS	1 444	3 183
8.0	Propulsion - OMS	1 032	2 275
9.0	Prime power	1 674	3 690
10.0	Electrical conversion & distribution	2 975	6 560
11.0	Hydraulic conversion & distribution	2 903	6 400
12.0	Surface controls	2 480	5 468
13.0	Avionics	2 096	4 622
14.0	Environmental control	1 836	4 048
15.0	Personnel provisions	499	1 100
18.0	Payload provisions	270	595
19.0	Margin	15 272	33 668
	Total dry weight	202 753	446 993
20.0	Personnel	1 199	2 644
23.0	Residuals and gases	3 691	8 137
	Landing weight	207 643	457 774
22.0	Payload	29 484*	65 000*
	Landing weight with payload	237 127	522 774
23.0	Residuals dumped	6 866	15 138
25.0	Reserve fluids	4 899	10 800
26.0	Inflight losses	1 613	3 555
27.0	Ascent propellant	1 660 998	3 661 873
28.0	Propellant - RCS	1 972	4 348
29.0	Propellant - OMS	11 179	24 647
	GLOW	1 924 654	4 243 136

Center of gravity: % of body length

Condition	X
Dry	73.315
Landing	73.029
Landing with payload	71.276
Liftoff	70.226

Moment of inertia:

Condition	Ix kg-m^2	Ix (slug-ft^2)	Iy kg-m^2	Iy (slug-ft^2)	Iz kg-m^2	Iz (slug-ft^2)
Dry	16 619 450	(12 257 880)	66 183 127	(48 814 179)	73 201 894	(53 990 957)
Landing	16 737 045	(12 344 613)	67 109 977	(49 497 789)	74 162 125	(54 699 187)
Landing with payload	17 410 479	(12 841 313)	70 684 913	(52 134 527)	77 140 569	(56 895 975)
Liftoff	57 289 016	(42 254 318)	183 726 609	(135 509 820)	229 046 721	(168 936 230)

Product of inertia:

Condition	Pxy kg-m^2	Pxy (slug-ft^2)	Pxz kg-m^2	Pxz (slug-ft^2)	Pyz kg-m^2	Pyz (slug-ft^2)
Dry	22 363	(16 494)	-136 353	(-100 569)	-4 344	(-3 209)
Landing	22 157	(16 342)	-288 865	(-213 056)	-4 307	(-3 177)
Landing with payload	20 884	(15 403)	-1 372 522	(-1 012 321)	-3 608	(-2 661)
Liftoff	20 123	(14 842)	-301 767	(-222 572)	-4 591	(-3 386)

*Revised aerodynamics capability is 32 493 kg (71 600 lb).

SLED LAUNCH VEHICLE (HTO) DESIGN

The design approach for the HTO vehicle was to use a rail-mounted sled to accelerate the HTO vehicle to its initial lift-off velocity of Mach 0.6. Vehicle thrust to weight, pullup acceleration, duration of constant load factor, inertial angle of attack rates, duration of angle of attack rates and inertial pitch rate were varied in the POST-ascent trajectory program to optimize the vehicle.

The engines of the flight vehicle are started at the same time as the sled starts so that they will be at full thrust before releasing from the sled. The sled is powered by two F-1 engines. The maximum acceleration during the sled run is 1.32 g and the vehicle thrust to weight was optimized at 0.95. The track length was set at 4267 m (14 000 ft) with half being used for acceleration and liftoff and the other half used as a water brake decelerator.

Sled Concept

The accelerator sled is designed as a flat low-drag body of 61.0 m (200 ft) length and 22.0 m (72 ft) width. It rides on two rails with lubricated slide shoes. RP-1 fuel and liquid oxygen tanks sized for 21 seconds thrust are provided in the sled for the two F-1 engines (See Figure 39.). The engines have a combined sea level thrust of 13.5 (10^6) N (3.04 (10^6) lbf). These engines operate at 6.9 (10^6) N/m^2 (1 000 psia) chamber pressure at an O/F ratio of 2.27. They deliver a sea level I_{sp} of 266 seconds with an expansion ratio of 16. The sled has three scoops and water ducts so the gradually down-sloping rail brings the brake scoops into the water in the three troughs and the water is deflected by the ducts to provide constant deceleration to the sled vehicle.

The HTO vehicle is towed onto the sled in the horizontal position on its landing gear. The main landing gear will be resting on platforms that can be moved laterally to align the aft vehicle supports with the erected aft tripods. The platforms will then be lowered and the vehicle locked in position in the aft supports. The forward inverted V-strut forms a scissors arrangement with an auxiliary strut and engages the forward support in the vehicle. With a cable winch mechanism, the HTO vehicle is erected to an 8 degree incidence angle, the support strut is locked in place, and the auxiliary strut is retracted. The landing gear assemblies are retracted into the vehicle and the propellant tanks are filled in this position. At launch, the forward strut swings out of the way as the two aft thrust mounts are released.

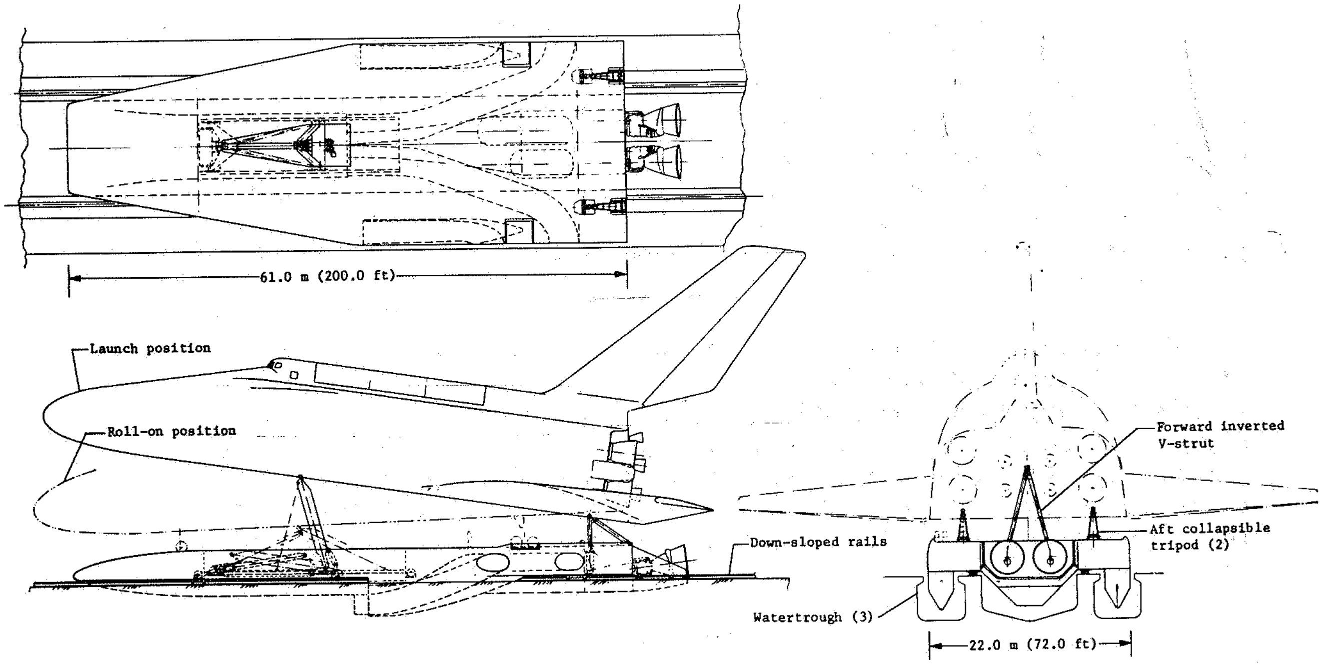


Figure 39.- HTO sled concept

Considerations of Propellant in Wings

Using the wing internal volume to store oxidizer propellants was evaluated for the HTO vehicle. Approximately 60% of the exposed wing total volume is usable for propellant loading. Major advantages of using this available volume for oxidizer propellant are to obtain wing bending load relief and to increase overall packaging efficiency. The wing bending load relief is most effective on the horizontal takeoff vehicles and reduces the overall wing weight by approximately 18%, resulting in a vehicle dry weight decrease of approximately 13%.

Critical design areas that should be further investigated for the use of cryogenic propellants in the wings are as follows:

(1) Tank ullage pressure plus the acceleration head of the propellants require increased wing shell unit weights over conventional wet-wing design with jet propellants.

(2) Flow of the propellant from the wing tanks to the engines is more complex, resulting in increased residual propellant weight.

(3) The dead weight of the wings with the large propellant load must be supported efficiently by the sled to enable the flight vehicle to profit by the load relief during flight.

(4) The external insulation of the wing tank area may require a subpanel mounted RSI concept to facilitate leakage inspection of the wing tankage after flights. This design requirement would add approximately 2720 kg (6000 pounds) of TPS weight to the vehicle.

Vehicle Design

The HTO dry-wing vehicle is shown in Figure 40. The vehicle is longer than the VTO by 18 m (59 ft) and has a 2.6 m (8.5 ft) greater wing span. Wing and vertical tail sweep angles are the same as the VTO. Due to the lower vehicle initial thrust to weight of 0.95, eight engines were sufficient.

The inboard profile drawing of the HTO (Figure 41) is similar to that previously shown for the VTO. The major components are identical in concept and equipment locations are relatively the same. The sled support points are located at Sections C-C and F-F. The feedline configurations have been changed to reflect the change from ten to eight engines.

The structural arrangement of the sled launched HTO vehicle is similar to the basic concept of the VTO vehicle and the same thermostructural concepts were used. The differences are reflected in Figure 42. The forward sled V-strut loads are introduced in the fuel tank as shown in Sections B-B and E-E. A structural bulkhead is located at this station. The aft sled tripod mounts are shown in Sections C-C and D-D. The thrust loads are introduced in the aft lower end of the wing carry-through torque box and are aligned with the vertical engine mount beams. The engine mount beams are configured for the eight engines as shown in Section C-C.

Propulsion

The main propulsion system uses eight engines. Four of the engines are fixed nozzle and four are dual nozzle, with the dual-nozzle engines gimbal-mounted. Engine data are given in Table 23.

The main LO_2 feedlines used in the VTO vehicle are eliminated and the individual engine feedlines are supplied for sumps in the bottoms of the two LO_2 tanks. A crossover line connects these two sumps.

Weight			C.G. % Ref Length
Payload	29 483 kg	(65 000 lb)	
Dry weight	225 121 kg	(496 307 lb)	
Landing without payload	230 486 kg	(508 134 lb)	73.25
Landing with payload	259 969 kg	(573 134 lb)	71.63
Ascent propellant	1 817 462 kg	(4 006 819 lb)	
Launch propellant	100 326 kg	(221 181 lb)	
Liftoff weight	2 106 196 kg	(4 643 368 lb)	70.09
Launch gross weight	2 206 522 kg	(4 864 549 lb)	69.60

Area		
Body plan area	1071.5 m^2	(11 534 ft^2)
Wing, theoretical	1225.9 m^2	(13 195 ft^2)
Wing, exposed	623.8 m^2	(6 715 ft^2)
Elevon	197.2 m^2	(2 123 ft^2)
Vertical tail	223.5 m^2	(2 406 ft^2)
Rudder	80.9 m^2	(871 ft^2)
Body wetted area	2869.3 m^2	(30 885 ft^2)

Volume		
LH_2 tank	3554.3 m^3	(125 520 ft^3)
LO_2 tank	1536.6 m^3	(54 266 ft^3)
Payload		
Diameter	4.572 m	(15 ft)
Length	18.288 m	(60 ft)
Payload bay clear opening		
Diameter	4.725 m	(15.5 ft)
Length	18.517 m	(60.75 ft)

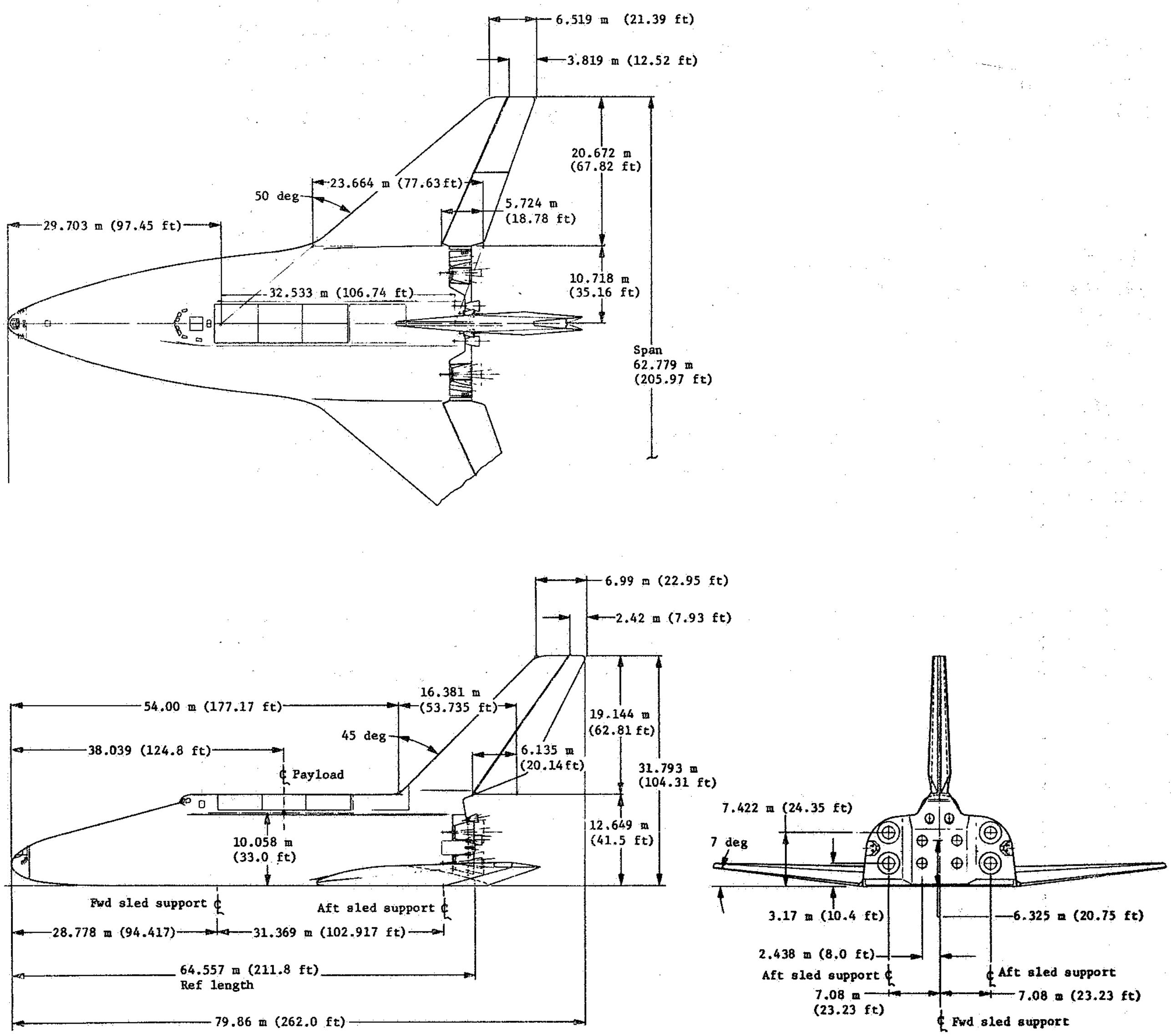


Figure 40.- HTO general arrangement

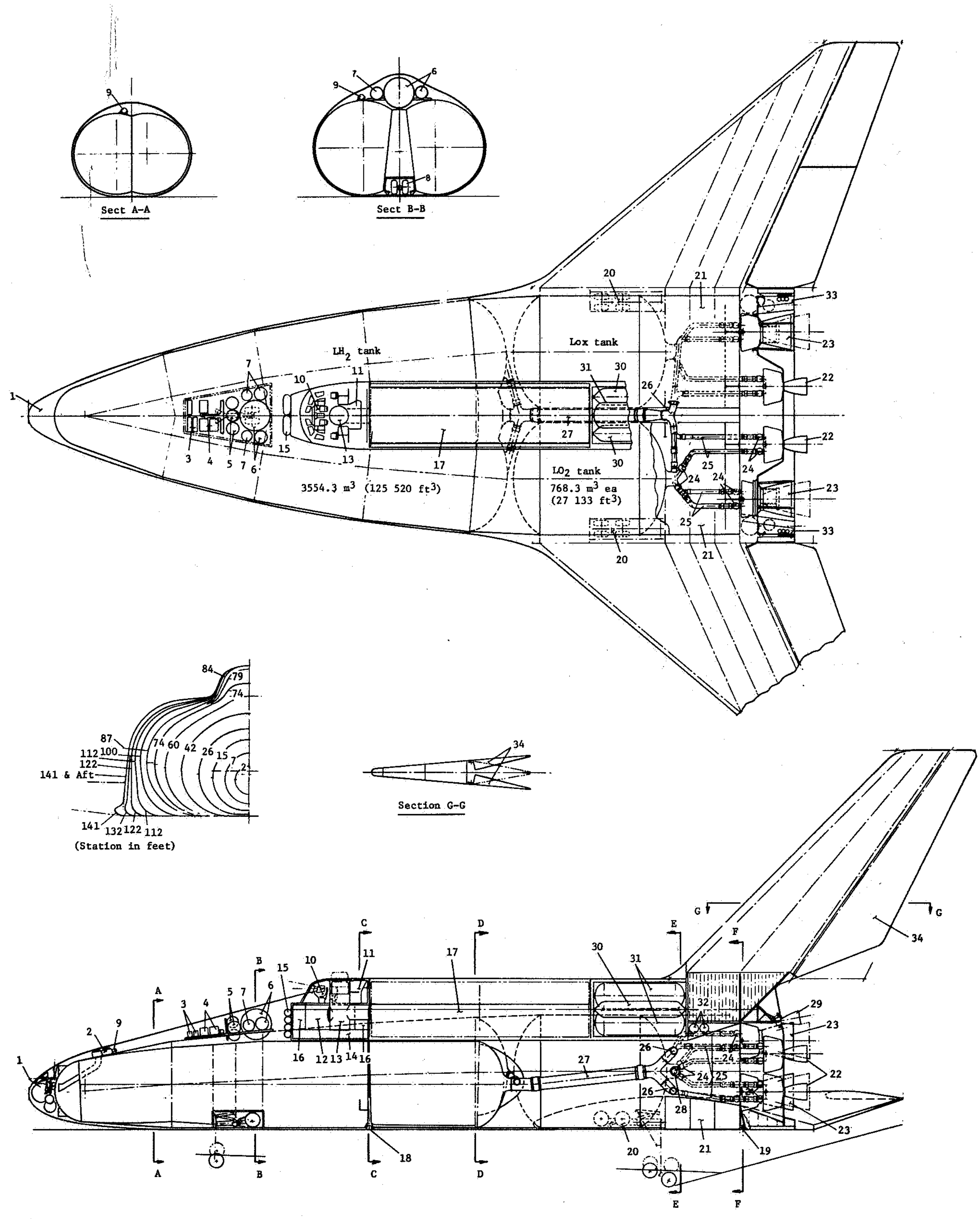
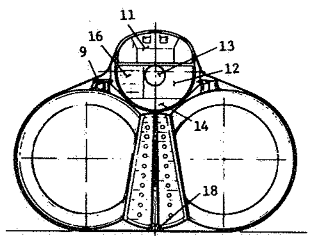
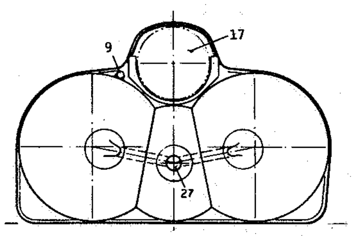


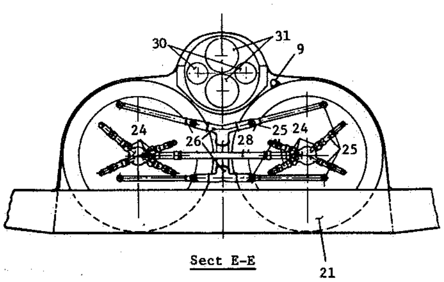
Figure 41.- HTO inboard profile



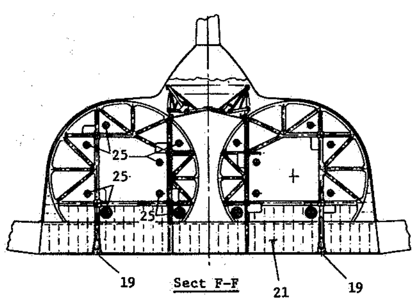
Sect C-C



Sect D-D



Sect E-E



Sect F-F

Legend:

1. Forward RCS module
2. LH_2 tank vent and pressurization valves
3. Electrical power system, fuel cells
4. Power system, APUs
5. Fuel cell propellants (LO_2 - LH_2)
6. APU propellant (LO_2 - LH_2)
7. Pressurants (He)
8. Nose landing gear
9. LH_2 tank pressurization line
10. Flight deck
11. Operations deck
12. Rest and passenger area
13. Airlock and docking module
14. ECLSS - system
15. ECLSS supply and purge gas tanks
16. Avionics
17. Payload bay
18. Forward sled support point
19. Aft sled support points
20. Main landing gear
21. Wing carrythrough structure
22. Main propulsion engine, ϵ = 35, fixed nozzle, not gimbaled
23. Main propulsion engine, ϵ = 55/160, extendable nozzle, gimbaled
24. Propellant prevalve
25. Propellant feedlines
26. LH_2, upper and lower feedline manifolds
27. LH_2 main feed line
28. LO_2 tank interconnect line
29. OMS engine, LO_2-LH_2
30. OMS propellant tank, LO_2
31. OMS propellant tank, LH_2
32. OMS pressurant tanks (He)
33. Aft RCS modules
34. Split rudder

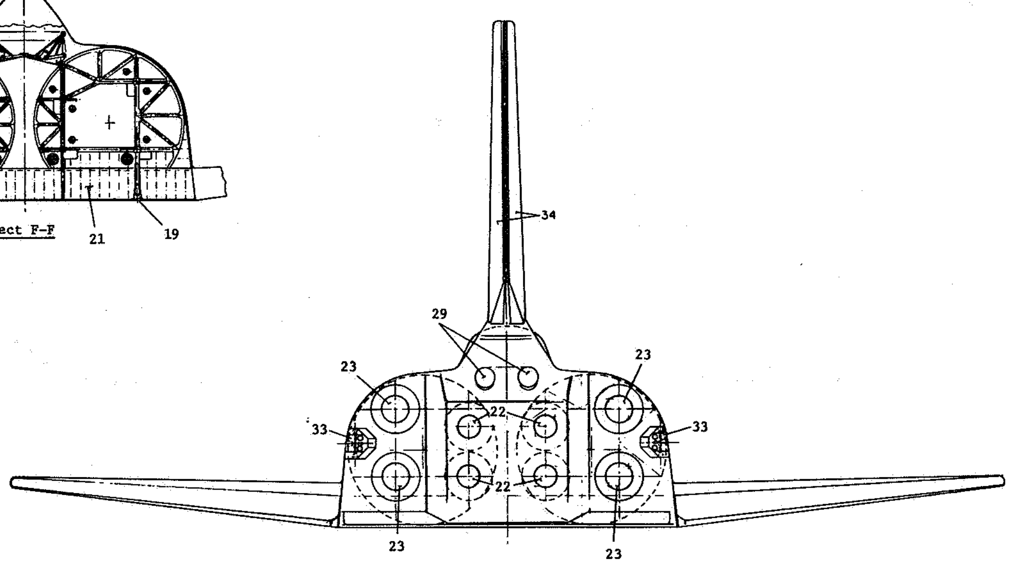


Figure 41.- Concluded

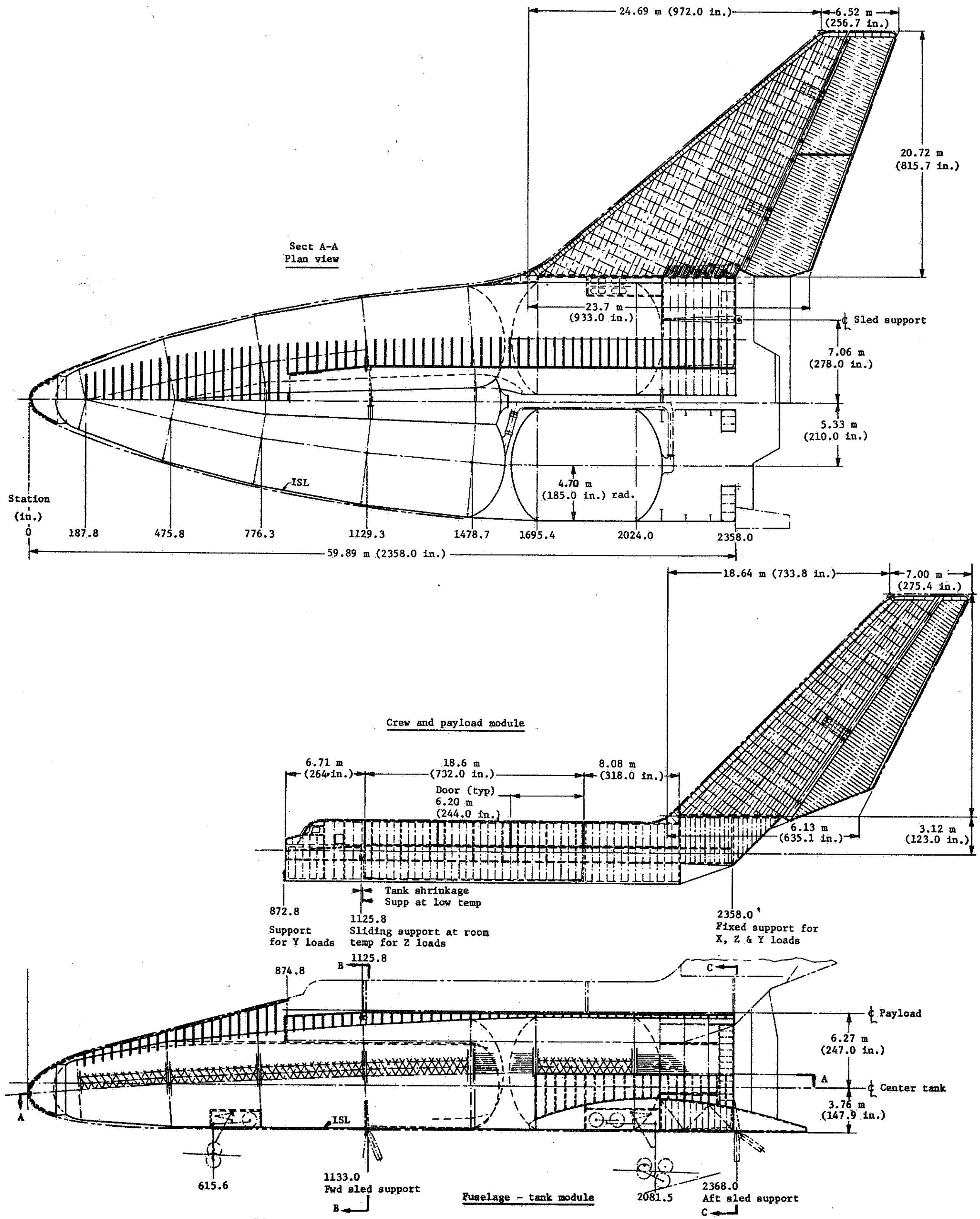


Figure 42.- HTO structural arrangement

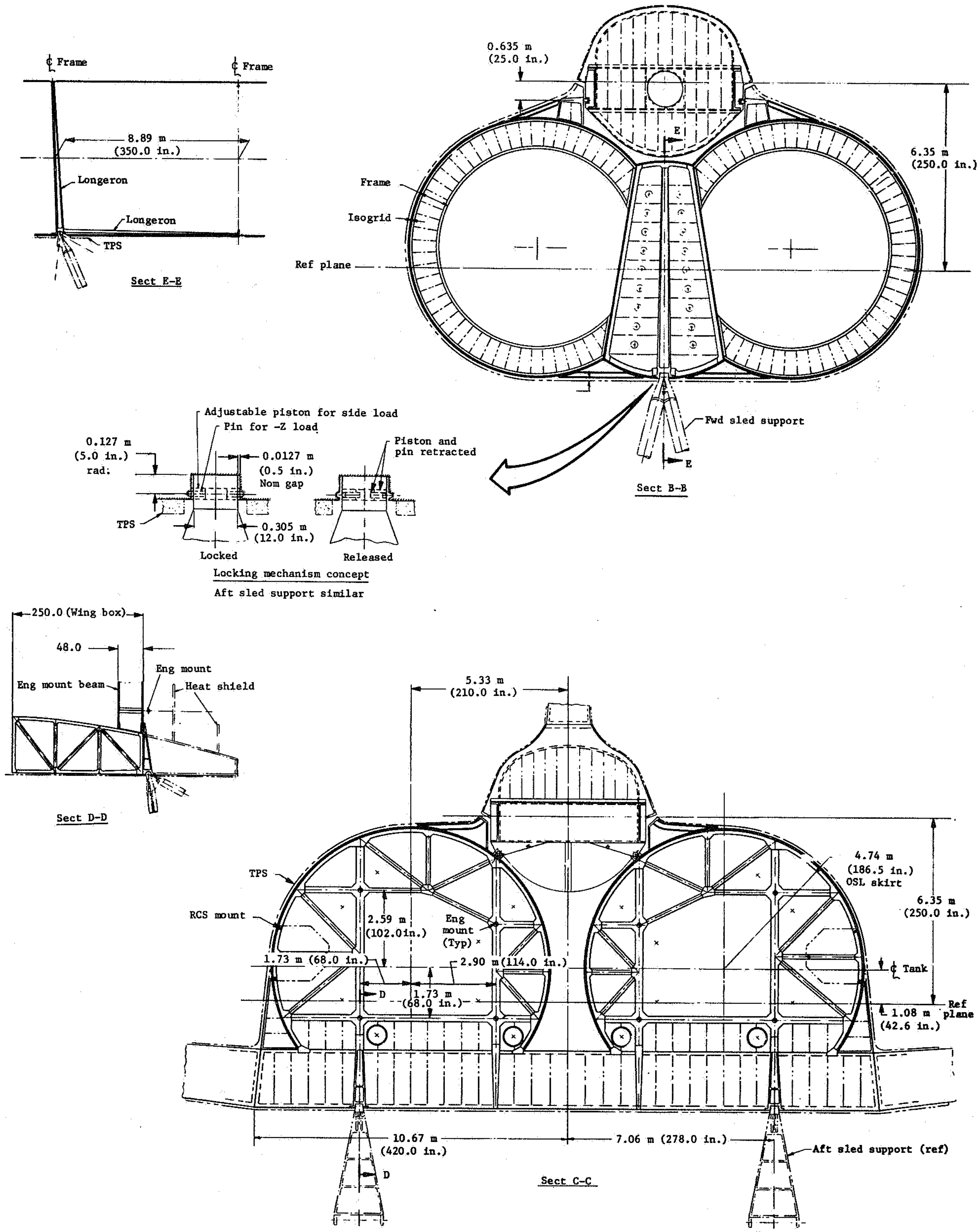


Figure 42.- Concluded

TABLE 23.- HTO ENGINE PERFORMANCE DATA

Nozzle type	Fixed	Dual	
Number per vehicle	4	4	
Engine weight - kg (lbm)	3077 (6783)	4127 (9098)	
Propellant flow rate - kg/sec (lbm/sec)	626 (1379)	626 (1379)	
LO_2 flow rate - kg/sec (lbm/sec)	548 (1207)	548 (1207)	
LH_2 flow rate - kg/sec (lbm/sec)	78 (172)	78 (172)	
Expansion ratio	35	55	160
Thrust, S.L. - 10^3 N (10^3 lbf)	2480 (557)	2430 (545)	----
Thrust, vac - 10^3 N (10^3 lbf)	2680 (601)	----	2840 (639)

The OMS and RCS requirements are as follows:

Tank	Propellant weight		Tank volume	
	kg	lbm	m^3	ft^3
OMS LO_2 (each tank)	10 470	23 090	9.6	340
OMS LO_2 (total)	20 950	46 180	19.2	680
OMS LH_2 (each tank)	2 100	4 620	30.9	1 090
OMS LH_2 (total)	4 200	9 240	61.8	2 180
OMS total propellant*	25 150	55 420	81.0	2 860
RCS LO_2 (each tank)	630	1 380	0.57	20
RCS LO_2 (total)	1 890	4 140	1.71	60
RCS LH_2 (each tank)	140	310	2.07	73
RCS LH_2 (total)	420	930	6.21	220
RCS total propellant	2 310	5 070	7.92	280
*OMS sized for ΔV = 381 mps (1250 fps)				

Mass properties of the dry-wing HTO vehicle are presented in Table 24. The larger vehicle, compared to the VTO, is a result of the heavier wing required for the loaded pullup maneuver after leaving the sled. The sled run propellant is due to the use of the main engines during sled acceleration. This propellant is loaded in the vehicle and affects the total size of the flight vehicle.

Consideration of a wet-wing HTO vehicle led to the mass properties summary shown in Table 25. Note the reduced wing weight and resulting vehicle dry weight.

TABLE 24.- HTO DRY WING MASS PROPERTIES

Code	System	Mass, kg		Weight, pounds	
1.0	Wing group	40 094		88 392	
2.0	Tail group	5 787		12 759	
3.0	Body group	58 855		129 752	
4.0	Induced environmental protection	41 652		91 828	
5.0	Landing and auxiliary systems	8 297		18 291	
6.0	Propulsion ascent	35 341		77 915	
	6.1 Engine accessories		2 162		4 767
	6.2 Propellant system		4 364		9 622
	6.3 Engines (8)		28 815		63 526
7.0	Propulsion - RCS	1 444		3 183	
8.0	Propulsion - OMS	1 071		2 361	
9.0	Prime power	1 674		3 690	
10.0	Electrical conversion and distribution	2 975		6 560	
11.0	Hydraulic conversion and distribution	2 903		6 400	
12.0	Surface controls	2 480		5 468	
13.0	Avionics	2 096		4 622	
14.0	Environmental control	1 836		4 048	
15.0	Personnel provisions	499		1 100	
18.0	Payload provisions	270		595	
19.0	Margin	17 846		39 344	
	Total dry weight	225 121		496 307	
20.0	Personnel	1 199		2 644	
23.0	Residuals and gases	4 165		9 183	
	Landing weight	230 486		508 134	
22.0	Payload	29 484*		65 000*	
	Landing with payload	259 970		573 134	
23.0	Residuals	5 929		13 071	
25.0	Reserve fluids	5 851		12 900	
26.0	Inflight losses	1 613		3 555	
27.0	Ascent propellants	1 817 463		4 006 819	
28.0	Propellant - RCS	2 301		5 072	
29.0	Propellant - OMS	13 071		28 817	
	GLOW	2 106 198		4 643 368	
30.0	Propellant - sled run	100 326		221 181	
	Gross weight	2 206 524		4 864 549	

Center of Gravity: Body length = 64.557 m (211.8 ft)

Condition	X % of body length	Z Meters (feet)
Dry	73.513	4.80 (15.75)
Landing	73.253	4.837 (15.87)
Landing with payload	71.629	5.395 (17.70)
Liftoff	70.090	4.922 (16.15)
Start	69.603	4.901 (16.08)

Moment of inertia:

Condition	I_x kg - m^2	(slug-ft^2)	I_y kg - m^2	(slug-ft^2)	I_z kg - m^2	(slug-ft^2)
Dry	22 799 233	(16 815 855)	76 546 022	(56 457 460)	87 164 550	(64 289 286)
Landing	22 938 915	(16 918 879)	77 573 943	(57 215 616)	88 235 484	(65 079 166)
Landing with payload	23 649 320	(17 442 847)	81 452 650	(60 076 404)	91 483 769	(67 474 979)
Liftoff	69 859 787	(51 525 945)	205 087 115	(151 264 524)	260 052 787	(191 805 131)
Start	72 506 784	(53 478 270)	223 590 215	(164 911 713)	280 907 527	(207 186 801)

Product of Inertia:

Condition	P_{xy} kg - m^2	(slug-ft^2)	P_{xz} kg - m^2	(slug-ft^2)	P_{yz} kg - m^2	(slug-ft^2)
Dry	871 461	(642 757)	-207 345	(-152 930)	144 885	(106 862)
Landing	880 602	(649 499)	-377 413	(-278 366)	142 887	(105 388)
Landing with payload	938 938	(692 525)	-1 566 391	(-1 155 311)	111 881	(82 519)
Liftoff	994 134	(733 236)	822 068	(606 326)	138 087	(101 848)
Start	1 011 628	(746 139)	2 437 127	(1 797 533)	139 283	(102 730)

*Revised aerodynamics capability is 41 277 kg (91 000 lb).

TABLE 25.- HTO WET WING MASS PROPERTIES

Code	System	Mass, kg		Weight, pounds	
1.0	Wing group	31 177		68 733	
2.0	Tail group	4 966		10 947	
3.0	Body group	47 176		104 005	
4.0	Induced environmental protection	38 096		83 987	
5.0	Landing and auxiliary systems	7 061		15 568	
6.0	Propulsion ascent	33 274		73 356	
	6.1 Engine accessories		1 900		4 188
	6.2 Propellant system		5 136		11 324
	6.3 Engines (8)		26 237		57 844
7.0	Propulsion - RCS	1 444		3 183	
8.0	Propulsion - OMS	1 015		2 238	
9.0	Prime power	1 674		3 690	
10.0	Electrical conversion and distribution	2 928		6 456	
11.0	Hydraulic conversion and distribution	2 903		6 400	
12.0	Surface controls	2 506		5 524	
13.0	Avionics	2 097		4 622	
14.0	Environmental control	1 836		4 048	
15.0	Personnel provisions	499		1 100	
18.0	Payload provisions	270		595	
19.0	Margin	15 268		33 661	
	Total dry weight	194 190		428 112	
20.0	Personnel	1 199		2 644	
23.0	Residuals and gases	3 816		8 413	
	Landing weight	199 205		439 169	
22.0	Payload	29 484*		65 000*	
	Landing with payload	228 689		504 169	
23.0	Residuals	10 820		23 855	
25.0	Reserve fluids	4 769		10 514	
26.0	Inflight losses	1 613		3 555	
27.0	Ascent propellant	1 642 748		3 621 640	
28.0	Propellant - RCS	1 921		4 234	
29.0	Propellant - OMS	10 881		23 989	
	GLOW	1 901 441		4 191 956	
30.0	Propellant - sled run	90 718		200 000	
	Gross weight	1 992 159		4 391 956	

Center of gravity: Body length - 60.26 m (197.7 ft)

Condition	X % of body length
Dry	73.637
Landing	73.424
Landing with payload	71.755
Liftoff	75.109
Start	74.659

*Revised aerodynamics capability is 44 452 kg (98 000 lb).

The vehicles sized in this study were based on initial aerodynamics estimates. The effects of revised aerodynamic characteristics are reported in the vehicle comparison summary. The mass properties summary tables indicated a payload of 29 484 kg (65 000 lb), with the increased performance capability, based on revised aerodynamics, shown as increased payload.

