NASA Contractor Report 2879

Technology Requirements for Advanced Earth Orbital Transportation Systems

Volume 2: Summary Report

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CONTRACT NAS1-13944
JULY 1978

NASA

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A. K. Hepler and E. L. Bangsund
Boeing Aerospace Company
Seattle, Washington

Prepared for
Langley Research Center
under Contract NAS1-13944

NASA
National Aeronautics
and Space Administration

Scientific and Technical
Information Office

1978

FOREWORD

This report presents the detailed results of a study of Technology Requirements for Advanced Earth Orbital Transportation Systems conducted by The Boeing Company under Contract NAS1-13944 from June 1975 through March 1976.

The work was performed by the Advanced High Speed Transportation group of the Space Systems Division, Boeing Aerospace Company, at its Kent Space Center.

Study Manager was Mr. E. L. Bangsund under the administration of Mr. A. K. Hepler. A listing of the Boeing personnel who contributed directly in the studies and preparation of the report is as follows:

M. J. Baker	Propulsion
B. E. Clingan	Loads
V. Deriugin	Aerothermal Analysis
D. M. Eckblad	Loads
H. N. Johnson	Configuration
R. T. Savage	Aerothermal Analysis
A. R. Swegle	Stress
W. H. Walker	Subsystems/Ground Accelerator
H. Zeck	Aerodynamics and Performance

Mr. F. Kirby of Rocketdyne provided support for the main propulsion subcontract.

Contract documentation is provided as follows:

Document		Boeing No.	NASA No.
Volume 1:	Executive Summary	D180-19168-3	CR-2878 Vol. 1
Volume 2:	Summary Report	D180-19168-4	CR-2879 Vol. 2
Volume 3:	Summary Report-Dual Mode Propulsion	D180-19168-5	CR-3037 Vol. 3

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SUMMARY

Presented in this report are the results of a four task effort to identify the technology requirements associated with advanced earth orbital transportation systems. Task I was directed at providing assessments of current technology and normal growth to 1986 in key system and subsystem technology areas as applied to future advanced earth orbital transportation systems. Data for this effort was obtained from recent literature, subcontractors, government and industry sources and in-house field specialists. The projected technology level increases based on normal growth in structures and subsystems are 17% and 12.5% respectively.

Task II consisted of the design and definition of performance potential of three different types of vehicle concepts. These concepts were a ground sled launched horizontal take-off (HTO) mode, a vertical take-off (VTO) mode, and an inflight fueled (IFF) mode consisting of both aerial refuel and air launch. Individual tasks consisted of defining the alternate configurations, integrating the technology data from Task 1, defining subsystem performance requirements and environments, selecting subsystem concepts, analyzing and sizing subsystems, and calculating total configuration weights. Aerodynamic characteristics, flight performance, operational requirements, and systems costs were developed for each study configuration.

The sled assisted, horizontal take-off (HTO) vehicle appears to offer the lowest practically attainable GLOW, 1.0×10^6 kg (2.2 million lb) and life cycle cost, 8.1 billion dollars. Operational costs of 1.35 million dollars per flight resulted in a transportation cost of 45.64 dollars/kg

(20.7 dollars/lb) based on a payload of 29.5 x 10^3 kg (65,000 lb). Estimated c.g. location and aerodynamic characteristics indicate a stable and trimmable vehicle both at hypersonic and subsonic speeds.

The vertical take-off vehicle GLOW is estimated at 2.01 x 10^6 kg (4.4 million lb). The primary increase in weight was caused by the difference in propulsion thrust to weight ratios (.77 for the HTO versus 1.31 for the VTO) and associated scaling effects. In addition, the VTO vehicle design concept was based on generic association with the HTO vehicle which utilized LO_2 in the wing during ascent for inertial load relief. This generic commonality of fuel location might have unduly penalized the VTO configuration. However, additional analysis and study indicate that the overall GLOW could not be reduced below 1.81 x 10^6 kg (4.0 million lb) even on an optimistic basis. Resultant life cycle cost for this vehicle was 12.6 billion dollars which reflect the size impact on the cost model as well as the operational differences associated with the vertical launch vehicle in comparison with a more aircraft like horizontal take-off. The 2.3 million dollar cost per flight results in transportation cost of 78.0 dollars/kg (35.4 dollars/lb).

The inflight fueled and air launch vehicle reduced take-off weights of .771x 10^6 kg (1.7 million lb) for each vehicle result from launching at altitudes of 6096 - 9144 m (20 - 30,000 ft). Overall life cycle costs for this concept are about one billion dollars more due mostly to the tanker development and unit costs, but the cost per flight approaches that of the horizontal take-off concept. As a result of the size and cost differences and the technical development difficulties affecting concept feasibility (cryogenic refueling, balance and stability, and large tanker development) associated with the inflight fueled concept, the sled assisted horizontal take-off vehicle was selected with Government concurrence for the advanced technology assessment in Task III.

The Task III activity was involved with defining advanced subsystems and technology areas where performance advancements reap the large payload gains for the R&D dollars invested. Structures and propulsion were determined as critical areas for eventual development of an all-metallic, completely reusable, cost effective earth orbital transportation system. This includes the nickel brazed Rene'41 and aluminum brazed titanium honeycomb thermal/structural concept which accomplishes the dual function of providing adequate cryogenic insulation properties during ascent while operating within the temperature capabilities of the materials during reentry.

The two-position nozzle for the Space Shuttle Main Engine (SSME) also has a significant impact on Single Stage to Orbit (SSTO) performance. Aerodynamic heating, trajectory optimization, operations, cost analysis, and certain configuration/systems programs are also recommended for future study.

The Task IV extended performance vehicle GLOW was reduced to $.855 \times 10^6$ kg (1.886 million lb) when updated with selected advanced technology programs. Overall program cost was reduced by approximately 600 million dollars resulting in a cost per flight of 42.8 dollars/kg (19.4 dollars/lb).

INTRODUCTION

The Space Shuttle program is currently in the final development stages, and hardware is being fabricated. It is anticipated that this vehicle system, together with the planned space tug, will provide the space transportation capability for most of the requirements to transport men and material between earth and earth orbit at least until the 1990 time frame and, more probably, for several years to follow. This program has provided a significant technology base (and will continue to do so throughout its lifetime) upon which to build for future aerospace transportation systems. For long range planning purposes, consideration of the lead times associated with major vehicle system programs and the assumption of a nominal fifteen year operational lifetime for the Space Shuttle gives a clue to the possible schedule for the development of more advanced systems. The lead time from an "Authority to Proceed" to an operational system is of the order of eight to ten years, based on both Apollo and Space Shuttle experience.

For study purposes, the assumption was made that a follow-on system to be available in the 1995 time frame based on a nominal schedule would require that the planning for and development of the necessary technology base must be accomplished within the next ten years. A fundamental assumption underlies any consideration of these more advanced systems; any new system must offer clear and significant cost/performance advantages over current systems.

Three operational concepts resulting in four configurations of a Single Stage to Orbit system using advanced hydrogen fueled rocket engines for the main propulsion system were examined under this contract. A detailed examination of these systems in light of both normal technology growth anticipated for the time frame of interest and focused growth in selected areas have provided clues as to which technology areas should and must be pursued on a cost/performance basis.

The fundamental objective of this study was to identify those areas of technology associated with future earth orbit transportation systems which are either critical to the development of such systems or which offer a significant cost and performance advantage as a result of their development. Additional objectives were to determine the most efficient operational mode for such systems and to define performance potential as a function of technology growth.

SYMBOLS

A.C.	Alternating Current or Aerodynamic Center
A&CO	Assembly and Checkout
AH	Ampere Hour
ALRS	Advanced Launch and Recovery Systems
APU	Auxilliary Power Unit
AR	Aspect Ratio
ATP	Authority to Proceed
BETA	Boeing Engineering Thermal Analysis
B.F.L.	Basic Factory Labor
B.L.	Buttock Line
C_D	Drag Coefficient
CDR	Critical Design Review
C.G.	Center of Gravity
C_f	Friction Coefficient
C_L	Lift Coefficient
C_M	Moment Coefficient
C_N	Normal Force Coefficient
CPF	Cost Per Flight
DC	Direct Current
DDT&E	Design, Development, Test and Engineering
E.P.L.	Emergency Power Level
E/T	External Tank
ETR	Eastern Test Range

SYMBOLS (Cont.)

FACI	First Article Configuration Inspection
F.O.M.	Figure of Merit
F_t	Applied Tension Stress
F_{th}	Induced Thermal Stress
F_{ty}	Tensile Yield Strength
g	Gravity
G&N	Guidance and Navigation
GH_2	Gaseous Hydrogen
GLOW	Gross Lift Off Weight
GSE	Ground Support Equipment
h	Altitude
H/C	Honeycomb
He	Helium
HTO	Horizontal Take Off
HTOHL	Horizontal Take Off Horizontal Landing
I/F	Interface
IFF	Inflight Fueled
IMU	Inertial Measurement Unit
IOC	Initial Operational Capability
I_{SP}	Specific Impulse - Seconds
K	Allowable Operating Stress Reduction Factor
KSC	Kennedy Space Center
kW	Kilowatts
LCC	Life Cycle Cost
LE	Leading Edge

LH_2	Liquid Hydrogen
LO_2	Liquid Oxygen
LSI	Large System Integration
M	Mach Number
MAX	Mean Aerodynamic Chord
M_e	Local Mach Number
MEPU	Monopropellant Emergency Power Unit
MMH	Monomethyl Hydrazine
MSBLS	Microwave Scanning Beam Landing System
MUX	Multiplexer
n	Load Factor
N/A	Not Applicable
NASA	National Aeronautics and Space Administration
N_2O_4	Nitrogen Tetroxide
N_2H_4	Hydrazine
NiCd	Nickel Cadimum
NPSH	Net Positive Suction Head
O/F	Oxidizer/Fuel Ratio
OMS	Orbit Maneuvering System
O.W.E.	Operating Weight Empty
P_c	Chamber Pressure
PCM	Program Cost Model
PDR	Preliminary Design Review
P/L	Payload
PTA	Propulsion Test Article
Pr	Prandtl Number
q	Dynamic Pressure
QC	Quality Control

RCUR	Receiver
RCS	Reaction Control System
R&D	Research and Development
RI/SD	Rockwell International Space Division
Re_θ	Momentum Thickness Reynolds Number
Re_{BEG}	Reynolds Number at Beginning of Transition
Re_{END}	Reynolds Number at End of Transition
SE&I	System Engineering and Integration
SOB	Side of Body
SSME	Space Shuttle Main Engine
SST	Supersonic Transport
SSTO	Single Stage to Orbit
S_t	Stanton Number
STA	Structural Test Article
S_w	Wing Reference Area
TACAN	Tactical Air Navigation
t/c	Thickness/Chord Ratio
Ti	Titanium
T.O.	Take Off
TPS	Thermal Protection System
T/W	Thrust/Weight Ratio
t	Thickness
TVC	Thrust Vector Control
UHF	Ultra High Frequency
V	Velocity
V.A.C.	Volts Alternating Current
Vac	Vacuum
VTO	Vertical Take Off
VTOHL	Vertical Take Off Horizontal Landing

W_{inj}	Injected Weight
Wp	Propellant Weight
XDCRS	Transducers
XFR	Transfer
α	Angle of Attack
μ	Friction Factor
λ	Mass Fraction = $\frac{\text{Propellant Weight}}{\text{Gross Lift Off Weight - Payload}}$
ε	Nozzle Expansion Ratio, Emissivity or Strain
ε_{th}	Induced thermal strain
ε_{press}	Net strain caused by limit pressure loading
$\varepsilon_{static\ loads}$	Net strain caused by limit static loads
ε_{allow}	Strain at allowable stress
$\varepsilon_{F_{tu}}$	Strain at allowable ultimate tension stress
ε_{total}	Total material tension strain (elongation)
ε_{F_c}	Strain at compression allowable stress
ΔV	Velocity Change
β	Angle of Yaw
ΔT	Temperature Change
Δn	Load Factor Increment

NORMAL TECHNOLOGY GROWTH - TASK 1

This task consists of providing assessments of current technology and normal growth to 1986 in key system and subsystem technology areas as applied to advanced earth-orbital transportation systems. For this purpose it was first necessary to define the required systems together with their operational environments and performance requirements generated in the course of the configuration development activities of Task II.