INFLIGHT FUELED VEHICLE (IFF) DESIGN

The IFF vehicle takes off from a runway on its own landing gear with enough propellant on board to climb, rendezvous with a tanker aircraft, refuel and ignite the ascent rocket engines. The vehicle design approach included trade studies of ascent propulsion and of refueling either LO_2 only or both LO_2 and LH_2 propellants. The refueling was selected at 4 572 m (15 000 ft) altitude and Mach 0.75, based on evaluations of turbofan engine performance and IFF aerodynamics.

Propulsion System Comparisons

Several propulsion alternatives were considered for the IFF vehicle. Initially, this vehicle was based on the inflight loading of LO_2 only. The propulsion systems considered for takeoff and ascent to tank rendezvous included all-rocket, turbojets, turbofans, and turbofans supplemented by one of the main rocket engines during the final LO_2 loading. The results of these studies, shown in Table 26, indicated that the turbofan plus rocket system had the lowest dry weight, followed by the all rocket system. Because all these vehicles were too heavy, a trade study was made using inflight loading of both LO_2 and LH_2. One vehicle used an all-rocket system and another used turbofans plus rocket engines. The results of this trade study showed that the dry weight of the all-rocket vehicle was 7.8% less.

TABLE 26.- LO_2 TRANSFER

System	Propellants	Δ dry weight, %	Δ takeoff weight, %
Rocket	LO_2/LH_2	Baseline	Baseline
	LO_2/RP-1	-0.5	+34
	LO_2/RJ-5	-0.8	+35
Turbojet	JP-4	+8.9	-45
	RJ-5	+8.9	-45
	LH_2	+10.3	-56
Turbofan	JP-4	+2.1	-56
	RI-5	+2.1	-56
	LH_2	+2.8	-62
Turbofan plus rocket	JP-4 plus LO_2/LH_2	-3.9	-59

Rocket-Takeoff Vehicle.- The rocket takeoff vehicle general arrangement is shown in Figure 43. The thermostructural concept is the same as the VTO and HTO vehicles. Eight rocket engines are used for vehicle propulsion with the takeoff-climb-accelerate-rendezvous-propellant transfer (TCART) mode using two rocket engines. This vehicle is sized for both LO_2 and LH_2 refueling to minimize vehicle size and dry weight.

Volume		
LH_2 tank	3171 m^3	(111 982 ft^3)
LO_2 tank	1370.9 m^3	(48 414 ft^3)
Payload		
Diameter	4.572 m	(15 ft)
Length	18.288 m	(60 ft)
Payload Bay clear opening		
Diameter	4.725 m	(15.5 ft)
Length	18.517 m	(60.75 ft)

Area		
Body plan area	1002.0 m^2	(10 785 ft^2)
Wing, theoretical	1146.4 m^2	(12 339 ft^2)
Wing, exposed	583.4 m^2	(6 280 ft^2)
elevon	184.4 m^2	(1 985 ft^2)
Vertical tail	209.0 m^2	(2 250 ft^2)
rudder	75.6 m^2	(814 ft^2)
Body wetted area	2685.3 m^2	(28 904 ft^2)

Weight		
Payload	29 483 kg	(65 000 lb)
Dry weight	217 994 kg	(480 595 lb)
Landing without payload	222 949 kg	(491 519 lb)
Landing with paylod	252 432 kg	(556 519 lb)
Ascent propellant	1 710 969 kg	(3 772 042 lb)
Gross launch weight	1 990 278 kg	(4 387 811 lb)
Propellant transfered	1 717 741 kg	(3 786 971 lb)
Takeoff propellant	144 587 kg	(318 761 lb)
Takeoff weight	418 640 kg	(922 943 lb)

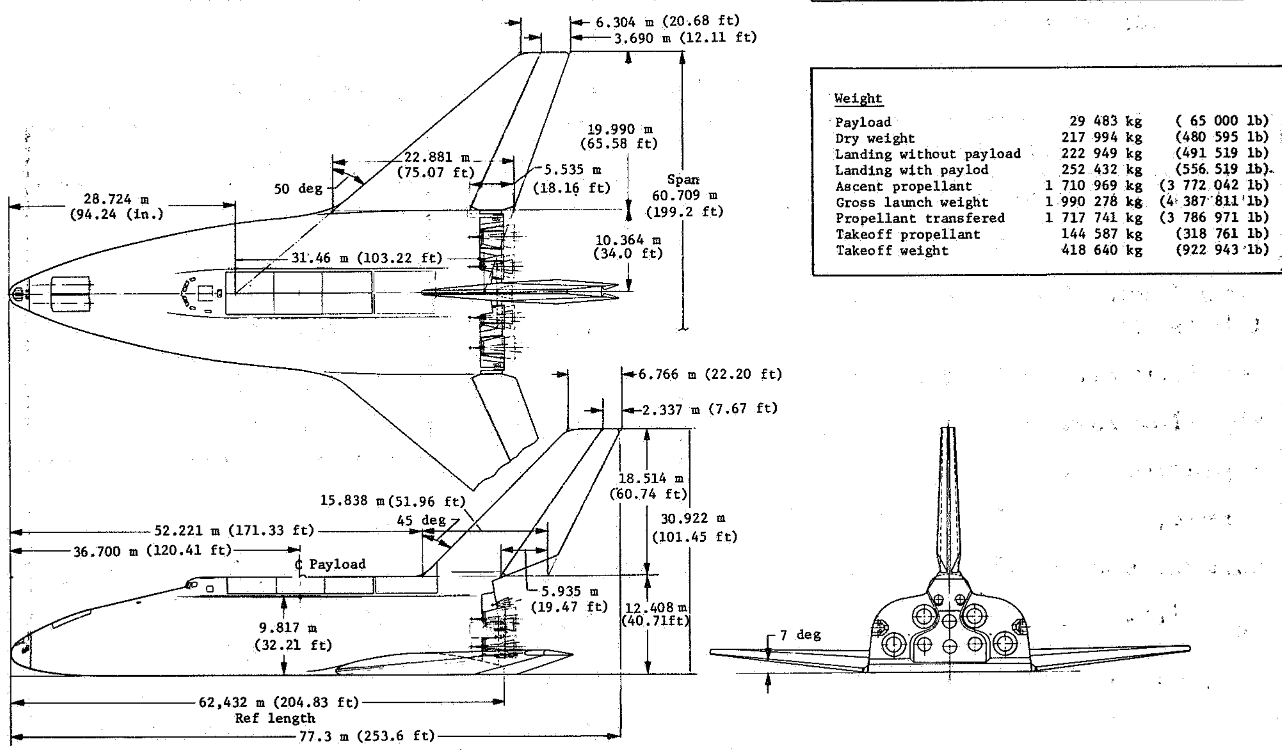


Figure 43.- IFF general arrangement, rocket engine takeoff

The inboard profile is shown in Figure 44. Due to the higher takeoff weight, an additional four wheel boogy main gear is provided on the center of the vehicle shown in Section D-D. The LO_2 refueling boom is connected first and is capable of maintaining a certain amount of tension on the boom. The LH_2 boom, which is attached to the LO_2 boom, is engaged subsequently and is not stressed. The propellant coupling is self-closing and redundant shutoff zero leak valves are provided. The refueling ports (for both propellants) are located in the nose section and the LO_2 line runs aft to the tanks as shown.

The structural arrangement of the rocket takeoff IFF vehicle is shown in Figure 45. The structural details peculiar to the IFF vehicle are the center main landing gear and support shown in Sections B-B and C-C, and the refueling boom receptacle support shown in Section A-A.

The rocket engines are four fixed nozzle, 50:1 expansion ratio and four dual nozzle, 55:1 - 160:1 expansion ratio configurations. Engine data are given in Table 27.

TABLE 27.- IFF ENGINE PERFORMANCE DATA

Nozzle type	Fixed	Dual	
Number per vehicle	4	4	
Engine weight - kg (lbm)	2880 (6350)	3790 (8360)	
Propellant flow rate - kg/sec (lbm/sec)	564 (1243)	564 (1243)	
LO_2 flow rate - kg/sec (lbm/sec)	493 (1087)	493 (1087)	
LH_2 flow rate - kg/sec (lbm/sec)	71 (156)	71 (156)	
Expansion ratio	50	55	160
Thrust, S.L. - 10^3 N (10^3 lbf)	2220 (499)	2183 (491)	----
Thrust, vac - 10^3 N (10^3 lbf)	2450 (551)	----	2295 (516)
I_{sp}, S.L. - sec	401.6	395.5	----
I_{sp}, vac - sec	443.6	----	463.5

The requirement for horizontal takeoff with a propellant load sufficient to complete the tanker rendezvous necessitates auxiliary outlets at the tank bottoms. These outlets use a dedicated set of lines to feed the lower center engine. Partial barriers are required in the tanks to control the liquid position before rendezvous. Isolation valves for these dedicated feedlines are located near the tank outlets to minimize residual trapped propellants.

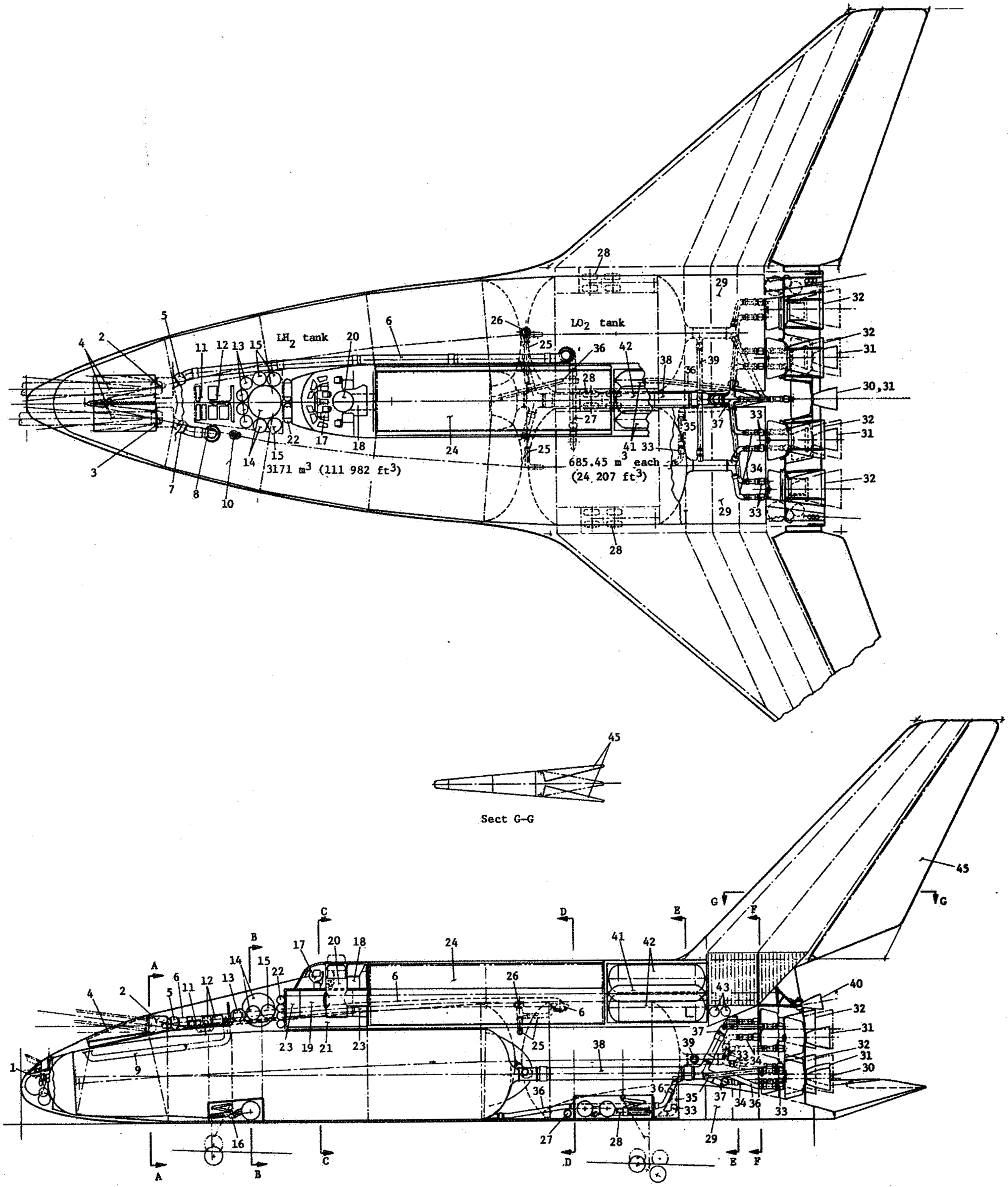
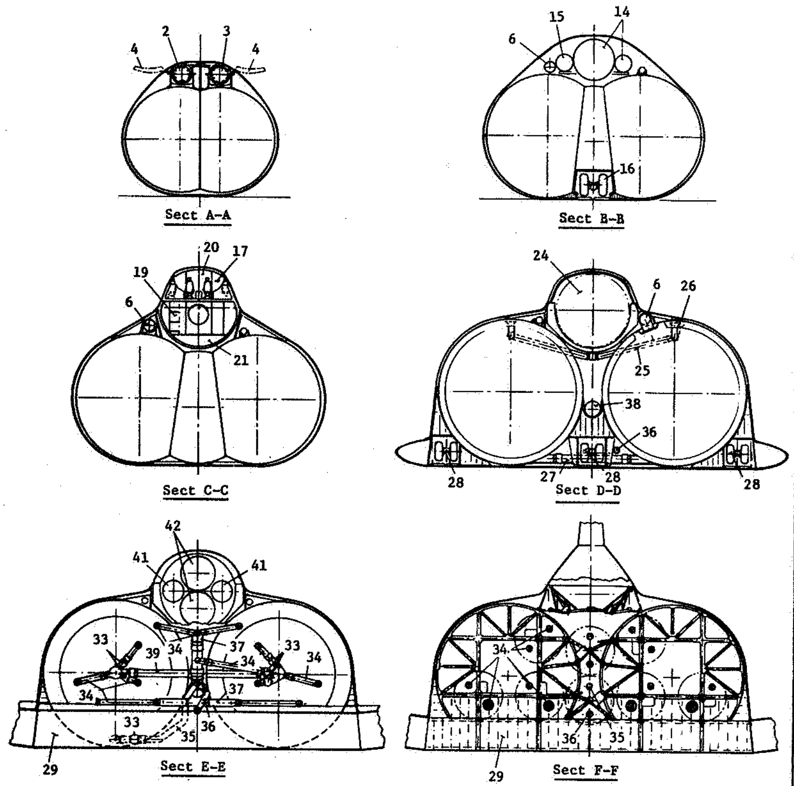


Figure 44.- IFF inboard profile, rocket engine takeoff



Legend:

1. Forward RCS module
2. Inflight refueling adapter, LO_2
3. Inflight refueling adapter, LH_2
4. Heat shield doors for refueling adapters
5. Refueling prevalve, LO_2
6. LO_2 refueling line
7. Refueling prevalve, LH_2
8. LH_2 refueling line
9. LH_2 tank vent and pressure relief line
10. LH_2 tank pressure relief/vent valve
11. Electrical power system, fuel cells
12. Power system APUs
13. Fuel cell propellants (LO_2 - LH_2)
14. APU propellants (LO_2 - LH_2)
15. Pressurants (He)
16. Nose landing gear
17. Flight deck
18. Operations deck
19. Rest and passenger area
20. Airlock and docking module
21. ECLSS system
22. ECLSS supply and purge gas tanks
23. Avionics
24. Payload bay
25. LO_2 tank vent interconnect and pressure relief line
26. LO_2 tank pressure relief/vent valve
27. LO_2 tank fill interconnect line
28. Main landing gears (outboard and center)
29. Wing carrythrough structure
30. Main propulsion/cruise engine, $\varepsilon = 50$, fixed nozzle not gimbaled
31. Main propulsion engine, $\varepsilon = 50$, fixed nozzle, not gimbaled
32. Main propulsion engine, $\varepsilon = 55/160$, extendable nozzle, gimbaled
33. Propellant prevalves
34. Propellant feedlines
35. LO_2 cruise engine feedline
36. LH_2 cruise engine feedline
37. LH_2 upper and lower feedline manifolds
38. LH_2 main feedline
39. LO_2 tank feed interconnect line
40. OMS engines
41. OMS propellant tank, LO_2
42. OMS propellant tank, LH_2
43. OMS pressurant tanks, He
44. Aft RCS modules
45. Split rudder

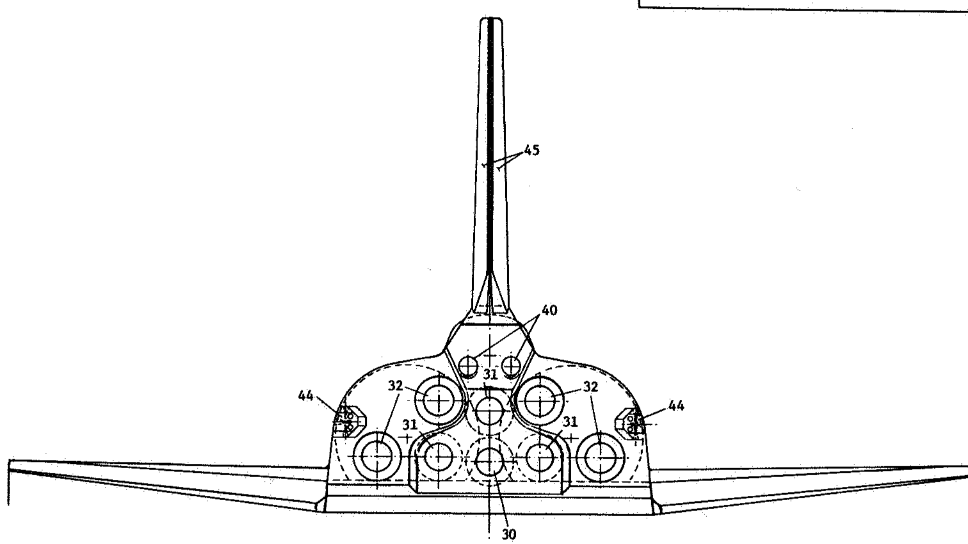


Figure 44.- Concluded

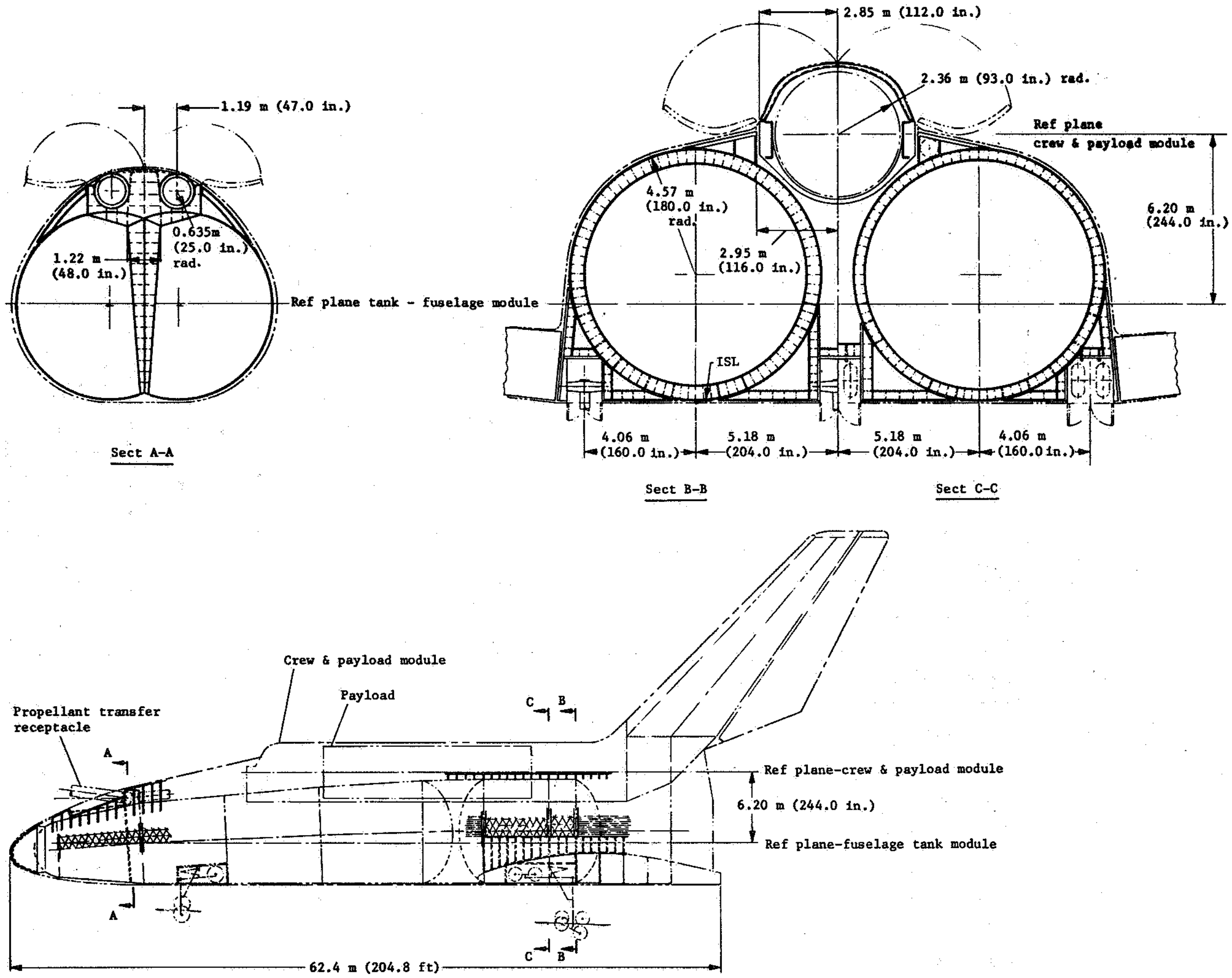


Figure 45.- IFF structural arrangement

The inflight propellant fill systems are designed for fill rates of four times the single engine flow rates. The fill lines enter the tanks at the top. For the LO_2 system, the fill line goes to one tank only with the other tank being filled via a crossover near the bottom. Tank vents are located near the top forward end. Although the tanks will be precooled on the ground, they will have some heat gain before inflight transfer. The vent systems must be adequate to handle these transient heat loads, the steady state heat input to the tanks and transfer lines, and the displaced vapor. The hydrogen vent system will exit aft of the vehicle to preclude damage in the event of accidental ignition.

The propellant weights and tank volumes required for OMS and RCS are given in the following tabulation:

Tank	Propellant weight		Tank volume	
	kg	lbm	m^3	ft^3
OMS LO_2 (each tank)	9 615	21 195	8.8	312
OMS LO_2 (total)	19 230	42 390	17.6	624
OMS LH_2 (each tank)	1 920	4 240	28.4	1 002
OMS LH_2 (total)	3 840	8 480	56.8	2 004
OMS total propellant*	23 070	50 870	74.4	2 628
RCS LO_2 (each tank)	576	1 270	0.54	19
RCS LO_2 (total)	1 728	3 810	1.62	57
RCS LH_2 (each tank)	128	283	1.90	67
RCS LH_2 (total)	384	850	5.70	201
RCS total propellant	2 112	4 660	7.32	258

*OMS sized for ΔV = 381 mps (1250 fps).

Turbofan takeoff vehicle.- The turbofan takeoff IFF vehicle is shown in Figure 46. The vehicle has eight high-bypass ratio turbofan engines with 222 411 N (50 000 lb) takeoff thurst. The engines are installed in the inter-tank bay in a retractable nacelle. The turbofan engines are retracted into the bay after the main rocket engines are ignited on completion of LO_2 refueling. This vehicle was configured with LO_2 refueling only because of the safety hazard introduced in refueling both propellants.

Ten rocket engines are used for the main propulsion system. Four dual position 55:1 - 160:1 expansion ratio nozzle engines and six 50:1 expansion ratio fixed nozzle engines are used. The refueling port for inflight transfer of LO_2 propellant is located between tanks on the left shoulder of the fuselage next to the payload bay. All the required LH_2 is loaded on the ground and, as the LO_2 is loaded, one rocket engine is ignited to supplement the turbofan thrust.

Mass properties.- The turbofan takeoff IFF vehicle with LO_2 propellant transfer results in a GLOW > 3 175 000 kg (7 000 000 lb). The rocket takeoff IFF vehicle using both LH_2 and LO_2 propellant refueling results in a vehicle of much more acceptable size and is the concept for which the mass properties are shown in Table 28.

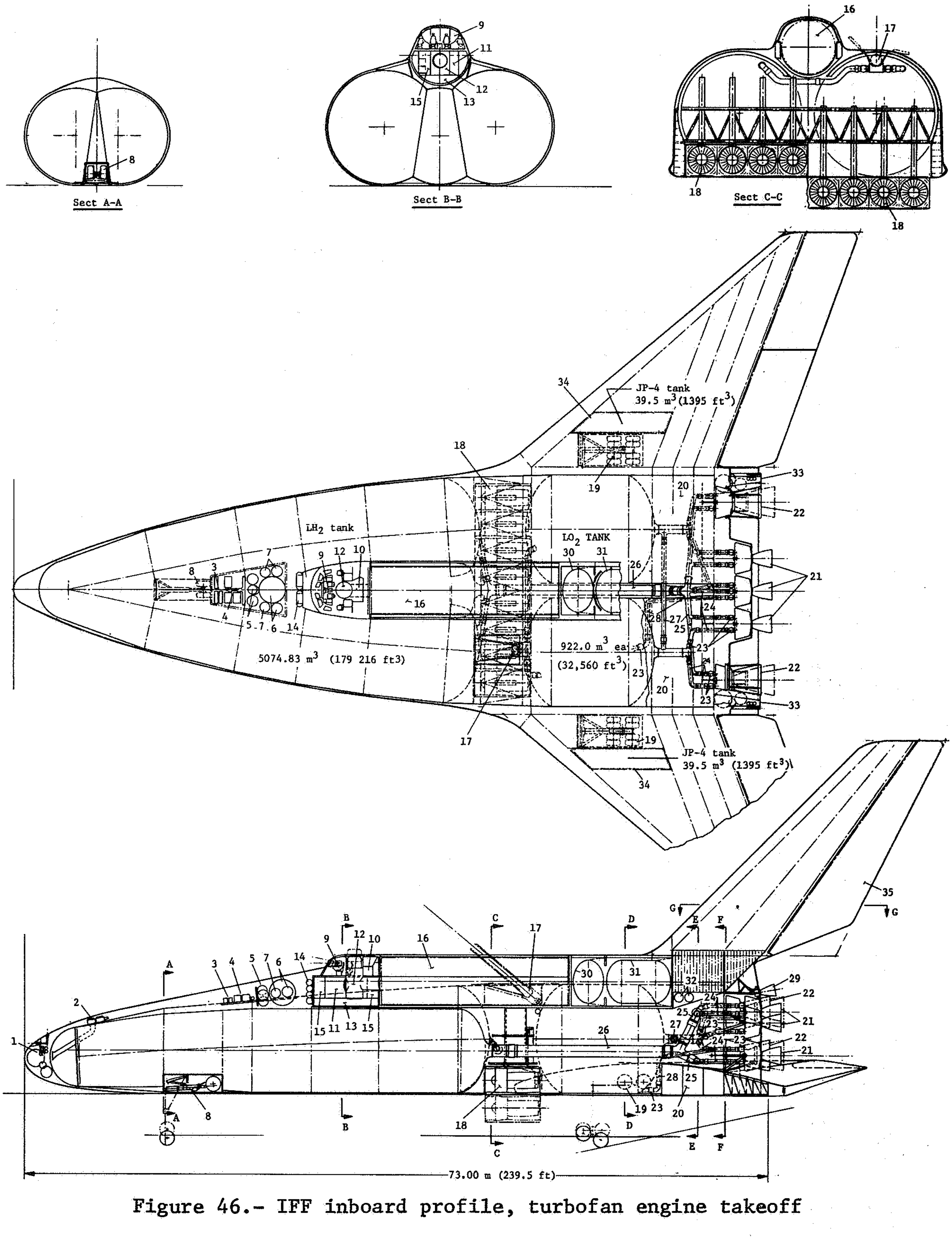
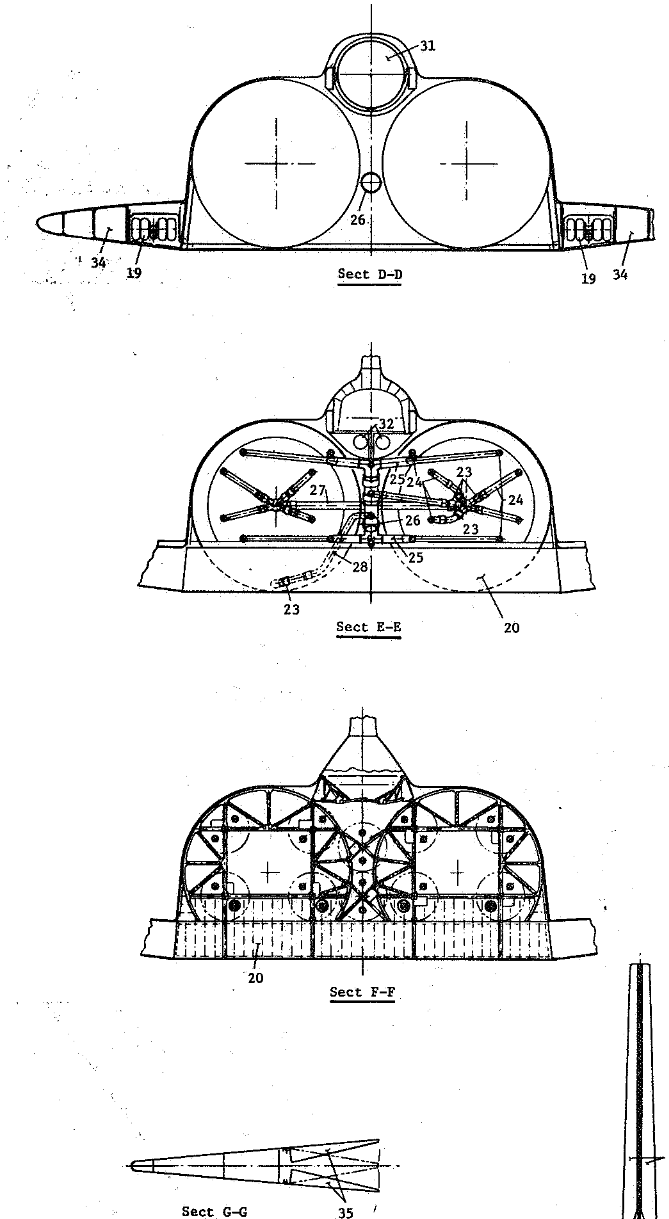


Figure 46.- IFF inboard profile, turbofan engine takeoff



Legend:

1. Forward RCS module
2. LH_2 tank vent and pressurization valves
3. Electrical power system, fuel cells
4. Power system, APUs
5. Fuel cell propellants (LO_2 - LH_2)
6. APU propellants (LO_2 - LH_2)
7. Pressurants (He)
8. Nose landing gear
9. Flight deck
10. Operations deck
11. Rest and passenger area
12. Airlock and docking module
13. ECLSS system
14. ECLSS supply and purge gas tanks
15. Avionics
16. Payload bay
17. Refueling receptacle (LO_2)
18. Eight turbofan engines and nacelles (retractable)
19. Main landing gear
20. Wing carrythrough structure
21. Main propulsion engine, ϵ = 50, fixed nozzle, not gimbaled
22. Main propulsion engine, ϵ = 55/160, extendable nozzle, gimbaled
23. Propellant prevalves
24. Propellant feedlines
25. LH_2 upper and lower feedline manifolds
26. LH_2 main feedline
27. LO_2 tank interconnect line
28. LO_2 cruise engine feedline
29. OMS engine
30. OMS propellant tank, LO_2
31. OMS propellant tank, LH_2
32. OMS pressurant tanks (He)
33. Aft RCS modules
34. JP-4 wing tanks
35. Split rudder

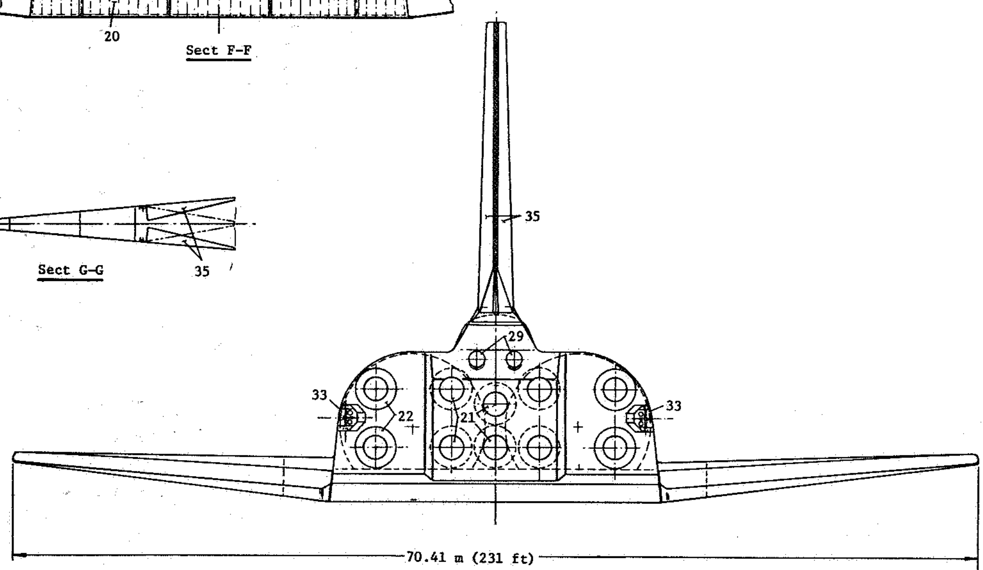


Figure 46.- Concluded

TABLE 28.- IFF MASS PROPERTIES

Code	System	Mass, kg	Weight, pounds
1.0	Wing group	36,988	81,544
2.0	Tail group	5 371	11 841
3.0	Body group	53 931	118 898
4.0	Induced environmental protection	39 910	87 987
5.0	Landing and auxiliary systems	14 730	32 474
6.0	Propulsion - ascent	32 460	71 562
	6.1 Engine accessories	1 898	4 183
	6.2 Propellant systems	3 887	8 570
	6.3 Engines (8)	26 675	58 809
7.0	Propulsion - RCS	1 444	3 183
8.0	Propulsion - OMS	1 065	2 347
9.0	Prime power	1 674	3 690
10.0	Electrical conversion and distribution	2 975	6 560
11.0	Hydraulic conversion and distribution	2 903	6 400
12.0	Surface controls	2 449	5 400
13.0	Avionics	2 096	4 622
14.0	Environmental control	1 836	4 048
15.0	Personnel provisions	499	1 100
18.0	Payload provisions	270	595
19.0	Margin	17 393	38 344
	Total dry weight	217 994	480 595
20.0	Personnel	1 199	2 644
23.0	Residuals and gases	3 756	8 280
	Landing weight	222 949	491 519
22.0	Payload	29 484	65 000
	Landing with payload	252 433	556 519
23.0	Residuals	5 897	13 000
25.0	Reserve fluids	5 256	11 587
26.0	Inflight losses	1 612	3 555
27.0	Ascent propellant	1 710 969	3 772 042
28.0	Propellant - RCS	2 114	4 661
29.0	Propellant - OMS	11 998	26 451
	GLOW	1 990 279	4 387 815
	Takeoff weight	418 640	922 943

Concerns.- The IFF concept was initially addressed because of seemingly potential benefits in reducing vehicle size by providing an airbreathing stage or higher energy (altitude and velocity) initial conditions. The study nevertheless shows no dry weight advantage of the IFF vehicle, and additional concerns, such as, very large size tanker aircraft to carry propellants to the IFF, severe requirements for rendezvous including short flight times with precise navigation, precise relative flight control between the two vehicles, and large flow rates for propellant transfer.

AERODYNAMICS

The initial trajectory analysis and vehicle sizing for SSTO configurations was made using estimated lift and drag aerodynamics based on Space Shuttle orbiter data. These estimates (Figure 47) represented the aerodynamics of preliminary SSTO configurations but were revised subsequently based on SSTO configuration developments. The VTO, HTO, and IFF vehicles have similar shapes and aerodynamics, except for small modifications such as wing and tail geometries, and locations to accommodate c.g. differences. Aerodynamic characteristics were therefore analyzed for the VTO configuration, and then applied with appropriate modifications to sizing the HTO and IFF vehicles. Parametric wing and tail sizing studies were conducted using the Hypersonic Arbitrary Body Program (HABS), the USAF stability and control DATCOM, and inhouse theoretical and empirical techniques. The geometries of the aerosurfaces were selected to satisfy the guideline requirements: hypersonic trim of 20 deg $\leq \alpha \leq$ 40 deg, 2% $\overline{c}$ or greater longitudinal subsonic stability, directional subsonic stability of $C_{n_\beta} \geqq 0.002$, and a maximum landing speed of 84.9 m/s (165 kts) at $\alpha = 15$ deg.

Analysis of the parametric wing studies showed that the hypersonic trim requirement was the determining factor in the wing size. As a compromise between aerodynamic effectivness and surface heating, a wing leading edge sweep of 50 deg and trailing edge sweep of 20 deg were selected. Figure 48 presents a summary plot of the hypersonic wing sizing requirements for the VTO configurations. The theoretical wing area required to trim for both $\alpha_{minimum}$ = 20 deg and 25 deg is given as a function of total configuration center of gravity. The summary VTO vertical tail sizing requirements to meet several levels of subsonic C_{n_β}, including the baseline $C_{n_\beta} = 0.002$, are given in Figure 49 as a function of configuration longitudinal c.g.

Based on the parametric data given in Figures 48 and 49, the aerosurfaces were sized for the VTO configuration and complete aerodynamic characteristics were generated for that configuration. This vehicle was designed with a length of 61.9 m (203 ft), a theoretical wing area of 1126 m^2 (12 120 ft^2) and exposed vertical tail area of 205 m^2 (2210 ft^2).

The critical longitudinal design requirement for this configuration is the hypersonic trim capability for a 73.0% (payload out) c.g. The selected wing provides the necessary trim range, as shown in Figure 50. An elevon deflection of +11 deg provides a 20 deg minimum angle of attack with neutral stability; the positive stability trim range extends well above the necessary 40 deg. Figure 51 presents the hypersonic trim characteristics with a 71.8% (payload in) c.g. An elevon deflection of +6 deg yields a minimum trim limit of 18 deg; the upper trim range still extends above 40 deg.

This configuration also satisfies the subsonic stability requirements. The longitudinal stability margins are 3.74% $\bar{c}$ and 8.64% $\bar{c}$ for the 73.5% c.g. and 71.8% c.g., respectively, both in excess of the required margin. The vertical tail is selected for this configuration so that the required total vehicle $C_{n_\beta} = 0.002$ is obtained for the worst c.g. condition (the forward c.g. location produced $C_{n_\beta} = 0.0024$).

The subsonic aerodynamic characteristics are given in Figure 52. For a required landing $\alpha = 15$ deg, these characteristics provide a minimum landing speed of 64.3 m/s (125 kts) for the payload-in condition and 60.2 m/s (117 kts) for payload-out, both speeds substantially below the maximum allowable.

The hypersonic L/D for the payload-out VTO configuration is presented in Figure 53. The maximum trimmed and longitudinally stable L/D is 1.8. Because rudder flare may be advisable to improve the hypersonic lateral stability, the degradation in L/D due to a rudder bias of 40 deg is also shown.

The complete ascent- and entry-trimmed lift and drag coefficients were determined for the VTO with a 73.5% longitudinal c.g. These data were then used in the final trajectory analysis and vehicle sizing iteration. The ascent characteristics are presented in Figures 54 and 55; the entry characteristics exhibited only minor changes.

AEROTHERMODYNAMICS

Aerothermodynamic tasks conducted to evaluate the candidate SSTO concepts included (1) predicting the ascent and entry aerodynamic heating environments, (2) determining the TPS thickness requirements, and (3) defining maximum temperature distributions. In addition, aerodynamic heating constraints were supplied for entry trajectory shaping studies and inputs were made to influence the configuration design; e.g., allowable nose and leading edge radii were specified.

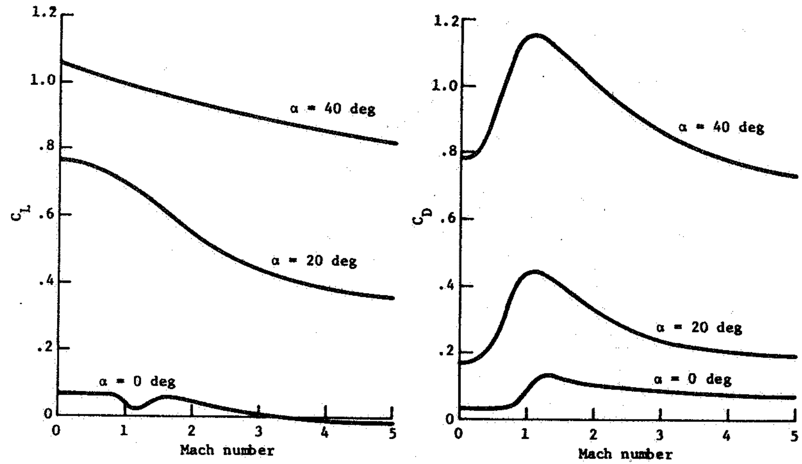


Figure 47.- Initial estimate of lift and drag coefficients

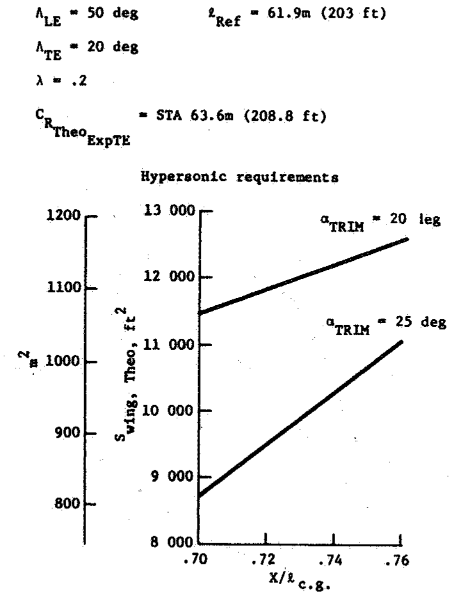


Figure 48.- Wing area requirements

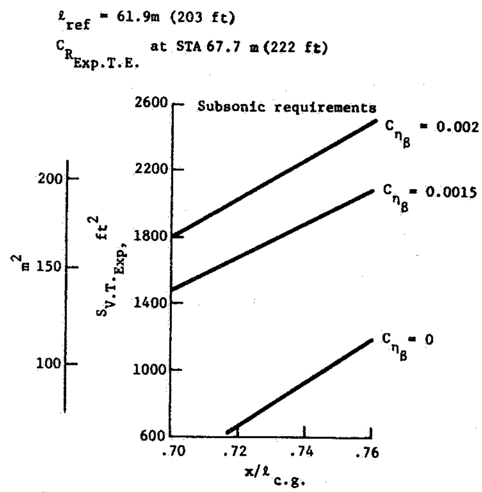


Figure 49.- Vertical tail area requirements

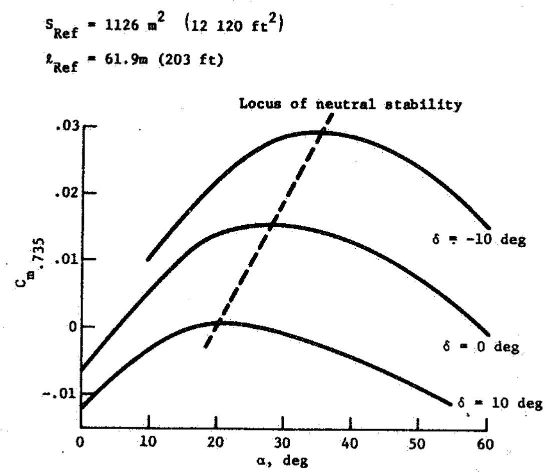


Figure 50.- Hypersonic trim capability, payload out

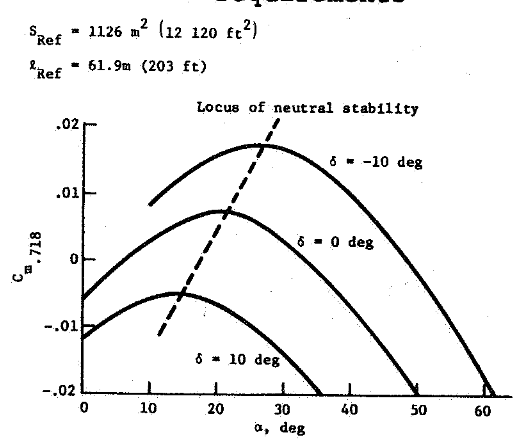


Figure 51.- Hypersonic trim capability, payload in

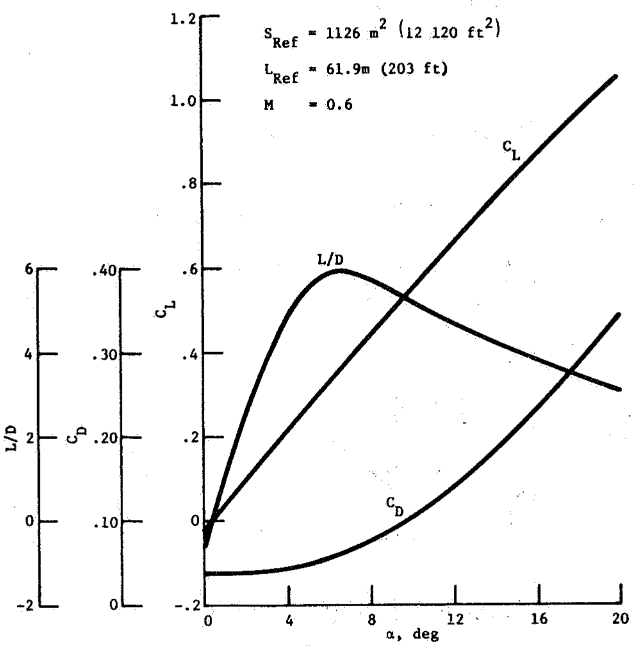


Figure 52.- Subsonic aerodynamics

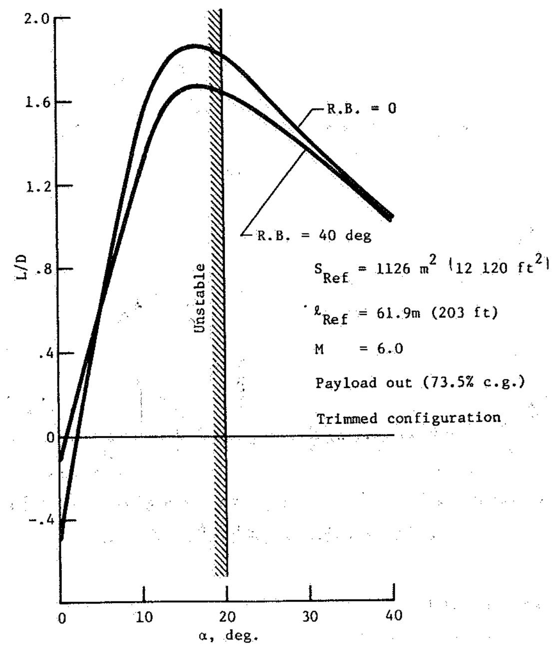


Figure 53.- Hypersonic lift/drag

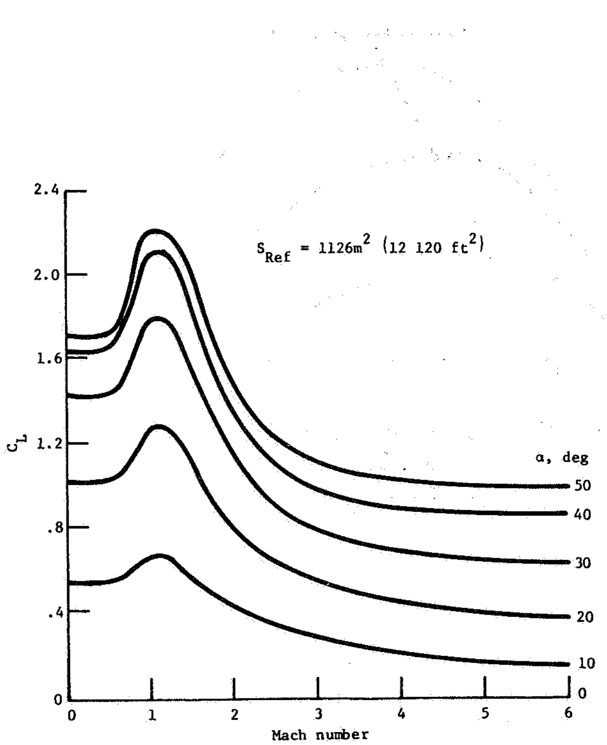


Figure 54.- Ascent lift coefficients

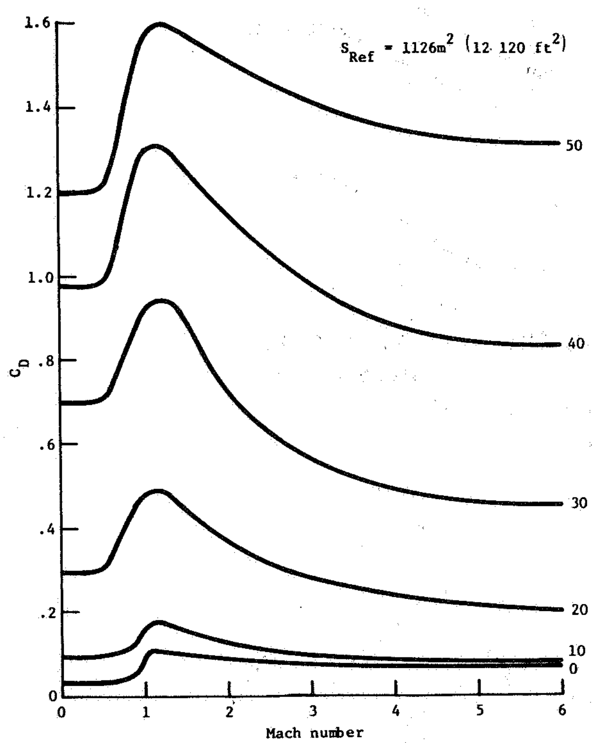


Figure 55.- Ascent drag coefficients

The methods used in the aerodynamic heating anaysis are similar to those currently employed on the Space Shuttle program. Flow field properties were determined using tangent cone theory for local surface pressure and boundary layer edge conditions. Heating rates were defined using Colburn's Reynolds analogy in conjunction with skin friction predictions. These predictions were based on Eckert's reference enthalpy method for laminar flow and the Spalding and Chi correlation for turbulent flow. Streamline divergence effects were included in all analyses. The onset of boundary layer transition was determined using a momentum thickness Reynolds number over local Mach number ratio $\left(Re_{\theta} / M_L \right)$ equivalent to the value of 225 used on the lower centerline of the Space Shuttle orbiter. All aerodynamic heating calculations were made using the MINIVER computer program.

Determination of the TPS thicknesses required to maintain the desired structural temperature limits was made using the FD202 Structural Heating Program. This program uses a lumped parameter system to describe any one-, two-, or three-dimensional heat transfer problem. The resulting heat balance equation is solved by finite difference techniques. All insulation thicknesses were determined using a 10-node system for the insulation. Body TPS thicknesses were sized to limit the interface between the RSI and the subpanel to a maximum temperature of 533 K (500°F). Wing and fin RSI requirements were determined by the thickness needed to limit a 3.175 mm (0.125 in.) thick aluminum skin to a maximum temperature of 450 K (350°F).

Aerothermal Influence on Entry Trajectory Shaping

For the baseline TPS, the primary aerothermal trajectory consideration was to minimize entry time and the total heat load, because past Space Shuttle studies have demonstrated that this minimizes insulative TPS weight. Initial studies, using a heating rate constraint compatible with the maximum projected allowable material temperature, resulted in a significant portion of the vehicle experiencing turbulent flow at the time of peak heating. Further analysis indicated that the total heat load could be reduced by maintaining laminar flow over the vehicle at the time of maximum heating, even though the entry time is increased. Figure 56 compares entry corridors on an altitude-velocity plot for two trajectories representing the extremes in aerodynamic heating investigated during the study. Also shown is a line denoting the onset of boundary layer transition at the aft end of the vehicle. From an aerothermodynamic viewpoint, the optimum trajectory for an insulative TPS concept is one that would fly along this line. However, deceleration limits and cross-range requirements force a departure from this line. The trajectories do not necessarily reflect fully optimized cases. It is anticipated that further studies could reduce, if not eliminate, the H-V spike at the end of the maximum heating period.

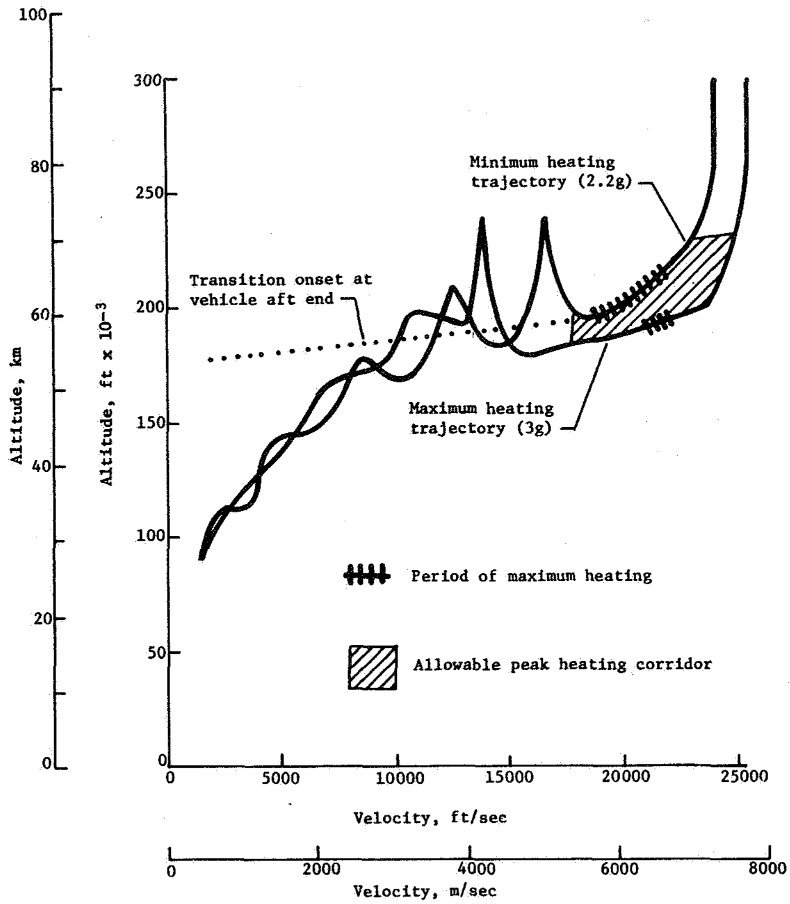


Figure 56.- Range of altitude-velocity profiles for entry trajectories evaluated

Sensitivity of Baseline TPS to Environmental Perturbations

Figure 57 gives the TPS thicknesses needed for the trajectories comprising the entry corridor shown on Figure 56. The RSI thickness requirements are shown for the total entry heat load associated with these trajectories at several lower centerline body locations. For a 100% increase in heat load, only 15% to 30% additional is required. This relative insensitivity to heat load is advantageous in that small heating perturbations caused by dispersions or uncertainties in aerodynamic heating methods have a negligible effort on the TPS design. For the same reason, an insulative thermal protection system for the SSTO can accommodate a relatively wide range of entry trajectories with a minimum impact on TPS weight.

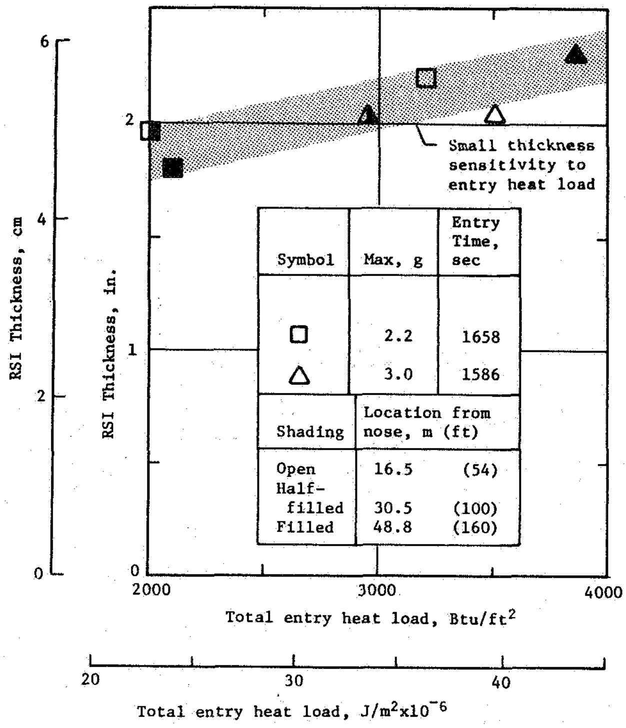


Figure 57.- Sensitivity of required RSI thickness to total entry heat load

VTO RSI Thickness and Maximum Temperature Distributions

The RSI thickness distributions required for the VTO vehicle together with maximum surface temperatures are shown in Figure 58. These thicknesses provide thermal protection for the most severe entry associated with the corridor of Figure 56. A typical transient temperature response for a representative location on the lower body centerline is given in Figure 59.

Even though the ascent environment produces higher surface temperatures on the upper portions of the vehicle than encountered during entry, it has no impact on the design of the insulation TPS. This is because the relatively short ascent heating period and small heat load result in much lower RSI backface temperatures than for entry.

Detailed investigations of the TPS thickness distributions for the sled launch vehicle and the inflight-fueled vehicle were not made. Because the entry trajectories were similar and the insulative TPS was found to be relatively insensitive to the entry heat load, TPS weights for these vehicles were determined by using the same unit weights and adjusting for the appropriate surface areas.

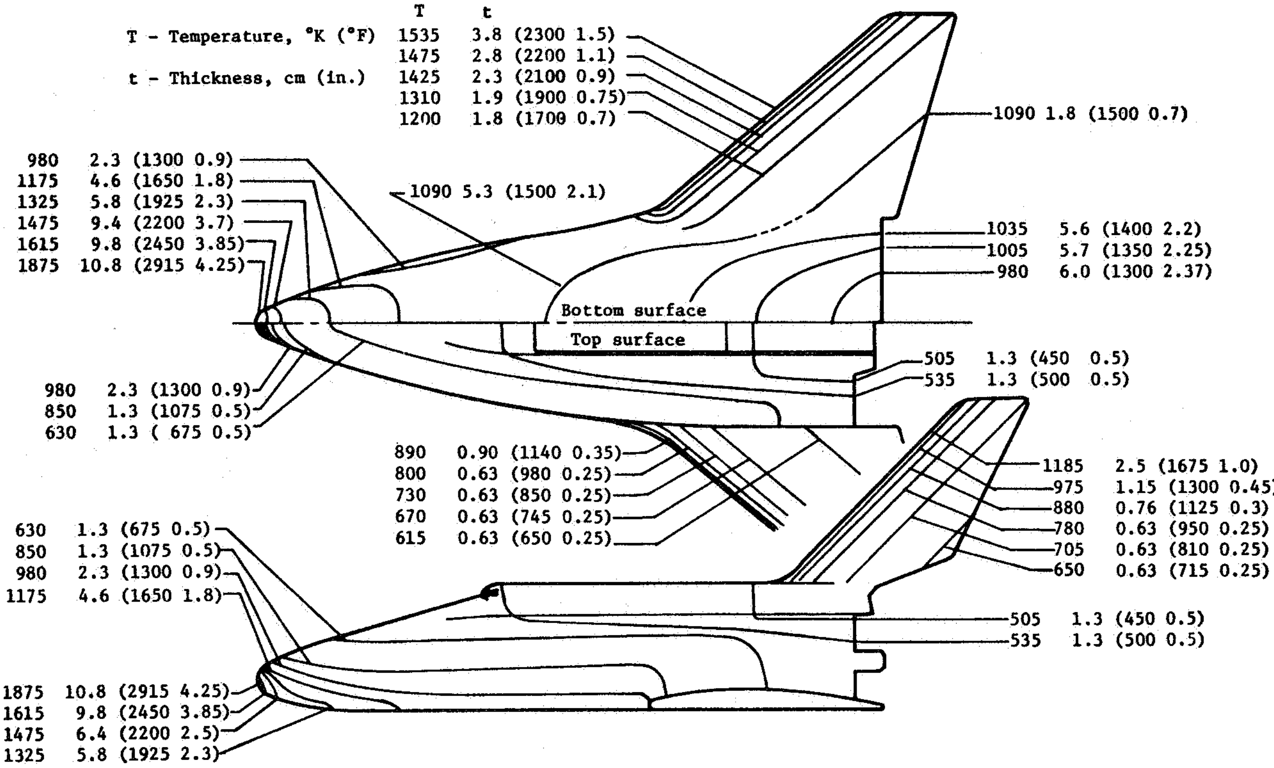


Figure 58.- SSTO-VTO entry surface isotherms and TPS thicknesses (2.2-g trajectory)

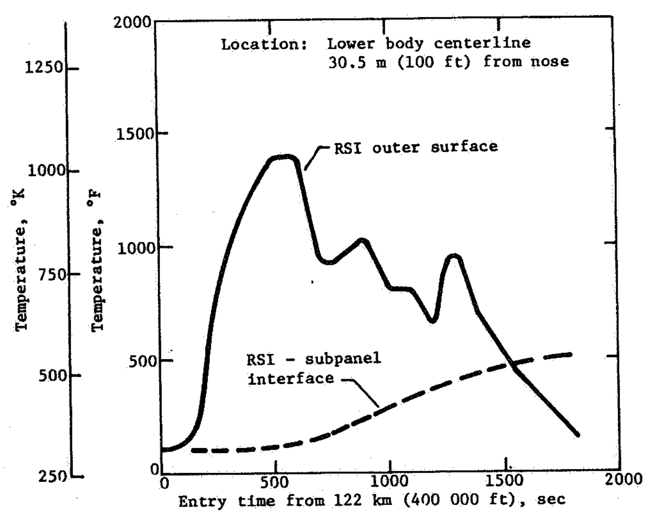


Figure 59.- Typical VTO entry temperature histories (2.2-g trajectory)

FLIGHT PERFORMANCE

Performance capability and trajectory characteristics of the SSTO vehicles were determined by trajectory simulation using the POST digital computer program. Boost trajectories were obtained for ETR east launch to the specified 92.6 km (50 n. mi.) perigee, 185 km (100 n. mi.) apogee elliptical orbit. For these trajectories, mass ratio performance requirements were defined.

Reentry trajectories were obtained beginning at 122 000 m (400 000 ft), the top of the sensible atmosphere, and terminating at 15 200 m (50 000 ft), which was considered the beginning of the landing approach. Initial conditions for reentry were consistent with deorbit from a 370 km (200 n. mi.) circular orbit inclined 28.6 deg to the Equator.

Various attitude control techniques were used for flight path definition, depending on the flight regime and the vehicle configuration, as described later. The time of application and the magnitude of these techniques were optimized as required. Thus the performance quotations and the trajectory characteristics described herein are considered near optimal.

VTO Vehicle Performance

Bell nozzle vehicles.- VTO vehicles were launched vertically from the Eastern Test Range. The pitch plane was aligned in an easterly direction to produce an orbit inclination of 28.5 deg. At a relative velocity of 45.7 m/sec (150 fps), a constant attitude rate (pitch down) was initiated. Some 10 seconds later, the vehicle was pitched up at a constant attitude rate until a specified angle of attack was reached. This angle of attack was maintained until reaching a Mach number of 0.6. Next, a period of constant lift was used by modulation of the angle of attack to improve performance. This period was terminated at approximately a Mach number of 3.5, where a constant angle of attack rate was used at 150 seconds. A period of constant attitude rate was started, ending at approximately 300 seconds. Here, another constant attitude rate began, terminating at orbit insertion.

All engines ignited at liftoff. When the atmospheric pressure had decreased to 15 500 N/m^2 (324 psf), the large expansion ratio nozzles were extended. The single expansion-ratio engine shutdown sequence began when the acceleration reached 3 g. To minimize control requirements, engines not on the vehicle longitudinal centerline were shut down in pairs. Each time the acceleration reached 3 g, another engine (or pair) was shut down until all single expansion-ratio engines were terminated. A similar sequence was used for the dual expansion-ratio engines.

Fundamental philosophy of this sequence was to maintain the highest possible vehicle thrust-to-weight ratio and effective specific impulse, thus minimizing velocity losses. Typical trajectory parameters are shown in Figure 60.

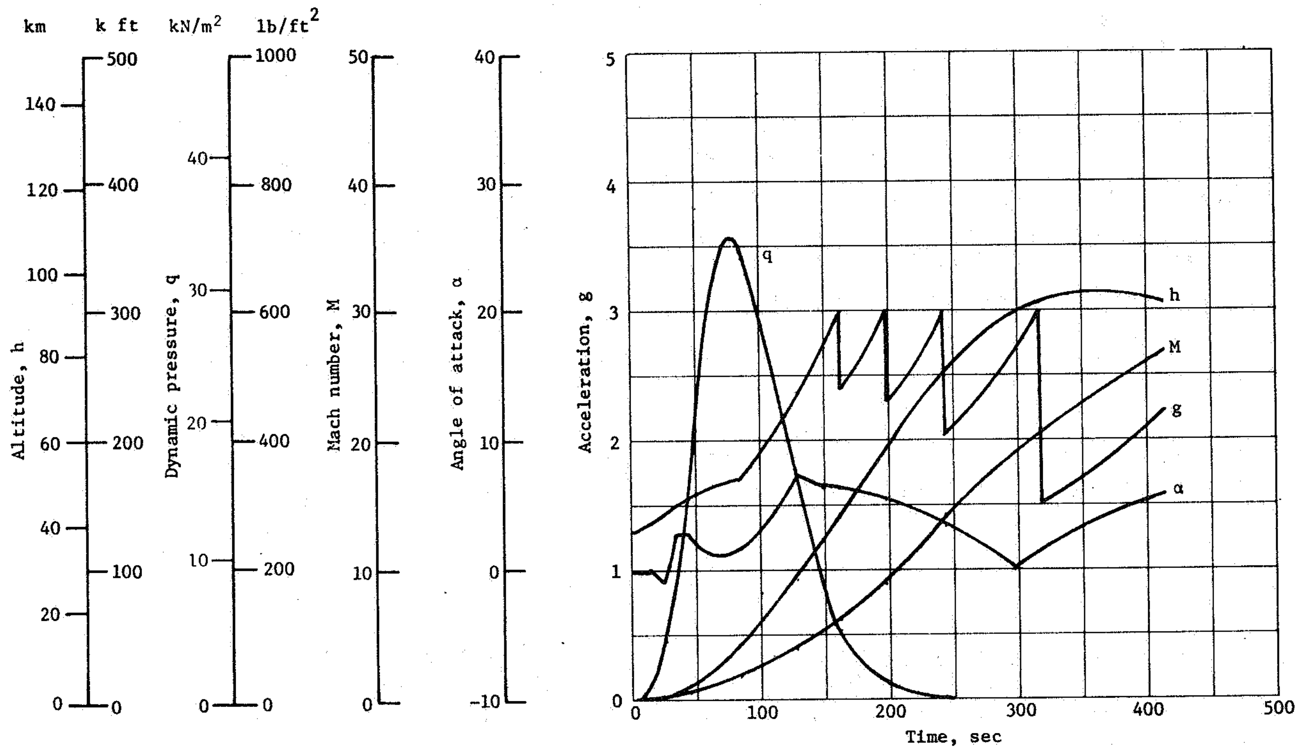


Figure 60.- VTO trajectory parameters

The numbers of single and multiple expansion ratio engines were selected on the basis of minimal vehicle dry weight holding the total number of engines constant. The required vehicle mass ratio for various engine combinations was determined using the POST trajectory program. Vehicles were sized to meet the required payload of 29 500 kg (65 000 lb). The dry weight comparison of Figure 61 shows that six single and four dual expansion ratio engines are at least 1360 kg (3000 lb) lighter in dry weight than other combinations. This engine combination was therefore selected for the VTO vehicle.

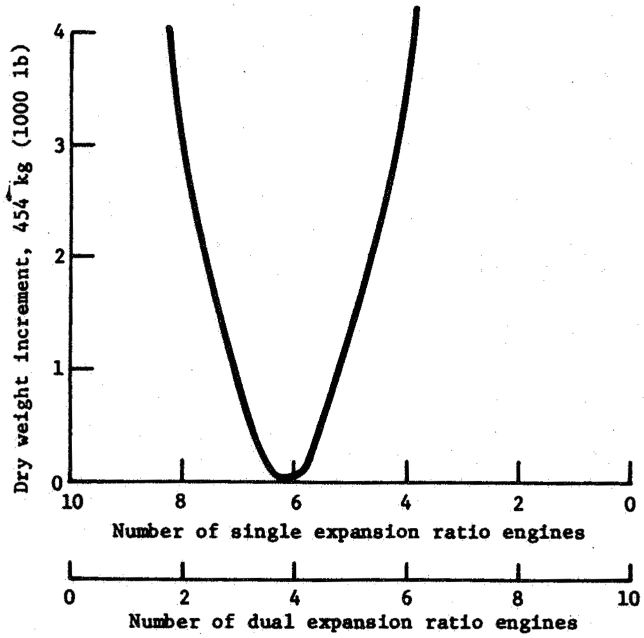


Figure 61.- Optimal engine characteristics

Effects of engine throttling were analyzed using the characteristics shown in Figure 62. Results shown in Figure 63 indicate that a lower mass ratio is required if the engines are not throttled. Virtually the same results were obtained for engines sized to provide liftoff thrust-to-weight ratios of both 1.25 and 1.30.

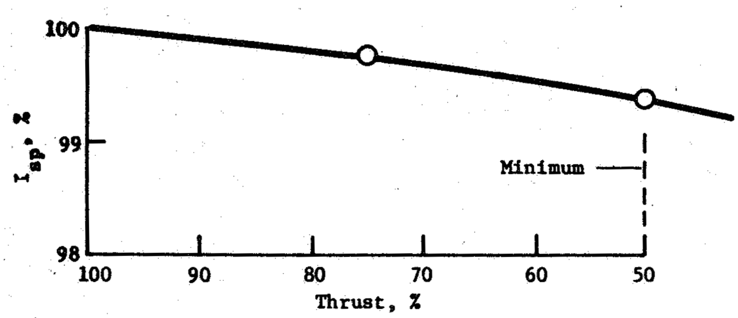


Figure 62.- Engine throttling characteristics

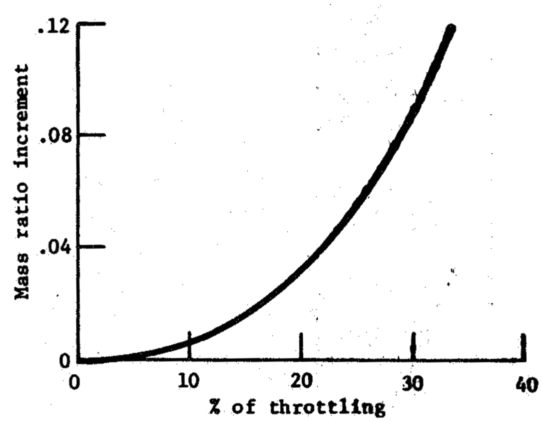


Figure 63.- Throttling effects

The effect of VTO liftoff acceleration on engine and vehicle performance was examined. The mass ratio required for a vehicle with a liftoff acceleration of 1.3 g was 0.097 less than that at an acceleration of 1.25 g. Furthermore, the corresponding dry weight and propellant weight reductions were 181 kg (400 lb) and 22 700 kg (50 klb), respectively, even though the total engine weight was approximately 1270 kg (2800 lb) heavier. The VTO was therefore designed for a 1.3 g liftoff.

Initial VTO trajectories were run at zero lift throughout the maximum dynamic pressure regime. Subsequent investigation indicates that the mass ratio requirements could be substantially reduced by a lifting trajectory. Results from the POST trajectory program indicated that the optimal value of lift was approxi-terminate constant lift was 3.6. The load imposed by the aerodynamic lift was within structural limits.

Linear nozzle vehicles.- Performance capability of VTO vehicles equipped with linear nozzle rocket engines was analyzed. To simulate the near-optimal expansion of this type of nozzle, thrust was described as a function of altitude. The fundamental trajectory shaping philosophy was identical to that of the bell nozzle configuration. The engines were throttled to maintain acceleration at or below the 3 g limit. In addition, the outer combustors were shut down at the optimum time. Results indicated that the required vehicle mass ratio was 7.893. This value was higher than that of a bell nozzle vehicle and was attributed to a lower average specific impulse caused by nonoptimum engine performance.

HTO Vehicle Performance

The HTO vehicles were launched horizontally from sea level in an easterly direction from the Eastern Test Range to produce an orbit inclination of 28.5 deg. Initial velocity at the end of the sled run was equivalent to Mach 0.6 and the relative flight path angle was 1 deg. After launch the vehicle was pitched with an angle of attack schedule for a constant g pullup. The magnitude of the pullup maneuver was varied to maximize vehicle performance but was constrained to be no greater than 1.3 g. After a specific flight path angle was reached, a constant rate of change of angle of attack was initiated and maintained until approximately 115 seconds from launch. At that time, a constant inertial pitch rate was specified, lasting until approximately 375 seconds. Here, another pitch rate was specified, lasting until burnout.

All engines were thrusting continuously after release from the sled launcher until the atmospheric pressure decreased to 15 500 N/m^2 (324 psf). The large expansion ratio nozzle was then extended. Each time the longitudinal acceleration reached 3 g, engines were shut down, beginning with the single expansion ratio engines. After all these engines were terminated, a similar sequence was used with the dual expansion ratio engines. Typical trajectory parameters for the Task 2 HTO vehicle are shown in Figure 64.

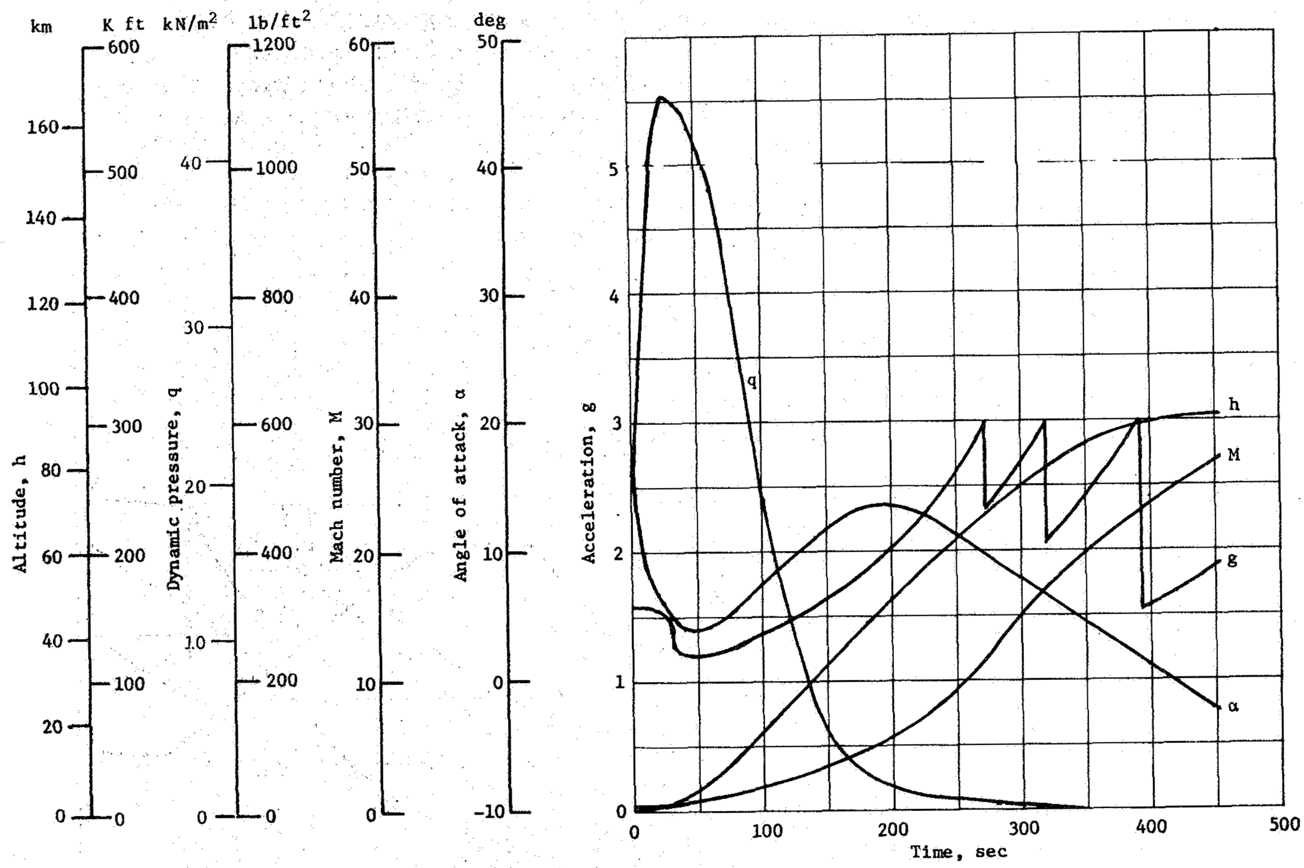


Figure 64.- HTO trajectory parameters

The sensitivity of mass ratio to initial acceleration, Δ MR/Δ $(F/W)_0$, equals -2.7. This is primarily caused by a decrease in gravity losses. However, at a launch acceleration of approximately 0.95 g, the increase in engine weight cancels the benefit of increased acceleration and vehicle dry weight. Thus the HTO vehicles were sized for 0.95 g.

An equal number of single and dual expansion ratio engines is near optimal for the HTO vehicles. Where an odd number of engines was required, the most favorable mix contained one more dual than the number of single expansion engines. This effect is attributed to an increase in average effective specific impulse. The HTO vehicle was sized for eight engines, four single and four dual.

IFF Vehicle Performance

The IFF vehicles were considered to be launched horizontally from 4570 m (15 000 ft) altitude. Initial velocity was equivalent to a Mach number of 0.75. Trajectory shaping variables and techniques were similar to those of the sled-launched vehicles. The pullup maneuver was limited to 1.05 g. Typical trajectory parameters for the Task 2 HTO inflight fueled vehicle are shown in Figure 65.