Components and Subsystems

The assessment included all major vehicle components and subsystems as shown in the following subsystem list:

- Structure
  - Body
  - Wing
  - Elevon
  - Vertical Tail
  - Crew Compartment
- Main Propulsion
  - Main Engine
  - Propellant Delivery
  - Pressurization
  - Pressurization - Reentry
- Aux Propulsion
  - RCS
  - OMS
- Secondary Power Generation
- Elec. Pwr. Gen. & Distrib.
- Hyd. Pwr. Gen. & Distrib.
- Flight Control Systems
- Avionics
  - Guidance, Navigation & Control
  - Comm. and Track
  - Displays and Controls
  - Instrumentation
  - Data Process and S-W
- Landing Gear
- Environmental Control
- Crew Accommodations
- Launch and Recovery
- Payload Accommodations

This list is the result of a selection of appropriate key components, subsystems and technologies on the basis of design and operational requirements applicable to the four SSTO vehicle configurations studied under this contract.

System Weight Relationships

In order to determine the leverage of the various vehicle elements and subsystems, it was necessary to determine their weight relationship with respect to the overall vehicle systems.

Figure 1 uses the horizontal takeoff vehicle as an example to illustrate the various vehicle weight breakdowns. The most significant item of Gross Lift Off Weight (GLOW) shown in Figure 1a is the usable ascent propellant, which makes up 85% of the total. Several areas associated with performance show potential for reducing the propellant weight, which in turn reduces the structures weight and GLOW.

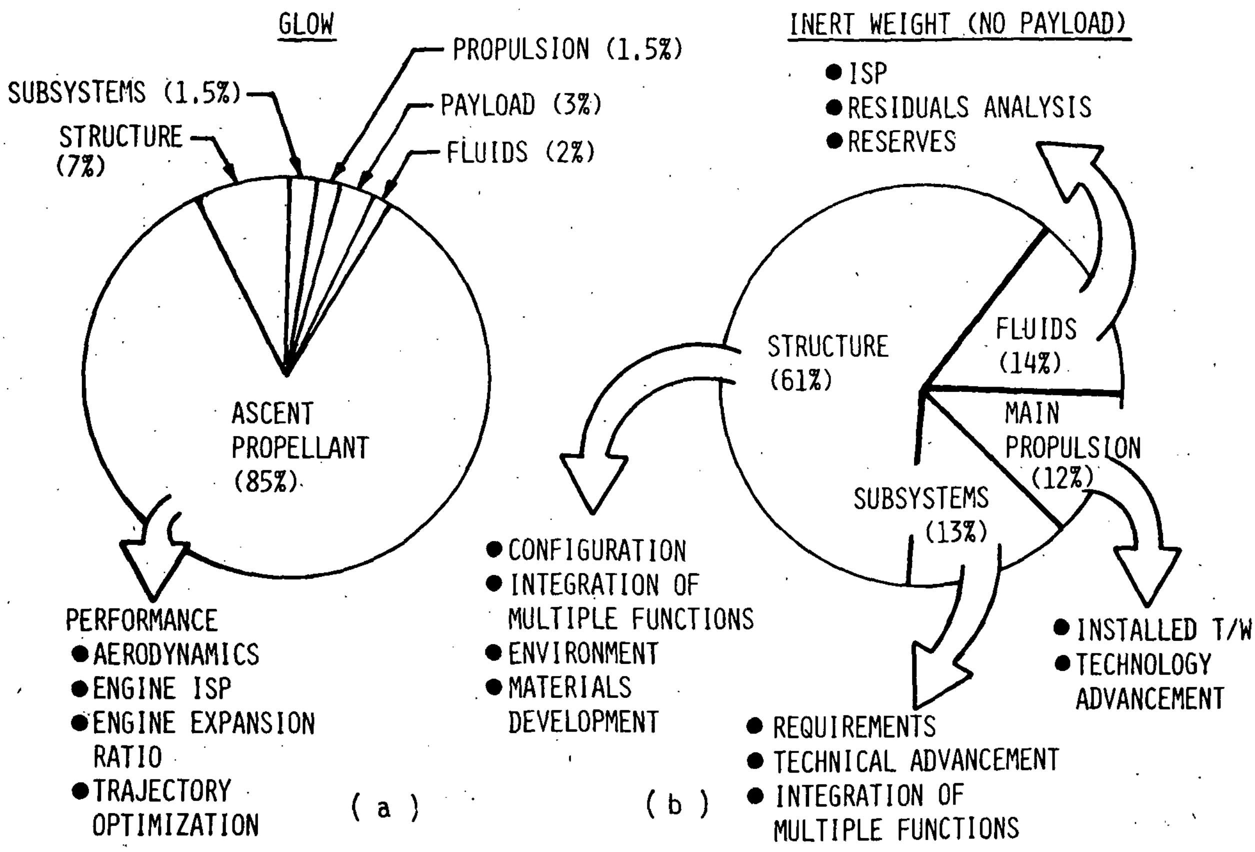


Figure 1 HTO Vehicle System Weight Distribution

Figure 1b details the breakout of vehicle inert weight to determine what elements are drivers. Structures is a key element at 61% of the total inert weight. Subsystems, main propulsion and fluids share nearly equally in making up the remaining weight.

A more detailed look at the horizontal takeoff vehicle shows how the various portions of the structure and subsystems make up the vehicle dry weight (Figure 2).

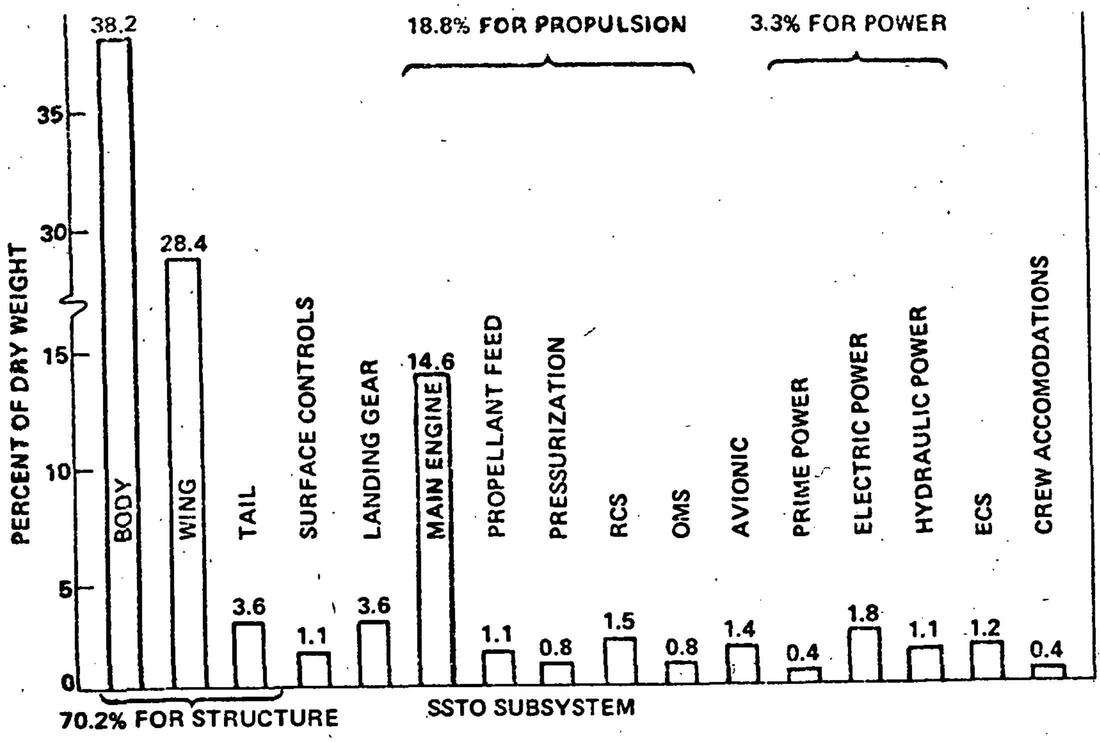


Figure 2 HTO Vehicle Dry Weight Distribution

Structure at 70.2% of dry weight remains very critical in the development of a single-stage-to-orbit system. The main engine at 14.6% of dry weight dominates the subsystem weight breakdown followed by the landing gear. Auxiliary propulsion subsystems by themselves are rather insignificant, but when combined as a total integrated propulsion system, make up 18.8% of vehicle dry weight. The power systems when combined make up 3.3% of the vehicle dry weight. In addition, the propulsion and power systems selection have a direct impact on the vehicle fluid weights.

Technology Projections

The 1986 technology projections thus reflect the results of detailed examinations of relative potential for advance in the various technology and subsystem areas as well as the leverage on vehicle performance that such advances provide. The 1986 technology projections are presented in a set of worksheets grouped by subsystems examples of which are shown in figures 3 through 21 .

The format selected for presentation of the study results includes in three separate columns a definition of the respective Single-Stage-to-Orbit vehicle (SSTO) requirements, the assessment of existing technology and technology projections for 1986.

SSTO Requirements: This column contains a brief definition of the purpose and function of the respective subsystem, a description of its elements, a definition of its operating environment as well as applicable remarks.

Existing Technology: This column contains a description of existing technology by subsystem element and the respective weight estimating rationale.

Projected Technology 1986: This column presents a brief description and the rationale for the respective technology projections including development programs required as a part of normal technology growth. Weight estimating relationships and rationale are also included.