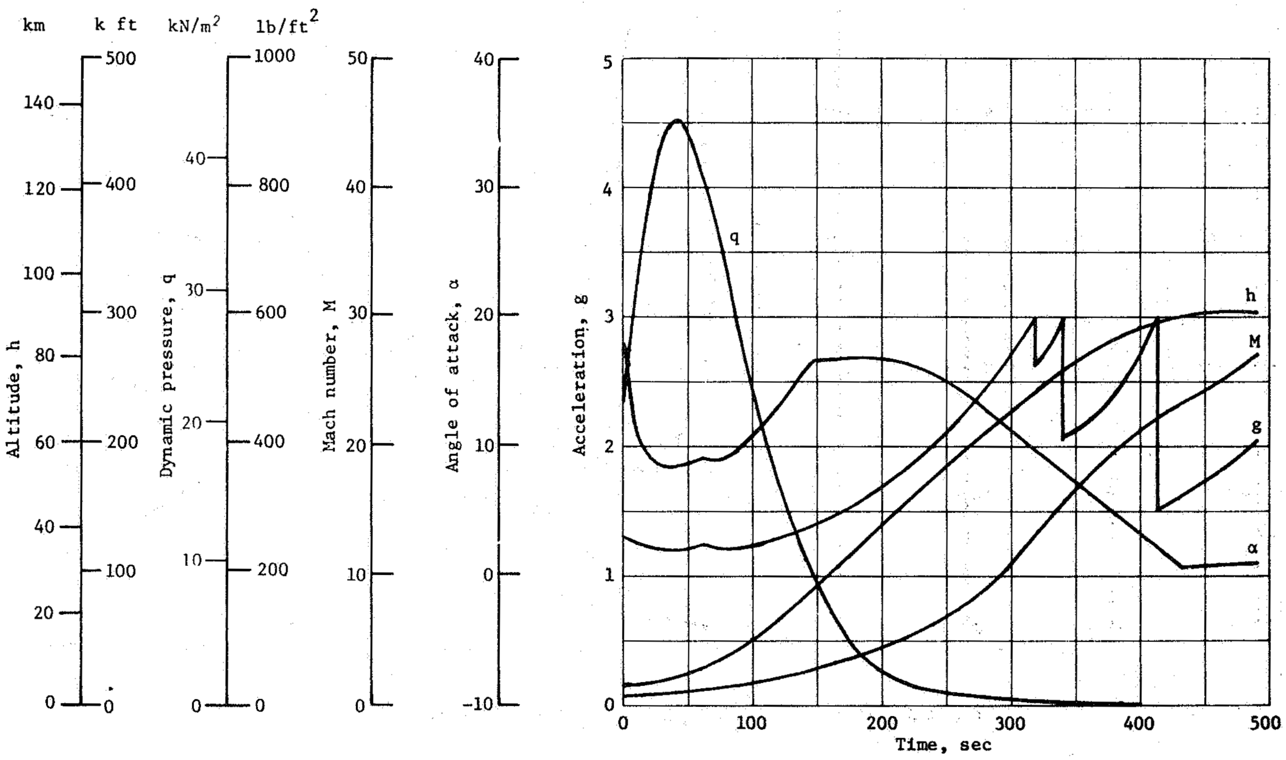


Figure 65.- IFF rocket takeoff trajectory parameters

When the expansion ratio of the single nozzle engine was increased from 35:1 to 50:1, the required mass ratio decreased by 0.06. The IFF vehicles therefore used the 50:1 expansion ratio engines.

Reentry Trajectory

Reentry was initiated at 400 000 feet at a velocity of 7800 m/s (25 600 ft/sec) and an inertial flight path angle of -0.8 deg. These conditions correspond to entry from a due East 28.5 deg inclination 370 km (200 n. mi.) circular orbit. An angle of attack of 30 deg was maintained from the initial entry until the velocity decreased to Mach 5. Angle of attack was then decreased linearly with time until a value of 20 deg was reached at Mach 4. This angle was held constant down to Mach 2, then decreased linearly to 6 deg.

An initial bank angle of 90 deg was maintained to 99 km (325 000 ft). Bank angle was then decreased to approximately 78 deg and was maintained until the Chapman heating rate parameter reached a value of 112.5. Bank angle was then modulated to maintain heating rate at this value. When the vehicle acceleration reached a level of 2.2 g, the bank angle was modulated to maintain the acceleration at 2.2 g.

After approximately 25 seconds, this mode was terminated and a series of three linear bank angle rates were initiated, each lasting 20 seconds and ending at a bank angle of 47.5 deg. This angle remained fixed until Mach 5 was reached and the vehicle was rolled out to level flight. A cross-range distance of 2070 km (1120 n. mi.) was achieved, slightly more than the required 2040 km (1100 n. mi.).

VEHICLE COMPARISON SUMMARY

The vehicles sized in the Task 2 study are summarized in Table 29. The initial aerodynamics characteristics used for vehicle trajectory analysis and vehicle sizing were revised based on the SSTO configuration and the effects on the VTO and HTO vehicles were determined.

The initially sized HTO vehicles using the revised aerodynamic characteristics have payload capabilities of 41 277 kg (91 000 lb) and 44 452 kg (98 000 lb) for the dry wing and wet wing respectively. The initially sized VTO vehicle has a payload capability of 32 493 kg (71 600 lb). The vehicles shown in Table 29 under the revised aero column are the vehicles that were resized for a payload capability of 29 484 kg (65 000 pounds).

The IFF vehicle was not resized based on the revised aerodynamics. The turbofan takeoff IFF vehicle that was sized using only LO_2 propellant refueling is not included in the table but requires a GLOW of over 3.2 million kg (7 million lb) and is not a competitive system.

TABLE 29.- VEHICLE CONCEPT COMPARISON SUMMARY - PAYLOAD = 29 500 kg (65 000 lb)

	VTO		HTO (dry wing)		HTO (wet wing)		IFF
	Initial aero	Revised aero	Initial aero	Revised aero	Initial aero	Revised aero	Initial aero
Vehicle dry							
kg	202 753	196 923	225 121	217 493	194 190	190 002	217 994
(lb)	(466 993)	(434 142)	(496 307)	(479 491)	(428 112)	(418 882)	(480 595)
Ascent propellant							
kg	1 660 998	1 626 277	1 817 463	1 681 808	1 642 748	1 502 256	1 710 969
(lb)	(3 661 873)	(3 585 326)	(4 006 819)	(3 707 751)	(3 621 640)	(3 311 907)	(3 772 042)
GLOW							
kg	1 924 654	1 883 631	2 106 198	1 960 291	1 901 441	1 752 275	1 990 279
(lb)	(4 243 136)	(4 152 695)	(4 643 368)	(4 321 701)	(4 191 956)	(3 863 105)	(4 387 815)
Sled acceleration propellant							
kg			100 326	93 172	90 718	83 415	
(lb)			(221 181)	(205 409)	(200 000)	(183 898)	
Vehicle loaded							
kg			2 206 524	2 053 463	1 992 159	1 835 703	
(lb)			(4 864 549)	(4 527 110)	(4 391 956)	(4 047 033)	

TECHNOLOGY CONSIDERATIONS

The IFF vehicle concept introduces unique concerns related to requirements for technology and flight operations. Present in-flight fueling techniques, although generally applicable, must be modified and updated to efficiently and safely rendezvous, hook up, and transfer the large quantities of cryogenic propellants. Timeline penalties or "holds" that reflect on the orbital vehicle size or design must be held to an absolute minimum. Efficient rendezvous and hook up require precision guidance and navigational integration of both vehicles, in addition to automatic deployment, positioning, and connection of the transfer lines. The two lines required will be more difficult to connect than one. They must be structurally rigid to carry unavoidable longitudinal loads. Towing, however, introduces unacceptable local structural penalties because of the large tension loads. The transfer lines must also be mutually aligned with suitable provisions for thermal contraction and heat transfer. The leak-proof disconnects required and the high capacity pumps must be developed. The IFF concept also requires development of a new tanker aircraft that is not only larger than any present aircraft, but also requires technology developments for transporting and transferring LO_2 and LH_2 propellants rapidly.

The HTO concept also introduces unique technology development requirements that are beyond "normal" growth potential. These requirements are related to design of cryogenic wet wing thermostructures and TPS integration, as well as to development of a large, high-speed, rocket-powered sled. The VTO concept offers no technology development concerns beyond "normal" growth expectations, and therefore has been selected for focusing studies of the merits of accelerated technology requirements. However, the HTO wet-wing concept is included with the VTO concept in the subsequent analyses of vehicles using accelerated technology.

PROGRAM COST ANALYSIS

Life-cycle costing techniques developed in various NASA and DOD programs were used to derive total system costs for the candidate vehicle concepts. A key element of the analysis was a highly organized data base structure originally developed during Space Shuttle Phase B studies. It consists of a fully integrated cost data bank encompassing a wide spectrum of programs from actual Martin Marietta history and other sources including NASA and DOD. The second key element was a proven, computerized cost model, COCOM II. This model, developed by Martin Marietta, includes cost estimating relationships that account for vehicle characteristics and DDT&E, production, and operations costs. Work breakdown structures, system development schedules, traffic models and operations schedules were established as bases for the cost analyses. Research costs were regarded as sunk costs and therefore were not included in the life-cycle costs.

WORK BREAKDOWN STRUCTURE

The Work Breakdown Structure (WBS) for the SSTO system is the same as used for the Space Shuttle system. This allows direct comparisons of the various WBS items to be made between the two systems. Table 30 summarizes the top level items in the SSTO system. A detailed statement on the WBS is presented in Appendix B.

TABLE 30.- WORK BREAKDOWN STRUCTURE

Level		
1	SSTO system	
2	Design and development Production	Operations
3	Program management Air vehicle	Systems engineering GSE, tests, facilities, etc
4-7 (Summary)	Structures Propulsion Avionics Life support system Power Crew Integration assembly/checkout	Management Systems analysis Test hardware Wind tunnel, Static fire, Flight: Tests Training Logistics

SYSTEM DEVELOPMENT SCHEDULE

The overall program schedule for the SSTO project, shown in Figure 66, has been designed to correlate with given milestones for the start of Phase A, the ATP, and the IOC. Milestones and activity periods are reflected on the schedule for the design, development, manufacture and test of the flight vehicle, the main engines, the launch processing system and the ground operations facilities. The activity periods for the fueling aircraft and the ground sled launch system options are also shown.

The design and development of the flight vehicles and the main engines begins at the time of Phase A go-ahead. During the period from Phase A to ATP, the design of the flight vehicle is developed, the list of materials is established, long lead orders are prepared and preorder procurement investigations are conducted. At the time of ATP, the detailed manufacturing is started; the first article (OV-1) is scheduled to be complete in early 1992 (4¼ to 5 years later). A 2-year test period, using OV-1 as the test article, is planned for a checkout of the SSTO system. Article OV-1 is later refurbished to be used as an operational flight vehicle. Ground test articles and a vehicle mockup are also scheduled to be manufactured for use in tests scheduled between early 1991 and 1994. The manufacturing of OV-2 follows OV-1, and is scheduled to be complete in late 1993 for use in the FMOF. The manufacture of OV-3, -4, and -5 is to be complete by mid-1998.

The design and development of the main engines is scheduled to start in 1983 and continues through 1991. Engine manufacturing is scheduled to start in 1989. An estimated delivery schedule based on a ten-engine VTO configuration is as follows:

Basic requirements, 5 vehicles x 10 engines per vehicle	50 engines
Spare engines, 20%	10 engines
Component spares, 20%	10 equivalent engines
Major overhaul, 50%	25 equivalent engines
Vehicle test articles, 1½ equivalent vehicles + 30% spares	20 engines
Total	115 engines and equivalent engines

	1989	1990	1991	1992	1993	1994	1995	1996	1997	1998	1999
Flight articles	----	2	4	14	14	14	14	10	10	8	5
Vehicle test articles	4	6	10	----	----	----	----	----	----	----	----
Totals	4	8	14	14	14	14	14	10	10	8	5

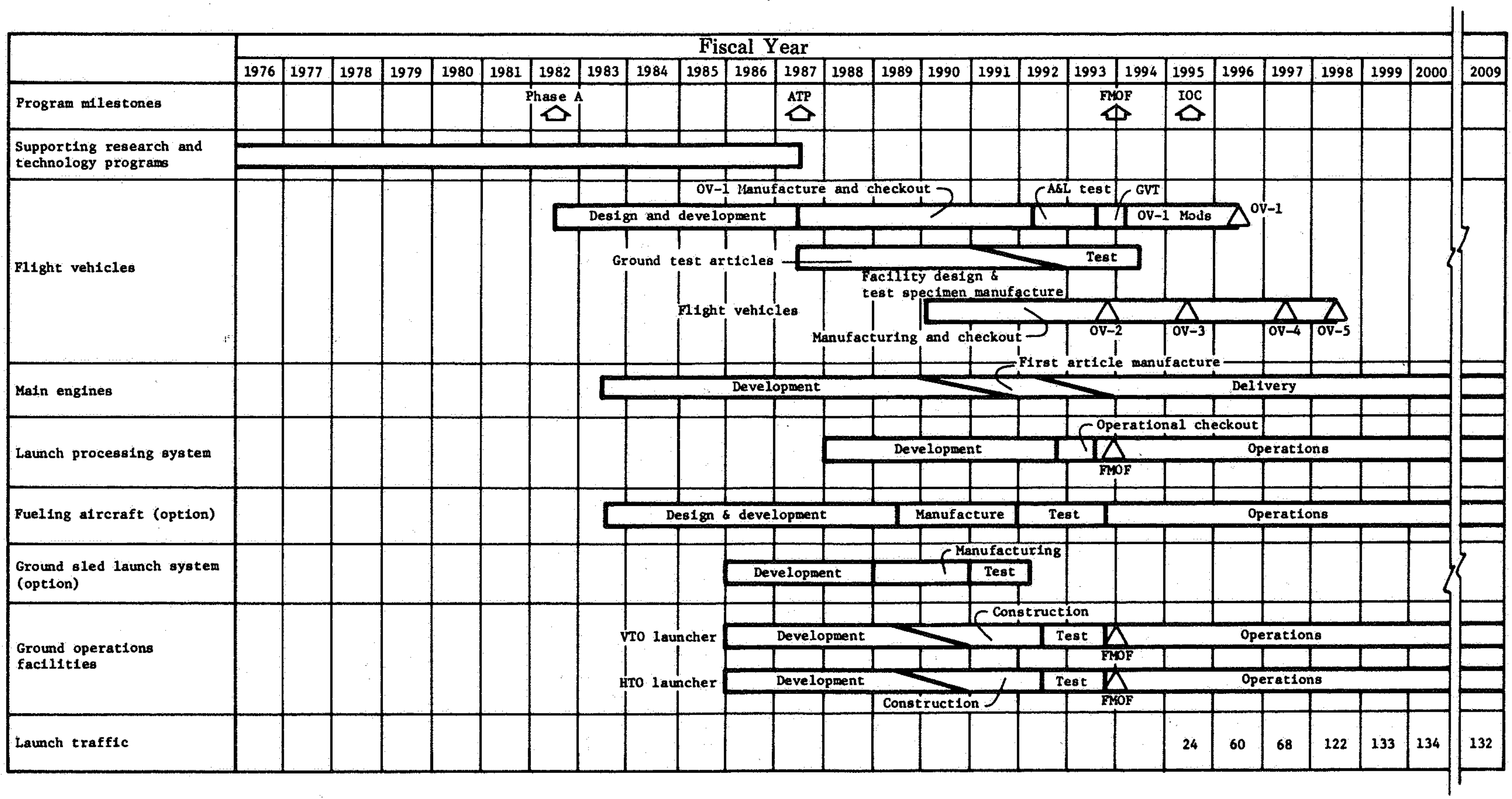


Figure 66.- System development schedule

The launch processing system development starts after the ATP and is to be complete in 1992. An operational checkout period is planned from mid-1992 through mid-1993. On completion of the checkout effort, the system will be available for operations beginning with the FMOF in 1993.

The Ground Operations Facilities require development of a vertical takeoff launcher or a horizontal takeoff launcher, and normal runways for landing and IFF takeoff. The initial development effort starts in early 1986. Construction extends from mid-1989 to mid-1992. A 1½-year test period has been scheduled before the FMOF. The SSTO system is to be completely tested and fully operational in 1995.

TRAFFIC MODEL

The October 1973 Space Shuttle Traffic Model is used as a basis for the SSTO traffic model. Table 6 (page 184) of Reference 5 illustrates a 12-year traffic summary. This 12-year summary, ending in 1991, was extended to 1994 to obtain a 15-year base representing the Space Shuttle program. This increased the total Space Shuttle traffic summary from 782 to 1061 Space Shuttle launch attempts. The number of flights per year of Space Shuttle was increased by the ratio of total SSTO flights to total Space Shuttle flights (1710/1016 = 1.6831) to obtain the number of flights per year of SSTO. The SSTO study guidelines defines the IOC data as 1995.

When the launch rate exceeds 114 launches per year, an improvement in the "average" turnaround time is expected. The number of launch attempts for the Space Shuttle and the SSTO resulting from this approach are as follows:

Traffic summary																
Space Shuttle																Total
Year	'80	'81	'82	'83	'84	'85	'86	'87	'88	'89	'90	'91	'92	'93	'94	
Launch Attempts	14	36	40	73	79	80	79	75	76	70	83	77	78	78	78	1016
SSTO																
Year	'95	'96	'97	'98	'99	'00	'01	'02	'03	'04	'05	'06	'07	'08	'09	
Launch Attempts	24	60	68	122	133	134	133	126	128	118	140	130	131	131	132	1710

GROUND OPERATIONS SCHEDULES

Launch and ground operations functions were analyzed to establish a basis for operations costs (Appendix C). The ground operations and timelines to refurbish and prepare the SSTO for succeeding launches are illustrated in Figure 67. The initial step in the flow is the safing and deservicing of the SSTO. This step has been estimated to be performed in the first 10 hours after landing. The payload removal and the maintenance activities can then begin. Systems retest and reverification is conducted in parallel immediately following the maintenance activity. The installation of new payloads then begins at the 22nd hour after landing over a nine hour period. After the installation, an integrated test is conducted in the orbiter processing facility, and the SSTO then is moved to the vertical assembly building (VAB) for mating with the launch platform. The SSTO and the launch platform interfaces are verified in the VAB. The SSTO is moved to the launch pad at the 43rd hour after landing. The remaining 17 hours are spent on the launch pad where the propellants and consumables are installed and the vehicle is prepared for relaunching 60 hours after landing.

Based on 114 launches per year and the 60 hour turnaround cycle, the ground operations can be performed as shown in Figure 68. There is an average of 18 hours between each ground operation activity. This period can be used to accomplish any activity that is not in the normal flow or to accommodate any anomalies that may occur.

The assumed mission model results in an average launch every 3.2 days or an average turnaround of 16 days for a 5-vehicle fleet. The requirement of a 60-hour turnaround for ground operations is driven by the assumption of a capability for processing only one vehicle at a time. By providing multiple facilities, the 24 hr/day pace could be relieved to a more reasonable schedule allowing for overtime to accommodate anomalies. The probable use of two launch sites (ETR and WTR) would in fact require at least two such facilities.

COST MODEL

The COCOM program calculates the cost of each WBS element using either preassigned algorithms or discrete costs assigned to selected elements. Equations and data are in an array matrix format enabling the program to draw on design and pricing spread coefficients, schedule, quantities, and other programmatic data as outlined in Figure 69. The costs are determined using Fiscal Year 1976 dollars and are later escalated and/or discounted as desired.

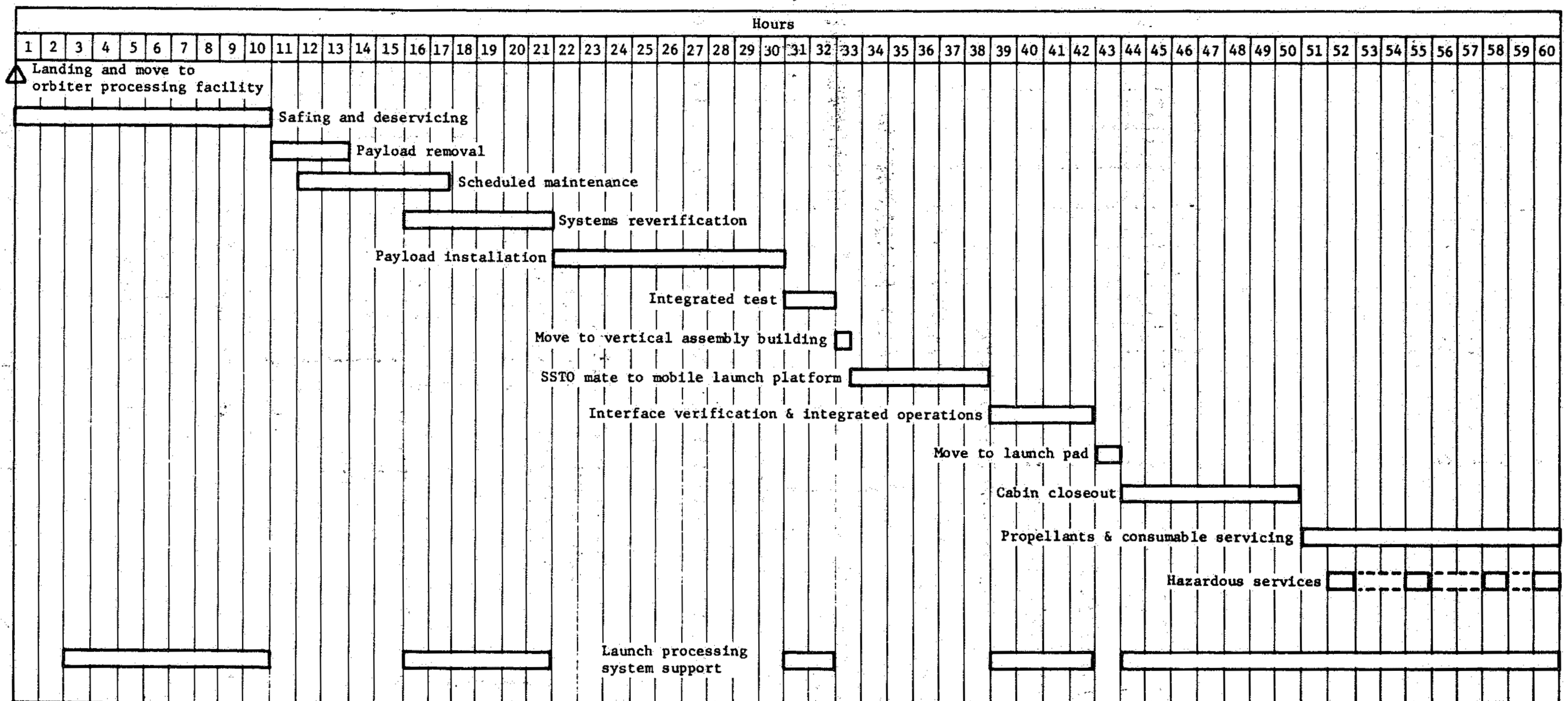


Figure 67.- Typical flow of ground operations

Discrete cost inputs were used for cost elements not significantly impacted by vehicle size. Examples are the avionics subsystem, batteries, horizontal flight test operations, and flight test instrumentation. Input data sources include Space Shuttle program costs and inhouse data based on aircraft and spacecraft experience.

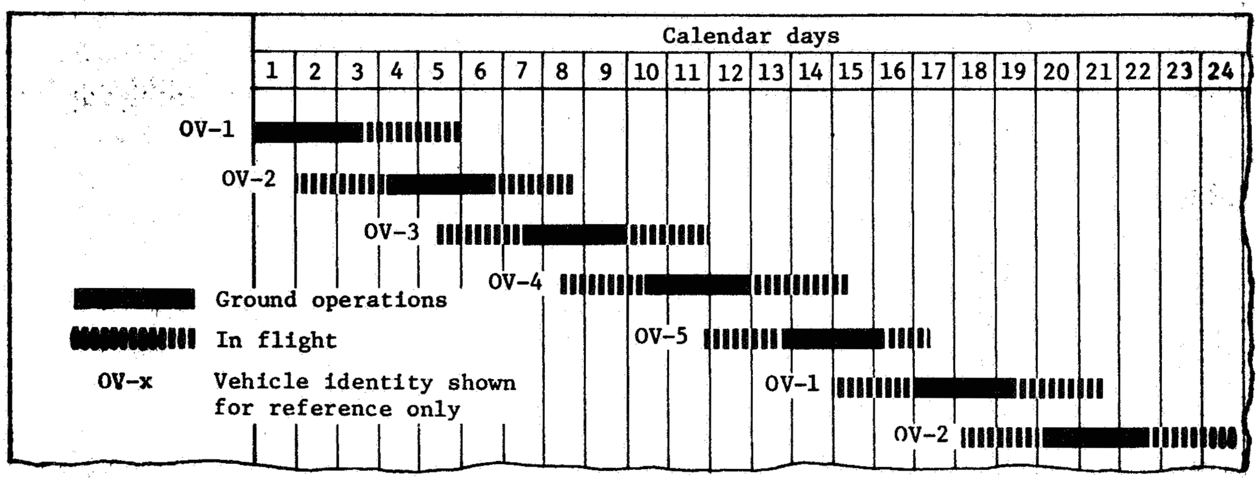


Figure 68.- Typical ground operations schedule

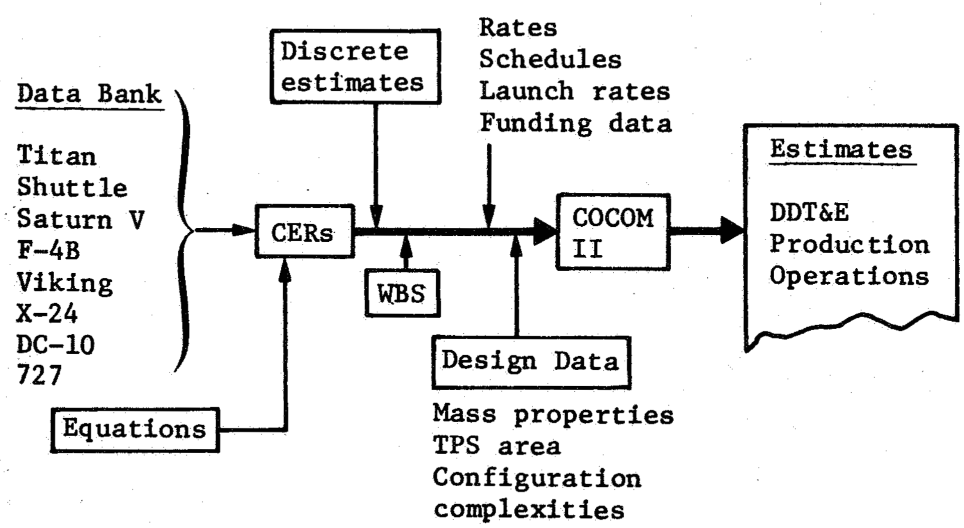


Figure 69.- Cost model flow diagram

The equations used in the cost model are:

$$C = F_1 \times F_2 \times F_3 \times F_4 \times R \times C_P \times (W)^{\alpha} \times (Q)^{\beta}$$

or

$$C = R \times C_P$$

where

F_1 = Access area complexity factor

F_2 = Density factor

F_3 = Configuration Complexity Factor

F_4 = Material complexity factor

R = Rate constant (labor and overhead rates)

C_P = Reference cost

W = Design parameter (weight or area)

α = Scaling exponent

Q = Production quantity

β = Learning Curve

The first four terms are further defined as follows:

$$F_1 = \left(\frac{4 \times \text{area of hatches and doors}}{\text{total wetted area}}\right) + 1$$

$$F_2 = \left(\frac{\text{total dry weight}}{\text{total moldline volume}}\right)^{0.25}$$

F_3 = 1 for launch vehicles
2 for transport aircraft

F_4 = 1 for aluminum structure
2 for composite structure

This equation represents the requirements of an engineered cost estimate. The total system cost is derived using as many elements as possible, with cost equations relating the elements and reflecting in detail the interaction of the elements when the system is developed, produced, operated, and supported.

GENERAL COST ESTIMATING GUIDELINES

Advanced CER methods relative to Space Shuttle technology were developed and improved during the Space Shuttle Phase A and B studies. The estimating relationships were validated against current Space Shuttle costs and applied to the SSTO costing. The prime contractor approach was assumed that allocates 50% of the total cost to materials and subcontracts, with the prime contractor retaining management, systems engineering, structures, landing gear, TPS, electrical, and final assembly checkout functions.

Separate cost classifications were identified for which labor, overhead, and G&A rates were developed. Rates typical of a large aircraft manufacturer were as follows:

Engineering	$23.80/hr
Tooling	$20.50/hr
Manufacturing	$18.55/hr
Materials & subcontract	27.5%
Major subcontract	3.5%

Engines and facilities were priced as GFE, without additional overhead or fee. The control document *Cost per Flight*, JSC Vol XVI, formed a baseline for costing purposes. Vehicle design life was set at 500 flights with an engine design life of 250 cycles. The launch interval of 16 days per vehicle requires an engine design life of only 172 cycles.

DDT&E COSTING

Guidelines

The following costing guidelines were used for DDT&E:

(1) The schedule was in no way restrictive;

(2) Program management was set at 6% of total program cost;

(3) Systems Engineering was 12% of total program cost, less program management;

(4) Facility construction assumed maximum use of existing facilities with the addition of two pads and one orbiter maintenance facility each at KSC and WTR;

(5) A nominal flight test program is assumed. Static fire, horizontal taxi tests, and vertical takeoff use a flight article;

(6) Three sets of AGE were deliverable;

(7) Flight test spares were delivered in this phase.

Cost Estimating Relations (CER)

Three groups of CERs that represent the basis for estimating design, tooling, test, and materials and subcontract costs are tabulated in Table 31. The body structure labor costs have been correlated with S-IV B LO_2 and hydrogen tankage and F4B data. The design complexity factors increase structures design costs by a factor of 2.4; tooling factors increase the tooling costs by a factor of 2.8 times the S-IV B baseline. The weight scaling exponents are 0.485 and 0.766 for design and tooling, respectively.

TABLE 31.- COST ESTIMATING RELATIONSHIPS*

Cost element	Area, m^2 (ft^2)	Labor, hr/m^2 (hr/ft^2)	Unit cost, $\$/m^2$ ($\$/ft^2$)	Total cost, $ Millions
Thermal protection system	4 192 (45 126)			
Design		220 (20)	5 000 (465)	21
Test		530 (49)	12 600 (1 174)	53
Tooling		250 (23)	5 000 (465)	21
Materials & subcontract			4 290 (399)	18
	Weight, kg (lb)	Labor, hr/kg (hr/lb)	Unit Cost, $/kg ($/lb)	
Body structure	52 753 (116 299)			
Design		46 (21)	1 100 (499)	58
Test		73 (33)	1 720 (782)	91
Tooling		255 (116)	5 270 (2 390)	278
Materials & subcontract			342 (155)	18
Aerodynamic control surface	28 767 (63 420			
Design		137 (62)	3 232 (1 466)	93
Test		82 (37)	1 947 (883)	56
Tooling		379 (172)	7 786 (3 432)	224
Materials & subcontract			485 (220)	14

*Task 2 VTO example vehicle.

Cost Results

Table 32 tabulates DDT&E costs for each of the vehicle concepts. Weight differences among the vehicles result in cost differences. The largest cost differences, however, result from considerations of the sled costs for the HTO concept, namely $122 Million for sled vehicle design and $328 Million for sled launch facilities.

TABLE 32.- DDT&E COSTS

Cost element	Dollars in millions			
	VTO	HTO		IFF
		Dry	Wet*	
Program management	$ 330	$ 347	$ 335	$ 332
Systems engineering and integration	590	619	599	591
Air vehicle design	2317	2491	2380	2441
Ground support equipment	296	296	296	296
Training	172	172	172	172
Systems test and evaluation				
Test hardware†	904	918	875	928
Test operations	390	390	390	390
HTO vehicle design		122	122	
Logistics	45	45	45	45
Facilities	466	756	756	466
Fee	458	483	466	459
Total	$5968	$6639	$6436	$6120

*LO_2 in wing
†2.5 equivalent air vehicles

PRODUCTION COSTS

Guidelines

Production cost CERs were developed for manufacturing, material, and labor. Sustaining engineering and tooling factors of 8% and 10% respectively were used. Four flight vehicles were priced for each configuration concept, applying a 95% learning curve. Due to schedule delays between deliveries, no learning credit was given for test article production. Production control, quality control, shipping, and other manufacturing departments were considered as overhead. Final assembly, installation and checkout was priced in accordance with historical data as 25% of total production costs

Production CERs

Costs in hours and dollars per unit weight, tabulated in Table 33, are results of design parameters, costs, and complexity factors using the general CER equation previously described. Derivation of hours per unit value can be determined by dividing the labor costs by the weight times the labor rate of $18.55 per hour. Comparisons of hours per pound among cost elements are invalid, however, because the equation relationships are exponential. The S-IVB structures cost per pound is displayed to provide a point of correlation with fuselage structures costs. A complexity factor of 1.8 for the fuselage structure was input to the cost model. With this factor, the data for both SSTO and S-IVB correlate to $190\ (W)^{0.766}$.

TABLE 33.- FIRST ARTICLE COST CERS*

Cost element	Area, m^2 (ft^2)	Labor, hr/m^2 (hr/ft^2)	Material unit cost, $\$/m^2$ ($\$/ft^2$)	Labor cost $ millions
TPS	4 192 (45 126)	323 (30)	1380 (113)	25.1
	Weight kg (lb)	hr/kg (hr/lb)	$/kg ($/lb)	
Crew station	1 450 (3 200)	207 (94)	275 (125)	5.6
Body structure	52 750 (116 299)	48 (22)	68 (31)	48.4
Aerodynamic control surfaces	28 770 (63 420)	48 (22)	55 (25)	25.6
Landing gear	6 960 (15 343)	46 (21)	48 (22)	5.8
S-IVB structures	8 690 (19 165)	40 (18)	35 (16)	6.4

*Example vehicle Task 2 VTO, FY 1976 dollars.

Cost Results

Variations of the production costs of the vehicle concepts are within 10% (Table 34). Concepts of construction are similar with the exception of the HTO concept with LO_2 tanks in the wings. Costs for avionics, ECLS, power, and hydraulics are the same for each concept. These costs exclude the aircraft tanker production costs. The sled costs are included in DDT&E.

TABLE 34.- PRODUCTION COSTS

Cost element	Dollars in millions			
	VTO	HTO		IFF
		Dry	Wet*	
Structures	$ 307	$ 363	$ 309	$ 346
Thermal protection	40	42	48	39
Landing gear	22	25	22	39
Propulsion	354	292	291	251
Avionics	101	101	101	101
ECLS	28	28	28	28
Power, hydraulics	149	144	153	150
Final assembly and checkout	197	209	198	195
Sustaining engineering	41	45	41	45
Sustaining tooling	52	56	52	57
Fee	108	115	108	108
Total	$1399	$1420	$1351	$1359
First article cost	$ 362	$ 367	$ 350	$ 371

*LO_2 in wing

OPERATION COSTS

Operations costs for SSTO systems were initially estimated using the approach of modifying present Space Shuttle operations cost projections for application to a 15-year 55% program. The primary modifications were to delete the Space Shuttle costs related to the external tank (ET) and the solid-rocket boosters (SRB). This approach led to a cost estimate of $6.6 million per launch for SSTO (VTO) compared to $13.9 million per launch for Space Shuttle, based on fiscal year 1976 dollars.

A second more fundamental approach was taken to reflect the potential simplification and combinations of launch and flight operations for an SSTO. This approach involved a functional analysis, anticipating that the next 15 years of Space Shuttle activities provide time for substantial cost reduction improvements. These projected improvements were based on considerations of the automation (computerization) of many functions, as well as the future Space Shuttle operations experience and the less complex SSTO flight vehicle with self-checkout capabilities. Guidelines and results of this approach are presented here.

The SSTO operations costs are based on 1710 total flight attempts over a 15-year period beginning in 1995. The number of flights each year (page 116) are estimated using the 12-year Space Shuttle traffic model extended to a 15-year period. Five flight vehicles are available, three at ETR and two at WTR. Costs are included for new launch pads, or sleds, on existing land. Costs of spares are based on Titan experience and projection for SSTO operations. Flight and launch operations are predominantly repetitive; ground based data systems and flight monitoring are largely automated. Most functions, therefore, can be performed by technicians rather than engineers, significantly minimizing launch and flight operations cost.

A result of the functional analysis was the 60-hour ground operations timeline shown in Figure 67. Manhours and costs to support these functions were estimated and used to develop the costs per flight shown in Table 35. This table shows Space Shuttle data for comparison, indicating significantly smaller costs projected for SSTO operations.

These smaller costs can be achieved with "normal" technology growth focused in improving onboard flight and ground support systems. Examples for operations technology emphasis are as follows:

(1) Onboard flight systems designed with automated self-test and checkout capabilities;

(2) Support systems designed with simplified prelaunch and on-orbit monitoring software and control-center staffing.

Space Shuttle operations costs (ref. 5) have been a basis for deriving SSTO operations costs. In deriving SSTO costs, the WBS (Appendix B) conforms to the cost element structure of Reference 5.

The Space Shuttle baseline program costs of $10.45 million was updated to Fiscal Year 1976 dollars by a factor of 1.32. Propellant quantity requirements were derived from NASA/KSC engineering information. Propellant and gas costs were derived from in-house data, Linde Corporation, and other sources. Costs were used for LO_2 and LH_2 were $0.08/lb and $1.00/lb, respectively. The operations costs of the SSTO concepts vary directly with the propellants required. Other cost variations depend on tanker operations, engine quantities or sled operations. An analysis of the launch and flight operations manpower requirements and costs is in Appendix C.

TABLE 35.- OPERATIONS COSTS PER FLIGHT

	Space Shuttle*		VTO	HTO FY '76 $		IFF
	FY '72 $	FY '76 $	FY '76 $	Dry	Wet†	FY '76 $
KSC civil service	0.51	0.67	0.092	0.092	0.092	0.092
Launch operations	2.00	2.75	0.858	0.875	0.815	0.937
Flight operations (JSC)	2.21	2.92	0.703	0.703	0.703	0.703
Refurbishment	0.42	0.55	0.077	0.077	0.077	0.077
Solid rocket booster	3.33	4.40				
External tank	1.75	2.31				
Engines	0.23	0.30	0.210	0.168	0.168	0.168
HTO				0.022	0.022	
Tanker						0.342
Totals	10.45	13.90	1.940	1.937	1.877	2.319

*Control document, JSC 07700, Volume XVI

†LO_2 in wing

LIFE-CYCLE COST RESULTS

A summary of projected total program life-cycle costs is shown in Table 36. Space Shuttle costs, shown for comparison, are based on 1,016 launches whereas SSTO costs are based on 1,710 launches, both over a 15-year period of operations. Discounted values are shown for Space Shuttle and VTO programs at a 10% rate. Space Shuttle costs were discounted from the 1973 start of DDT&E; SSTO costs were discounted from 1976. These dates are selected as being the years of decision making.