Figures 3 through 6 show examples of worksheets dealing with the structure. The selected structural concept for all SSTO vehicle configurations uses a single structural system to serve functions which previously had required four separate systems: thermal protection, airframe, cryogenic tankage and cryogenic insulation thus significantly reducing structural weight. These figures also show that development effort is required in improving brazed aluminum and Rene'41 honeycomb, joining of Rene'41 and titanium materials and in improving metal matrix and other composite materials.

Figures 7 through 14 show examples of worksheets for the other SSTO components and subsystems.

Figures 15 through 21 are examples of technology projections for the landing gear and its elements and the impact of the design requirements on these elements. Figure 15 compares the design requirements for the various loading conditions, as specified by FAR 25; as applied to the Boeing Jet Transports; as specified for the SSTO studies; and finally, the recommendations to achieve a minimum weight landing gear. Figure 21 illustrates the continuing lower weight trends for landing gear. This effect is dominated by the use of higher strength materials, i.e. 1240 MPa (180 ksi) to 2210 MPa (320 ksi)

1986 TECHNOLOGY PROJECTION

NAME

SUBSYSTEM / TECHNOLOGY STRUCTURE - BODY/LH_2 TANK - UPPER SURFACE

SSTO REQUIREMENT	EXISTING TECHNOLOGY	PROJECTED TECHNOLOGY 1986
SUBSYSTEM PERFORMANCE REQUIREMENTS o CONTAIN LH_2 AT 0.1055 MPa o SUPPORT BODY LOADS: NOSE GEAR, LANDING, FLIGHT, THRUST, REDISTRIBUTE MAIN GEAR LANDING LOADS AND SUPPORT PAYLOAD BAY.	DESCRIPTION BY ELEMENT	DESCRIPTION AND RATIONALE
DESCRIPTION OF ELEMENTS o UPPER SURFACE ALUMINUM BRAZED TITANIUM H/C TITANIUM H/C PANELS WILL BE 30.48 mm DEEP THE OUTER SKIN WILL BE SLOTTED AT 0.254 m O.C. FOR .508m ONLY ABOVE THE RENE'41 - TITANIUM H/C JOINT. ABOVE THAT POINT THE OUTER SKIN WILL BE CONTINUOUS. FRAME JOINTS WILL BE SAME AS DISCUSSED INNER AND OUTER SKINS OF ADJACENT PANELS WILL BE WELDED ON ASSEMBLY.	o ALUMINUM BRAZED TITANIUM H/C TECHNOLOGY HAS BEEN DEVELOPED BY BOEING FOR THE DEPT. OF TRANSPORTATION FOR SST APPLICATION. o ALLOWABLES & PROCESSES ARE AVAILABLE. BASIC AL. BRAZED TI H/C ALLOWABLES ARE AVAILABLE. o BASIC TITANIUM WELD ALLOWABLES WERE DEVELOPED DURING THE SST PROGRAM AND ARE AVAILABLE. TECHNOLOGY IS AT THE FULL SCALE SUBASSEMBLY STAGE o RENE'41 TO TITANIUM JOINING FOR LH_2 APPLICATION MUST BE DEVELOPED.	o PROJECTED ALUMINUM BRAZED TITANIUM H/C PROPERTIES ARE AVAILABLE. o ADDITIONAL ALUMINUM BRAZED H/C COMPONENT DEVELOPMENT PROGRAMS ARE REQUIRED. WER AND RATIONALE o WEIGHT ESTIMATES REQUIRE DESIGN AND ANALYSIS USING THE ABOVE MENTIONED DEVELOPED STRUCTURAL ALLOWABLES.
ENVIRONMENT UPPER SURFACE o TEMPERATURE EQUILIBRIUM: 20.2K TO 738K INCLUDING INTERNAL RADIATION: 20.2K TO 650K REMARKS	WER o PRELIMINARY SIZING ANALYSIS	

Figure 3 Technology Projection - Upper Surface Body Structure

1986 TECHNOLOGY PROJECTION

NAME

SUBSYSTEM / TECHNOLOGY STRUCTURE - BODY LH_2 TANK - LOWER SURFACE

SSTO REQUIREMENT	EXISTING TECHNOLOGY	PROJECTED TECHNOLOGY 1986
SUBSYSTEM PERFORMANCE REQUIREMENTS o CONTAIN LH_2 AT 0.1055 MPa o SUPPORT BODY LOADS: NOSE GEAR, LANDING, FLIGHT, THRUST, REDISTRIBUTE MAIN GEAR LANDING LOADS AND SUPPORT PAYLOAD BAY.	DESCRIPTION BY ELEMENT	DESCRIPTION AND RATIONALE
DESCRIPTION OF ELEMENTS o LOWER SURFACE RENE' 41 BRAZED HONEYCOMB (H/C) RENE' 41 H/C PANELS WILL BE 30.48 mm DEEP. THE OUTER SKIN WILL BE SLOTTED AT 0.254m. O.C. TO RELIEVE ASCENT THERMAL STRESS. INTEGRAL FRAME WEBS WILL BE BUILT INTO THE INNER SKINS AT 0.762m O.C. FOR FRAME WEB ATTACHMENT. THE INNER SKINS OF ADJACENT PANELS WILL BE WELDED ON ASSEMBLY.	o RENE' 41 H/C BRAZING TECHNOLOGY IS IN BRAZE ALLOY SELECTION PHASE. CURRENT BASIC RENE'41 MATERIAL AND H/C PROPERTIES ARE AVAILABLE. INITIAL H/C ALLOWABLES DEVELOPMENT IS FUNDED BY NASA LANGLEY RESEARCH CONTRACT o RENE'41 WELDING ALLOWABLES AND PROCESSES WERE DEVELOPED ON THE X-20 PROGRAM FOR HOT STRUCTURE DESIGN. RENE'41 WELDED ALLOWABLES ARE AVAILABLE. RENE'41 WELD ALLOWABLES AND PROCESSES MUST BE ADAPTED TO H/C PANEL TO PANEL WELDING AND JOINING OF FRAME WEBS TO BASIC H/C PANEL. o ESTIMATED BRAZED RENE'41 H/C PROPERTIES ARE AVAILABLE. THESE ESTIMATES ARE BASED ON SIMILAR DATA OBTAINED ON PRECIPITATION HARDENED STAINLESS STEELS AND BASIC PROPERTY DATA. o RENE'41 TO TITANIUM JOINING FOR LH_2 APPLICATION MUST BE DEVELOPED.	o ADDITIONAL RENE'41 H/C ALLOWABLES AND COMPONENT DEVELOPMENT PROGRAMS ARE REQUIRED. WER AND RATIONALE o WEIGHT ESTIMATES REQUIRE DESIGN AND ANALYSIS USING THE ABOVE MENTIONED DEVELOPED STRUCTURAL ALLOWABLES.
ENVIRONMENT LOWER SURFACE o TEMPERATURE EQUILIBRIUM: 20.2K TO 1210K INCLUDING INTERNAL RADIATION: 20.2K TO 1155K o MAXIMUM INNER/OUTER SURFACE THERMAL GRADIENT (ΔT) = 836K ON ASCENT REMARKS	WER o PRELIMINARY SIZING ANALYSIS	

Figure 4 Technology Projection - Lower Surface Body

1986 TECHNOLOGY PROJECTION

NAME

SUBSYSTEM / TECHNOLOGY STRUCTURES - BODY/LH_2 TANK INTERNAL STRUTS AND FRAMES

SSTO REQUIREMENT	EXISTING TECHNOLOGY	PROJECTED TECHNOLOGY 1986
SUBSYSTEM PERFORMANCE REQUIREMENTS	DESCRIPTION BY ELEMENT	DESCRIPTION AND RATIONALE
°STRUTS AND FRAMES STABILIZE AND SUPPORT BODY CONTOUR FOR FUSELAGE BENDING AND HOLD BODY CONTOUR DURING LH_2 PRESSURIZATION.		
DESCRIPTION OF ELEMENTS °STRUTS - BOLTED TO TOP AND BOTTOM FRAMES, 6-2-4-2-Ti TUBES WITH SWAGED ENDS AND WELDED END FITTINGS. °ALTERNATE - BORON ALUMINUM TUBES WITH SWAGED 6-2-4-2-Ti ENDS AND WELDED END FITTINGS. °FRAMES - LOWER SURFACE RENE'41 - INTREGRAL WEBS AND INNER FLANGE BOLTED OR WELDED TO NIBS PROTRUDING FROM SANDWICH PANELS UPPER SURFACE 6-2-4-2-Ti - ATTACHED AS ABOVE RENE'41	°THE DEVELOPMENT OF TITANIUM MATERIAL AND COMPONENT ALLOWABLES AND ALL REQUIRED MANUFACTURING PROCESSES WAS COMPLETED ON 6AL-4V-Ti ALLOY ON THE SST BY BOEING FOR THE DEPT. OF TRANSPORTATION. °6-2-4-2-Ti TEMPERATURE LIMIT IS 811K °MODEST DEVELOPMENT IS REQUIRED TO BRING THIS ALLOY TO THE EQUIVALENT STATUS OF 6AL-4V-Ti. °THE 6-2-4-2-Ti DEVELOPMENT STAGE IS AT THE LEVEL OF LOADS AND THERMAL TEST OF A FULL SCALE COMPONENT. °6-2-4-2-Ti MATERIAL PROPERTIES ARE AVAILABLE. °BORON ALUMINUM TECHNOLOGY DEVELOPMENT IS TO THE ASSEMBLY DEVELOPMENT STAGE. BORON ALUMINUM PROPERTIES ARE AVAILABLE. °RENE'41 TECHNOLOGY - CURRENT BASIC RENE'41 MATERIAL PROPERTIES ARE AVAILABLE. °RENE'41 WELDING ALLOWABLES AND PROCESSES WERE DEVELOPED ON THE X-20 PROGRAM FOR HOT STRUCTURE DESIGN. RENE'41 WELDED ALLOWABLES ARE AVAILABLE. RENE'41 WELD ALLOWABLES AND PROCESSES MUST BE ADAPTED TO JOINING OF FRAME WEBS TO H/C PANEL. °RENE'41 TO TITANIUM JOINING FOR LH_2 APPLICATION MUST BE DEVELOPED.	°PROJECTED TITANIUM PROPERTIES ARE AVAILABLE. °ADDITIONAL REQUIRED DEVELOPMENT PROGRAMS FOR A SKIN STIFFENED TITANIUM STRUCTURAL SYSTEM ARE AVAILABLE. °COMPOSITE MATERIAL STRUTS MAY UTILIZE VARIOUS COMBINATIONS OF MATERIALS SUCH AS BORON FILAMENTS IN A TITANIUM MATRIX TO ACHIEVE HIGHER TEMPERATURE USAGE AND/OR IMPROVED STRENGTH TO WEIGHT RATIOS. °PROJECTED DESIGN PROPERTIES FOR COMPOSITE MATERIALS ARE AVAILABLE. °REQUIRED COMPOSITE MATERIALS STRUCTURAL DEVELOPMENT PROGRAMS ARE AVAILABLE. °ADDITIONAL REQUIRED RENE'41 H/C ALLOWABLES AND COMPONENT DEVELOPMENT PROGRAMS ARE AVAILABLE. WER AND RATIONALE °WEIGHT ESTIMATES REQUIRE DESIGN AND ANALYSIS USING THE ABOVE MENTIONED DEVELOPED STRUCTURAL ALLOWABLES.
ENVIRONMENT TEMPERATURE °RENE'41: ~ 922K °6-2-4-2-Ti: 811K °ALUMINUM BORON: 700K ALUMINUM BORON AND TITANIUM TEMPERATURES ARE CONTROLLED BY MATERIAL THICKNESS TO MAXIMUM TEMPERATURES NOTED:	WER °PRELIMINARY SIZING ANALYSIS	

Figure 5 Technology Projection - Internal Body Structure

1986 TECHNOLOGY PROJECTION

NAME

SUBSYSTEM / TECHNOLOGY STRUCTURE - VERTICAL TAIL

SSTO REQUIREMENT	EXISTING TECHNOLOGY	PROJECTED TECHNOLOGY 1986
SUBSYSTEM PERFORMANCE REQUIREMENTS	DESCRIPTION BY ELEMENT	DESCRIPTION AND RATIONALE
o VERTICAL TAIL IS REQUIRED FOR DIRECTIONAL STABILITY.		
DESCRIPTION OF ELEMENTS		
o ALUMINUM BRAZED TITANIUM H/C WITH MULTIPLE RIBS COMPOSED OF TITANIUM STRUTS AND/OR WEBS o ALTERNATE STRUCTURAL SYSTEM: PRIMARY STRUCTURE SKIN STIFFENED TITANIUM AND MULTIPLE TITANIUM RIBS RUDDER ALUMINUM BRAZED TITANIUM H/C	o ALUMINUM BRAZED TITANIUM H/C TECHNOLOGY HAS BEEN DEVELOPED BY BOEING FOR THE DEPT. OF TRANSPORTATION FOR SST APPLICATION. o ALLOWABLES AND PROCESSES ARE AVAILABLE. BASIC ALUMINUM BRAZED Ti H/C ALLOWABLES ARE AVAILABLE. o THE DEVELOPMENT OF TITANIUM MATERIAL AND COMPONENT ALLOWABLES AND ALL REQUIRED MANUFACTURING PROCESSES WAS COMPLETED ON 6AL-4V-Ti ALLOY ON THE SST BY BOEING FOR THE DEPT. OF TRANSPORTATION. o 6-2-4-2-Ti TEMPERATURE LIMIT IS 811K. o MODEST DEVELOPMENT IS REQUIRED TO BRING THIS ALLOY TO THE EQUIVALENT STATUS OF 6AL-4V-Ti. o THE 6-2-4-2-Ti DEVELOPMENT STAGE IS AT THE LEVEL OF LOADS AND THERMAL TEST OF A FULL SCALE FIN STRUCTURAL BOX. o 6-2-4-2-Ti MATERIAL PROPERTIES ARE AVAILABLE.	o PROJECTED ALUMINUM BRAZED TITANIUM H/C PROPERTIES ARE AVAILABLE. o ADDITIONAL ALUMINUM BRAZED H/C COMPONENT DEVELOPMENT IS REQUIRED. o PROJECTED TITANIUM PROPERTIES ARE AVAILABLE. o ADDITIONAL DEVELOPMENT PROGRAMS FOR A SKIN STIFFENED TITANIUM STRUCTURAL SYSTEM ARE REQUIRED. WER AND RATIONALE o WEIGHT ESTIMATES REQUIRE DESIGN AND ANALYSIS USING THE ABOVE MENTIONED DEVELOPED STRUCTURAL ALLOWABLES.
ENVIRONMENT TEMPERATURE o ALUMINUM BRAZED TITANIUM H/C MAXIMUM. TEMPERATURE 700K o SKIN STIFFENED TITANIUM MAXIMUM TEMPERATURE 811K	WER o PRELIMINARY SIZING ANALYSIS	
REMARKS o ALUMINUM BRAZED TITANIUM H/C IS LIMITED TO 700K FOR SERVICE LIFE. o TITANIUM ALLOY IS LIMITED TO 811K SERVICE LIFE. A GREATER EXTENSION OF HIGH TEMPERATURE MATERIALS AFT OF THE LEADING EDGE IS REQUIRED FOR H/C THAN FOR THE SKIN STIFFENED DESIGN		

Figure 6 Technology Projection - Vertical Tail Structure

1986 TECHNOLOGY PROJECTION

NAME

SUBSYSTEM / TECHNOLOGY LANDING GEAR

SSTO REQUIREMENT	EXISTING TECHNOLOGY	PROJECTED TECHNOLOGY 1986
SUBSYSTEM PERFORMANCE REQUIREMENTS o LANDED WT. 119,750 kg o LANDING SPEED 84.9 m/s o α AT LANDING 15° MAX o SINK SPEED 3.05 m/s o CROSS WIND 12.9 m/s o STOPPING DISTANCE LESS THAN 1829m o MUST STEER & TURN OFF ON ROLL OUT o EXTEND TIME - 10 SECS MAX	DESCRIPTION BY ELEMENT a - 2069 MPa STEEL AIR/OIL STRUT WITH INTEGRAL DAMPENING - 27.6MPa MAX. AIR PRESSURE b - 414-552 MPa ALUMINUM WITH TEFLON FABRIC BEARINGS c - 2069 MPa STEEL - 27.6MPa INTERNAL LOCKING ACTUATOR d - 414-552 MPa ALUMINUM WITH TEFLON FABRIC BEARINGS e - SEE a f - R_c65 BALL & ROLLER CAGED g - ALUMINUM AND MAGNESIUM CASTINGS h - LOW ASPECT RATIO - LIMITED BY RUNWAY CAPABILITY TO APPROX. 2.07 MPa FOOTPRINT PRESSURE i - CARBON-BERYLLIUM HEAT SINK j - HYDRAULIC - 27.6 MPa INTEGRAL DAMPENING & VALVING	DESCRIPTION AND RATIONALE a - 1. 2413 MPa MARAGING STEEL 2. BORON/ALUMINUM COMPOSITES b - BORON/ALUMINUM COMPOSITES c - REDUCED SIZE DUE TO INCREASED PRESSURE d - BORON/ALUMINUM COMPOSITES e - SEE a 1. and a.2. f - NO SIGNIFICANT CHANGE g - LOW BRAKE ENERGY REQUIREMENTS OF VEHICLE PERMIT RELATIVELY LOWER WEIGHT WHEELS. h - NO SIGNIFICANT CHANGE LIMITED BY RUNWAY i - NO SIGNIFICANT CHANGE j - REDUCED SIZE WITH INCREASED PRESSURE
DESCRIPTION OF ELEMENTS a - SHOCK DAMPER STRUT b - DRAG STRUT & BRACES c - EXTEND ACTUATORS d - TORSION LINKS e - AXLES f - BEARINGS g - WHEELS h - TIRES i - BRAKES AND ANTI-SKID j - NOSE GEAR STEERING		WER AND RATIONALE a - 2.018×10^{-3} kg OF STRUT/kg OF VEHICLE & STRUT LENGTH m (6.15×10^{-4} LB OF STRUT/LB OF VEHICLE & STRUT LENGTH b - 18% WT. REDUCTION DUE TO BORON/ALUMINUM WT = STRUT WT c - 7% REDUCTION DUE TO 27.6MPa WT = .4 X TIRE WT d - SEE b e - SEE b f - SEE b g - WT = .46 BRAKE WT OR .79 TIRE WT h - 1.2328×10^{-3} kg/kg OF VEHICLE & MPa (8.5×10^{-6} LB/LB of VEHICLE & psi) i - 1.64N/l X 10^{-5} Nm OF ENERGY (5 LB/l X 10^{6} FT LB OF ENERGY) j - 3.2% = CONTROLS TOTAL
ENVIRONMENT o 219 K TO 366K o AMBIENT PRESS 6.9 KN to 101 KN (0 PSIA TO 14.7 PSIA) REMARKS o 500 LANDING LIFE ON ALL ELEMENTS EXCEPT - TIRES 10 LANDINGS - BRAKES 10 LANDINGS	WER a - kg/m EXT. STRUT - kg LOAD & % OF TOTAL & FACTOR OF BRAKE & TIRE b - % OF STRUT WT. c - % OF TOTAL WT. d - % OF STRUT WT. e - FACTOR OF WHEELS & TIRES WEIGHT f - % g & h g - % OF BRAKE OR TIRE h - % MAX LANDING WEIGHT i - EMPIRICAL FACTOR SPEED WT & DISTANCE j - % GEAR WEIGHT & STEERING ANGLES	

Figure 7 Technology Projection - Landing Gear

1986 TECHNOLOGY PROJECTION

SUBSYSTEM / TECHNOLOGY PROPULSION MAIN ENGINE

NAME

SSTO REQUIREMENT	EXISTING TECHNOLOGY	PROJECTED TECHNOLOGY 1986
SUBSYSTEM PERFORMANCE REQUIREMENTS PROVIDE THE PROPULSIVE THRUST FOR INSERTION TO THE REFERENCE 92.7 X 185.3 km (50 X 100 n.mi) ORBIT. MUST BE THROTTLABLE. PROVIDE THRUST VECTOR CONTROL FOR PITCH, YAW, AND ROLL	**DESCRIPTION BY ELEMENT** SSME - STAGED COMBUSTION, HIGH CHAMBER PRESSURE, MODERATELY LOW NPSH, FIXED BELL NOZZLE LO_2/LH_2 ENGINE. 20.7 MPa (3000 psi) = P_c - NPSH = 13.8 kPa (2 psi) LH_2 55.2 kPa (8 psi) LO_2. TWO POSITION NOZZLES - LIGHT WEIGHT ARE NONREUSABLE - REUSABLE INVOLVE SIGNIFICANT WEIGHT PENALTY ZERO NPSH IN LOW FLOW RATE, LOW HEAD PUMPS, BUT TECHNOLOGY EXISTS FOR HIGH FLOW RATES. AERO SPIKE ENGINES FOR J-2 TURBO MACHINERY PERFORMANCE HYDRAULIC ACTUATOR FOR GIMBAL OF $\pm 9^\circ$ IN EACH OF 2 AXES	**DESCRIPTION AND RATIONALE** STAGED COMBUSTION CYCLE LIMITS WITH H_2 COOLING P_c = 27.6 MPa (4000 psi) MR = 6:1 $\pm$ 1 THRUSTS TO 4.45 MN (1 MILLION POUNDS) REDUCED WEIGHT PENALTY FOR REUSABLE TWO-POSITION NOZZLES RESULTING FROM DEVELOPMENT WORK FOR SRM'S. ZERO NPSH PUMPS DEMONSTRATED TECHNOLOGY ON CONTRACT NAS 8-29189
DESCRIPTION OF ELEMENTS COMBUSTION CHAMBER NOZZLE PUMPS ACTUATORS		
ENVIRONMENT	**WER** $W = \frac{T}{31.47}$ - WET ENGINE WEIGHT - SSME $W = \frac{T}{442.25}$ - ACTUATOR WEIGHT ENGINE - SHUTTLE	**WER AND RATIONALE** ROCKETDYNE DATA -
REMARKS LINEAR ENGINE MUST BE CONSIDERED A CANDIDATE IF TVC CAN BE WORKED OUT.		