Technology growth provided by the Space Shuttle program is, of course, a prerequisite for the development of the SSTO program. Also, significant reductions in Space Shuttle operations costs should be anticipated as repetition of mission functions and more automation is experienced. The SSTO costs, however, being considerably less than Space Shuttle costs, indicate that R&T focused on advanced transportation systems will have an important payoff.

TABLE 36.- LIFE CYCLE COSTS

	Space Shuttle		VTO		HTO					
					Dry		Wet*		IFF
	FY '76 $	Discounted	FY '76 $	Discounted	FY '76 $	Discounted	FY '76 $	Discounted	FY '76 $
DDT&E	5 499	3976	5 968	1 777	6 639	1 979	6 436	1 906	6 120
Production	1 000	655	1 399	281	1 420	285	1 351	271	1 359
Operations	14 052	3699	3 317	249	3 312	248	3 210	253	3 965
Totals	20 551	8270	10 684	2 307	11 371	2 512	10 997	2 430	11 444

*LO_2 in wing

Perturbations on SSTO costs were examined from several aspects. If, for example, the production learning curve is reduced from 95% to 85%, approximately $283 million would be saved. Production of one less vehicle (four instead of five) would save $300 million. Increasing the mission success ratio from 92.5% to 95% would reduce the number of launch attempts required, thereby reducing the operations costs over 15 years by $86 million.

The cost analysis has reflected the advantages of "normal" growth in technology that will result from both continued research focused on SSTO requirements and from related future Space Shuttle and aircraft experience. Selection of thermostructural designs that use aluminum tanks as well as lightweight composites has allowed us to calculate costs without introducing any abnormal cost-complexity factors. Costs of TPS have been based on, in part, our background with projecting costs of RSI in many other applications. The cost analysis has used a rational approach and provided meaningful results.

Major results of the vehicle design weight analyses and program cost analyses are shown on Table 37. The weights of the VTO and HTO concepts are for vehicles sized using revised aerodynamics. Dry weight is a figure of merit for comparing concepts and this parameter is least for the HTO vehicle. Other figures of merit are total program costs and the cost per pound of payload in orbit; these are least for the VTO vehicle. For comparison, the Space Shuttle merit index is \$509/kg (\$231/lb) and \$134/kg (\$60.9/lb) based on fiscal year 1976 and discounted dollars respectively.

TABLE 37.- COMPARISON OF VEHICLE CONCEPTS, WEIGHTS, AND COSTS

	VTO	HTO		IFF
		Dry wing	Wet wing	
Dry weight				
kg	196 923	217 493	190 002	217 994
(lb)	(434 142)	(479 491)	(418 882)	(480 595)
GLOW				
kg	1 883 631	1 960 291	1 752 275	1 990 279
(lb)	(4 152 695)	(4 321 701)	(3 863 105)	(4 387 815)
Total program costs dollars in billions				
Fiscal year 1976	10.7	11.4	11.0	11.4
Discounted 10%	2.3	2.5	2.4	2.5
Merit index* dollars/kg (dollars/pound)				
Fiscal year 1976	69.3 (31.4)	71.0 (32.2)	68.8 (31.2)	85.1 (38.6)
Discounted 10%	5.3 (2.4)	5.3 (2.4)	5.2 (2.3)	6.4 (2.9)

*(Operations costs)/(mission success factor) (no. of flights) (payload)

A mission success factor of 0.925 was used for the HTO and the IFF concepts because the sled or the tanker aircraft introduce risks that may degrade success similar to the Space Shuttle ET/SRB stages. A mission success factor of 0.95 was used for the VTO based on the following expected improvements:

(1) SSTO will have an additional 15 to 20 years experience in technology and Space Shuttle flights;

(2) SSTO will have a higher flight rate than Space Shuttle, thereby exposing and solving flight problems in a shorter time span;

(3) SSTO will use Space Shuttle technology in various subsystems, thereby minimizing new high risk technology items;

(4) The VTO is a single stage flight system.

Based on the assessments of vehicle cost-performance merits, the VTO and the wet-wing HTO concepts were pursued during the Extended Performance Studies. Advanced technology assessments were focused on the VTO concept.

ADVANCED TECHNOLOGY ASSESSMENT

The Advanced Technology Assessment task identifies technology areas offering the greatest potential cost/performance/ benefits for SSTO VTO vehicles that can result from focused R&T and additional funding. The additional funding represents R&T funding above the "normal" level previously defined. Technology parameters were selected that offered a potential for significant improvement in vehicle dry weight. These parameters related to the primary technology areas of materials, structures, and propulsion, as well as secondary technology areas taken as a whole and vehicle design criteria and design margin requirements. Research and technology programs were then identified that could be implemented to pursue the improvements in the technology parameters. These R&T activities were selected using the following general guidelines:

(1) Each program represents a definable set of R&T activities that lead to improvements in related parameters;

(2) Each program is essentially independent of other programs in terms of its goals and activities, although combinations of programs may lead to common vehicle objectives;

(3) Each program is defined in sufficiently general terms to include a broad scope (matrix) of related R&T activities;

(4) Each program is considered as a major candidate for identification in the NASA RTOPs, and can include subsets of RTOPs that support the program.

The goals of the R&T programs in terms of vehicle parameter improvements and the associated man-years of effort were estimated using delphi techniques for a 95% total confidence interval, i.e., the tolerances for the parameters and funding levels were estimated so the total intervals included 95% of the anticipated total range. The manloadings for these tasks for the years 1975-1988 were converted to 1975 (Fiscal Year 1976) dollars. The costs of any additional materials and facilities expenditures also were included.

Each technology improvement for the various R&T programs was used to calculate its overall effects on vehicle size and weights. These perturbed vehicle data were incorporated in a cost model to determine the total life cycle costs (LCC) for the improved operational vehicle, assuming start of the DDT&E phase in 1982 and last operational flight in 2009. Both the R&T funding and the life cycle costs were expressed in 1975 dollars and then discounted at a nominal rate of 10%.

Cost/performance/benefit figures of merit for the various technology improvements were defined using combinations of the discounted and undiscounted R&T and LCC values and the improvements in vehicle weights. These data were a basis for assessments of the merit of advanced technologies.

IDENTIFICATION OF PERTURBED PARAMETERS

The first step in the Advanced Technology Assessment was to identify the technology parameters that could offer a significant reduction in SSTO dry weight. These parameters, identified in Tables 38 and 39, were selected based on the previous two task activities, as well as awareness of possible new technology programs. The improved values of these parameters, which may result with accelerated R&T funding, were then based on the projection for "normal" technology growth as well as judgements of further technology growth potentials.

IDENTIFICATION OF RESEARCH AND TECHNOLOGY PROGRAMS

Based on the preceding selection of perturbed parameters, twelve research programs were selected for assessment of the potential benefits of accelerated funding and emphasis. Seven of the twelve relate to advancements in the materials, structures, and system support areas and the remaining five relate to the propulsion areas. These twelve areas are summarized in Table 40.

The funding levels and required overall activities for each selected R&T program are given in Figure 70. The materials, structures, and system support programs are planned to start in 1977 and to encompass a 10 to 12 year period. With the exception of the integration engineering program, each of the programs consists of a period for an analysis of the design and materials, optimization of the design, development of material characteristics and manufacturing techniques, small scale tests, and large scale tests. The five propulsion technology advancement programs are scheduled to start in 1976 and to be completed by 1984. Each of these programs will consist of an analysis of the design concept, materials characterization or laboratory tests, and component and subsystem tests. The objectives, activities, and test programs of each of the twelve programs are given in the following subsections.

TABLE 38.- PROPULSION PARAMETERS

Parameter to be perturbed \ Basis for Improvement	Improved combustion efficiency	Improved cooling	Advanced design pumps	Better nozzle efficiency	Higher chamber pressure	Triple point and slush propellants	Reduced design margin requirements
Main engine specific impulse	X	X		X	X		X
Main engine thrust/weight		X	X		X		X
Propellant density						X	X
Reaction control and orbit maneuvering specific impulse	X	X			X		X

TABLE 39.- MATERIALS, STRUCTURES, AND DESIGN OPTIMIZATION PARAMETERS

Parameter to be perturbed \ Basis for Improvement	Improved materials and fabrication	Improved thermostructural design technology	Flight trajectory optimization	Reduced design margin requirements - More thorough R&T testing - Improved design criteria	Improved aerodynamic configuration, systems/subsystems integration, vehicle design arrangements
Thermal protection system weight	X	X	X	X	X
Propellant tank weight	X	X		X	X
Structure weight other than tanks - wings and vertical tail, thrust structure, skirts, payload doors, crew compartment, etc	X	X		X	
Systems/subsystems weight	X		X	X	X
Reduction in dry weight margin requirements			X	X	

TABLE 40.- ADVANCED TECHNOLOGY PROGRAMS SELECTED FOR ASSESSMENT

Materials, structures, and design optimization	Propulsion
1. Thermal protection systems	6. Main engine injectors/chambers/ nozzles
2. Propellant tanks	7. Main engine pumps
3. Wing and vertical tail structures	8. Main engine cooling
4. Thrust structures	9. OMS/RCS systems
5. Miscellaneous structures	10. Triple point propellants
Secondary technologies	Design criteria
11. Subsystems weight reduction	12. Integration engineering

Thermal Protection Systems (TPS)

This R&T program will concentrate on accelerated research to improve the vehicle thermal protection system (TPS) in terms of (1) maximizing performance, reliability, and reuse, and (2) minimizing the complexity associated with design, analyses, fabrication, installation, maintenance, and quality assurance. The R&T emphasis will be placed on, but not limited to, reusable surface insulation systems improvements. Advancements in the characteristics of thermal protection systems using reusable nonmetallics, high temperature metallics, and combinations thereof will be pursued with the focus on SSTO applications. Activities are enumerated in the following paragraphs.

TPS analysis and design.-

(1) Improve analytical methods for evaluating TPS performance using materials characteristics, laboratory, and flight data.

(2) Develop TPS design concepts including interfaces with vehicle structures. Analyze performance as related to various vehicle configurations and aerothermodynamic flight environments, and operational environments.

(3) Provide goals and approaches toward developing new TPS materials and improving known materials.

(4) Analyze alternative manufacturing and quality assurance techniques and facility requirements.

Research and laboratory tests.-

(1) Obtain TPS materials and subsystem characteristics using wind tunnel, plasma arc, and mechanical test facilities. Upgrade wind tunnel and plasma arc facilities to more closely represent flight environments.

(2) Develop new and improved material compositions and formulation techniques.

(3) Evaluate applicability of non-destructive test methods and equipment.

Subsystem tests.-

(1) Perform structural/environmental tests on small and full-scale TPS panels. Include ground tests and flight tests (Space Shuttle and aircraft such as YF-12 and X-24C).

(2) Perform verification of non-destructive evaluation techniques.

(3) Perform work/time studies to support cost analyses on maintenance, repair and refurb activities affecting turnaround time.

(4) Develop manufacturing, assembly, and maintenance processes.

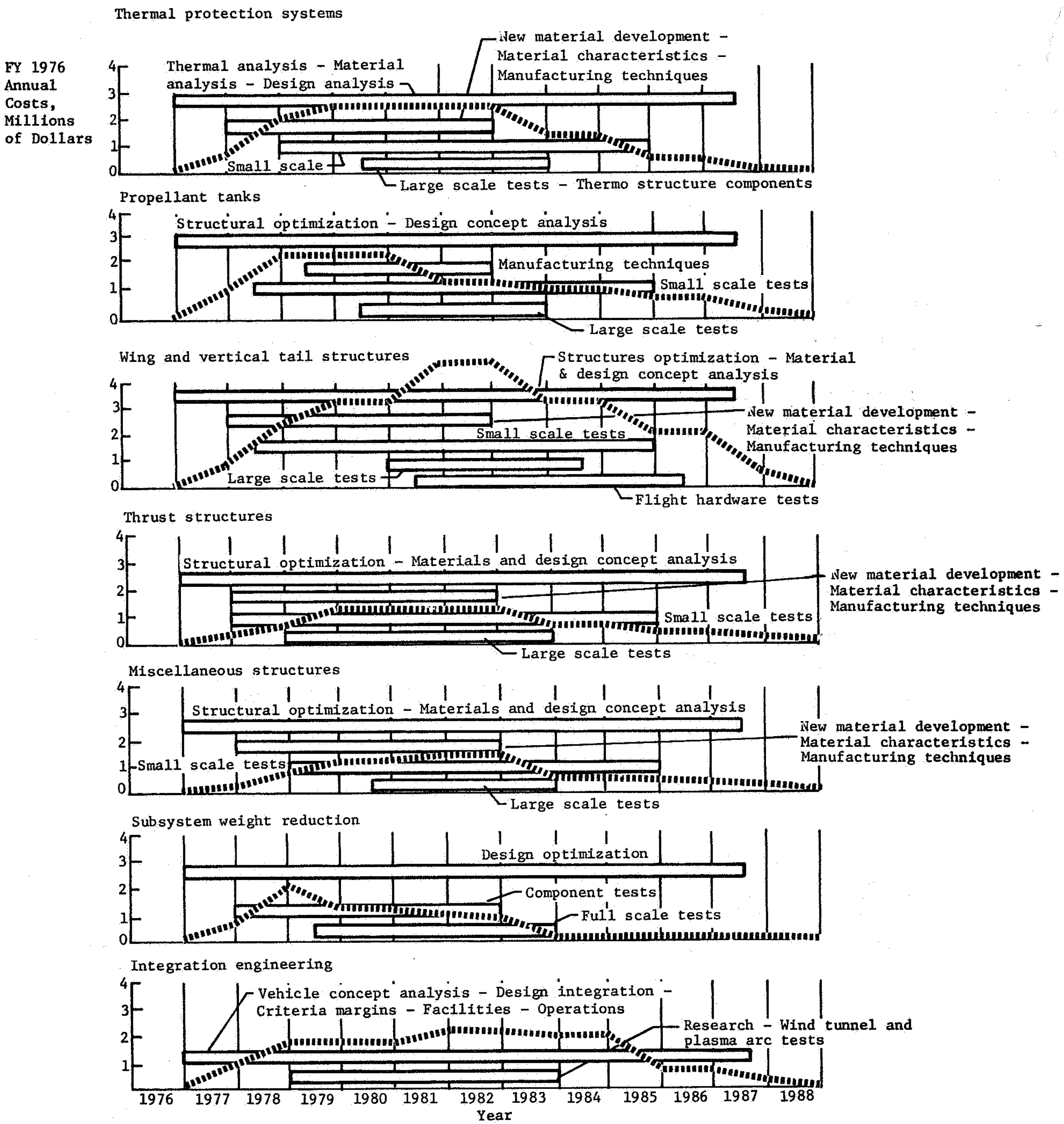
Propellant Tanks

The objective of this program will be to improve the propellant tank design technology level. This development will include such areas as main propellant tank, RCS/OMS, and propellant feed systems. Activities are listed in the following paragraphs.

Structural optimization and design.-

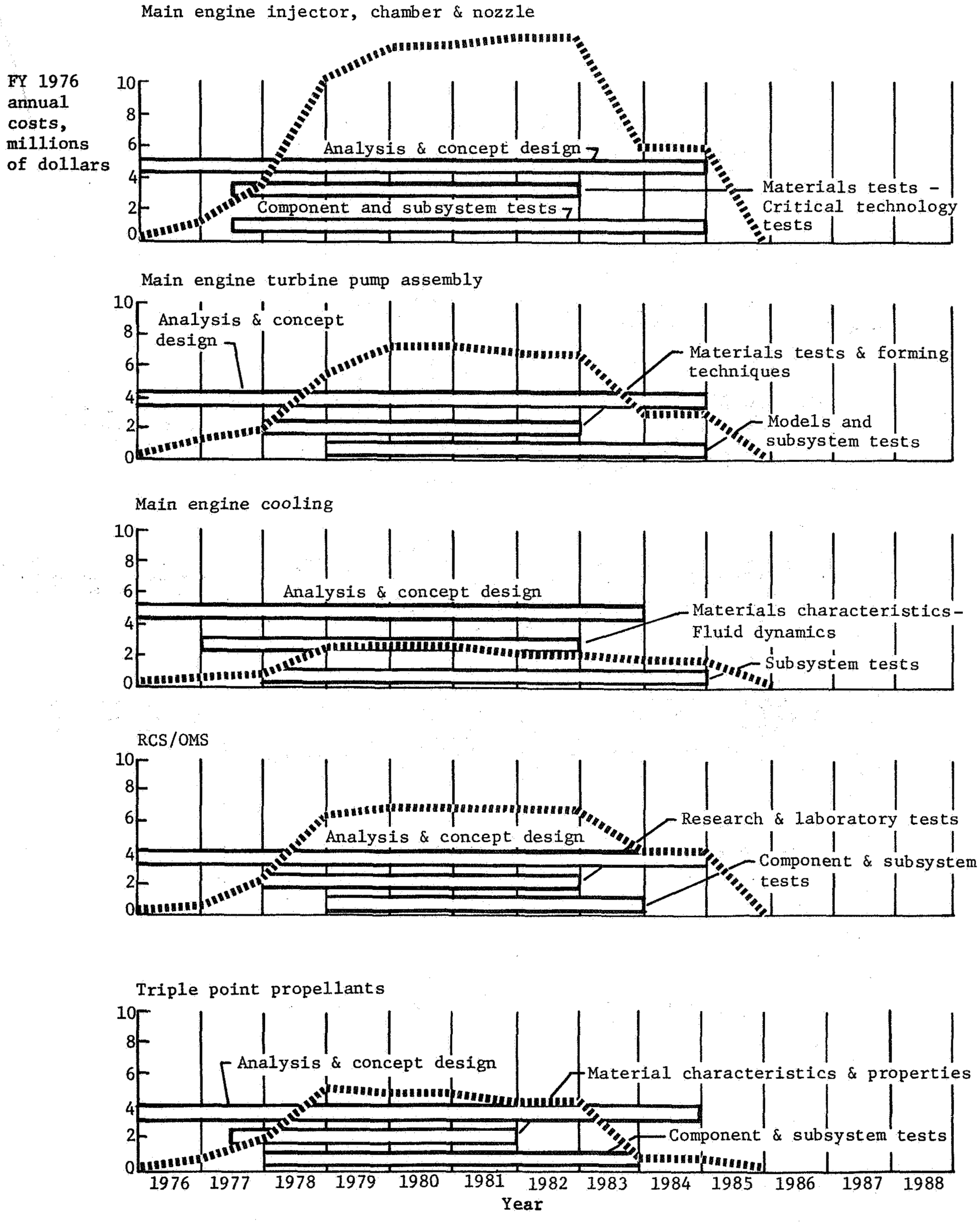
(1) Focus on propellant tank design and optimization to improve analytical methods for predicting failure modes.

(2) Design propellant tanks to improve the basic structural layout and construction, as well as the feed systems, propellant utilization features, and interfaces with the TPS and other structures. Apply advanced composite materials when applicable.



Materials, structures, and support technology areas

Figure 70.— Accelerated R&T programs and annual costs.



Propulsion technology areas

Figure 70.- Concluded

Research and laboratory tests.-

(1) Determine the characteristics of the tank materials in SSTO flight environments so optimal use can be made of them, minimizing design margin requirements.

(2) Develop the manufacturing technology required to use materials of interest combined with tankage configurations.

(3) Accelerate material testing to increase tankage reliability in the area of fracture mechanics.

Subsystem tests.- Conduct small scale and large scale structural and environmental tests on selected tank structural concepts.

Wing and Vertical Tail Structures

This program will improve the structures technology area for application to the wing and vertical tail structural assembly. These improvements will encompass such items as control surfaces, control actuators, fuselage interfaces, carrythrough structure, wing propellant tanks, composite materials, and TPS integration. Activities are as follows:

Structural optimization and design analysis.-

(1) Define and analyze alternative concepts for structural materials and optimization. Materials with high strength to weight and high modulus to weight properties, such as the advanced composite filaments - graphite, boron, borsic and Kevlar families - will be analyzed in various matrices to provide minimum weight structures. Design optimization will include design layouts, finite element thermostructural modeling, external load, and TPS design.

Research and laboratory tests.-

(1) Accelerate development of advanced composite material for both higher efficiencies and lower costs. Material characteristics will be determined for application to the SSTO environment.

(2) Determine updated manufacturing technology to handle the new materials of interest.

Subsystem test.-

(1) Conduct both small and full-scale structural and environmental tests of typical wing and vertical tail structural sections.

(2) Flight test selected designs to be used for the development/verification process. Test platforms such as the YF-12, X-24C, and Space Shuttle will be available for these tests.

Thrust Structures

This R&T program will improve thrust structure design concepts leading to reduced weight, using advanced materials, design concepts, and manufacturing techniques. Activities are detailed in the following paragraphs.

Structural optimization and design.-

(1) Develop concept designs for thrust structures using advanced composite materials such as the graphite/epoxy and boron/epoxy families, integrated with alternative engine/airframe/tank arrangements.

(2) Establish environmental criteria (loads, vibration, noise, thermal, life) for SSTO thrust structures.

(3) Perform loads analyses of concept designs with improved computer synthesis models.

(4) Analyze potential manufacturing techniques and requirements.

Research and laboratory tests.-

(1) Accelerate advanced composite material development to increase efficiency and lower costs. Determine material characteristics.

(2) Fabricate thrust structure samples and perform structural and environmental tests. Test various fabrication techniques to improve manufacturing technology.

Subsystem tests.-

(1) Fabricate small and large scale thrust structure elements using selected advanced materials and manufacturing techniques.

(2) Perform structural and environmental tests as a basis for evaluation of design concepts, techniques, and analysis methods.

Miscellaneous Structures

The objective of this program will be to improve the design technology level of a number of secondary structural systems.

These systems will include nontank structures, access doors, landing gear interfaces, subsystem interfaces, the payload compartment, the crew compartment with docking mechanisms, and the internal heating control. The following activities will be performed.

Structural optimization and design.- Define and analyze alternative concepts for structural materials and optimization. Materials with high strength to weight and high modulus to weight properties, such as the graphite/epoxy and boron/epoxy advanced composite families, will be analyzed to provide minimum weight structures. Design optimization will include design layouts, finite element thermostructural modeling, loads and environmental effects.

Research and laboratory tests.-

(1) Accelerate advanced composite material development to increase efficiency and lower costs. Determine material characteristics.

(2) Develop the manufacturing technology required to use advanced materials in the design of these structures.

Subsystem tests.-

(1) Conduct small scale and large scale structural and environmental tests on selected structural concepts.

(2) Some flight test verification may be required. High speed aircraft such as the YF-12, X-24C, and Space Shuttle can be used in the test program.

Main Engine Injectors/Chambers/Nozzles

The objective of this program will be to improve the main engine technology level through more intensive development of the components that comprise the thrust chamber assembly. Activities are outlined in the following paragraphs.

Thrust chamber assembly analysis and design.-

(1) Develop injector pattern to improve performance, reduce pressure drop, improve combustion stability, and reduce required chamber length.

(2) Develop injector structural design to accommodate pattern changes and to minimize weight. This effort will include investigation of new manufacturing techniques, combustion chamber size, shape and structural configuration to reduce weight, improve performance, and maintain sufficient cooling.

(3) Explore applicable engine cycles to improve performance and, in particular, to extend engine life and reusability. The design optimization will include examination of oxidizer and fuel-rich preburners or gas generators and component integration to reduce size and weight of valves, lines, etc.

(4) Evaluate the injector and combustion chamber technology improvements derived for primary thrust chambers as applied to gas generators and preburners. In addition, investigate higher performing fuel-rich and oxidizer-rich designs. Injector pattern development with reduced pressure drop will contribute to higher subsystem efficiency and reduced weight.

Research and laboratory tests.-

(1) Investigate higher strength metals and composite materials to establish applicability, material characteristics, and design criteria.

(2) Develop new manufacturing and forming techniques paralleling the design concepts.

Subsystem tests.-

(1) Build and test components and subassembly hardware representing the most promising concepts and cycle features.

(2) Although no new major facilities will be necessary, test fixtures, new instrumentation and modification of existing facilities will be required.

Main Engine Pumps

This R&T program will be directed toward turbine and propellant pump improvements that increase efficiencies, improve component life, and reduce weight. Activities are as follows.

Turbopump assembly design analysis.-

(1) Optimize propellant impeller, diffuser, and blade design. Particular emphasis on cavitation phenomena definition and suppression will be required. Technology of low NPSH pumps is emphasized.

(2) Investigate turbine cooling extensively to extend life and to improve performance by allowing higher turbine inlet gas temperatures.

(3) Pursue pump bearing development and seals improvements (possibly through seal elimination).

Research and laboratory tests.-

(1) Accomplish new materials research for application to pumps, turbines, and drive mechanisms.

(2) Investigate new manufacturing and forming processes.

Subsystem tests.-

(1) Manufacture and test components and subassembly test hardware using existing facilities.

(2) Some modification of existing facilities, some new fixtures, and additional instrumentation will be required.

Main Engine Cooling

The objective of this program will be to reduce weight through improved thrust chamber and turbine cooling. Activities are detailed in the following paragraphs.

Thrust chamber assembly and turbine design analysis.-

(1) Reduce system pressure losses by developing better cooling techniques. Lower pressure losses reduce pump discharge pressures and power requirements, resulting in smaller lighter pumps, turbines, and preburners or gas generators.

(2) Investigate oxidizer or both propellants as the coolant. Because of density, higher liquid oxygen pump discharge pressures are easier to attain than those with liquid hydrogen. The system can be optimized for minimum engine weight or higher chamber pressures.

(3) Research new materials and coatings toward minimizing the heating effects on engine hardware thus reducing cooling requirements and giving longer life.

Research and laboratory tests.-

(1) Test new materials and coatings for effectiveness and to establish design criteria.

(2) Test propellants to better define their fluid properties, heat transfer characteristics, and cooling capabilities.

(3) Conduct model heat transfer tests of representative cooling configurations.

Subsystem tests.- Conduct single component and subassembly tests of the best designs using LO_2 and/or both propellants as coolants.

OMS/RCS

The objective of this program is to establish advanced engine and propellant system performance and design criteria for orbit maneuvers and reaction control systems using LO_2/LH_2. Activities are listed in the following tabulation.

OMS/RCS analysis and conceptual design.-

(1) Pursue LO_2/LH_2 pressure-fed and pump-fed engine and/or thruster development using the technology developed from larger scale hardware as well as new concepts tailored to fast acting small impulse bit thrusters. Additional research into pulsing LO_2/LH_2 attitude control thrusters will develop high-performance, low-weight auxiliary propulsion systems.

(2) Continue studies and development on gaseous propellant supply systems common to OMS/RCS and/or auxiliary power systems.

(3) Focus particular emphasis on cryogenic liquid propellant, used in either liquid or gaseous phase, employing a common, relatively small, accumulator or boost service tank to reduce overall system weight and minimize residuals.

(4) Zero-g propellant acquisition techniques will continue to be developed.

Research and laboratory tests.-

(1) Evaluate and test new materials to establish design criteria.

(2) Evaluate new manufacturing and forming techniques.

Subsystem tests.-

(1) Test thrust chamber, turbopump, and storage and feed system components and subsystems.

(2) No significant increase in facilities requirements are foreseen.

Triple-Point Propellants

This program will establish ground and flight system concepts, design criteria, and processes necessary to develop complete large-size oxygen-hydrogen propulsion systems that use cryogenic propellants that are stored at pressures and temperatures near their triple-point. Activities are enumerated in the following paragraphs.

Propellant system and engine analysis and design.

(1) Conduct propellant storage, feed, loading, and pressurization subsystems analyses to determine their respective operating and performance characteritics. Define thermal influences on tank and system design. Determine the effects of triple-point and slush propellant fluid properties on line pressure drop, valve design, and pump power requirements.

(2) Establish the impact of dense cryogenic fluids on engine pumps, bearings, seals, cooling passages, and engine performance. The lower propellant enthalpy level will result in somewhat lower total effective system performance.

(3) Evolve the most economical method for producing, maintaining, and using triple-point or slush propellants.

Research and laboratory tests.-

(1) Develop new materials for insulation, bearings, and seals.

(2) Determine propellant characteristics and fluid properties.

Subsystem tests.-

(1) Build and test engine and propellant system components and subassemblies.

(2) Demonstrate and evaluate pilot facilities for producing the propellants.

Subsystems Weight Reduction

This R&T program will address performance and weight reduction potentials in subsystems such as electrical, hydraulics, pneumatics, life support, avionics, and communications. Detailed activities are as follows.

Subsystems design optimization.-

(1) Perform weights/cost/performance benefits analysis covering all secondary technology areas.

(2) Establish weight goals.

(3) Evaluate designs and advanced concepts for cost and weight effectiveness.

Configuration analysis.-

(1) Evaluate configuration alternatives.

(2) Perform system trades.

Subsystem tests.- Perform test-bed demonstrations of improved subsystem components.

Integration Engineering

This R&T program will consist of systems engineering, design engineering, and costing activities to provide technical focusing and integration of SSTO-related research programs. The activities include continuing efforts toward establishing research goals, guidelines, design criteria and margin requirements, and cost/performance benefits of SSTO vehicle and program concepts. Activities are listed in the following paragraphs.

Research program development and technical management.-

(1) Identify and prioritize research activities (RTOPS) including their goals, schedules, and funding based on continued analysis of cost/performance/benefits.

(2) Provide design goals, design criteria, and design margins for the advanced technology programs.

(3) Develop mission models and traffic models for SSTO vehicles.

(4) Analyze functional and facility requirements for DDT&E, production, and operations.

(5) Perform total program cost analyses and figure-of-merit analyses. Include potential budgetary limitations and payload cost considerations.

Support technology and configuration analysis.-

(1) Perform design engineering functions using updated technology projections and improved analysis techniques.

(2) Evaluate configuration alternatives, considering mission/payload models, flight performance optimization, flight stability augmentation, main propulsion system characteristics, and cost/performance benefits.

(3) Improve analysis techniques including aerothermodynamics, computer-aided design, and performance optimization with operational constraints (e.g., mission profiles for standard and emergency flight situations, aerodynamic and aerothermodynamic optimizations). Improve computer program capabilities for SSTO vehicle and program synthesis for more sophistication in optimizing and modeling.

(4) Perform parametric wind tunnel tests and plasma arc tests of flight configurations, and evaluate Space Shuttle data as a basis for better analytic capabilities (e.g., viscous effects, boundary-layer transition). Upgrade test facilities to better simulate flight environments.

PERTURBED PARAMETERS AND EFFECTS ON VEHICLE

The technology improvements for each of the twelve R&T programs selected were expressed in terms of subsystem weight reductions for the materials and structural programs and in terms of component weight reduction and I_{SP} improvement for the propulsion programs. With the exception of the integration engineering program, the system improvements are tabulated in Table 41 along with the resultant improvements in SSTO dry weight and gross liftoff weight. As can be observed, all the improved parameters result in significant savings to both vehicle dry weight and GLOW.

Each row of data in Table 41 pertains to the given technology program, each taken individually as if it were the only accelerated program that would be given the required additional funding. In a subsequent section (Figures of Merit), example results of implementing meaningful combinations of programs are shown.

The integration engineering task proved to be the most subjecttive of all the technology improvement analyses. This task, which included the reduction of design criteria and margin requirements for all phases of the vehicle design, produced a weight saving that was significantly larger than any of the other programs.

The revised vehicle weights that were based on each technology improvements were used to determine the perturbed life cycle costs expressed in FY 1976 dollars and then discounted using a 10% rate. The Δ life cycle costs, obtained by subtracting the baseline VTO costs from the perturbed costs, are shown in Table 41.

FIGURE OF MERIT ANALYSIS

The R&T funding levels, the technology improvements, and the life cycle costing are all important parameters of the Advanced Technology Assessment task. The problem was to combine these parameters in the most effective manner so that the net benefits from the twelve research programs could easily be discerned. A number of figures of merit were selected as meaningful, including the savings in technology parameters for a given R&T cost input, the net cost savings of the combined R&T and life cycle costs, and the savings in life cycle costs for a given R&T cost level.

The improvements in technology parameters are plotted in Figure 71 as a function of the total discounted R&T funding for each program with the exception of the Integration Engineering task. The range of expected values for each R&T program, as obtained from the original 95% confidence interval estimates, are also plotted. These values are also given as Δ Technology and $\Delta \$ R_D$ in Table 41.

The saving in discounted life cycle costs as a function of the discounted R&T total funding for each technology program is shown in Figure 72, along with the associated variances. The slopes of the nominal and upper and lower limit values (i.e., $\Delta \$ LCC_D / \Delta \$ R_D$) have been plotted in Figure 73. Any program with a slope less than one will not save as much in LCC as it cost in R&T dollars. These slopes for both the discounted and undiscounted values are tabulated in Table 41.

A third figure of merit is the net cost of the program expressed in discounted dollars; i.e., the saving in life cycle costs minus the additional expenditures required for the associated accelerated R&T technology program. These net savings figures are tabulated in Table 41. Several of the propulsion programs have the potential for a net loss on the technology programs.

TABLE 41.- FIGURES OF MERIT

Technology program		ΔTechnology: ΔI_{sp} – sec, ΔW, kg (lbm)	ΔTechnology: Δ% ± Tolerance	$\Delta\$ R_D$, <R> $M	max	min	ΔW_{dry}, kg (lbm)
1. Thermal protection systems	ΔW	-2970 ± 450 (-6550 ± 1000)	-7.5 ±1.1	10.5 <18.1>	14.2	9.0	-9 510 ± 1 450 (-20 960 ± 3 200)
2. Propellant tanks	ΔW	-2940 ± 1360 (-6480 ± 3000)	-10 ± 4.6	9.0 <15.3>	12.6	7.3	-9 230 ± 4 270 (-20 350 ± 9 420)
3. Wing & vertical tail structures	ΔW	-3750^{+1470}_{-1730} $\left(-8260^{+3250}_{-3820}\right)$	$-13^{+5.1}_{-6.0}$	16.4 <30.8>	22.9	12.6	$-12\ 470^{+4\ 910}_{-5\ 770}$ $\left(-27\ 500^{+10\ 820}_{-12\ 720}\right)$
4. Thrust structures	ΔW	-590^{+360}_{-500} $\left(-1300^{+800}_{-1100}\right)$	$-8.1^{+5.0}_{-6.9}$	4.5 <8.2>	7.2	3.3	$-2\ 990^{+1\ 840}_{-2\ 540}$ $\left(-6\ 600^{+4\ 060}_{-5\ 590}\right)$
5. Miscellaneous structures	ΔW	-1360^{+360}_{-1350} $\left(-3000^{+800}_{-2970}\right)$	$-12.0^{+3.2}_{-11.8}$	4.5 <8.0>	6.8	3.5	$-3\ 910^{+1\ 040}_{-3\ 870}$ $\left(-8\ 630^{+2\ 300}_{-8\ 540}\right)$
6. Main engine (injectors/ chambers/ nozzles)	ΔI_{sp}	$+6^{+2}_{-5}$	$+1.3^{+.4}_{-1.0}$	47.8 <78.8>	66.3	37.8	$-11\ 010^{+8\ 650}_{-3\ 640}$ $\left(-24\ 280^{+19\ 060}_{-8\ 020}\right)$
	ΔW	-45 ± 14/Eng (-100 ± 30/Eng)	-1.1 ± 0.33				
7. Main engine pumps	ΔI_{sp}	+2.5 ± 1	+0.53 ± 0.21	26.3 <40.0>	35.4	18.7	-4 670 ± 1 820 (-10 300 ± 4 010)
	ΔW	-23 ± 7/Eng (-50 ± 15/Eng)	-0.55 ± 0.16				
8. Main engine cooling	ΔI_{sp}	+1.5 ± 1	+0.32 ± 0.21	10.5 <17.2>	13.7	8.3	-3 300 ± 1 830 (-7 280 ± 4 030)
	ΔW	-45 ± 9/Eng (-100 ± 20/Eng)	-1.1 ± 0.22				
9. OMS/ RCS	Propellant ΔW	-830^{+120}_{-140} $\left(-1830^{+260}_{-320}\right)$	$-6.4^{+.92}_{-1.1}$	26.8 <44.4>	36.0	20.0	$-1\ 920^{+270}_{-330}$ $\left(-4\ 240^{+600}_{-720}\right)$
	Dry ΔW	-90 ± 10 (-190 ± 22)	-3.4 ± 0.4				
10. Triple-point propellants	ΔW	-1810^{+420}_{-1150} $\left(-4000^{+930}_{-2530}\right)$	$-6.2^{+1.4}_{-3.9}$	17.5 <27.1>	23.2	12.7	$-15\ 380^{+3\ 570}_{-9\ 770}$ $\left(33\ 910^{+7\ 830}_{-21\ 380}\right)$
11. Subsystem weight reduction	ΔW	-1360 ± 680 (-3000 ± 1500)	-9.7 ± 4.9	4.8 <7.3>	6.5	4.3	-4 140 ± 2 070 (-9 130 ± 4 570)
12. Integration engineering		Refer to text.					

Note: The symbols < > indicate undiscounted nominal values of added R&T funding <R> and resulting LCC savings <LCC>.