Figure 8 Technology Projection - Propulsion - Main Engine

1986 TECHNOLOGY PROJECTION

NAME

SUBSYSTEM / TECHNOLOGY HYDRAULIC POWER GENERATION AND DISTRIBUTION

SSTO REQUIREMENT	EXISTING TECHNOLOGY	PROJECTED TECHNOLOGY 1986
SUBSYSTEM PERFORMANCE REQUIREMENTS PROVIDE POWER FOR: - FLIGHT CONTROL SURFACE ACTUATORS - THRUST VECTOR CONTROL - LANDING GEAR EXTEND STEER AND BRAKE - ENVIRONMENTAL CONTROL FANS & PUMPS - PAYLOAD BAY DOORS AND MANIPULATORS	DESCRIPTION BY ELEMENT a) 4000 PSI VARIABLE DISPLACEMENT, CAST ALUM OR STNLS STEEL, SOLENOID OPERATED UNLOADING (LOW PRESSURE) CONTROL VALVE b) PRESSURIZED BELLOWS OPERATED PISTON WITH ENTRAINED AIR REMOVER c) BELLOWS OPERATED PISTONS WITH INTERNAL DAMPENING d) HIGH HARDNESS R_C70 ABRASION RESISTANT LOW LEAKAGE VALVES, METAL MESH FILTERS e) TITANIUM & STAINLESS STEEL WITH BRAZED OR WELDED CONNECTIONS f) MODIFIED PETROLEUM BASE OR SYNTHETIC POLYESTER FLUIDS COMPOUNDED TO ENHANCE PARTICULAR QUALITIES g) STNLESS STL/OIL TO COLD PLATE TO CRYOGEN OR WATER BOILER h) ALUMINUM, TITANIUM, OR STAINLESS STEEL BODIES, STEEL-PLATED RODS, TEFLON FABRIC BEARINGS	DESCRIPTION AND RATIONALE a) INCREASE PRESSURE TO 34.5 TO 41.4 MPa (5-6000 PSI) b) INCORPORATE COMPOSITES - NO SIGNIFICANT CHANGE c) NO SIGNIFICANT CHANGE d) NO SIGNIFICANT CHANGE e) COMPOSITES OFFER WEIGHT REDUCTION f) NO SIGNIFICANT CHANGE g) NO SIGNIFICANT CHANGE h) REDUCTION DUE TO REDUCED SIZE WITH INCREASED PRESSURE
DESCRIPTION OF ELEMENTS a) PUMPS b) RESERVOIRS c) ACCUMULATORS d) VALVES & FILTERS e) PLUMBING f) FLUID g) HEAT EXCHANGES h) ACTUATORS	WER a) kg/kw, kg/GPM-RPM, LB/IN^3 kW/IN^3 b) kg/m^3 CONTAINED VOLUME - % SYSTEM f(PRESS) c) kg/m ^{3} CONTAINED VOLUME =/m kg d) % SYSTEM, kg/SYSTEM kW e) % SYSTEM, kg/kWm f) % SYSTEM, kg/SYSTEM kW g) kg/kW h) kg/m kg, kg/kg -m · kg/HM (m kg) - ANGLE	WER AND RATIONALE a) .54 $\frac{kg}{k2}$ (.45 LB/HP) @ 20.7 MPa (3000 psi) & 3200 RPM, 10% TO 20% SYSTEM WT. b) .12 to .3 kg/SYSTEM kW (.2 LB TO .5 LB/SYSTEM HP) c) .12 to .55 kg/SYSTEM kW (.2 LB TO .9 LB/SYSTEM HP), 5% TO 15% SYSTEM WT. d) 20% to 50% PUMP WT., 2% TO 7% SYSTEM WT., .12 kg/kW (.2 LB/HP) e) 25% to 40% SYSTEM WT., .91 TO 1.52 kg/SYSTEM kW (1.5 TO 2.5 LB/SYSTEM HP) 1.2 $\frac{kg}{kWm}$ TO .06 $\frac{kg}{kWm}$ (.60 LB/HP FT to .03 LB/HP FT) AT 52.2 kW and 61m (70 HP & 200 FT) f) 15% to 20% SYSTEM WT. .45 kg/SYSTEM kW (1 LB/SYSTEM HP) g) 1.1 kg/SYSTEM kW (1.82 LB/SYSTEM HP) $\frac{.36\ MJ\ HR}{SYSTEM\ kW\ HR}$ (254.64 BTU HRS/SYSTEM HP HR.) 6.88×10^7 $\frac{kg\ hr}{J}$ (1.6×10^{-3} LB/BTU/HR) h) .17 kg/kW (.28 LB/HP) MOTORS 1 kg/152.4 mkg (1 LB/6 x 10^3 LB IN) ACTUATOR SERVO ACTUATOR LB = .016 $\frac{mkg}{MPa}$ $\left[\left(\frac{HM\theta}{104.18P}\right)\left(\frac{LB/IN^0}{psi}\right)\right]$ HM, kg. m θ, ° OF TRAVEL P, MPa (psi) HYDRAULIC PRESSURE
ENVIRONMENT AMBIENT PRESS		
REMARKS 6000 HRS LIFE 500 FLIGHTS 733 HRS AT FULL PRESSURE.		1) OPTIMUM PRESSURE FOR A HYDRAULIC SYSTEM, CONRAD COCKE, MARTIN CO., 1956 2) SELECTION OF AIRBORNE AUXILIARY POWER TRANSMISSION SYSTEMS, VICKERS INC., 1958 3) HYDRAULIC TRANSMISSION LINE WEIGHT AS A FUNCTION OF SYSTEM OPERATING PRESSURE, E. T. RAYMOND, BOEING, 1957 4) EFFECTS OF WORKING PRESSURE ON AIRCRAFT HYDRAULIC SYSTEMS, R. H. WALSH, THE AERONAUTICAL JOURNAL OF THE ROYAL AERONAUTICAL SOCIETY, MARCH, 1970

Figure 9 Technology Projection - Hydraulic Power

NAME

1986 TECHNOLOGY PROJECTION

SUBSYSTEM / TECHNOLOGY SECONDARY POWER GENERATION

SSTO REQUIREMENT	EXISTING TECHNOLOGY	PROJECTED TECHNOLOGY 1986
SUBSYSTEM PERFORMANCE REQUIREMENTS HYDRAULIC POWER o TVC DURING ASCENT o FLT CONTROL - ASCENT & DESCENT o CIRCULATION ON ORBIT o P/L DOORS - DEPLOYMENT MECHANISMS TOTAL LOAD 2.94 kW 40.4 k W- HR ELECTRICAL POWER o AVIONICS o MISCELLANEOUS TOTAL LOAD 4KW/50KW HR DESCRIPTION OF ELEMENTS a) ENERGY CONVERSION DEVICE MECHANICAL POWER b) ENERGY CONVERSION DEVICE FOR ELECTRICAL POWER c) ENERGY STORAGE	DESCRIPTION BY ELEMENT MECHANICAL a) CONVERSION ELEMENTS IN COMMON USE TODAY INCLUDE: 1) DYNAMIC CYCLES (TURBINE) 2) ELECTRIC MOTOR 3) FLYWHEELS b) ELECTRICAL CONVERSION ELEMENTS IN COMMON USE TODAY INCLUDE: 1) SOLARY ARRAYS 2) FUEL CELLS 3) TURBINE DRIVEN ALTERNATOR 4) RTG 5) BATTERIES 6) FLY WHEELS c) SEPARATE ENERGY STORAGE DEVICES ARE ASSOCIATED WITH CERTAIN PARTICULAR CONVERSION ELEMENT 1) TURBINES a) CRYOGENIC b) STORAGE N2H4 2) FOR FUEL CELLS - CRYOGENIC LH2/LO2	DESCRIPTION AND RATIONALE MAJORITY OF SYSTEMS REQUIRE DIRECTED TECHNOLOGY EFFORT. MOST PROMISING CANDIDATE(H2/O2 APU) SHOULD BE OPERATIONAL BY 1985 WITH PRESENT EFFORT OF INDUSTRY SUPPLEMENTED BY FUNDING VIA NASA/LEWIS IN SUPPORT OF PRESENT ENVIRONMENTAL/ENERGY PUSH. a) TURBINE APU DRIVING HYDRAULIC PUMPS VIA GEAR BOX b) ALTERNATOR DRIVEN FROM APU VIA GEAR BOX c) VARIABLE QUALITY LH2/LO2 STORED IN METAL TANKS AT SUB-CRITICAL PRESSURES. SCREEN ACQUIRED LH2 AT LOW PRESSURES .34 M Pa ($\leq$ 50 psi) WITH PUMP AND LIQUID ACCUMULATOR. SCREEN ACQUIRED LO2 AT HIGH PRESSURE.
ENVIRONMENT 500 REUSES 6000 HOURS REMARKS	WER a) MECHANICAL CONVERSION 1) SHUTTLE TURBINE N2H4 .43 $\frac{kg}{kw}$ 2) ELECTRIC MOTOR 3) FLYWHEELS (IN DEVELOPMENT) b) ELECTRICAL CONVERSION 1) SOLAR ARRAYS 22 W/kg 2) FUEL CELLS 33 W/kg 3) SEE ELECTRICAL POWER 4) RTG 3.8 W/kg 5) BATTERIES 6) FLYWHEELS (IN DEVELOPMENT) c) STORAGE 1) N2H4 STORAGE (SHUTTLE) 4.32 $\frac{kg}{KW}$ 2) SHUTTLE FUEL CELLS	WER AND RATIONALE NASA LEWIS STUDIED, VIA A GARRETT CONTRACT, H2/O2 APU'S FOR SHUTTLE. WTS FOR THE APU ARE BASED ON THAT STUDY. WTS FOR PROPELLANT SYSTEM BASED ON A SIMILAR STUDY FOR LIQUID H2O2 RCS SYSTEM BY MDAC FOR LEWIS. a) 0.6 LBS/HP FOR APU (FOR PUMPS SEE HYDRAULIC POWER) b) SEE ELECTRICAL POWER FOR ALTERNATOR WEIGHTS c) 1.86 $\frac{kg}{KW - HR}$ (3.06 LB/HP-HR) (2.0 IS H2O2, 1.06 IS TANKAGE PRESSURIZATION AND FEED).