TABLE 41.- Concluded

ΔGLOW, kg (lbm)	Δ$ $DDT\&E_D$, $M	Δ$ $Prod_D$, $M	Δ$ OPS_D, $M	Δ$ max LCC_D, min <LCC> $M	Δ$$LCC_D$ − Δ$$R_D$, $M	Δ$LCC / Δ$R	Δ$$LCC_D$ / Δ$$R_D$ max min
−77 950 ± 11 900 (−171 850 ± 26 240)	15.8	4.5	2.8	23 26.6 <121> 19.6	12.5 17.6 5.4	6.67	2.19 2.9 1.4
−75 700 ± 35 050 (−166 890 ± 77 260)	30.0	8.6	4.1	43 62.6 <203> 22.8	34.0 55.3 10.2	13.33	4.76 8.5 1.8
−102 240 +40 250 −47 310 (−225 400 +88 730 −104 290)	80.7	11.1	6.1	98 143.3 <405> 59.4	81.6 130.7 36.5	13.16	5.99 11.4 2.6
−27 630 +17 230 −23 380 (−60 920 +37 990 −51 550)	12.9	4.2	2.6	20 36.4 <403> 7.6	15.5 33.1 0.4	12.66	4.40 11.0 1.0
−32 050 +8 550 −31 710 (−70 660 +18 840 −69 900)	21.1	6.0	4.1	31 61.9 <161> 22.7	26.5 58.4 15.9	20.0	6.90 17.7 3.3
−137 330 +45 510 −110 210 (−302 750 +100 330 −242 960)	34.9	9.4	6.5	51 91.6 <276> 34.0	3.2 53.8 −32.3	3.51	1.07 2.4 0.5
−58 660 ± 22 580 (−129 320 ± 49,780)	11.2	2.8	2.5	17 23.1 <87> 9.9	−9.3 4.4 −25.5	2.02	0.65 1.2 0.3
−39,460 ± 22 020 (−86,900 ± 48 550)	8.8	2.8	1.4	13 20.3 <68> 5.7	2.5 12.0 −8.0	3.95	1.24 2.4 0.3
−22 000 +3 120 −3 740 (−48 510 +6 870 −8 240)	6.0	2.0	1.0	9 10.5 <46> 7.7	−17.8 −9.5 −28.3	1.04	0.34 0.5 0.2
−133 330 +30 970 −84 080 (−293 940 +68 270 −185 360)	44.8	11.0	−7.0	49 79.6 <117> 37.5	31.5 66.9 14.3	4.31	2.80 6.3 1.6
−33 980 ± 16 990 (−74 910 ± 37 450)	12.0	3.4	1.1	17 25.0 <79> 8.0	12.2 18.5 1.5	10.87	3.55 5.9 1.3

Propulsion

6 Main engine injectors/chambers/nozzles

7 Main engine pumps

8 Main engine cooling

9 OMS/RCS

10 Triple-point propellants

Structures and TPS

1 TPS

2 Propellant tanks

3 Wing & vertical tail structure

4 Thrust structure

5 Miscellaneous structure

11 Subsystem weight reduction

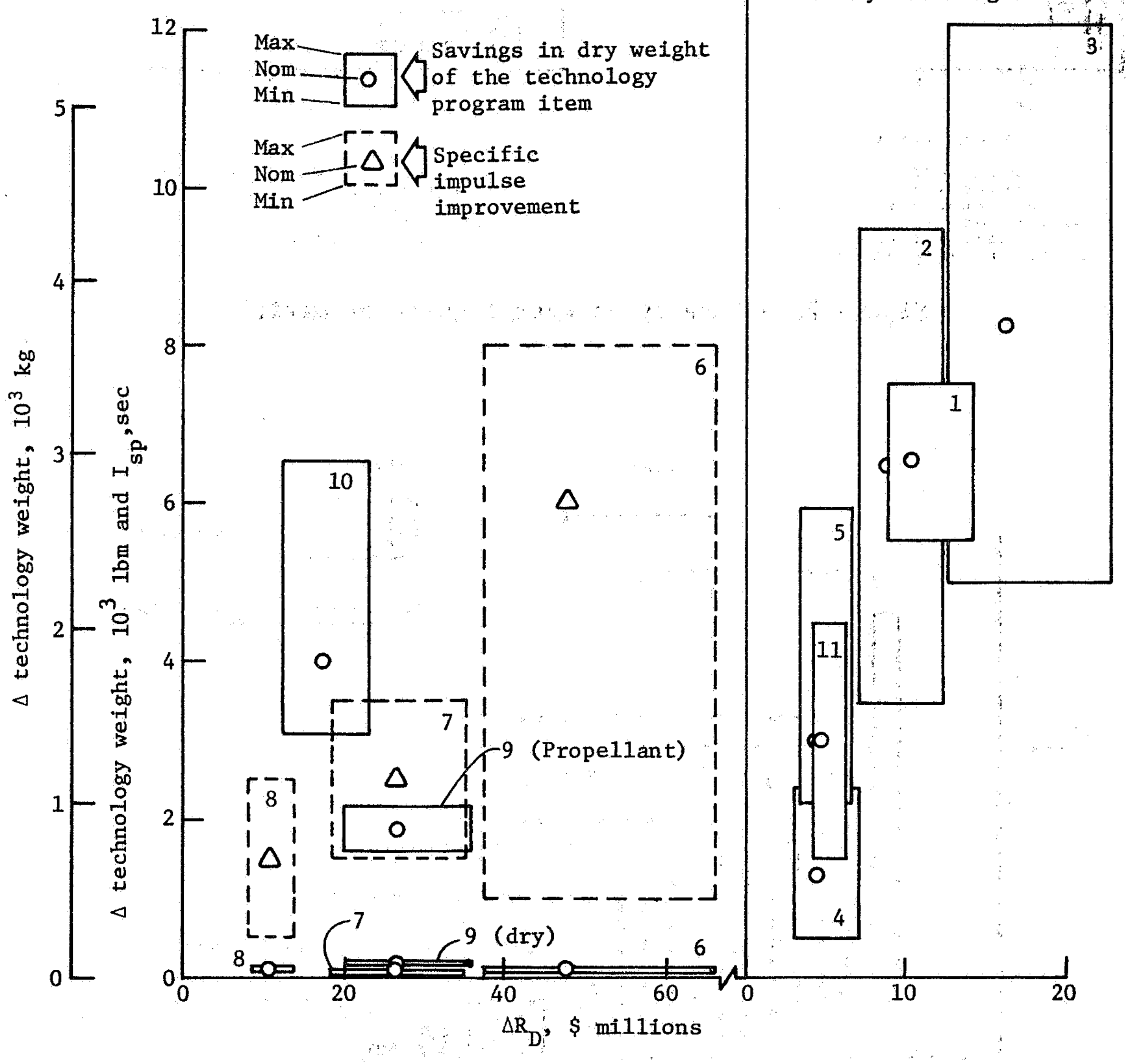
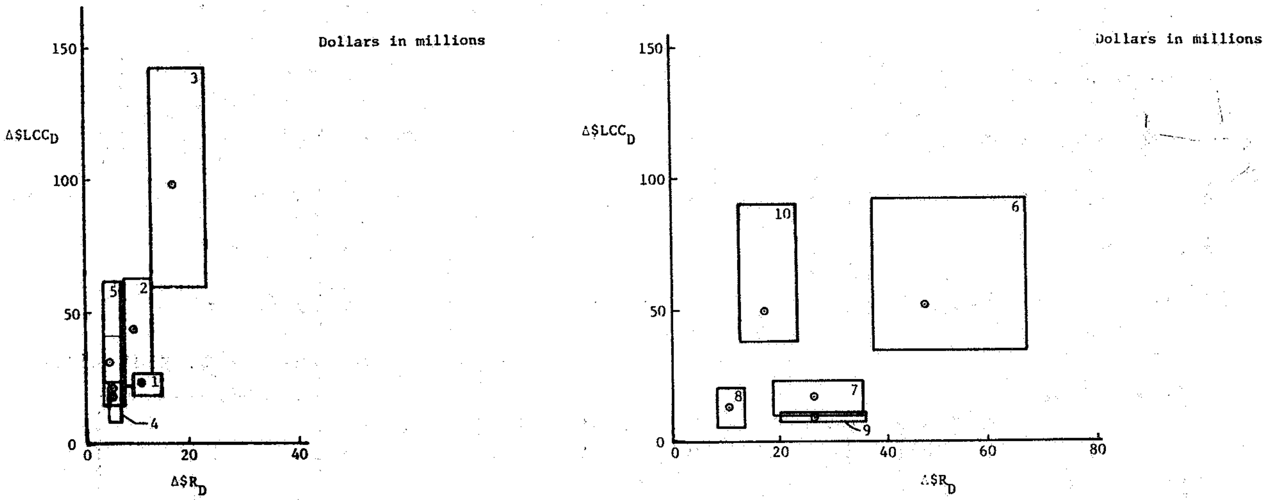


Figure 71.- Accelerated technology costs and direct impact on weights and specific impulse.



1. Thermal protection system
2. Propellant tanks
3. Wing and vertical tail structures
4. Thrust structures
5. Miscellaneous structures
11. Subsystem weight reduction

6. Main engine injectors/chambers/nozzles
7. Main engine pumps
8. Main engine cooling
9. OMS/RCS systems
10. Triple point propellants

Figure 72.- Life cycle cost figures of merit

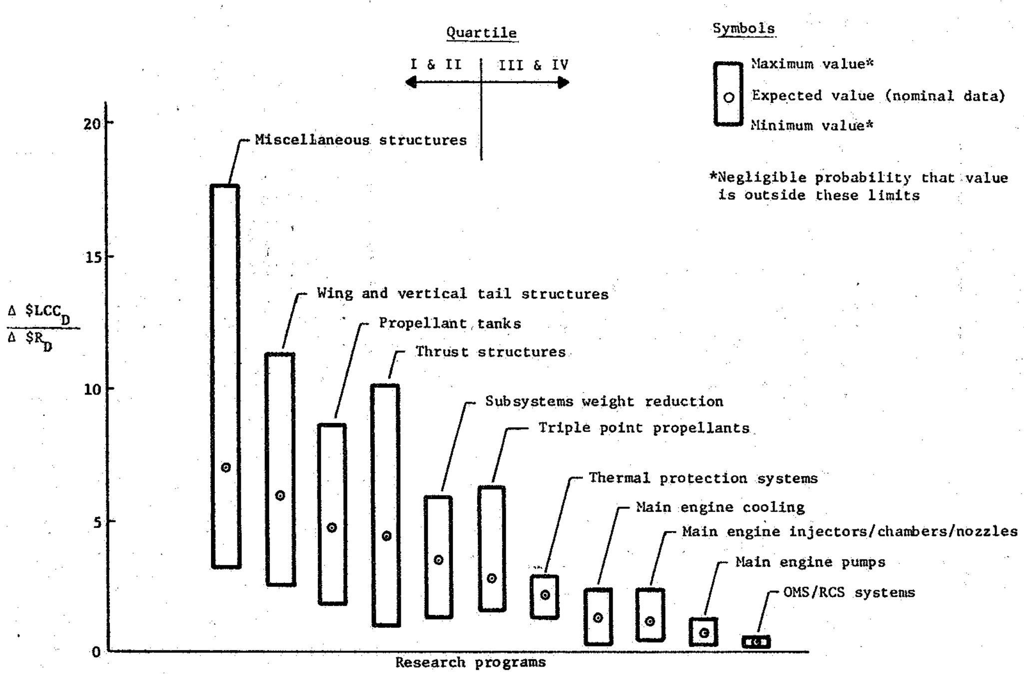


Figure 73.- Figures of merit comparison

The four figures of merit discussed previously (i.e., $\Delta\$LCC_D/\Delta\$R_D$, $\Delta\$LCC/\Delta\$R$, ΔW_{dry}, and $\Delta\$LCC_D - \Delta\$R_D$) have been normalized and ranked according to their relative nominal values in Table 42. The normalizing value for each FOM is the highest nominal value for each category, excluding the integration engineering program. The ΔTechnology parameter has had the mixed inputs of weight and I_{SP} converted to total equivalent system weight for this comparison. In addition to the obvious value of determining the relative merits of the technology programs, Table 42 also provides two other significant conclusions by examining the quartile rankings of each of the four FOMs. The first is that there are definitive groupings of the programs in each quartile, indicating that the quartile ranking would not be different even if there were changes of 10% or more in the cost or weight estimates. The second result is that the quartile rankings are almost the same regardless of the FOM used.

The structures, TPS, and triple-point propellant programs are primary candidates for accelerated activities. The advanced propulsion programs are not expected to have reasonable payoffs from accelerated funding, although "normal" activities in these research areas are required. Advanced propulsion programs in this study were limited to LH_2/LO_2 systems for main propulsion and OMS/RCS, and this conclusion is valid for these LH_2/LO_2 rocket systems. Systems with other propellants may show payoffs.

TABLE 42.- RANKING OF ADVANCED TECHNOLOGY PROGRAMS

Figures of merit		A. $\Delta\$LCC_D/\Delta\$R_D$			B. $\Delta\$LCC/\Delta\$R$			C. $\Delta W_{dry}/\Delta\$R_D$			D. $\Delta\$LCC_D - \Delta\$R_D$		
Research programs													
No.	Title	Rank	Relative Value	Quartile	Rank	Relative Value	Quartile	Rank	Relative Value	Quartile	Rank	Relative Value	Quartile
12.	Integration engineering	0	3.13	I	0	2.78	I	0	5.42		0	2.27	I
5.	Misc structures	1	1.00	($\Sigma\$R_D$	1	1.00		3	0.88	I	4	0.33	III
3.	Wing & vertical tail structures	2	0.87	= 29.9)	3	0.66		6	0.75		1	1.00	I
2.	Propellant tanks	3	0.69	II	2	0.66	II	1	1.00		2	0.41	III
4.	Thrust structures	4	0.64	($\Sigma\$R_D$	4	0.63		7	0.65	II	5	0.20	IV
11.	Subsystem weight reduction	5	0.51	= 18.3)	5	0.54		5	0.85		6	0.16	
10.	Triple point propellants	6	0.41	III	7	0.22	IV	4	0.86	I	3	0.39	III
1.	Thermal protection systems (TPS)	7	0.32	($\Sigma\$R_D$ = 28.0)	6	0.33	III	2	0.89		7	0.15	
8.	Main engine cooling	8	0.18	IV	8	0.20	IV	8	0.31	III	9	0.04	
6.	Main engine injectors/ chambers/nozzles	9	0.15	($\Sigma\$R_D$ = 111.4)	9	0.18		9	0.23		8	0.04	IV
7.	Main engine pumps	10	0.09		10	0.10		10	0.17	IV	10	-0.11	
9.	OMS/RCS systems	11	0.05		11	0.05		11	0.07		11	-0.21	

The Integration Engineering Technology program, although difficult to precisely quantify, is the most important of the R&T programs. As shown in Table 42, it is expected to have FOMs more than twice as large as any other program. The estimates of the merits of this program were based on assumptions for relaxed stability requirements, reduced design margin requirements, improved aerothermodynamic and design analysis techniques, and further design optimization. The outcome of this program is difficult to assess quantitatively, as it depends on the expectation of excellent and efficient talent applied to design and operations philosophy, critéria and integration. It is characterized by great cost avoidance with relatively low R&T costs. Because these activities have the potential for substantial program saving, this program should be vigorously pursued.

RISK ASSESSMENT

Inherent in the figure of merit analysis is an assessment of the risk associated with each R&T program. There are several ways to view the risk associated with each technology. The variances on technology parameters, R&T funding levels, and life cycle costs were all derived from the 95% confidence interval assessment of improved vehicle parameters. Thus, there is a low risk that any technology parameter or cost level will fall outside the tolerance ranges given in Table 41.

If the net funding levels of both R&T and LCC are considered for each program then the parameter $\Delta\$LCC_D - \Delta\$R_D$ is of interest. If the tolerance range for a given program is completely positive, there is little risk of that program not producing positive program cost benefits. Based on this rationale, Programs 1, 2, 3, 4, 5, 10, 11, and 12 should be emphasized. The other programs all include a high possibility of costing more in R&T dollars than they save in life cycle costs.

Another approach is to consider the R&T dollars as being sunk and including only the life cycle costs in the selection. Assuming that a technology program should be undertaken only if it results in an approximate 1% savings in life cycle costs compared to the baseline VTO (i.e., \$22.2M ΔLCC_D), Table 41 indicates that the programs with a high probability of meeting these returns are 2, 3, 4, 5, 6, 10 and 12. Because the 1% is somewhat arbitrary, Program 1 is also included for it is close to the cutoff.

EXTENDED PERFORMANCE STUDIES

The impact of focused advanced technology programs on vehicle characteristics was developed using both VTO and HTO vehicle concepts. The accelerated technology goals of the Advanced Technology Assessment were applied to these concepts, except that the "normal" technology of the main-engine and OMS/RCS propulsion systems was used. As a representation of program goals of the Integration Engineering R&T program, the static stability guidelines were re-reduced; the minimum angle for hypersonic trim was changed from 20 deg to 25 deg, and the minimum subsonic lateral directional derivative was changed from 0.002 to 0.0015. These values are representative of current technology and are conservative, yet yield significant vehicle dry-weight reductions. The extended performance vehicle designs were a basis for merit analysis that led to identification of high-yield and critical technology areas.

VEHICLE DESIGN USING ACCELERATED TECHNOLOGIES

This phase of the vehicle study used the figure-of-merit rationale of Task 3 to define the R&T programs to be applied to the extended performance vehicles. The VTO and HTO vehicles have been sized using the R&T programs listed below:

Program No.	Description
1	Thermal protection system
2	Propellant tank structures
3	Wing and vertical tail structures
4	Thrust structures
5	Miscellaneous structures
10	Triple-point propellants
11	Subsystems weight reduction
12	Integration engineering

Using the combined R&T program weight advantages in addition to the Task 2 vehicle projections, the VTO and HTO vehicles were resized. Recalculated aerodynamic characteristics were included in ascent performance optimization conducted on the Program to Optimize Simulated Trajectories (POST). The Vehicle Integrated Sizing Program (VISP) was used to obtain near optimum requirements for both the VTO and HTO vehicles. The final vehicle sizing is shown in Figure 74 with the Task 2 vehicles shown for reference.

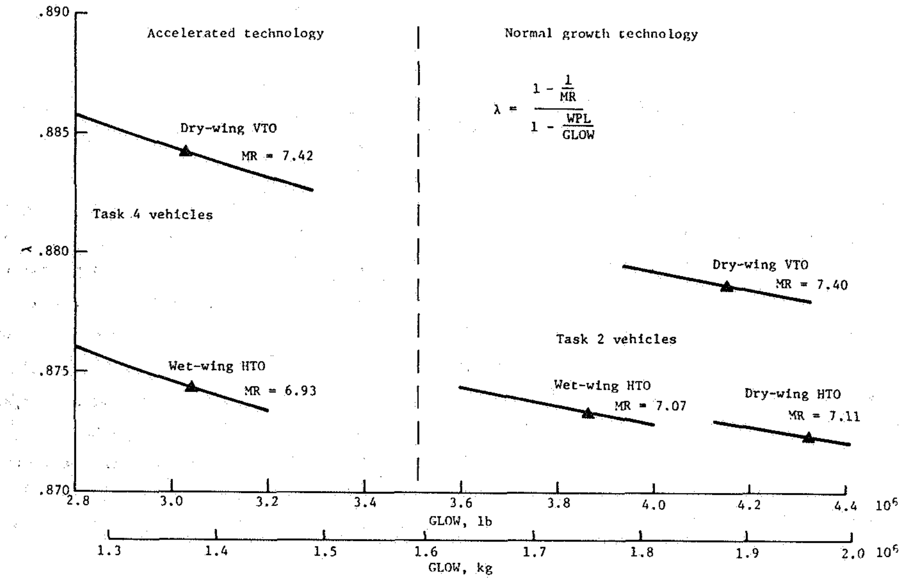
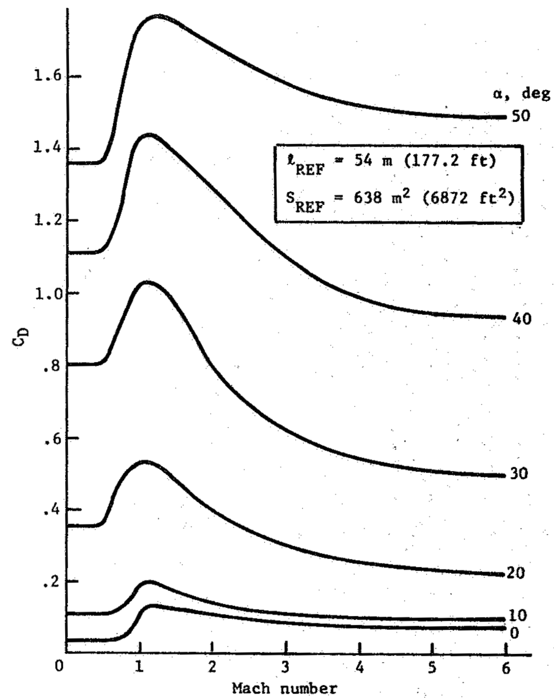


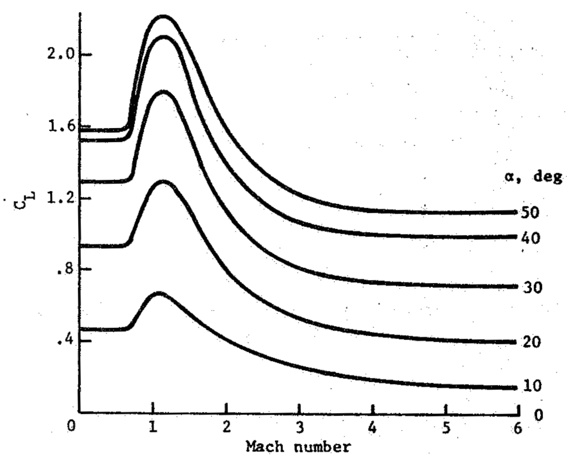
Figure 74.- Vehicle sizing

Design Information

The VTO and HTO vehicle preliminary sizes were based on Task 2 revised aerodynamics and then vehicle aerodynamics were recalculated to reflect these configurations. The final vehicle aerodynamic characteristics are shown in Figures 75 through 77 for both vehicles. These aerodynamic characteristics were used in the ascent trajectory optimization POST program to determine the required mass ratio. The higher densities of the triple-point propellants have a significantly favorable effect on vehicle size and resulting dry weight. The densities used in the analysis are as follows:

Liquid hydrogen	72.1 kg/m^3 (4.5 lb/ft^3)
Liquid oxygen	1304 kg/m^3 (81.4 lb/ft^3)





Hypersonic trim capability, payload out

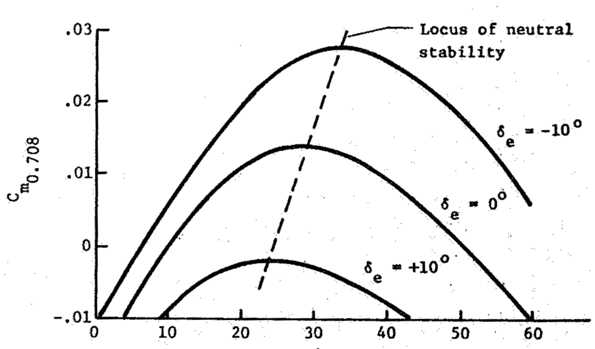


Figure 75.- VTO aerodynamics

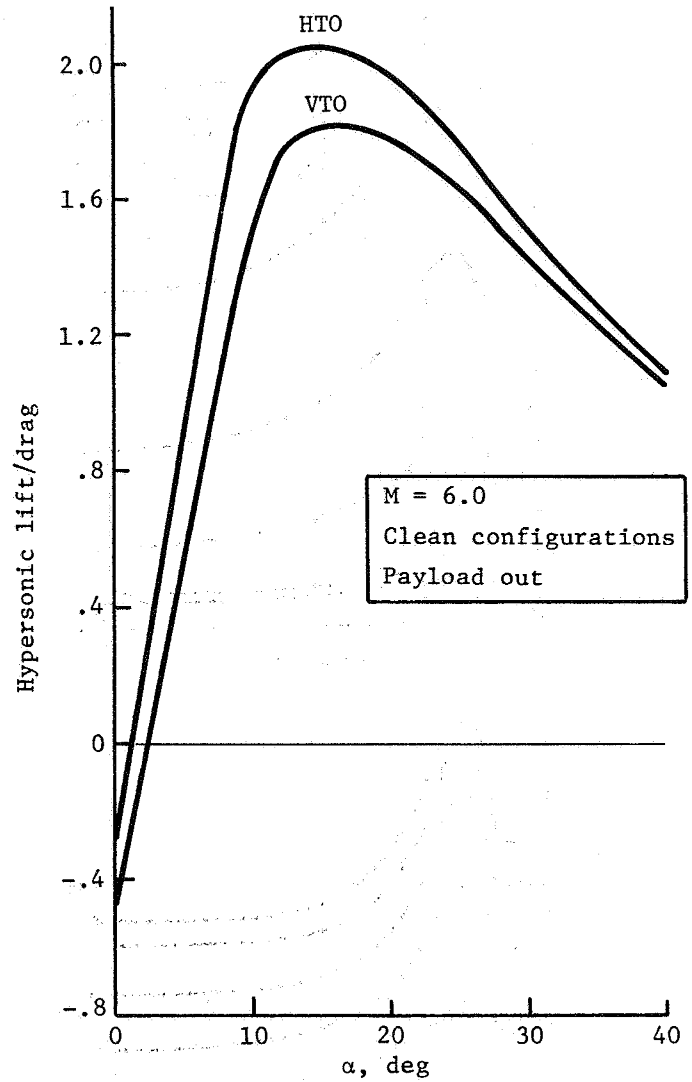


Figure 76.- Extended performance hypersonic lift/drag

VTO Inboard Profile

The inboard profile of the Task 4 VTO vehicle is shown in Figure 78. The vehicle is similar in concept to the Task 2 VTO vehicle except that the wing and vertical tail areas are smaller relative to the body. The thickness-to-chord ratio has been increased to 0.10 at the root of the exposed wing. The vehicle has three dual-position nozzle engines and four fixed-position nozzle engines.

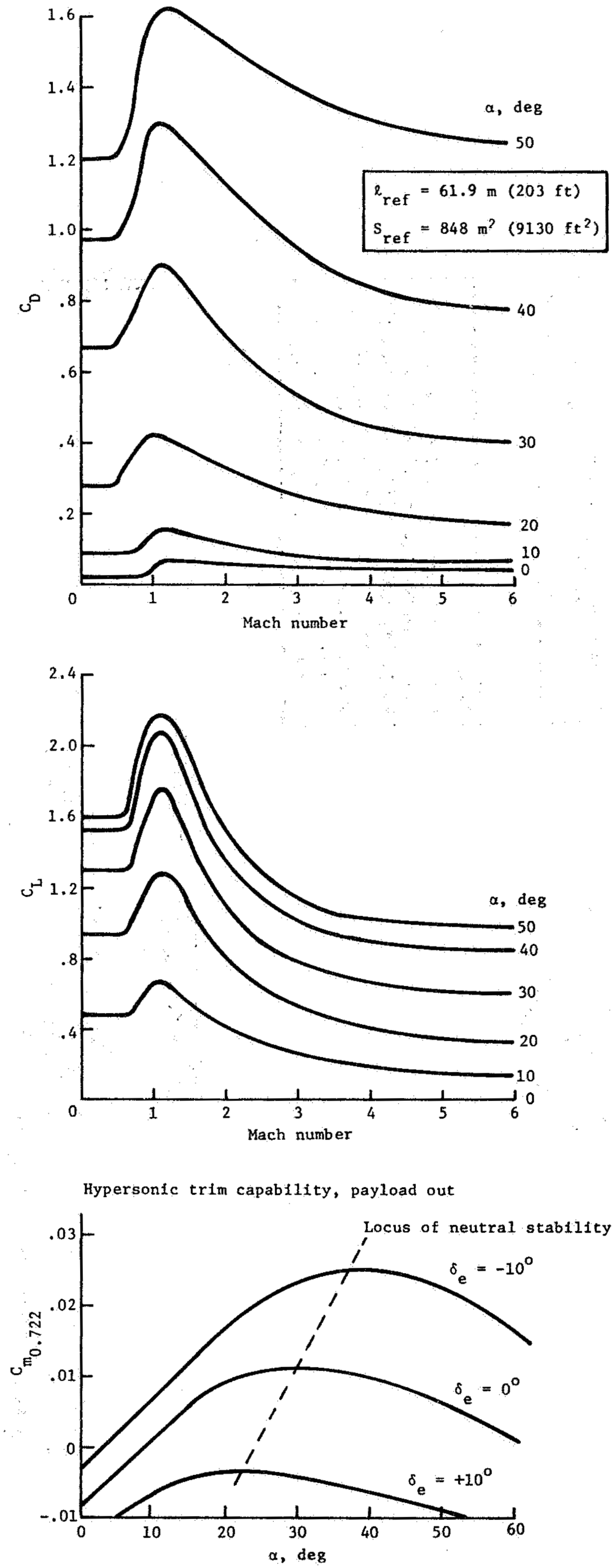
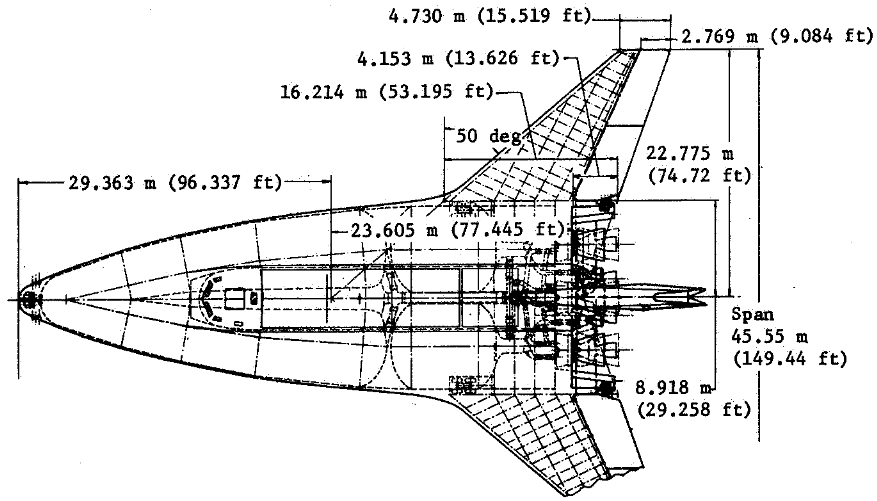


Figure 77.- HTO aerodynamics



Weight			cg % Ref Length
Payload	29 484 kg	(65 000 lb)	49.02
Dry Weight	134 985 kg	(297 588 lb)	
Landing W/O Payload	138 638 kg	(305 643 lb)	70.9
Landing with Payload	168 122 kg	(370 643 lb)	68.8
Ascent Propellant	1 185 441 kg	(2 613 450 lb)	
Gross Liftoff Weight	1 372 710 kg	(3 026 308 lb)	69.9

Volumes		
LH_2 Tank	2137.9 m^3	(75 500 ft^3)
LOX Tank	827.4 m^3	(29 218 ft^3)
Payload, Diameter	4.572 m	(15 ft)
Length	18.288 m	(60 ft)
Payload Bay Clear Opening		
Diameter	4.725 m	(15.5 ft)
Length	18.517 m	(60.75 ft)

Areas		
Body Plan Area	756.7 m^2	(8 145 ft^2)
Wing, Theoretical	645.3 m^2	(6 946 ft^2)
Wing, Exposed	287.5 m^2	(3 094 ft^2)
Elevon	95.9 m^2	(1 032 ft^2)
Vertical Tail	112.0 m^2	(1 205 ft^2)
Rudder	41.0 m^2	(441 ft^2)
Body Wetted Area	2074.5 m^2	(22 330 ft^2)

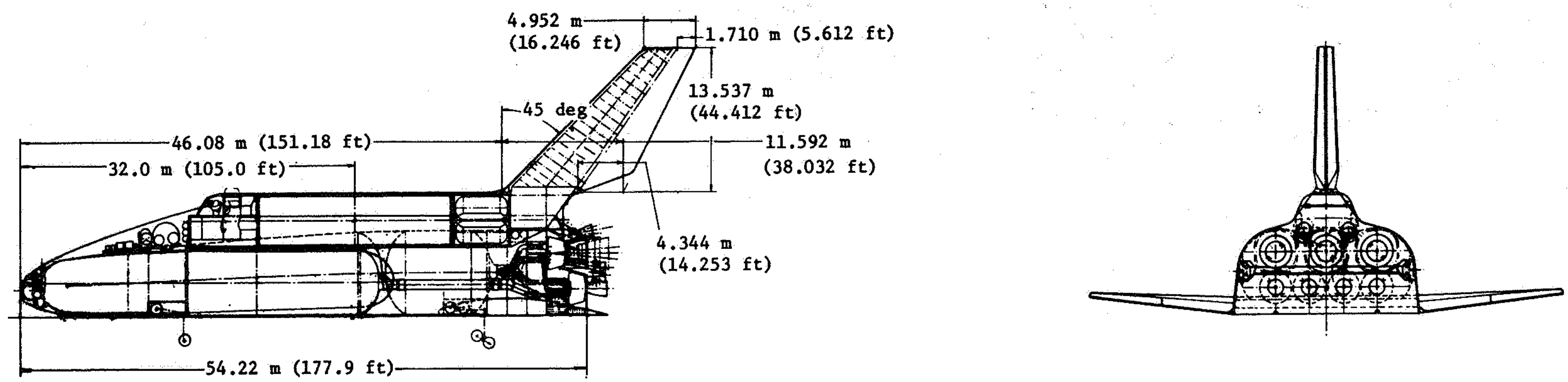
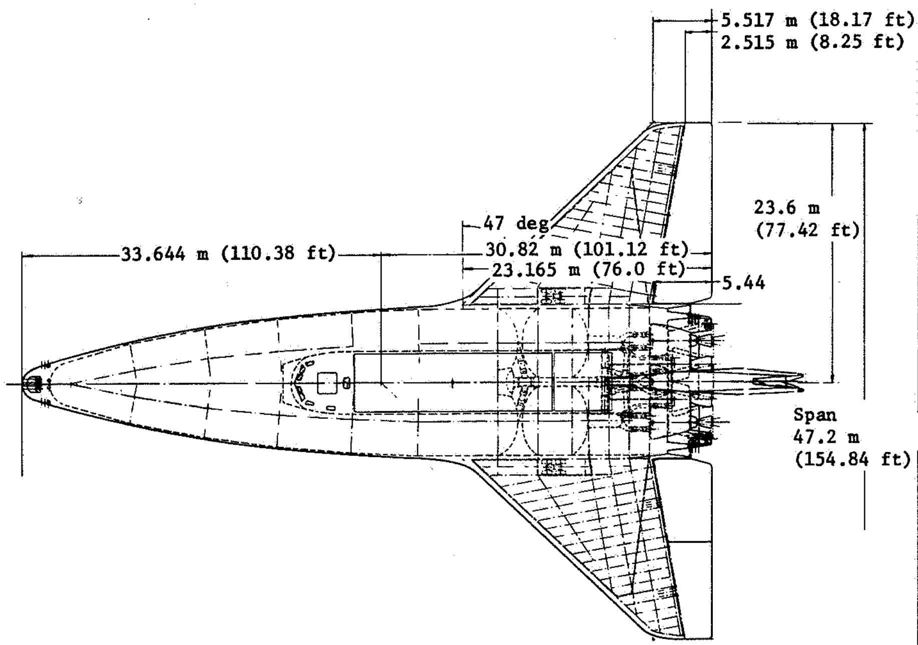


Figure 78.- Extended performance, VTO inboard profile

HTO Inboard Profile

The sled launched HTO vehicle shown in Figure 79 is a wet-wing design concept. Approximately 62% of the oxidizer propellant is in the wing and wing carrythrough box. The oxidizer propellant is transferred to the body tanks by pumps and transfer lines from the aft end of the wing carrythrough box. The wing is configured with a 47 deg leading edge sweep and a 0 deg trailing edge sweep to facilitate the transfer of LO_2 propellants. The main landing gear is housed in the wing structure adjacent to the fuselage oxidizer tanks. The aft fuselage is boat tailed on the sides to match the base-rocket engine packaging requirements. The rocket engines are three dual-position nozzle and two fixed-nozzle configurations.

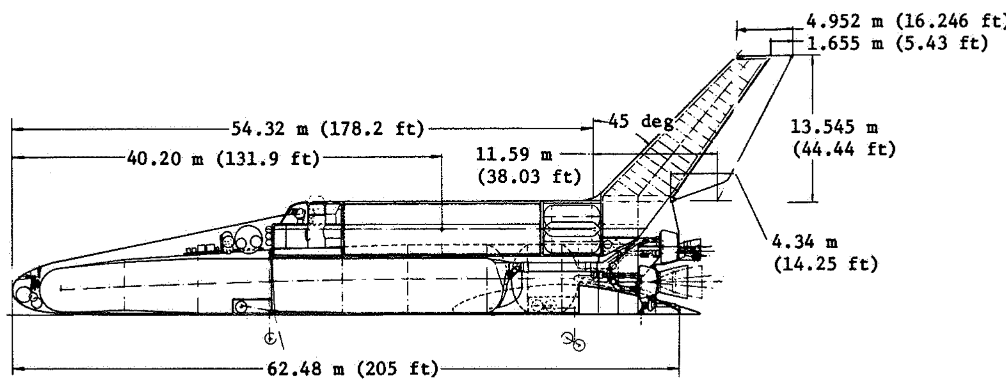
A vehicle thrust-to-weight value of 0.95 was used based on Task 2 optimization analyses. The vehicle is sized with main engines firing for six seconds during the sled acceleration phase compared to the 20-second firing used for the Task 2 vehicle.



Weights		cg % Ref Length
Payload	29 484 kg (65 000 lb)	64.3
Dry Weight	147 054 kg (324 196 lb)	
Landing W/O Payload	150 741 kg (332 326 lb)	73.6
Landing with Payload	180 224 kg (397 326 lb)	72.1
Ascent Propellant	1 181 416 kg (2 604 577 lb)	
Gross Liftoff Weight	1 382 551 kg (3 048 004 lb)	78.4

Areas	
Body Plan Area	751.1 m^2 (8 085 ft^2)
Wing, Theoretical	857.5 m^2 (9 230 ft^2)
Wing, Exposed	472.6 m^2 (5 087 ft^2)
Elevon	131.2 m^2 (1 412 ft^2)
Vertical Tail	112.0 m^2 (1 206 ft^2)
Rudder	40.6 m^2 (437 ft^2)
Body Wetted Area	]0]2.0 m^2 (21 775 ft^2)

Volumes	
LH^2 Tank	2130.6 m^3 (75 243 ft^3)
LO_2 Tank	824.6 m^3 (29 119 ft^3)
Payload, Diameter	4.572 m (15 ft)
Length	18.288 m (60 ft)
Payload Bay Clear Opening	
Diameter	4.725 m (15.5 ft)
Length	18.517 m (60.75 ft)



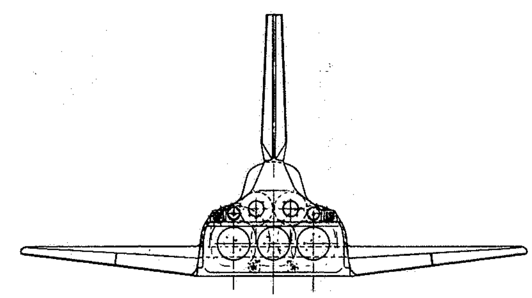


Figure 79.- Extended performance, HTO inboard profile

Mass Properties

The Task 4 vehicle mass properties summary is presented in Tables 43 and 44 for the VTO and HTO vehicles respectively. The primary difference in vehicle dry weight between the two concepts is in the wing and body weights. The wing of the HTO vehicle is heavier because of requirements to carry propellants and the larger wing area to accommodate the vehicle center of gravity, which is 5.2% farther aft. The final results indicate that the selected thermostructural concept is as efficient for the extended performance vehicles as it was for the normal technology vehicles.