Figure 10 Technology Projection - Secondary Power

1986 TECHNOLOGY PROJECTION

NAME

SUBSYSTEM / TECHNOLOGY ELECTRICAL POWER GENERATION & DISTRIBUTION

SSTO REQUIREMENT

SUBSYSTEM PERFORMANCE REQUIREMENTS

GENERATE, DISTRIBUTE CONTROL AND CONVERT KW OF ELECTRICAL POWER REQUIRED FOR SSTO:

- AVIONICS
- SUBSYSTEM CONTROLS
- HEATERS
- LIGHTING
- SUBSYSTEM ACTUATORS (SQUIBS)

DESCRIPTION OF ELEMENTS

- ALTERNATOR/INVERTER
- DC & AC BUSSES
- DISTRIBUTION BOXES
- RELAYS
- REGULATORS
- CABLES
- CONNECTOR
- UMBILICALS
- PROTECTIVE DEVICES

ENVIRONMENT
275K – 336K (35°F - 145°F)

REMARKS

500 MISSIONS
6000 HR MAXIMUM

EXISTING TECHNOLOGY

DESCRIPTION BY ELEMENT

ALTERNATOR - OIL COOLER ALTERNATORS 115 3 ϕ 400 Hz
EFFICIENCY = 80% MAXIMUM

INVERTER - 24 - 32 UDC INPUT/115V 3 ϕ 400 Hz
EFFICIENCY = 80% MINIMUM - SOLID STATIC

DISTRIBUTION BOXES - SPACE SHUTTLE ORBITER
RELAYS - D.C. POWER
REGULATORS - 28 UDC NOMINAL, TWO WIDE, STRUCTURAL GROUND
CABLES - A.C. POWER
CONNECTOR - 400 Hz, 115/200V, 3 ϕ, 4 WIRE WYE CONNECTED WITH STRUCTURAL GROUND
UMBILICALS - WIRING
KAPTAN OR TEFLON DEPENDENT UPON IMPEDANCE REQUIREMENTS

PROTECTIVE ELEMENTS- VARIOUS OVERLOAD DEVICES INCLUDE DIODES, FUSES, CIRCUIT BREAKERS, REMOTE POWER CONTROLLERS, AND REMOTE POWER CONTROLLER CIRCUIT BREAKERS. ALSO INCLUDED ARE OVER AND UNDERVOLTAGE DEVICES FOR AC & DC.

WER (BASED ON SPACE SHUTTLE)

CONTROL UNITS	416.4	(918.0)
DISTRIBUTION EQUIPMENT	246.8	(544.0)
LIGHTING SYSTEM	67.1	(148.0)
ELEC. EQUIP. SUPPORTS	125.2	(276.0)
INSTALLATION	337.9	(745.0)
TOTAL ELEC. PWRT. CONV. & DIST.	1193.4	(2631)
ELECTRICAL CABLING	1191.5	(2626.8)
ELECTRICAL CONNECTORS	157.0	(346.1)
SUPPORTS/INSTALLATION	497.6	(1097.1)
TOTAL CIRCUITRY	1846.1	(4070)
TOTAL ELECTRICAL SYSTEM	3039.5 kg	(6701 LB)

PROJECTED TECHNOLOGY 1986

DESCRIPTION AND RATIONALE

SYSTEM SELECTED IS A.C. POWER SYSTEM PROVIDED BY 115V. 3 ϕ 400 Hz ALTERNATORS. D.C. POWER PROVIDED BY BATTERIES AND TRANSFORMER RECTIFIER UNITS. SOLID STATE POWER CONDITIONING EQUIPMENT. SYSTEM SELECTION BASED ON WEIGHT SAVINGS OF A.C. POWER SYSTEM IN DISTRIBUTION EQUIPMENT AND CIRCUITRY OVER THAT OF D.C. SYSTEM.

SPACE SHUTTLE DRIVEN TO D.C. POWER SYSTEM OF FUEL CELLS WITH INVERTERS FOR POWER CONDITIONING. FUEL CELL SELECTION BASED ON MISSION DURATIONS UP TO 28 DAYS.

WER AND RATIONALE

REQUIREMENTS

AVIONICS IMPROVEMENTS IN SPACEBORNE COMPUTER PERFORMANCE, MASS & BUBBLE MEMORIES (SEE AVIONICS PROJECTIONS) REDUCES OVERALL POWER LEVEL DIRECTLY PROPORTIONAL TO WEIGHT (SEE JPL "FORECAST OF SPACE TECHNOLOGY" PG. 3-87, 3-88)

$\frac{2880 \text{ LB}}{5411 \text{ LB}} = .5322$

POWER SYSTEM CIRCUITRY SHALL BE REDUCED BY SAME PROPORTION + WIRE WEIGHT SAVINGS FOR A.C. POWER

TECHNOLOGY

SOLID STATE POWER CONDITIONING EQUIPMENT COULD ACHIEVE A SPECIFIC MASS IMPROVEMENT OF 21% BY 1980 (SEE JPL "FORECAST OF SPACE TECHNOLOGY" PG 1-42)
THIS FORECAST INCLUDES SHIELD AGAINST EMI AND REDUNDANCY TO ENHANCE RELIABILITY. AN ALLOWANCE FOR WIRING AND CABLING COMMON TO A POWER DISTRIBUTION SYSTEM IS ALSO INCLUDED.

Figure 11 Technology Projection - Electrical Power

NAME

1986 TECHNOLOGY PROJECTION

SUBSYSTEM / TECHNOLOGY AUXILIARY PROPULSION - REACTION CONTROL SYSTEM

SSTO REQUIREMENT	EXISTING TECHNOLOGY	PROJECTED TECHNOLOGY 1986
SUBSYSTEM PERFORMANCE REQUIREMENTS PROVIDE PITCH, YAW, AND ROLL CONTROL DURING ORBIT AND RE-ENTRY OPERATIONS. SIZE FOR 30.5 m/s (100 ft/sec) CAPABILITY WHICH INCLUDES RESERVES.	DESCRIPTION BY ELEMENT N_2O_4/MMH PRESSURE FED THRUSTERS WITH 289 SEC. SPECIFIC IMPULSE PERFORMANCE IN PULSING MODE. ALUMINUM FOIL LINED KEVLAR FILAMENT-WOUND TANKAGE FOR PRESSURANT AND PROPELLANTS. TUBING AND VALVES FOR PROPELLANT DELIVERY AND CONTROL.	DESCRIPTION AND RATIONALE LO_2/LH_2 - PRESSURE-FED THRUSTERS WITH 380 TO 400 SEC. SPECIFIC IMPULSE *PERFORMANCE IN PULSING MODE. FEASIBILITY OF PULSING MODE* OPERATION DEMONSTRATED WITH SPECIFIC IMPULSES OF 250 TO 350 SEC. MAY OR MAY NOT PROVE OUT AT REQUIRED PERFORMANCE. TITANIUM TANKAGE AND ACTUATORS.
DESCRIPTION OF ELEMENTS THRUSTERS TANKAGE TUBING FOR PROPELLANT DELIVERY		WER AND RATIONALE LO_2/LH_2 - 30.5 m/s (100 ft/sec) INC. RESERVES BASED ON MDAC APS STUDY ADJUSTED FOR MODULAR AND PUMPED LO_2. $W = 816.5 + .0047\ W_{REENTRY}$
ENVIRONMENT	WER N_2O_4/MMH - 30.5 m/s (100 ft/sec) INC. RESERVES BASED ON MDAC APS STUDY ADJUSTED BY SHUTTLE WEIGHT. $W = 554.3 + .0058\ W_{RE-ENTRY}$	
REMARKS N_2O_4/MMH IS STATE-OF-THE-ART SYSTEM. LO_2/LH_2 IS FEASIBLE BUT REQUIRES FURTHER DEVELOPMENT. HOWEVER, BASED ON TOTAL IMPULSE REQUIREMENTS, N_2O_4/MMH IS SOMETIMES A COMPARABLE (WEIGHT EFFECTIVE) SYSTEM. FOR THE HORIZONTAL TAKE-OFF VEHICLE, THIS IS THE CASE.		

Figure 12 Technology Projection - Auxiliary Propulsion

1986 TECHNOLOGY PROJECTION

NAME

SUBSYSTEM / TECHNOLOGY AVIONICS - GUIDANCE, NAVIGATION, AND CONTROL SYSTEM

SSTO REQUIREMENT	EXISTING TECHNOLOGY	PROJECTED TECHNOLOGY 1986
SUBSYSTEM PERFORMANCE REQUIREMENTS THE GN&C SYSTEM PROVIDES GUIDANCE, NAVIGATION, AND CONTROL FOR ALL FLIGHT PHASES WITH BOTH AUTOMATIC (COMMAND) AND MANUAL (CONTROL STICK STEERING) MODES. INJECTION ACCURACIES: POSITIVE < 37 km (20 n.mi) RMS VELOCITY < 3 m/s (10 fps) RMS RCS ATTITUDE ACCURACIES 0.5 TO 1.0 DEGREE, EACH AXIS DESCRIPTION OF ELEMENTS THE GN&C SYSTEM INTERFACES WITH THE MAIN ENGINES AND THE OMS FOR TVC AND WITH THE RCS & CONTROL SURFACES FOR ATTITUDE CONTROL. RATE GYROS AND ACCELEROMETERS PROVIDE SENSING FOR DAMPING AND LOAD RELIEF. INERTIAL MEASUREMENT UNIT PROVIDES ATTITUDE INFORMATION. AIR DATA PROVIDED BY VEHICLE NOSE PORTS AND PROBES.	DESCRIPTION BY ELEMENT GIMBAL PLATFORM, AND STRAPDOWN INERTIAL MEASUREMENT UNITS USED IN CONJUNCTION WITH LANDMARK TRACKER/STAR SENSOR. RATE GYRO ASSEMBLY CONTAINS THREE SINGLE-DEGREE-OF-FREEDOM INTEGRATED GYRO'S MOUNTED ORTHOGONALLY AND OPERATED IN CAGED MODE TO PROVIDE ANGULAR RATE OUTPUT INFORMATION. IMU CONSISTS OF AN ALL ATTITUDE, FOUR-GIMBAL, INERTIALLY STABILIZED PLATFORM AND PLATFORM ELECTRONICS, POWER SUPPLY, INTERFACE ELECTRONICS, BUILT-IN TEST EQUIPMENT AND THERMAL CONTROL CIRCUITRY. STAR SENSORS AND IMU'S ARE TRIPLE REDUNDANT. SINGLE LANDMARK TRACKER WITH FOUR-ANTENNA ARRAY TO ALLOW ONE ANTENNA OUT OPERATION. AUTONOMOUS OPERATIONAL CAPABILITY. TACAN AND MSBLS. VELOCITY ERRORS 15 m/s AND POSITION ERRORS 3.7 km AT INSERTION IN ALL INERTIAL MODE. LANDMARK TRACKERS CAN PROVIDE VELOCITY ERRORS .3 m/s (1 FT/SEC) AND POSITION ERRORS 305 m (1000 ft) FOR LOW EARTH ORBIT. ATTITUDE ERRORS 0.5^{o}. AIR DATA SUBSYSTEM CONSISTS OF RETRACTABLE PROBE ASSEMBLIES, AIR DATA TRANSDUCER ASSEMBLIES, WIRE HARNESS AND PNEUMATIC LINES AND THE ELECTRICAL CONTROLS FOR HEATERS AND DEPLOYMENT/RETRACTION.	DESCRIPTION AND RATIONALE STRAPDOWN INERTIAL MEASUREMENT UNITS, LANDMARK TRACER, AND STAR SENSORS. MODULAR REDUNDANCY IN IMU AT INERTIAL INSTRUMENT LEVEL. LIGHTER WEIGHT TUNED-SUSPENSION DRY GYROS. TACAN & MSBLS. TECHNOLOGY GROWTH IN THIS AREA SHOULD NOT REQUIRE SPECIAL FUNDING, AS SYSTEMS ARE FUNCTIONALLY ADEQUATE AND PROVIDE LITTLE LEVERAGE IN OVERALL SSTO WEIGHT REDUCTION IN POUNDS PER DOLLAR.
ENVIRONMENT CABIN ATMOSPHERE	WER SPACE SHUTTLE WEIGHT STATEMENT 428.6 kg (945 lb) 340 kg (750 lb) INCLUDING 59 kg (130 lb) OF INSTALLATION HARDWARE. WEIGHT IS BASED ON A STRAPDOWN IMU WITH LOWER WEIGHT THAN PRESENT ORBITER IMU.	WER AND RATIONALE 550# INCLUDING INSTALLATION HARDWARE. RATIONALE: REDUNDANCY AT MODULE LEVEL INSTEAD OF UNIT LEVEL. GREATER USE OF LSI CIRCUITRY. LIGHTER WEIGHT INSTRUMENTS ARE UNDER DEVELOPMENT. JPL FORECAST OF SPACE TECHNOLOGY.
REMARKS		