TABLE 43.– VTO EXTENDED PERFORMANCE MASS PROPERTIES SUMMARY

Code	System	Mass, kg		Weight, pounds	
1.0	Wing group	8 552		(18 854)	
2.0	Tail group	2 316		(5 107)	
3.0	Body group	35 388		(78 017)	
4.0	Induced environmental protection	30 508		(67 258)	
5.0	Landing and auxiliary systems	4 690		(10 339)	
6.0	Propulsion ascent	30 097		(66 352)	
	6.1 Engine accessories		2 007		(4 424)
	6.2 Feedlines		1 829		(4 032)
	6.3 Engines		26 261		(57 896)
7.0	Propulsion-RCS	1 444		(3 183)	
8.0	Propulsion-OMS	1 086		(2 395)	
9.0	Prime power	1 674		(3 690)	
10.0	Electrical conversion and distribution	1 569		(3 458)	
11.0	Hydraulic conversion and distribution	1 666		(3 672)	
12.0	Surface controls	1 656		(3 650)	
13.0	Avionics	1 965		(4 333)	
14.0	Environmental control	1 721		(3 795)	
15.0	Personnel provisions	499		(1 100)	
18.0	Payload provisions	270		(595)	
19.0	Margin	9 884		(21 790)	
	Dry weight	134 985		(297 588)	
20.0	Personnel	1 199		(2 644)	
23.0	Residuals and gases	2 454		(5 411)	
	Landing weight	138 638		(305 643)	
22.0	Payload	29 484		(65 000)	
	Landing and payload	168 122		(370 643)	
23.0	Residuals dumped	4 786		(10 552)	
25.0	Reserve fluids	3 464		(7 637)	
26.0	Inflight losses	1 613		(3 555)	
27.0	Ascent propellant	1 185 441		(2 613 450)	
28.0	Propellant-RCS	1 400		(3 086)	
29.0	Propellant-OMS	7 886		(17 385)	
	GLOW	1 372 710		(3 026 308)	

Center of gravity: Body length = 54.2 m (177.9 ft)

Condition	$X_{c.g.}$ % of body length
Dry	71.2
Landing	70.9
Landing with payload	68.8
Liftoff	69.9

TABLE 44.- HTO EXTENDED PERFORMANCE MASS PROPERTIES SUMMARY

Code	System	Mass, kg		Weight, pounds	
1.0	Wing group	24 097		(53 124)	
2.0	Tail group	3 755		(8 279)	
3.0	Body group	31 462		(69 362)	
4.0	Induced environmental protection	32 540		(71 738)	
5.0	Landing and auxiliary systems	5 382		(11 866)	
6.0	Propulsion ascent	22 831		(50 333)	
	6.1 Engine accessories		1 523		(3 357)
	6.2 Feedlines		1 872		(4 128)
	6.3 Engines		19 436		(42 848)
7.0	Propulsion-RCS	1 444		(3 183)	
8.0	Propulsion-OMS	1 080		(2 381)	
9.0	Prime power	1 674		(3 690)	
10.0	Electrical conversion and distribution	1 849		(4 076)	
11.0	Hydraulic conversion and distribution	2 612		(5 758)	
12.0	Surface controls	2 271		(5 006)	
13.0	Avionics	1 965		(4 333)	
14.0	Environmental control	1 721		(3 795)	
15.0	Personnel provisions	499		(1 100)	
18.0	Payload provisions	270		(595)	
19.0	Margin	11 602		(25 577)	
	Dry weight	147 054		(324 196)	
20.0	Personnel	1 199		(2 644)	
23.0	Residuals and gases	2 488		(5 486)	
	Landing weight	150 741		(332 326)	
22.0	Payload	29 484		(65 000)	
	Landing with payload	180 224		(397 326)	
23.0	Residuals dumped	5 599		(12 343)	
25.0	Reserve fluids	3 721		(8 204)	
26 0	Inflight losses	1 613		(3 555)	
27.0	Ascent propellant	1 181 416		(2 604 577)	
28.0	Propellant-RCS	1 502		(3 312)	
29.0	Propellant-OMS	8 476		(18 687)	
	GLOW	1 382 551		(3 048 004)	
30.0	Sled acceleration propellant	19 751		(43 543)	
	Gross weight	1 402 302		(3 091 547)	

Center of gravity: Body length = 62.5 m (205 ft)

Condition	$X_{c.g.}$ % of body length
Dry	73.9
Landing	73.6
Landing with payload	72.1
Liftoff	78.4

Technology Requirements

The designs of extended performance vehicles are based on the nominal research goals projected for selected, focused, advanced technology programs. The selected areas, based on potentials for high yield, were sources for weight reductions of TPS, structures (both tank and nontank), propellants, subsystems (power, electrical, hydraulics, surface controls, environmental control, and avionics) and aerodynamic surfaces using relaxed stability criteria.

The technology requirements are to attain the projected advanced goals before DDT&E, as presented in the Advanced Technology Assessment. The advanced HTO sled-launched vehicle uses cryogenic wet-wing technology, which is assumed to be addressed with advanced R&T. Also, the main engines are ignited while the sled is accelerating near the end of the sled run. Advanced technology is required to develop this technique and to confirm the reliability of ignition in this acceleration environment.

MERIT ANALYSIS AND RISK ASSESSMENT

Research Activity Assessments

The nominal schedules relating to the accelerated technology programs (Figure 70) can accept some delays in startup if funding levels are increased later in the program. However, there is a limit to startup delays beyond which the total program output becomes jeopardized. An analysis was conducted to determine the maximum schedule compressions that could be allowed without incurring high program risks. Table 45 summarizes the maximum delays in program start time that could be allowed before schedule compression would become unrealistic. This analysis was conducted by first estimating the variances in R&T and DDT&E program schedules that could be expected if the programs were operated at a low risk concentrated level of effort. These total time variances were then subtracted from the ATP date of 1987 to determine the expected variance in start time. If any start dates were determined to be before 1976, they were set to 1976. Then the maximum allowable slip in start dates was calculated by subtracting 1976 from the latest year in each category. Some programs, such as the wing and vertical tail structures, could slip their start dates to 1977 without incurring high program risk, and others, such as the thrust and miscellaneous structures and subsystem weight reduction tasks, could start as late as 1980 before a high probability of jeopardizing the program would be incurred.

TABLE 45.- RISK ASSESSMENT OF ACCELERATED TECHNOLOGY AREAS

	Technology area	Time span, years			Start dates	Max delay time w/o high risk, (years)
		R&T	DDT&E	Total		
1.	TPS	4-7	4-5	8-12	1976-1979	3
2.	Propellant tanks	4-6	4-5	8-11	1976-1979	3
3.	Wing and vertical tail structures	6-8	4-5	10-13	1976-1977	1
4,5,11.	Thrust structures, miscellaneous structures, subsystem weight reduction	4-5	3-4	7-9	1978-1980	4
6,7,8.	New main engine propulsion systems	4-6	4-8	8-14	1976-1979	3
9.	OMS/RCS	4-6	4-5	8-11	1976-1979	3
10.	Triple point propellants	5-6	4-6	9-12	1976-1978	2
12.	Integration engineering	5-7	4-5	9-12	1976-1978	2

System Development Schedule Assessments

Possibilities for accelerating the SSTO system development schedule (Figure 66) are discussed in this section. The perturbed schedule that reflects the results of the assessment is shown in Figure 80. The accelerated schedule is based on considerations of the timing of advanced research programs, development, test, and production without incurring high risks of schedule delays. Possibilities for condensing these schedules are discussed here with the assumption that the cumulative funding for these activities is maintained.

Research Programs.-

(1) TPS.- The TPS research program could be accelerated from a 10-year project to a 6-year project without significant risk. The effort could start in 1978 with peak activity complete before 1983 and the large scale tests complete before 1984. Any added refinements could parallel the design and development effort. The design development of the flight vehicle could be started in mid-1981 and the material/procurement activity could be initiated in mid-1982. Manufacturing effort could still take advantage of the results of the large scale tests to be completed in late 1983.

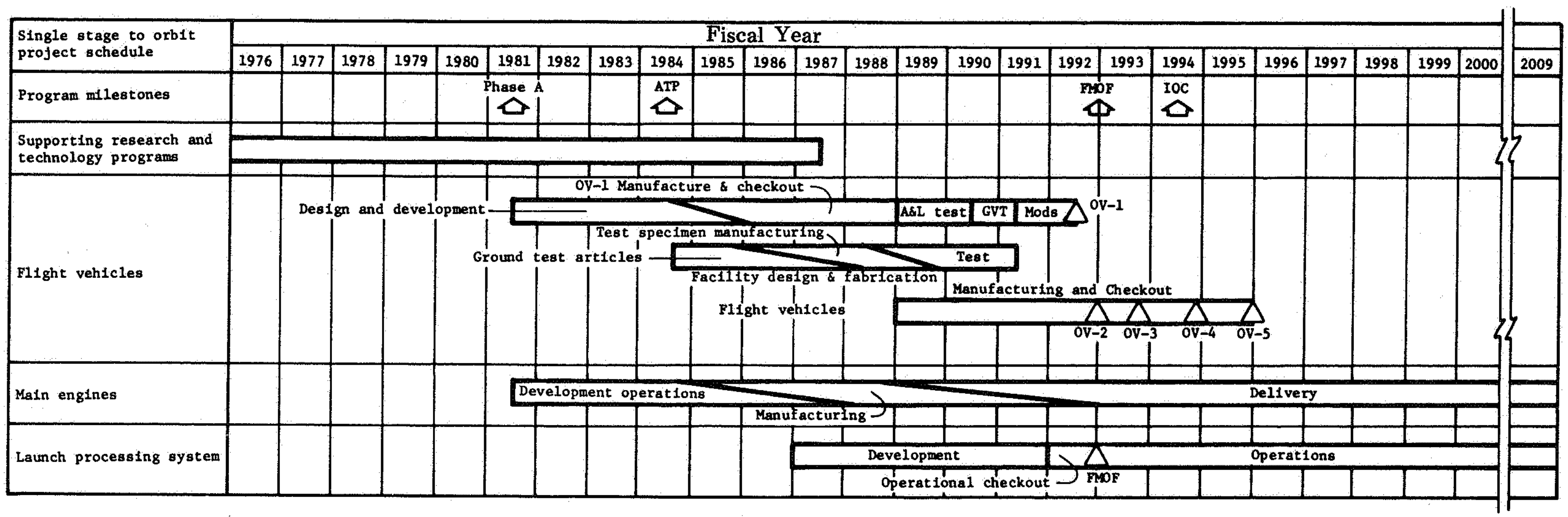


Figure 80.- Accelerated total system development plan

(2) Propellant tanks.- The effort in this research area can be accelerated from a 10-year period to a 6-year period without significantly increasing the risk. The effort could begin in early 1977 and would allow a beneficial start for the design and development effort in mid-1981. Refinement in the technology improvements could parallel the design effort. This statement is supported by completion of the large scale tests in 1983 and peaking of the technology effort in early 1981. Material/procurement activity could be initiated in mid-1982 and manufacturing effort would be supported by the results of the large scale tests.

(3) Wing and vertical tail structures.- Improvement efforts in this area could be accelerated from the planned 10 years to 7 or 8 years. Starting in 1977, the major thrust of the effort would be complete in 1984. A beneficial start of the design and development effort could occur in 1982. The testing with flight hardware starting in 1981 will provide three years of test data and will support a possible commitment to start manufacturing in 1983.

(4) Miscellaneous and thrust structures.- Schedules for these research programs support the start of the design effort in 1981 and the start of the manufacturing effort in 1983. The completion of the manufacturing techniques developed by the end of 1982 and the completion of the large scale tests by the end of 1983 supports this conclusion.

(5) Subsystem weight reduction.- Effort in this area is related to advancements in the other areas and the tradeoffs available in the design. Effective research effort in this area would be worked before design and development. Adequate results could be achieved to support start of the design effort in 1981 without increasing the risk.

(6) Propulsion technology.- Research effort in this area is planned to achieve its major goals by the end of 1982. The peak of the effort is concentrated over a 5-year period from 1978 through 1982. Achievements from 1978 through 1981 allow a start of the design effort for the main engines and propulsion system in 1981.

DDT&E and production.-

(1) Flight vehicles.- System development schedules as presently planned (Figure 66) represent a low risk schedule. The project can be accelerated by starting the design and development in mid-1981, based on the assessments of the technology schedules. Manufacturing of flight vehicles structure could be started in mid-1984 and could be completed in four years. Development effort would overlap the manufacturing by one and one half years and would allow for an adequate period for incorporating any modifications without impacting the manufacturing process. Test efforts

should not be condensed. The approach and landing tests (A&L) and the ground vibration tests (GVT) should remain scheduled over a two-year period.

By accelerating the manufacture of OV-1, OV-2 could be either accelerated similarly or could be delayed until after OV-1 tests are complete. It would be more advantageous to hold manufacture of OV-2 until one year after the OV-1 tests and deliver within six months after OV-1. The OV-3 could be delivered one and one half years after OV-2 and OV-4, and OV-5 waterfalled in one year increments. Ground test articles would have to be scheduled for delivery before the flight vehicle A&L tests.

With a minimum increase in risk, manufacture of the first article can be accelerated by three years and delivery of the vehicles can be arranged to eliminate any risks because of modifications. The flight vehicles could be totally delivered two and one half years earlier than presently planned.

(2) Main engines.- Development of the main engines could be started in 1981, two years earlier than shown on the guideline schedule. The span time could conceivably be reduced one and one half years, overlapping manufacture of the first flight articles. The manufacturing period of three years appears to be realistic and should include some testing and modifications. Engines might be selected that are basic SSMEs with moderate performance uprating, but not requiring new components. The development time for SSTO main engine modifications then could be reduced three years.

(3) Launch processing system, ground operations facilities.- These operations will be scheduled to relate to the flight vehicle and the main engine schedules. Acceleration of these schedules is feasible without any increase in schedule risks.

Conclusion.- Relative to the guideline schedule, the start of DDT&E could be advanced approximately three years, whereas the FMOF and IOC could be advanced one year, without incurring any significant increase in risk. Without an acceleration of the total program, the first article test effort could be moved three and one half years earlier. The test results would then be used in the initial build effort, and the possibility of inline modifications and later retrofitting of OV-2 would be reduced.

Life Cycle Costs

The life cycle costs for the two advanced vehicle systems were determined in the same manner as for vehicles described previously. The results given in Table 46 are about 10% less than before primarily as a result of the smaller vehicle sizes. The first-article costs for the SSTO vehicles are about the same as for the Space Shuttle Orbiter.

TABLE 46.- LIFE CYCLE COSTS IN MILLIONS OF DOLLARS

	VTO		HTO	
	FY 76	Discounted	FY 76	Discounted
DDT&E	5 288	1 577	5 589	1 663
Production	1 118	225	1 145	229
Operations	3 305	248	3 268	245
Totals	9 711	2 050	10 002	2 137
First article cost	258		279	

Figures of Merit

Figures of merit (FOM) were developed for the Task 4 vehicles using goals of the recommended advanced technology programs 1, 2, 3, 4, 5, 10, 11, and 12. The decreases in dry weight and GLOW for both the revised VTO and revised wet-wing HTO were calculated. The total research costs and the discounted life cycle cost improvements based on the Task 2 baselines were determined. Two figures of merit were then applied: (1) the improvement in LCC divided by the increase in R&T funds and (2) the net savings of the combined programs, i.e., $\Delta LCC - \Delta R$. The FOMs were derived using the nominal parameter values expressed in discounted dollars.

As a basis of comparison, two other possible combinations of accelerated R&T programs were analyzed. The first approach was to apply all twelve of the programs to the VTO vehicle. Investigation of Table 41 shows that, of the aforementioned recommended programs, Programs 4 and 11 had the highest risk of not achieving a 1% improvement in life cycles costs compared to the Task 2 baseline. Therefore, the second approach excluded these two programs and applied goals of programs 1, 2, 3, 5, 10, and 12 to the VTO vehicle.

Table 47 summarizes the weight saving, R&T costs, LCC savings and FOMs for the vehicles. The original technology combinations produced better FOMs when applied to the Task 2 wet-wing HTO than when applied to the Task 2 VTO. The revised combination of 1, 2, 3, 5, 10, and 12 produced a better return than the original selection, but this could be expected because two of the lesser effective programs (for that combination) were eliminated. The combination, including all twelve programs, produced much lower returns because some of the programs individually had negative returns (Table 41). This analysis shows that the total program return is a function of the advanced R&T programs that are applied. Therefore, when a level of total R&T funding is a constraint, consideration must be given to the best distribution of those funds among advanced technology areas. Meaningful combinations (Table 47) show that total R&T funding would be increased about $12M/year (undiscounted) over normal funding (cf. Figure 26).

TABLE 47.- FIGURES OF MERIT FOR ADVANCED PROGRAM COMBINATIONS

Combined technology programs	ΔW_{dry} kg (pounds)	Δ GLOW kg (pounds)	$\Delta\$R_D$ <Δ$R> $M	Δ$LCC <Δ$LCC> $M	$\frac{\Delta\$LCC_D}{\Delta\$R_D}$	Savings, $\Delta\$LCC_D - \Delta\$R_D$	Total Cost $LCC_D + \Delta\$R_D$
1, 2, 3, 4, 5, 10, 11 and 12 applied to VTO	-63 451 (-139 885)	-542 272 (-1 195 506)	76.2 <131.0>	257 <1206>	3.37	180.8	2126.2
1, 2, 3, 4, 5, 10, 11 and 12 applied to wet-wing HTO	-42 949 (-94 686)	-369 724 (-815 101)	76.2 <131.0>	293 <1376>	3.85	216.8	2213.2
1, 2, 3, 5, 10, and 12 applied to VTO	-60 584 (-133 565)	-526 434 (-1 160 589)	66.9 <115.5>	255 <1197>	3.81	188.1	2073.9
All applied to VTO	-73 228 (-161 440	-681 430 (-1 502 296)	187.6 <314.4>	300 <1408>	1.60	112.4	2239.6

Note: The symbols < > indicate undiscounted nominal values of added R&T funding <R> and resulting LCC savings <LCC>.

For comparing the benefits of the advanced technology programs to the total SSTO program costs, an additional figure of merit was determined. The net program cost for the advanced SSTO vehicle was calculated as the sum of the life cycle cost and the additional R&T funds required, assuming the normal technology funding represents sunk costs. As shown in Table 47 the net discounted investment is less for the VTO system than for the HTO system. On this basis, the VTO system would continue to be selected as the perferred system. The FOMs here reflect the compounding effects of simulataneous application of the combined accelerated programs.

TECHNOLOGY RECOMMENDATIONS

An advanced earth-orbital transportation system has been shown to be feasible from both technological and life-cycle cost saving viewpoints. The "normal" technology growth, when focused on SSTO requirements, will provide the basis for DDT&E of these systems using thermostructural and propulsion concepts presented in our vehicle designs. The "normal" technology goals will be achieved without additional projected NASA R&T funding, although some re-allocation of budgets among RTOPS will develop. The advanced technology growth, supported with additional NASA R&T funding, will provide the basis for DDT&E of systems that have significantly better vehicle weight and program cost saving than with "normal" technology. Recommendations of "normal" technology and advanced technology areas that should be vigorously pursued are discussed in this section.

THERMOSTRUCTURES

Development of lightweight composite materials and structures are important for application to aerosurfaces, thrust structures, miscellaneous structures, and subsystems. Research activities should address material improvements, material characterization, design analysis, and fabrication technology.

Integral, load carrying, membrane propellant tanks can be developed using normal aluminum alloys applied to multilobe and isogrid structural designs. Research related to tank design should focus on improving loads and failure prediction analysis and testing techniques applied to multilobe designs with the goal of minimizing weight by reducing design margin requirements and nonoptimum factors. Environmental criteria for research and concept analysis should include triple-point propellant requirements.

Advanced research programs related to thermostructures generally have high figures-of-merit (Quartiles I and II) and should be vigorously pursued. Near-term accelerated research should be applied to technology related to composites for use in primary structures, and subsystem interfaces.

THERMAL PROTECTION SYSTEMS

Thermal protection systems (TPS) will require materials such as reuseable surface insulation (RSI), reinforced carbon-carbon (RCC), flexible RSI, ceramics and metallics. Development of these materials, and application to entry vehicle designs are being vigorously pursued in the Space Shuttle program. Research related to SSTO should focus on improved materials, characterization of physical properties, fabrication techniques, testing techniques, structural design (including interfaces with primary structures), performance analysis methods, maintenance, and refurbishment. A critical requirement is to demonstrate reusability of TPS for 100 to 500 reentry cycles.

Accelerated research in TPS technologies should focus on reducing weight as well as DDT&E and production costs of RSIs, and on efficient interfaces among various sections of the vehicle that have different TPS materials or TPS thicknesses. This technology area exhibits a dry-weight FOM in Quartile I, although cost FOMs are in Quartiles III and IV. Accelerated funding could be delayed without high risk, until further SSTO studies and Space Shuttle designs are completed. Inasmuch as RSI technology is relatively young, the possibility of unforeseen payoffs from accelerated research should be considered in evaluating allocations of advanced funding.

PROPULSION SYSTEMS

Main-engine propulsion systems are being developed for the Space Shuttle system that will be an important base for SSTO engine developments. In addition, research is under way or planned for parametric characterizations of dual-mode and linear engine systems and their potential application to SSTO vehicles. Normal propulsion research in the near future should provide sufficient engine parametrics applied to SSTO vehicle concepts to establish the merits of concepts other than LO_2/LH_2 bell-nozzle engines. New engine research, however, requires accelerated funding for component and systems tests following analytic characterization and designs. Normal technology growth goals, therefore, are considered to be related only to long-term (about eight years) R&T applied to LO_2/LH_2 bell-nozzle engines. Normal product improvements are expected from SSME activities including developments in materials and designs of components leading to better thermodynamic efficiency and systems performance. Normal propulsion research should focus on technology related to multiposition bell nozzles, which are critical to SSTO performance. Specific areas of concern are fast-acting extension-retraction mechanisms operable without engine shutdown, cooling methods for extended nozzles, seals for the interface between nozzle segments, and dynamic loads during extension.

Research has been applied to use of triple-point propellants, but the activity is small. Normal growth in this area, therefore, is considered to be inadequate for SSTO applications.

Auxiliary propulsion systems, such as for OMS and RCS, should be improved by research focused towards LO_2/LH_2 systems for SSTO applications. Normal technology growth is projected to be adequate.

Several advanced concepts for main engine systems have been identified including aerospike engines, dual-mode engines, engines integrated with airframe, multinozzle engines with linear arrays of nozzles, and engines with rolled-diaphragm extendable nozzles. Such concepts may find application to SSTO, but their performance characteristics and their merit have yet to be studied. The merit of advanced propulsion concepts would depend greatly on mission and payload definitions, and on development costs of new engines. Within the guidelines of the present study in which hydrogen-fueled rocket engines were to be used, cost/performance benefits of accelerated technology would have been small, with figures-of-merit in Quartiles III and IV.

INTEGRATION ENGINEERING

The R&T activities are being pursued to identify SSTO program concepts and technology requirements and assessments, as exemplified by the present study. Other supporting research activities in NASA and DOD have been identified and projected to continue, including integrated-computer-aided-design synthesis, wind tunnel testing of configuration concepts, aerothermodynamic analysis, and performance optimization. Additional near-term, normal activities should focus on further assessments of dual-mode engines, controlled-configured vehicles (CCV) and payload effects on vehicle design. Future activities should focus on establishing in-depth mission and payload requirements, design criteria and design margins, research requirements, and cost/performance/benefit assessments.

Transportation systems which will launch vehicles at rates projected for Shuttle and SSTO require research for cost savings in operations. Normal Shuttle program developments during the next 14 years will provide an efficient operations system base for SSTO operations. R&T activities should focus on improved computerization and software techniques for automating repetitive and redundant functions, and on improved data-link systems. Inasmuch as liquid hydrogen costs have inflated so much (more than doubled in 1975), research should also focus towards achieving low-cost hydrogen production.

Activities represented by this R&T program are extremely important. Present assessments have indicated exceptionally good figures-of-merit. In the near term, more focus than normal should be applied for in-depth assessments of NASA research activities and goals related to advanced transportation systems of national interest. Advanced R&T in the near term should also focus on parametric design, research requirements, and cost analyses for SSTO systems using alternative guidelines to those anticipated with "normal" technology funding to support detailed recommendations for allocating relevant R&T resources. More support than normal should be given to improving analytic techniques for aerodynamics, aerothermodynamics, performance optimization, configuration development, mass properties, and cost and mission models.

HIGH-YIELD AND CRITICAL TECHNOLOGIES

Assessments of technology have led to the conclusions summarized in Table 48. High-yield technologies are those with potential for large improvements in cost-performance benefits. Critical technologies are those that are required for SSTO success using the guidelines of this study. The high yield and criticality of normal technology have already been discussed. The triple-point propellants program also has potential for high yield. It is considered an advanced program because activities in this area have not been continually and vigorously pursued. Critical aspects are the technology for large scale production, storage, and transfer to the flight vehicles. The flight vehicle technology itself, however, is not critical, as it is available with normal technology.

TABLE 48.- HIGH-YIELD AND CRITICAL TECHNOLOGY ASSESSMENTS

Technology area	"Normal" growth (focused)		Accelerated growth	
	High yield	Critical	High yield	Critical
1 Thermal protection systems				
Reusable surface insulation	X	X Reusability for more than 100 missions must be demonstrated	X	
2 Propellant tanks				
Dry wings	X		X	
Wet wings (applied to HTO)	X	X Large wet wing cryogenic tank technology must be developed Lightweight pressurized structures Propellant utilization	X	
3 Wing and vertical tail structures				
Composite materials	X		X	
4 Thrust Structures				
Composite materials	X		X	
5 Miscellaneous structures				
Composite materials	X		X	
6,7,8 Main engine propulsion				
Multiposition nozzles	X	X 2-position nozzle development is required Extension/retraction Nozzle cooling Seals Dynamic loads		
9 RCS/OMS	Research not high yield nor critical			
10 Triple-point propellants	Not being vigorously pursued at present time		X	X (Based on timeliness) Technology for large scale applications must be developed Manufacture and storage
11 Subsystems weight reduction	X		X	
12 Integration engineering Design integration Design criteria	X	X Continued focusing of technology and evaluations of SSTO concepts are needed	X	

High yield: 1) Attractive cost/performance/benefits and/or dry weight improvements.
2) Technology not highly developed at present (1975-1976).

Critical: 1) Technology development is necessary for SSTO cost and performance success.
2) Timely, near future, focus on SSTO-related research is recommended.

CONCLUSIONS

A fundamental goal of this study was to identify important areas of technology associated with future earth-orbit transportation systems. These systems were represented by reuseable, single-stage-to-orbit vehicle concepts with vertical and horizontal (sled-launched and inflight-fueled) takeoff capabilities. Payload and mission requirements were similar to Space Shuttle, which the SSTO system could replace in 1995.

The study goal was pursued by a sequence of analyses that included projecting "normal" technology growth over the next ten to fifteen years, applying the technology to vehicle designs, and calculating total program (life cycle) costs. Assessments of advanced technology were then made, projecting goals that could be achieved in accelerated research programs. Assessments were aided by developing figures of merit that reflected cost-performance benefits. The advanced goals were then applied to vehicle designs and program costs, providing a basis for assessments of high-yield and critical areas of technology.

The major results of the study are as follows:

(1) Single-stage-to-orbit concepts have exceptionally worthwhile cost-performance merits as advanced earth-orbital transportation systems using "normal" technology growth.

(2) Guidelines of this study led to the specific design concepts of this report. Changes to the guidelines such as reduced dry-weight margins, relaxed stability criteria, and other main-engine and propellant combinations can lead to smaller and lighter vehicles for the same payload requirement. Such guideline changes, however, would not affect the major conclusions that identify technology requirements, except for propulsion.

(3) Assessments of the potential benefits of advanced technology indicate the high-yield areas that should be vigorously pursued are thermal protection systems, propellant tanks, wing and vertical tail structures, thrust structures, miscellaneous structures, triple-point propellants, subsystem weight reduction, and integration engineering.

(4) Critical areas of technology are the reusability demonstration of RSI materials for more than 100 missions, the development of main engines with multiposition nozzles, and the continuing evaluations of vehicle concepts and supporting technology. Also, wet-wing technology is critical for HTO concepts. Advanced growth technology requires timely emphasis on large-scale applications of triple-point propellants, in particular, their manufacturing and storage requirements.

(5) Projections of "normal" technology growth over the next ten years indicated that overall improvements over today's technology will result in a vehicle dry-weight saving of 16% or more. Advanced growth in selected technologies would increase this saving to 27% or greater.

(6) Evaluations of thermostructural concepts indicated weight and technology advantages for using primary fuselage structures composed of integral, multilobe, load-carrying aluminum propellant tanks, protected from entry heating by RSI materials. Advanced composites were selected for primary structures in the wings and vertical tail, as well as for other structural elements. This concept is applicable to both VTO and HTO vehicles.

(7) Comparison of linear and bell-nozzle main engines for the SSTO resulted in selection of high-pressure staged combustion bell-nozzle engines, similar to the SSME. The selected configurations include both fixed nozzle and dual-position nozzles. Unless future studies by engine manufacturers show significant potential improvements in linear engine performance and weight, they do not appear to be competitive with bell-nozzle engines in SSTO applications.

(8) The VTO vehicle is optimized better with dry wings, whereas the HTO vehicle is better with wet wings.

(9) The inflight-fueled vehicle concept is not feasible because it requires unique technology for rendezvous and for large-scale propellant transfer, and it requires development of tanker aircraft that would be considerably larger than heavy aircraft now in use.

(10) Additional studies are required to establish the cost-performance benefits of linear and dual-mode engines. Other advanced main engine concepts, such as vehicle-integrated nozzles (e.g., body flaps) and air-augmented (composite) engines, appear to be beyond the time span of Space Shuttle follow-on vehicles.

(11) Future low-recurring costs can be achieved by continued and expanded emphasis on use of new operations technology that includes automation, computerization and combinations of functions of flight and mission operations.

(12) Further studies of SSTO concepts applied to other payload and mission models, and with control-configured vehicle concepts, are recommended to demonstrate their payoffs as advanced transportation systems.

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APPENDIX A

SECONDARY TECHNOLOGIES

Technology projections have been addressed quantitatively in the materials, structures, and propulsion areas because of their primary influence on vehicle dry weight and c.g. location. Other technology areas have been addressed, but with less depth of study because of their lesser influence on the overall vehicle design. These secondary disciplines included aerothermodynamics, performance optimization, computer technology, aerodynamics, control systems, and auxiliary power. The general approach in studying these areas consisted of first identifying the present activities and their associated level of technology and then identifying the projected 1990 technology status and its impact on SSTO vehicle design.

Table A-1 summarizes the analysis for the aerothermodynamics discipline. It is believed that emphasis on catalytic wall effects and lee surface heating could result in significant TPS weight reductions.

Performance optimization (Table A-2) will allow weight reductions because of improved trajectories and increased speed, accuracy, and reliability of flight controls, guidance, and navigation systems. The major determinant in these improvements will be the implementation of optimal or near-optimal real-time onboard guidance systems.

Advancements in computer technology (Table A-3) will be closely associated with the performance optimization. The major impact on the SSTO will be in terms of advanced onboard computers. These advancements are projected based on recent breakthroughs in large-scale integration, microprocessors, distributed computer architecture, and reusable software libraries. The improvements in flight computers will allow for the implementation of advanced guidance systems that will be used for the performance optimization task. Main frame computer technology will see increases in computer power; however, this technology is already advanced and major impacts on ground based operations are not forecast.

TABLE A-1.- AERODYNAMICS

Technology area	Current activity	Current status	Projected status	Impact on SSTO
Catalytic Wall Effects	Limited theoretical and experimental parametric studies on simple shapes	Results to date show that non-catalytic walls could reduce aeroheating input by 50% or more. Current TPS designs do not take advantage of this.	Studies needed for SSTO shapes. Also should examine possibility of trajectory shaping to maximize time spent in non-equilibrium flow (where catalytic effects occur).	Could reduce TPS weight substantially.
Lee Surface Heating	Primarily experimental	Lack of theoretical models to correlate with test results force conservative estimates of leeward heating.	Careful contouring to avoid vortices could reduce leeward heating environment significantly. Methods and test data are needed to support this technology area.	Because almost 50% of SSTO surface area is on the leeward side, reductions in leeside environments due to more accurate prediction techniques and knowledgeable contouring could impact TPS weight appreciably.
Boundary Layer Transition	Primarily experimental	Data are configuration dependent, use for other shapes is questionable.	Careful configuration shaping could delay transition onset and thereby alleviate turbulent heating.	Moderate reduction in TPS weight.
		Transition criteria are selected conservatively but require flight evaluations.	Flight evaluation could be a basis for revising design criteria.	

TABLE A-2.- PERFORMANCE OPTIMIZATION

Technology area	Current activity	Current status	Projected status	Impact on SSTO
Performance Optimization	<u>Applications</u>			
	Trajectory Shaping	Several operational programs exist for trajectory optimization	Approximate solutions to the equations of motion offer potential for an order of magnitude reduction in cost of shaping trajectories.	Reduced weights because of improved trajectories, guidance, and vehicle configurations. Optimal guidance will enable the actual vehicle to fly trajectories that are closer to the optimum. This will result in small propellant reserves, and hence lighter weight vehicles.
	Guidance and Control	Linear tangent is the only near optimal scheme now being used.	Optimal iterative guidance may be required for a SSTO and would be feasible with the new developments in onboard computers.	
	Vehicle Design	Little work done in this area with modern optimization algorithms.	Decomposition approach should be used to coordinate the optimal design of aerospace vehicles.	
	Aerodynamic Shapes	Some progress has been made on simple configurations.		
	Propulsion Nozzle Design	Some progress has been made.		
	Structural Weight Minimization	A lot of current interest in this area with substantial progress being made.		
	<u>Technology</u>			
	Algorithms	Projected gradient and variable metric methods are most popular.	Not much advancement expected because of high level of development in this area.	
	Problem Formulation	Discrete parameter methods are most popular with little current work on variational methods.	Decomposition formulation will probably appear for more types of problems.	
	Theory	Most theoretical work centers on decomposition techniques and nonlinear programming algorithms.	Not much advancement expected.	

TABLE A-3.- COMPUTER TECHNOLOGY

Technology area	Current activity	Current status	Projected status	Impact on SSTO
Computers				
Spaceborne Computers	Hardware Technology			
	Microprogramming	Microprogrammed chips have not been flown on any vehicle.	Will be used on advanced flight computers to perform basic mathematical functions and operations.	High performance flight computers will increase the speed, accuracy, and reliability of flight controls, guidance, and navigation. This will enable propellant margins to be reduced.
	LSI	Extremely advanced technology	Will not change much.	
	Technology Areas			
	Power, Weight Size	Highly developed	Will not change much.	
	CPU Control	Wired sequencer used in majority of computers.	Advances will be made in ROM packaging density, speed, and power.	
	Accuracy/ Reliability	Single precision/ 20000 MTBF	Development of more DP instructions	
	Memory Techniques	Core	Use of plated wire	
	Speed	1-5 μsec arithmetic 400K instructions per second.	Use of LSI circuits, semiconductor memories, floating point arithmetic	
Ground Based Computers	Performance (speed, size)	IBM 360/195, CDC 7600 are indicative of present status.	Near-term development of CDC STAR is typical of the trend that will result in speed increase by a factor of 10 to 50.	Use of new computer technology in all areas of vehicle design and mission analysis will result in more optimal configurations developed at less cost.
	Processors	Now being used as components.	This trend will continue.	
	Storage	Effort concentrated on magnetic technology.	Magnetic bubble or charged coupled devices will be used by 1985.	
	Batch I/O	Advanced	Little change. Costs will remain high.	
	Program Development Aids	FORTRAN is still the basic language for most scientific applications.	Structured programming.	Reduce cost of supporting software.
Computer-Aided Design	Integrated Programs for Aerospace Design (IPAD)	In feasibility study phase	Implementation of an IPAD system is feasible within four years.	Improved design procedures and techniques will result in more optimum configurations at less cost.
	Optimal Design Integration (ODIN)	Operational at LRC and JSC	Minor improvements will be made.	

Improvements in the ability to estimate aerodynamic characteristics are being paced by fundamental problems in fluid mechanics. The most significant advances are being made in the area of computational flow simulation. The 2D flows were solved in 1975 with extension of 3D expected as early as 1978. Complete solutions to the viscous, time-dependent Navier-Stokes equations are expected by the mid-1980s. This technology, which is being paced by development of advanced computers and improvements to turbulent models, coupled with the development of high Reynolds number wind tunnel facilities, will allow for the development of optimal aerodynamic configurations and will especially increase the ability for more accurate analysis in the early stages of SSTO design. Table A-4 summarizes the aerodynamics technology levels.

TABLE A-4.- AERODYNAMICS

Technology area	Current activity	Current status	Projected status	Impact on SSTO
Computerized Aerodynamic Solutions	Inviscid linear Inviscid nonlinear	Continued refinement	Viscous time dependent Navier-Stokes equations	Lower time and cost for design Reduced need for wind tunnels
	Viscous time-averaged Navier-Stokes equation.	Under development		
Wind Tunnel Development	High Reynold's number facilities. Larger test sections. Minimize interferences.	Under development/construction: Aeropropulsion test facility High Reynold's number transonic tunnel Full-scale subsonic wind tunnel	High ReN facility	Reduced need for flight test substantiation. Decreased data uncertainty.
Configuration Development	Wing body blending Control configured vehicles High-lift devices Low-drag shaping	Under development	Application to SSTO shapes	Low weight solutions to stability problems. Reduced wing areas. Increased performance.
Flight Test Substantiation of Theoretical and Wind Tunnel Data	SR 71 X-24B Current fighter bomber aircraft	Continuing programs	X-24C Shuttle New military aircraft	Lower margins applied to aero predictions and reduced weight penalties.
Fluid Mechanics	Boundary layer flow Separation Interference aero Vortex flow	Continuing analysis	Application to SSTO shapes.	Used in computerized aero solutions and configuration development.

As shown in Table A-5, advanced control techniques will have significant impact on SSTO control system performance, cost, and reliability. The key to achieving this new level of performance is integrated onboard digital systems, relaxed static stability, and flight path/attitude coupling.

TABLE A-5.- CONTROL SYSTEMS

Technology area	Current activity	Current status	Projected status	Impact on SSTO
Flight Controls				
Digital Fly-by-Wire (FBW)	NASA F-8 FBW Experimental Aircraft	Phase I system flight tested in May 1972.	Phase II-A will replace Apollo hardware with commercial aircraft hardware. Extended testing of performance handling qualities, fault detection, autopilot function of (M, h) holds, and CCV control laws.	45% savings in control system weight. 14% reduction in production costs. Reductions are relative to mechanical control systems.
	Space Shuttle	Under development - 1975	Will be completed. Primary emphasis is to obtain reliability via redundancy, not expensive quality control.	
	YF-16	Flight tested (prototype) 1975. Successful program	Operational to be sold to NATO as a result of the fly-off win over the YF-17.	
	SST	USA program cancelled 1971.		
Control Configured Design	Space Shuttle	Under development 1975.	Will be successfully completed.	10% - 20% reduction in dry weight as a result of saving in control surface weights (results from the "snowball" effect of control system weights on total vehicle weight).
	YF-16	Flight tested 1975	Will be operational 1977 - 1985.	
	SST	USA program cancelled 1971.		

Table A-6 summarizes the status of auxiliary power systems. Substantial hydraulic system weight and volume savings are obtainable by raising the operational pressure. A wider temperature range capability for the hydraulic fluid can reduce the cooling requirements during entry. Hot gas actuation systems can convert prime power to useful power directly, thereby saving prime fuel, and eliminating the requirement for hydraulic fluid. A cryogenic fueled APU has demonstrated significantly better performance for SSTO applications than the conventional storable fueled APU. Also, advancements in fuel cell technology may result in significant system weight reductions.

TABLE A-6.- AUXILIARY POWER

Technology area	Current activity	Current status	Projected status	Impact on SSTO
Hydraulic Power	Pressure	27.6×10^6 N/m^2 (4000 psi)	55.2×10^6 N/m^2 (8000 psi)	Minor weight reduction potentially eliminates need for cooling during reentry
	Temperature	394°K (250°F)	533°K (500°F)	
Hot Gas Actuation	Application	Small Missiles	Flight control actuation system lighter than hydraulic	Increased payload, reduced sensitivity to environment
	Rotary Actuator Leakage	High-pressure rotary actuators not feasible	High-pressure rotary actuators developed	Reduced power for actuation
	Hot Gas Generators	Usually full flow	High degree of throttleability	Reduced power for actuation
APU	H_2/O_2 Reactants	Rejected for Shuttle due to risk	SFC $0.46/10^6$ kg/Joule $(1.5 \frac{\text{lb}}{\text{hp-hr}})$ fully developed	Lower launch weight, higher landing weight compared to storable APU
Fuel Cell	System Power Density	11.3 kg/kW (25 lb/kW)	9.1 kg/kW (20 lb/kW)	Reduced weight for prime electrical power
	Solid Electrolyte	Fully developed for Shuttle	4.5 kg/kW (10 lb/kW)	Reduced weight for prime electrical power

The data for the secondary technology projections were collected from a number of diverse sources. The main source for each disciplines's analysis are summarized in Table A-7.

TABLE A-7.- SOURCE MATERIAL FOR TECHNOLOGY ASSESSMENT

Discipline	Sources
Aerothermodynamics	H. A. Stine: *Effects of Surface Catalysis on Heat Transfer to Shuttle Orbiters.* NASA Ames TMX-62, 016, March 15, 1971.
	W. B. Olstad: "Computational Analysis and Flight Experience." *Astronautics and Aeronautics,* December 1974.
Performance Optimization	J. L. Kamm: *Development of a Shuttle Optimal Abort Program (SOAP).* TRW Systems Group, MSC/TRW Task A-521, unpublished.
Computer Technology	Task Group, JPL: *A Forecast of Space Technology, 1980-2000.* NASA SP-387, 1976.
	"Spaceborne Digital Computer Systems. NASA SP-8070. *Guidance and Control Design Criteria,* March 1971.
	F. G. Withington: *Beyond 1984: A Technology Forecast.* Datamation, January 1975.
Aerodynamics	D. R. Chapman; H Mark; M. W. Pirtle: "Computers versus Wind Tunnels." *Astronautics and Aeronautics,* April 1975.
Control Systems	Advanced Control Technology and Its Potential for Future Transport Aircraft. NASA Symposium, July 1974.
	Task Troup, JPL: *A Forecast of Space Technology, 1980-2000.* NASA SP-387, 1976.
	"Present U.S. Fly-by-Wire Programs." *Astronautics and Aeronautics,* July/Aug 1974.
Auxiliary Power	*High Temperature Polyimide Hydraulic Actuator Rod Seals for Advanced Aircraft.* LeRC Development Program, SAE #700790.
	Testing of a Pneumatic Servomechanism. Bendix Dynavector, AFFDL-TR-71-146, Feb 1972.
	Fuel Cell Technology Program. PWA Final Report, CR-135002, 25 Jul 1973.
	Advanced Development Fuel Cell Program. GE Final Report, LPR-023, 20 Aug 1974, unpublished.

APPENDIX B

WORK BREAKDOWN STRUCTURE

A listing of the Work Breakdown Structure (WBS) used in the SSTO costing analysis is presented in this appendix. This WBS is similar to the Space Shuttle WBS. Costs for each of these items was computed based on either system weights and areas, or input as discrete values.

Item No.	WBS No.	Item
1	01-00-00-00-00	S.S.T.O. R.D.T.AND E
2	01-00-00-00	PROGRAM MANAGEMENT
3	02-00-00-00	SYSTEMS ENG, INT
12	03-00-00-00-00	AIR VEHICLE DESIGN
13	03-00-01-00-00	STRUCTURE
14	03-00-01-01-00	CREW SECTION
15	03-00-01- -01	TOOLING
16	03-00-01- -02	MATERIAL + SUBCONTRACT
17	03-00-01- -03	DESIGN
18	03-00-01- -04	TEST
19	03-00-01-02-00	CARGO/PROPULSION
20	03-00-01- -01	TOOLING
21	03-00-01- -02	MATERIAL + SUBCONTRACT
22	03-00-01- -03	DESIGN
23	03-00-01- -04	TEST
24	03-00-01-03-00	AERO CONTROL SURFACES
25	03-00-01- -01	TOOLING
26	03-00-01- -02	MATERIAL + SUBCONTRACT
27	03-00-01- -03	DESIGN
28	03-00-01- -04	TEST
29	03-00-02-00-00	THERMAL PROTECTION
30	03-00-02- -01	TOOLING
31	03-00-02- -02	MATERIAL + SUBCONTRACT
32	03-00-02- -03	DESIGN
33	03-00-02- -04	TEST
34	03-00-03-00-00	LANDING GEAR
35	03-00-03- -01	TOOLING
36	03-00-03- -02	MATERIAL + SUBCONTRACT
37	03-00-03- -03	DESIGN
38	03-00-03- -04	TEST
43	03-02-00-00-00	PROPULSION
44	03-02-01-00-00	MAIN
45	03-02-01-01-00	L.V.M.
46	03-02-01-02-00	INTEGRATION
47	03-02-01- -01	DESIGN
48	03-02-01- -02	TEST
49	03-02-01-00-03	G.F.E. ENGINES
50	03-02-02-00-00	DROP TANKS
55	03-02-02- -01	PROPULSION MAT + SUB
56	03-02-02- -02	PROPULSION DESIGN
57	03-02-02- -03	PROPULSION TEST
73	03-02-03-00-00	ATTITUDE CONTROL SYS.
74	03-02-03-01-00	ENGINES
75	03-02-03-02-00	L.V.M.
76	03-02-03-03-00	TANKS AND PODS
77	03-02-03-04-00	INTEGRATION
78	03-02-03- -01	DESIGN
79	03-02-03- -02	TEST

Item No.	WBS No.	Item
80	03-02-04-00-00	CRUISE PROPULSION
81	03-02-0 -01-00	ENGINES
82	03-02-0 -02-00	L.V.M.
83	03-02-0 -03-00	TANKS
84	03-02-0 -04-00	INTEGRATION
85	03-02-0 - -01	DESIGN
86	03-02-0 - -02	TEST
87	03-02-05-00-00	ORBIT MANEUVERING SYS.
88	03-02-05-01-00	ENGINES
89	03-02-05-02-00	L.V.M.
90	03-02-05-03-00	INTEGRATION
91	03-02-05- -01	DESIGN
92	03-02-05-04-00	TANKAGE DESIGN
93	03-02-06-00-00	AUXILIARY PROPULSION
94	03-02-06-01-00	ENGINES
95	03-02-06-02-00	INTEGRATION
96	03-02-06- -01	DESIGN
97	03-02-06-03-00	TANKAGE
103	03-03-00-00-00	AVIONICS
104	03-03-00- -01	AVIONICS MAT + SUB
105	03-03-00- -02	AVIONICS DESIGN
106	03-03-00- -03	AVIONICS TEST
107	03-03-01-00-00	GUIDANCE + NAVIGATION
108	03-03-01- -01	MATERIAL + SUBCONTRACT
109	03-03-01- -02	DESIGN
110	03-03-01- -03	TEST
111	03-03-02-00-00	FLIGHT CONTROL ELEMENT
112	03-03-02- -01	MATERIAL + SUBCONTRACT
113	03-03-02- -02	DESIGN
114	03-03-02- -03	TEST
115	03-03-03-00-00	DATA MANAGEMENT
116	03-03-03- -01	MATERIAL + SUBCONTRACT
117	03-03-03- -02	DESIGN
118	03-03-03- -03	TEST
119	03-03-04-00-00	COMMUNICATION + NAVIG.
120	03-03-04- -01	MATERIAL + SUBCONTRACT
121	03-03-04- -02	DESIGN
122	03-03-04- -03	TEST
123	03-03-05-00-00	CREW STATION CONTROLS
124	03-03-05- -01	MATERIAL + SUBCONTRACT
125	03-03-05- -02	DESIGN
126	03-03-05- -03	TEST
127	03-03-06-00-00	SOFTWARE
128	03-03-06- -01	MATERIAL + SUBCONTRACT
129	03-03-06- -02	DESIGN
130	03-03-06- -03	TEST
131	03-04-00-00-00	ECLS GROUP
132	03-04-01-00-00	ECS, CRYOGENIC

Item No.	WBS No.	Item
133	03-04-01- -01	MATERIAL + SUBCONTRACT
134	03-04-01- -02	DESIGN
135	03-04-01- -03	TEST
136	03-04-02-00-00	CREW SYSTEMS
137	03-04-02- -01	MATERIAL + SUBCONTRACT
138	03-04-02- -02	DESIGN
139	03-04-02- -03	TEST
140	03-05-00-00-00	POWER SUPPLY GROUP
141	03-05-01-00-00	ELECTRICAL POWER
142	03-05-01-01-00	ELECTRICAL DISTRIBUTN.
143	03-05-01- -01	MATERIAL + SUBCONTRACT
144	03-05-01- -02	DESIGN
145	03-05-01- -03	TEST
146	03-05-01-02-00	FUEL CELLS
147	03-05-01- -01	MATERIAL + SUBCONTRACT
148	03-05-01- -02	DESIGN
149	03-05-01- -03	TEST
150	03-05-02-00-00	HYDRAULIC + PNEUMATIC
151	03-05-02- -01	TOOLING
152	03-05-02- -02	MATERIAL + SUBCONTRACT
153	03-05-02- -03	DESIGN
154	03-05-02- -04	TEST
160	-02-00-00-00-00	FIRST UNIT COST
161	-02-00-00- -01	PRODUCTION
162	-02-00-00- -02	MATERIAL + SUBCONTRACT
164	-02-01-00-00-00	AIRFRAME
165	-02-01-01-00-00	STRUCTURE
166	-02-01-01-01-00	CREW SECTION
167	-02-01-01- -01	PRODUCTION
168	-02-01-01- -02	MATERIAL + SUBCONTRACT
169	-02-01-01-02-00	CARGO/PROPULSION
170	-02-01-01- -01	PRODUCTION
171	-02-01-01- -02	MATERIAL + SUBCONTRACT
172	-02-01-01-03-00	AERO CONTROL SURFACES
173	-02-01-01- -01	PRODUCTION
174	-02-01-01- -02	MATERIAL + SUBCONTRACT
175	-02-01-02-00-00	THERMAL PROTECTION
176	-02-01-02- -01	PRODUCTION
177	-02-01-02- -02	MATERIAL + SUBCONTRACT
178	-02-01-03-00-00	LANDING GEAR
179	-02-01-03- -01	PRODUCTION
180	-02-01-03- -02	MATERIAL + SUBCONTRACT
181	-02-02-00-00-00	PROPULSION
182	-02-02-00- -01	PROPULSION PRODUCTION
183	-02-02-00- -02	PROPULSION MAT + SUB
184	-02-02-01-00-00	MAIN
185	-02-02-01-01-00	L.V.M.
186	-02-02-01-02-00	INTEGRATION

Item No.	WBS No.	Item
187	-02-02-01-03-00	ENGINE
204	-02-02-03-00-00	ATTITUDE CONTROL SYS.
205	-02-02-03-01-00	ENGINES
206	-02-02-03-02-00	L.V.M.
207	-02-02-03-03-00	TANKS
208	-02-02-03-04-00	INTEGRATION
209	-02-02-04-00-00	CRUISE PROPULSION
210	-02-02-04-01-00	ENGINES
211	-02-02-04-02-00	L.V.M.
212	-02-02-04-03-00	TANKS
213	-02-02-04-04-00	INTEGRATION
214	-02-02-05-00-00	ORBIT MANEUVERING SYS.
215	-02-02-05-01-00	ENGINES
216	-02-02-05-02-00	L.V.M.
217	-02-02-05-03-00	INTEGRATION
218	-02-02-06-00-00	AUXILIARY POWER UNIT
219	-02-02-06-01-00	ENGINES
220	-02-02-06-02-00	INTEGRATION
225	-02-03-00-00-00	AVIONICS
226	-02-03-00- -01	AVIONICS PRODUCTION
227	-02-03-00- -02	AVIONICS MAT + SUB
228	-02-03-01-00-00	GUIDANCE + NAVIGATION
229	-02-03-01- -01	PRODUCTION
230	-02-03-01- -02	MATERIAL + SUBCONTRACT
231	-02-03-02-00-00	FLIGHT CONTROL ELEMENT
232	-02-03-02- -01	PRODUCTION
233	-02-03-02- -02	MATERIAL + SUBCONTRACT
234	-02-03-03-00-00	DATA MANAGEMENT
235	-02-03-03- -01	PRODUCTION
236	-02-03-03- -02	MATERIAL + SUBCONTRACT
237	-02-03-04-00-00	COMMUNICATION + NAVIG.
238	-02-03-04- -01	PRODUCTION
239	-02-03-04- -02	MATERIAL + SUBCONTRACT
240	-02-03-05-00-00	CREW STATION + CONTROL
241	-02-03-05- -01	PRODUCTION
242	-02-03-05- -02	MATERIAL + SUBCONTRACT
243	-02-04-00-00-00	ECLS GROUP
244	-02-04-01-00-00	ECS, CRYOGENIC
245	-02-04-01- -01	PRODUCTION
246	-02-04-01- -02	MATERIAL + SUBCONTRACT
247	-02-04-02-00-00	CREW SYSTEMS
248	-02-04-02- -01	PRODUCTION
249	-02-04-02- -02	MATERIAL + SUBCONTRACT
250	-02-05-00-00-00	POWER SUPPLY GROUP
251	-02-05-01-00-00	ELECTRICAL POWER
252	-02-05-01-01-00	ELECTRICAL DISTRIBUTN.
253	-02-05-01- -01	PRODUCTION
254	-02-05-01- -02	MATERIAL + SUBCONTRACT

Item No.	WBS No.	Item
255	-02-05-01-02-00	FUEL CELLS
256	-02-05-01- 01	PRODUCTI/ON
257	-02-05-01- -02	MATERIAL + SUBCONTRACT
258	-02-05-02-00-00	HYDRAULIC + PNEUMATIC
259	-02-05-02- -01	PRODUCTION
260	-02-05-02- -02	MATERIAL + SUBCONTRACT
261	-02-06-00-00-00	FINAL ASSEMBLY + C/O
262	-02-06-00- -10	PRODUCTION
263	-02-06-00- -01	TOTAL PRODUCTION
266	-02-07-00-00-00	SUSTAINING ENGINEERING
267	-02-07-00- -01	ENGINEERING
269	-02-07-00- -06	SUSTAINING TOOLING
380	-04-00-00-00 -01	GROUND SUPPORT EQUIP
381	-04-01-00-00 -01	HANDLING + TRANS.EQ.
382	-04-02-00-00 -01	SERVICING EQUIP
383	-04-03-00-00 -01	TEST + C/O EQUIP
384	-05-00-00-00 -01	TRAINING SERV. + EQ.
385	-05-01-00-00 -01	TRAINING SERVICES
386	-05-02-00-00 -01	TRAINING SIM. +EQUIP
387	-06-00-00-00 -01	SYS. TEST + EVAL.
388	-06-01-00-00 -01	GROUND TEST HARDWARE
389	-06-02-00-00 -01	WIND TUNNEL TESTS
390	-06-03-00-00 -01	STATIC FIRE TESTS
391	-06-04-00-00 -01	MOCK-UPS
392	-06-05-00-00 -01	FLIGHT TESTS
393	-06-05 01-00 -01	FLIGHT TEST HARDWA
394	-06-05 02-00 -01	HORIZONTAL(TAXI)TEST
395	-06-05 03-00 -01	VERTICAL TESTS
396	-06-05 04-00 -01	MISC MODS + REFURB
397	-06-05 05-00 -01	FLT. TEST INST RUM.
398	-07-00-00-00 -01	LOGISTICS
399	-07-01-00-00 -01	TRANSPORTATION
400	-07-02-00-00 -01	SPARES
401	-11-00-00-00 -01	FACILITIES
402	-11-00-00- 01-0	FEE

272	-03-00-00-00-06	PRODUCTUON
273	-03-00-00- -01	PRODUCTION
274	-03-00-00- -02	MATERIAL + SUBCONTRACT
275	-03-01-00-00-00	AIRFRAME
276	-03-01-01-00-00	STRUCTURE
277	-03-01-01-01-00	CREW SECTION
278	-03-01-01- -01	PRODUCTION
279	-03-01-01- -02	MATERIAL + SUBCONTRACT
280	-03-01-01-02-00	CARGO/PROPULSION

Item No.	WBS No.	Item
281	-03-01-01- -01	PRODUCTION
282	-03-01-01- -02	MATERIAL + SUBCONTRACT
283	-03-01-01-03-00	AERO CONTROL SURFACES
284	-03-01-01- -01	PRODUCTION
285	-03-01-01- -02	MATERIAL + SUBCONTRACT
286	-03-01-02-00-00	THERMAL PROTECTION
287	-03-01-02- -01	PRODUCTION
288	-03-01-02- -02	MATERIAL + SUBCONTRACT
289	-03-01-03-00-00	LANDING GEAR
290	-03-01-03- -01	PRODUCTION
291	-03-01-03- -02	MATERIAL + SUBCONTRACT
292	-03-02-00-00-00	PROPULSION
293	-03-02-00- -01	PROPULSION PRODUCTION
294	-03-02-00- -02	PROPULSION MAT + SUB
295	-03-02-01-00-00	MAIN
296	-03-02-01-01-00	L.V.M.
297	-03-02-01-02-00	INTEGRATION
298	-03-02-01-03-00	ENGINE
310	-03-02-03-00-00	ATTITUDE CONTROL SYS.
311	-03-02-03-01-00	ENGINES
312	-03-02-03-02-00	L.V.M.
313	-03-02-03-03-00	TANKS
314	-03-02-03-04-00	INTEGRATION
315	-03-02-04-00-00	CRUISE PROPULSION
316	-03-02-04-01-00	ENGINES
317	-03-02-04-02-00	L.V.M.
318	-03-02-04-03-00	TANKS
319	-03-02-04-04-00	INTEGRATION
320	-03-02-05-00-00	ORBIT MANEUVERING SYS.
321	-03-02-05-01-00	ENGINES
322	-03-02-05-02-00	L.V.M.
323	-03-02-05-03-00	INTEGRATION
324	-03-02-06-00-00	AUXILIARY POWER UNIT
325	-03-02-06-01-00	ENGINES
326	-03-02-06-02-00	INTEGRATION
327	-03-03-00-00-00	AVIONICS
328	-03-03-00- -01	AVIONICS PRODUCTION
329	-03-03-00- -02	AVIONICS MAT + SUB
330	-03-03-01-00-00	GUIDANCE + NAVIGATION
331	-03-03-01- -01	PRODUCTION
332	-03-03-01- -02	MATERIAL + SUBCONTRACT
333	-03-03-02-00-00	FLIGHT CONTROL ELEMENT
334	-03-03-02- -01	PRODUCTION
335	-03-03-02- -02	MATERIAL + SUBCONTRACT
336	-03-03-03-00-00	DATA MANAGEMENT
337	-03-03-03- -01	PRODUCTION
338	-03-03-03- -02	MATERIAL + SUBCONTRACT
339	-03-03-04-00-00	COMMUNICATION + NAVIG.

Item No.	WBS No.	Item
340	-03-03-04- -01	PRODUCTION
341	-03-03-04- -02	MATERIAL + SUBCONTRACT
342	-03-03-05-00-00	CREW STATION + CONTROL
343	-03-03-05- -01	PRODUCTION
344	-03-03-05- -02	MATERIAL + SUBCONTRACT
345	-03-04-00-00-00	ECLS GROUP
346	-03-04-01-00-00	ECS, CRYOGENIC
347	-03-04-01- -01	PRODUCTION
348	-03-04-01- -02	MATERIAL + SUBCONTRACT
349	-03-04-02-00-00	CREW SYSTEMS
350	-03-04-02- -01	PRODUCTION
351	-03-04-02- -02	MATERIAL + SUBCONTRACT
352	-03-05-00-00-00	POWER SUPPLY GROUP
353	-03-05-01-00-00	ELECTRICAL POWER
354	-03-05-01-01-00	ELECTRICAL DISTRIBUTN.
355	-03-05-01- -01	PRODUCTION
356	-03-05-01- -02	MATERIAL + SUBCONTRACT
357	-03-05-01-02-00	FUEL CELLS
358	-03-05-01- -01	PRODUCTION
359	-03-05-01- -02	MATERIAL + SUBCONTRACT
360	-03-05-02-00-00	HYDRAULIC + PNEUMATIC
361	-03-05-02- -01	PRODUCTION
362	-03-05-02- -02	MATERIAL + SUBCONTRACT
365	-03-06-00-00-00	FINAL ASSEMBLY +C/O
366	-03-06-00-00-00	INTEGRATION ASSEM.
367	-03-07-00-00 00	SUSTAINING ENGR
368	-03-08-00-00-00	SUSTAINING TOOLING
369	-03-08-00-00-01	FEE

404	03-00-00-00-00 -0	OPERATIONS	
405	08-00-00-00 -0	LAUNCH OPERATIONS	
406	08-01-00-00 -0	KSC CIVIL SERVICE	
407	08-02-00-00 -0	PROPELLANTS	
408	08-02 01-00 -0	LH 2	
409	08-02 02-00 -0	LOX	
410	08-02 03-00 -0	LN-2	
411	08-02 04-00 -0	LIQUID AIR	
412	08-02 05-00 -0	GHE	
413	08-02 06-00 -0	GO-2	
414	08-02 07-00 -0	GH-2	1
415	08-02 08-00 -0	FREON,AMMON,HYD.FLUID	
416	08-03-00-00 -0	GROUND SYS. CONTRACT	
417	08-04-00-00 -0	ORBITER SPARES	
418	09-00-00-00 -0	FLIGHT OPERATIONS(JS)	
419	10-00-00-00 -0	REFURBISHMENT(GD.SYS)	

APPENDIX C

LAUNCH AND FLIGHT OPERATIONS FUNCTIONAL AND COST ANALYSIS

The largest cost items that have contributed to operational costs of previous manned orbital vehicles are launch operations at ETR and WTR and flight operations at JSC.

Significantly lower costs for SSTO in the 1995 through 2009 time period can be expected. A realistic approach to estimating these projected costs has been taken by addressing the potential simplification and combination of operational functions and by anticipating automated (computerized) techniques for mission planning and operations. This approach provides results that project cost reductions based on having acquired substantial operational experience and technology improvements during the next 15 years, as well as on having a less complex flight vehicle available. Costs in this appendix are expressed in FY 1971 dollars.

Space Shuttle Orbiter Baseline Launch Operations

The baseline launch oprations costs per flight (CPF) for the Space Shuttle program are taken from Reference 1 and are as follows:

	Dollars in millions
KSC civil service	0.51
Propellants	0.31
Ground operations	0.42
Secondary landing site	0.06
Orbiter ferry oprations	0.01
Ground systems support	0.78
Orbiter spares (including GSE)	0.84
TOTAL	2.93

The SSTO study requires no secondary landing site or ferry operations, and the propellant and spares categories do not include personnel costs. The remaining three areas are therefore important focal points for the cost reduction analyses described in the following paragraphs.

(1) KSC civil service ($0.51 million per flight).- These costs include spare parts inventory maintenance, sustaining engineering, and ferry kit installation and removal. Because the ferry kit effort is not required, spare parts inventory maintenance can be automated and sustaining engineering can be reduced using a technical representative approach with specific discipline specialists on call, and combining LCC and MCC functions.

(2) Ground operations element ($0.42 million per flight).- This element includes costs related to refurbishment, maintenance, and operators of orbiter-peculiar GSE and Main Engine assembly and disassembly with the vehicle, and are based on an average of 60 flights per year. For the SSTO, this area represents a potential for significant cost saving.

(3) Ground systems support ($0.78 million per flight).- This element includes contractor support for vehicle software maintenance, launch processing system (LPS) to vehicle interfaces, vehicle systems monitoring and control for prelaunch and launch activities in support of integrated systems tests and LCC console engineers. These activities represent significant potential savings. Other activities in this element include surveillance of GSE handling equipment, crew equipment, launch site storage and maintenance of cleaning equipment. These activities do not represent significant areas for potential savings, although some automation and improved efficiency gains are foreseen.

SSTO Launch Operations

In analyzing SSTO launch operations the Space Shuttle Turnaround Analysis Report (STAR) 0008 functional flow was used to simplify these activities. The SSTO functional flow (Figure C-1) illustrates estimated time for each activity, which produces vehicle turnaround time of 60 hours compared with the current 160 hours for Space Shuttle. The most significant saving is estimated to occur in the activities that are checkout related, as illustrated by the LPS interfaces in Figure C-1.

Experience gained in the analysis of these activities during our DOD/STS Ground Operations Study (Reference 2) was drawn upon to combine or reduce those functions that logically are potentially within "normal" technology growth. Specific crew sizes related to each function were drawn from Study Report (MCR-74-309), entitled *Recommended Concept, Siting Arrangement and Acquisition Plan* for Western Test Range operations in the 1980s. Those functions relating only to solid rocket booster (SRB) or external tank (ET) were excluded. These crew sizes and the projected crew size for SSTO operations are illustrated in Table C-1.

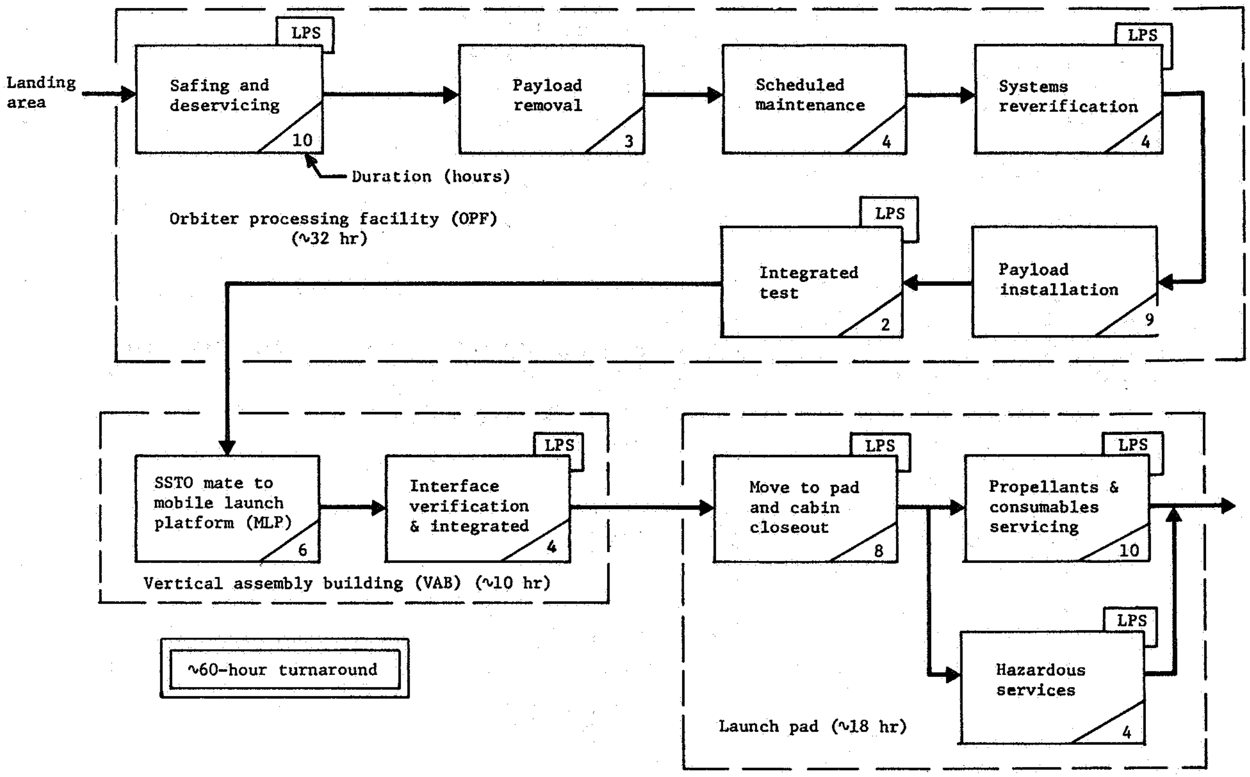

Figure C-1.- SSTO streamlined ground operations flow

TABLE C-1.- SUPPORT CREW SIZING COMPARISON FOR LAUNCH OPERATIONS

Support activity or area	Vehicle	Work force: Basic	LPS	RPIE Maint	SE Maint	Tng	Total	Support force: Basic	Logistics: Equip	Logistics: Facil	Total	Total staff	Manhours per flight
Landing & runway support	Orbiter	1					1	1			1	2	2
	SSTO	1					1	1			1	2	2
Safing & Deservice (OPF)	Orbiter	5	1		1		7	4	2		6	13	130
	SSTO	5	1		1		7	4	2		6	13	121
Other operations (OPF)	Orbiter	37	11	4	8	3	63	32	12	4	48	111	10 464
	SSTO	20	4	4	4	2	34	18	12	4	32	66	1 217.5
Vertical assembly building & launch pad (LCC)	Orbiter	65	26	7	15	6	119	61	22	8	91	210	13 230
	SSTO	24	2	7	8	2	43	24	10	6	40	83	759.5

Orbiter force per WTR ground operations study based on 18 flights/year and 160 hour turnaround.

RPIE = real property installed equipment

SE = support equipment

Landing and runway support.- The two-man orbiter and runway crew is required for one hour per the current STAR. Because there are 114 flights per year, this operation is performed approximately once every third day. No significant reductions are foreseen due to personnel physical limitations and, therefore, the activity is carried in the SSTO at the same cost per flight (two manhours per flight) using technician manpower.

Safing and deservicing.- The orbiter crew is charged with venting, draining, and purging residual RCS and main propellants and environmental control and life support systems consumables and fuel cell tanks, as well as removing hypergolic modules. These activities are allocated 10 hours with a crew of 13 men (11 mechanical, one electrical technican, and one engineer for the LPS console). This SSTO operation is estimated to take 10 hours due to offsetting factors of improved efficiency and increased propellant tank size. However, the portion of the operation requiring LPS monitoring by an engineer is reduced to one hour. The traffic rate results in a facility use of 1140 hours per year. Approximately 50% of a one-shift operation can be accommodated by the existing OPF with only a single cell. However, provisions for parallel operations using two cells would provide better schedule flexibility. The same crew size as needed for Space Shuttle is considered adequate for SSTO (12 technicians, 120 manhours per flight).

Other Operations in OPF

The remaining orbiter operations in the Orbiter Processing Facility (OPF) require 86 hours turnaround time and require a crew of 109 for support, including 11 engineers manning the LPS consoles. By automating and combining this function, the turnaround can be reduced from 86 to 20 hours, and LPS operations reduced to three hours.

The total time in the OPF of 32 hours indicates that adequate work stations are desirable to accommodate two SSTOs in the facility at one time. The crew size work force for performing the functions indicated by the streamlined flow of Figure C-1 are reduced to a total of 66 men, which include four LPS and two training engineers.

VAB and launch pad support.- In these activities the SSTO is mated to the Mobile Launch Platform and the vehicle is moved to the pad by a crawler-transporter for final preparations, propellant loading, and launch. In the orbiter's 160 hour turnaround cycle, VAB activities used 39 hours and the vehicle spent 24 hours on the pad. For SSTO, the combination of functions and automation results in reducing these times to 10 and 18 hours respectively.

Functions requiring LPS monitoring and support for the two areas are reduced to 14 hours. For the 10-hour period of propellant and consumables loading, the LPS function requires only one engineer. A second engineer can cover the parallel hazardous service operations. In this approach, prelaunch console manning is done by the flight operations team, with control handover after propellant servicing in lieu of tower-clear. The range safety function is assumed to remain at KSC but is to be based on onboard automated checkout plus flight crew intelligence. The LPS support then becomes three engineers for six hours each for a total of 18 engineering manhours per flight. Training requies 17.5 manhours per flight. Technician time for these functions is based on a support crew of four for an 8-hour roll-out and cabin closeout giving 32 manhours per flight, plus the full 83 man support crew for the propellant and hazardous service period of four hours giving 332 manhours per flight, plus 60 men for the remaining six hours or 360 manhours per flight. The total requirements are 724 manhours per flight for technicians and 35.5 manhours per flight for engineers.

Launch operations manpower and cost summary.- The manpower comparisons resulting from applying the foregoing rationale are shown in Table C-2.

TABLE C-2.- MANHOUR PER FLIGHT COMPARISON

Functional area	Support manpower, mh/flt	
	Orbiter	SSTO
Landing & runway support	2	2
Safing & deservice (OPF)	130	121
Other operations (OPF)	10 464	1217.5
VAB & launch pad	13 230	759.5
TOTAL	23 826	2100
mh/flt - manhours per flight		

Applying 36% reduction to orbiter values for ET and SRB deletions (Reference 2) results in 15 249 manhours per flight for orbiter. The relative factor for SSTO relatable to reduced turnaround time and automation is 0.138. Applying this factor to the data of Reference 1 yields the following costs for the SSTO:

	Dollars in millions
KSC civil service	0.51 x 0.138 = 0.070
Ground operations	0.42 x 0.138 = 0.058
Ground systems support	0.78 x 0.138 = 0.107

The total launch operations cost includes the foregoing costs related to manpower as well as cost for propellants and spares.

Space Shuttle Orbiter Baseline Flight Operations

The baseline flight operations costs per flight (CPF) for the Space Shuttle program are listed below and discussed in succeeding paragraphs.

	Dollars in millions
JSC civil service	0.15
Mission control & crew operations	0.70
Program support	1.62
Allowance for growth	0.445
Total	2.915

(1) JSC civil service ($0.15 million per flight).- These costs include spares inventory maintenance for mockups, trainers, and posttask simulators. Crew training and crew procedures documentation maintenance is also included. Civil service personnel (JSC) man the consoles in the Mission Control Center (MCC) Mission Operations Control Room (MOCR) and these personnel are also included in this element. Historically, this support force has been sizeable and manned the MOCR consoles 24 hours per day. Training has been extensive to allow crewmen to become proficient in onboard systems and scientific experiments. These areas are foreseen as potential cost reductions by onboard automation and function combination techniques.

(2) Mission control and crew operations ($0.70 million per flight).- This element includes the operation of mockups, trainers, part-task simulators and mission simulators. This includes contractor personnel as were used previously for Mercury, Gemini, Apollo, and Skylab programs. An on-call function is provided in the Multi-Purpose Support Room (MPSR).

(3) Program support element ($1.62 million).- This element consists of the equipment, food, cameras, and biomedical equipment for crew personnel. Mockups, trainers, training aircraft costs, and spares are included for mission simulator and training aircraft. For this element, the SSTO flight rate allows spreading the costs of training, aircraft, and mockups, and thus reduces the cost per flight. This can be accomplished by more automation and self-test by onboard systems. The development of these capabilities is expected to occur with technology advancements in the current orbiter program and should not require excessive additional SSTO costs. The orbiter onboard computer capability has inherent flexibility to accommodate new technology

developments and reduce the crew training requirements.

(4) Allowance for growth ($0.445 million per flight).- This element is anticipated to be unnecessary for SSTO.

The JSC Baseline Operations Plan (BOP) is used as a source for functional definitions. The BOP summarizes the flight operations responsibilities for Space Shuttle orbiter operations as follows:

Concept: Small team of flight controllers for real-time support in MCC

Provide communication management and central voice interface to orbiter crew

Consult with orbiter and ground support, including NRT support for systems, trajectory and medical problems

Coordinate support facilities for effective data retrieval

Provide required mission support services

SSTO Flight Operations

Support team positions.- Table C-3 tabulates the number of positions defined by the current BOP for each support team area.

TABLE C-3.- SHUTTLE SUPPORT TEAM POSITIONS

Support area	Team size
Master operations control room (MOCR)	15
Mission control rooms (MCR)	6
Multipurpose support rooms (MPSR)	On call
Data retrieval and analysis	
Vehicle systems support	
Contamination, radiation, and weather	
Trajectory operations	
Training	
Documentation and distribution	

The MOCR function is decision making, similar to the flight operations management room (FOMR) during Skylab. The MCR corresponds to the mission operation control room for Apollo/Skylab in which the flight director and his staff of senior civil service engineers manned consoles for each discipline. Each console engineer was supported by specialists in the staff support rooms (SSR), which correspond to the MPSR support except that support was provided for three shifts during the previous programs instead of the "on call" support planned for Space Shuttle.

Several assumptions must be made to establish relative support manpower levels for SSTO. The MOCR and MCR team of 21 persons provides 24 hours per day for two days (average), or 1008 manhours per flight. For the on call functions, a team of 24 is estimated with a use of 25% for a total of 32 832 manhours per year. This support reduces to 547 manhours per flight for 60 flights per year, yielding a total of 1555 manhours per flight. Space Shuttle orbiter MCC activity is reflected in the JSC civil service and mission control and crew support categories.

Assuming a commercial airlines approach, managerial decisions are made on call, MCR functions are reduced to "control tower" functions, and the MPSR functions remain on call. Assuming a MCR support level of two men for two days per flight and 24 hours per day gives 96 manhours per flight. By automation, the management and MPSR support crew can be reduced to 12 with the same 25% use for a total of 16 416 manhours per year. This support reduces to 144 manhours per flight for the on call functions or a total of 240 manhours per flight. This compares with the 1555 manhours per flight for orbiter, yielding a reduction factor of 0.15. The reduced training requirements for SSTO also affect these areas, but produce a major impact in program support, which is addressed in the following paragraph.

Functional allocations.- A comparison of allocation of operational functions for current and projected levels of autonomy was made. The tradeoff between onboard and ground performance of these functions as well as methods of implementation was examined. Some functions are allocated and implemented onboard regardless of the level of autonomy, as illustrated in Table C-4, whereas others depend on the level, shown in Table C-5. The major impact of automation is in reduced training and simulation requirements for onboard and MCC crews. The reduced number of manual functions results in both support crew and training cost reductions.

TABLE C-4.- FUNCTIONS WHICH DO NOT CHANGE WITH AUTONOMY LEVEL

Function	Apollo concept Shuttle application				Highly autonomous concept			
Functional allocation	Onboard		Ground		Onboard		Ground	
Implementation method	Auto	Man	Auto	Man	Auto	Man	Auto	Man
Orbit determination	X				X			
Guidance and navigation	X				X			
Guidance voting		X				X		
Cruise and flight control	X				X			
Backup cruise/flt control		X				X		
Flight safety failure detection and isolation	X				X			
Backup and alternative systems implementation		X				X		
Routine systems maintenance		X				X		

TABLE C-5.- FUNCTIONS WHICH VARY WITH AUTONOMY LEVEL

Function	Apollo concept Shuttle application				Highly autonomous concept			
Functional allocation	Onboard		Ground		Onboard		Ground	
Implementation method	Auto	Man	Auto	Man	Auto	Man	Auto	Man
Vehicle ephemeris determination			X		X			
Real time mission planning				X			X	
Alternative return planning				X			X	
Solar flare, weather, etc.				X			X	
Performance trend analysis				X	X			
Routine scheduled maintenance				X	X			
Routine calibration			X		X			
Consumables planning			X		X			
Failure diagnosis				X			X	
Postflight data collecting				X			X	

Another area expected to reduce cost for program support is crew equipment. This is achieved by standardization and new manufacturing processes that minimize the uniqueness of space-related equipment. Cost per flight for this element can conservatively be reduced to 25% of the present projected cost.

Flight operations manpower and cost summary.- Applying the reduction factors developed in the previous paragraphs, the SSTO flight operations CPF becomes:

	Dollars in millions
JSC civil service	0.15 x 0.15 = 0.023
Mission control and crew operations	0.70 x 0.15 = 0.105
Program support	1.62 x 0.25 = 0.405
Total	0.533 per flight

REFERENCES

1. *Space Shuttle Program, Space Shuttle System Cost Per Flight Control Document*, NASA JSC 07700, Volume XVI, Change 1, September 1975.

2. *DOD/STS Ground Operations Study*, AF-SAMSO-TR-234, Book 2, October 1974.

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