Figure 14 Technology Projection - Avionics

1986 TECHNOLOGY PROJECTION

SUBSYSTEM / TECHNOLOGY FLIGHT CONTROLS SYSTEM - AERODYNAMIC

NAME

SSTO REQUIREMENT	EXISTING TECHNOLOGY	PROJECTED TECHNOLOGY 1986
SUBSYSTEM PERFORMANCE REQUIREMENTS PROVIDE AERO CONTROL FOR: PITCH ROLL YAW STRUCTURAL LOAD RELIEF SURFACE DAMPENING ACTIVE VEHICLE AERO STABILITY VEHICLE ENTRY ENERGY MANAGEMENT FLIGHT PROFILE	DESCRIPTION BY ELEMENT a) - RUDDER/BRAKE PEDALS & TRANSDUCERS - HAND CONTROLLER & TRANSDUCERS - SPEED BRAKE CONTROLLER b) SHIELD AND GROUNDED WITH INTEGRAL AMPLIFIERS FOR LONG RUNS c) SOLID STATE COMPUTER/AMPLIFIER SUMS AND SELECTS PREPROGRAMMED MODE ACCORDING TO INPUT FOR SURFACE CONTROL, TVC, & RCS MIXED OPERATION d, e, f) STAINLESS STEEL OR TITANIUM CYLINDERS, HOUSINGS, SHAFTS, AND RODS. HIGH HARDNESS VALVING ELEMENTS. g) SOLID STATE DIGITAL	DESCRIPTION AND RATIONALE a) ALUM. &/OR MAG. CASTINGS & REINFORCED PLASTICS - SOME IMPROVEMENT POSSIBLE WITH CORED ALUM/CARBON COMPOSITES. b) NO SIGNIFICANT CHANGE c) NO SIGNIFICANT CHANGE. d, e, f) INCORPORATE COMPOSITES IN COMPONENTS - SOME IMPROVEMENT WITH INCREASE IN PRESSURE g) NO SIGNIFICANT CHANGE
DESCRIPTION OF ELEMENTS a) COCKPIT MECHANICAL/ELECTRICAL CONTROLLERS b) SIGNAL TRANSMISSION ELEMENTS c) SIGNAL MODE-SELECT & MIXER d) ELEVON ACTUATORS e) RUDDER ACTUATORS f) SPEED BRAKE ACTUATORS g) ACTUATOR SERVO-AMP CONTROLLER (AIR DATA, CENTRAL COMPUTERS, INERTIAL REFERENCE, ETC., IN GN&C)		
ENVIRONMENT 200 K to 478 K (-100^0F to $+400^0$F) AMBIENT PRESS 0 to .1 MPa (0 psia to 14.7 psia) 0 g to 5 g	WER a) - WT/STA - WT/CONTROLLER - WT/CONTROL HANDLE INST. b) kg/SINGLE CHANNEL - m c) MIN. WEIGHT -kg/100 FUNCTIONS d, e, f) kg/HM - ANGLE g) kg/FUNCTIONS	WER AND RATIONALE a) .11 kg (.25 LB) STA. + 5.4 kg (12 LB) FOR FEEL AND CENTERING 13.6 kg/STA 6.8 kg/FUNCTION b) .069 kg/m CHANNEL(4.25 LB/100 FT) CHANNEL c) PART OF GN&C COMPUTERS d, e, f) ACT. WT. kg(LB)= $.016 \frac{mkg}{MPa} \left(\frac{HM}{104.12P}\right) \left(\frac{LB/IN^0}{psi}\right)$ g) .9 kg (2 LB)/INST.
REMARKS 6000 HRS. LIFE 500 FLIGHTS 733 HRS. OPERATION		

Figure 13 Technology Projection - Flight Controls System

	LANDING IMPACT	LANDING CONDITIONS SPIN UP	3 PT. LEVEL	DRIFT	TAXI/ROLLOUT BRAKED ROLL	TURNING	TOWING	LOAD DISTRIBUTION
FAR 25	(10 ft/sec) 3.048 m/s	.8μ x MAX. VERT. FORCE	SINK LOADS	SINK LOADS + .8 IN & .6 OUT	1.2 X BRAKED ROLL	.5 g SIDE OR OVERTURN	TAXI WT. X .15	VARIOUS FLAT TIRE COMBINATIONS
COMMERCIAL TRANSPORTS 707,727,737 747	✓	.7μ	SINK LOADS + PITCH-OVER ON NOSE	✓	BRAKE TORQUE LIMITED (ANTI-SKID)	✓	✓	✓
SSTO SEE CRITERIA	✓	✓ ± WITH SPRING-BAC @ 20% STROKE	✓	✓ LIMITED TO .5 x 3.048 m/s (10 ft/sec) LOADS	BRAKE TORQUE LIMITED (ANTI-SKID)	✓	TOW WT. X .15	✓
MINIMUM WEIGHT RECOMMENDATIONS	✓ * NOTE 2.06° GLIDE-SLOPE @ 84.9 m/s (165 KTS	✓	✓	REVIEW TO LIMIT TO REASONABLE CROSSWINDS	BRAKE TORQUE LIMITED	DRIFT LANDING TO SIZE CAPABILITY	CAPABILITY SIZED BY LANDING	✓

Figure 15 Landing Gear Design Requirements

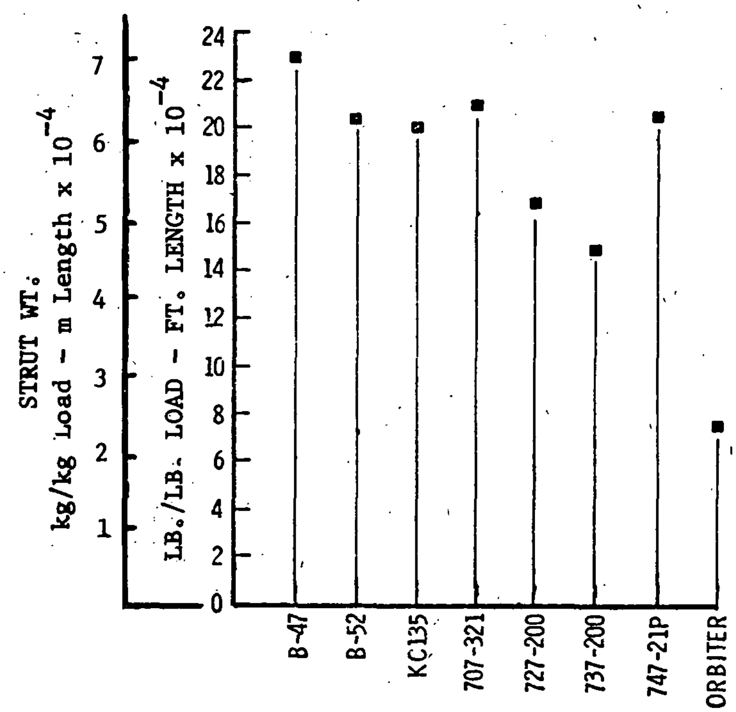


Figure 16 Landing Gear Strut Weight Relationships

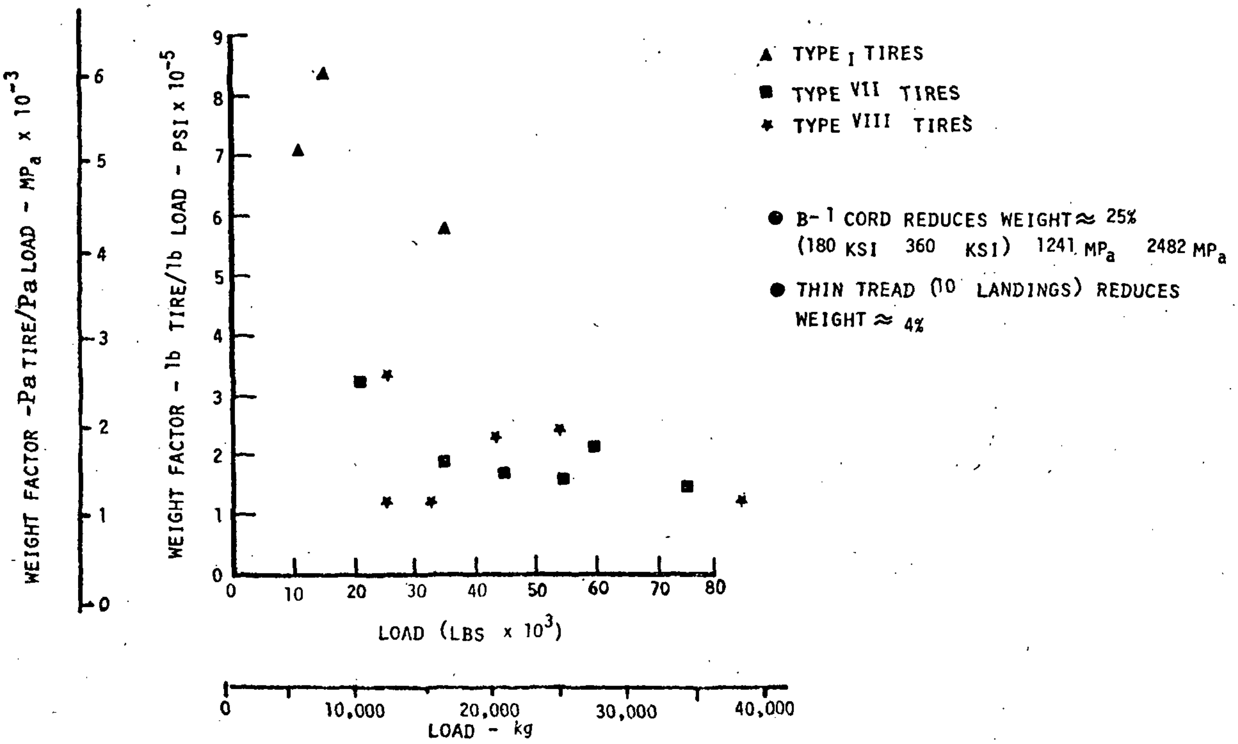


Figure 17 Landing Gear Tire Weight Relationships

WEIGHT APPROX. 20% OF GEAR WT.

WEIGHT A FUNCTION OF ENERGY ABSORBED AND MAX ΔT ALLOWED

- BRAKES ABSORB 38% MV^2 AT LANDING

- ΔT	NORM	RTO
	(# BRAKE/1 X 10^6 ft lb) kg/1.3558 X 10^6 JOULES	
1387K (1680°F)	3.47 kg (7.66 lb)	1.86 kg (4.1 lb)
1169K (1286°F)	4.54 kg (10.0 lb)	2.18 kg (4.8 lb)
1030K (1037°F)	5.62 kg (12.4 lb)	2.36 kg (5.2 lb)

- 62% MV^2 ABSORBED BY AERO DRAG, SPOILERS, THRUST REVERSER, ETC.
- ORBITER 1.38 kg/1.3558 X 10^6 JOULES AT 84,640 kg, 313 km & 38%
(3.05 LBS x 10^6 ft lb) (186,600 LBS, 169 kts)

Figure 18 Brake Weight Estimation

SSTO @ 119,748 kg (264,000 LB

TIRE WT.	502.6 kg (1108 lb) X .71 =		357 kg (787 lb)
BRAKE WT.	@ 119,748 kg & 259 KPH (264K & 140 KTS) @ .38 & 2.27 kg /1.3558 X 10^6 JOULES (5 lb / 1 X 10^6 ft lb)		395 kg (870 lb)
WHEEL WT.	.46 BRAKE WT. =	450 kg (992 lb)	423 kg (933 lb)
	OR .79 TIRE WT. =	397 kg (875 lb)	
RUN GEAR			1175 kg (2590 lb)(.98%)
STRUCTURE			
STRUT		1329 kg (2930 LB)	
IF STRUT 50% OF TOTAL STRUCTURE			2658 kg (5860 lb)(2.2%)
IF BORON/ALUM. - 18% REDUCTION			2180 kg (4805 lb)(1.8%)
IF 2.06 X 10^9 Pa TO 2.413 X 10^9 Pa (300) TO (350 KSI) - 7.8% REDUCTION			2450 kg (5402 lb)(2.0%)
CONTROLS WT.			
TIRE WT. OR 8% OF TOTAL GR WT. RETRACT SYSTEM 60% OF TOTAL WITH GSE RETRACT = 357 kg (787 lb) x .4			142 kg (314 lb)
			3354 kg (7395 lb)(2.8%)

Figure 19 HTO Vehicle Landing Gear Weight Analysis

STRUT	391 kg (862 lb)
PISTON	174 kg (384 lb)
INTERNAL VALVING	107 kg (236 lb)
TORSION	51 kg (112 lb)
OIL	38.1 kg (84 lb)
	761 kg (1678 lb)

SERVO ACTUATOR 27.6×10^6 Pa (4000 psi)	181 kg (400 lb)
VERTICAL	363 kg (800 lb)
SWING LINK	136 kg (300 lb)
	680 kg (1500 lb)

11% WEIGHT SAVINGS BY DESIGN CHANGE TO SERVO

SERVO STRUTS MORE COMPATIBLE WITH BORON/ALUMINUM CONSTRUCTION

20% TO 25% WEIGHT SAVINGS

Figure 20 Landing Gear Oleo Versus Servo Strut

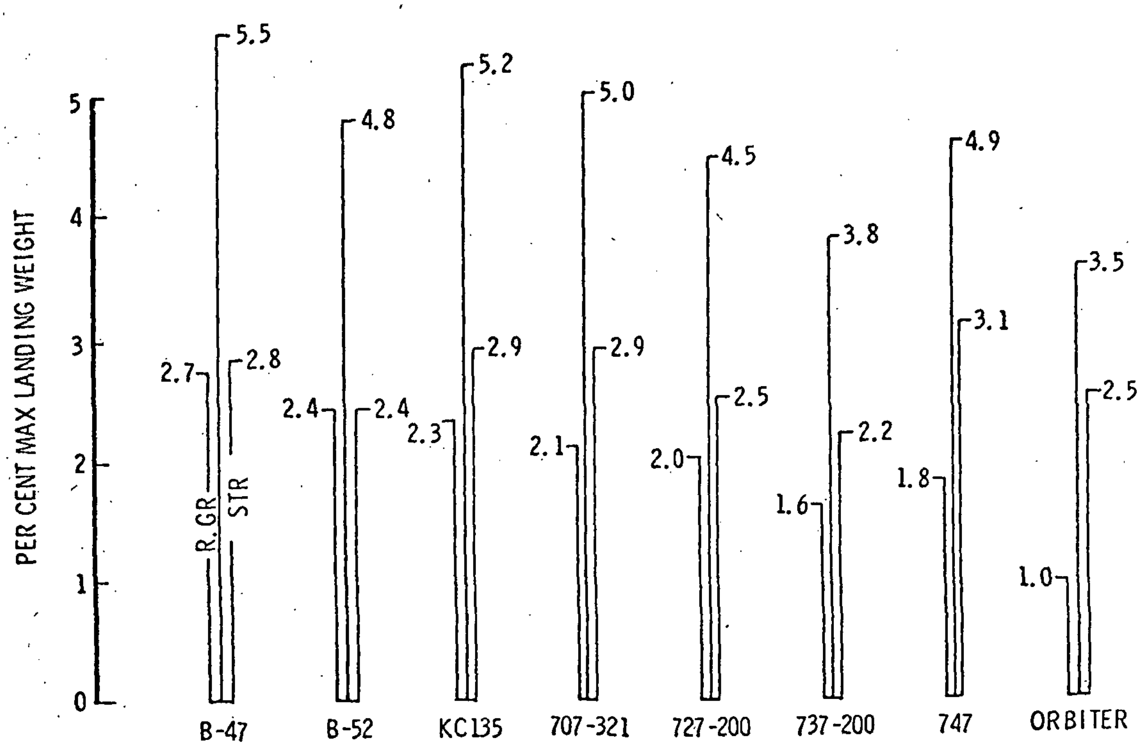


Figure 21 Landing Gear Weight Trends

steel and improved tire designs. Figure 16 illustrates weight trends for the strut as influenced by load and length. Figure 17 illustrates similar trends for tires. Figure 18 depicts the relationships associated with sizing and weighing brakes. From these then, the weight of a landing gear can be developed as shown on figure 19 . Also, shown is the impact of improvements such as Boron-Aluminum, 18% weight reduction, a 2410 MPa (350 ksi) steel, a 7.8% weight reduction.

Figure 20 illustrates the weight savings potential of a configuration change in the landing gear design through the separation of the hydraulic and structural functions in the oleo. These savings would require a detailed design analysis to verify, however the potential is evident.

Figure 22 summarizes the "normal technology projections" for the structures subsystems of a Single-Stage-to-Orbit vehicle. The illustration lists the various structural elements, the technology growth area or program which will drive the technology improvement and the result in terms of weight and/or performance capability. The main criterion used to determine if a

Structural elements	Technology growth area	Result
Surface panels Rene' 41 honeycomb	Basic braze alloy/process dev.	Decrease cost, improve braze toughness
	Allowables development Panels/joints/dev. & test Assemblies/dev. & test	Low density/insulative structure with 20 K (-423°F) to 1,144 K (1600°F) operational capability
Titanium honeycomb	Basic braze alloy/process dev.	Improve temp cap. from 699 K (800°F) to 811 K (1000°F)
	Allowables development Panels/joints/assy dev. & test Assemblies (see above)	Provide low density/insulative structure with 20 K (-423°F) to 811 K (1000°F) operational capability
Truss/frames/thrust structure Metal matrix composites	Process/manufacture dev.	Provide low density/high strength structure
	Allowables development Design/joints/assys dev. & test	Provide structure with significant weight savings over metallic structure for temperature of 33 K (-400°F) to 755 K (900°F)
Leading edges Refractory & super-alloy metals	Design/analysis development Assembly dev. & test	Provide lightweight, long life leading edges with temp capability to 1,589 K (2400°F)
Components	Tooling, joining and inspection	Capability to manufacture advanced structural system for cryogenic fuel containment

Figure 22 Structures Technology Growth Summary

technology would be available by 1986 without special funding was: "Would the program exist if an SSTO type program were not available?" In-house structural programs at Boeing and other aerospace and aircraft companies as well as supplemental government funding indicate the application of the structural concepts to areas outside the interest of an SSTO vehicle (i.e., SST, Space Shuttle improvements, hypersonic research vehicle, etc.).

Figure 23 summarizes the "normal technology projections" for the subsystem elements of a Single-Stage-to-Orbit vehicle. The illustration lists the subsystem, the technology growth area or program which will drive the technology improvement and the result in terms of weight and/or performance. As indicated, several subsystems utilize the existing technology because perturbed or special funding would be required so that the subsystem program presently projected would not be weight competitive with the present performance requirement.

Subsystem	Technology growth area	Result
● Landing gear	● 2.4×10^9 Pa (350 ksi) maraging steel ● Boron/aluminum composites ● 2.7×10^7 Pa (4,000 psi) hydraulics	● System weight reduced from 3.5 to 2.8% landed weight
● Main propulsion	● Nozzle extension ● 2.4×10^7 Pa (3,500 psi) chamber pressure ● Zero NPSH pumps	● Increased performance with improved T/W ● Reduced ullage pressures
● Surface controls	● 3.45×10^7 Pa (5,000 psi) hydraulics ● Composite materials	● Reduced system weight in actuators
● Hydraulic conversion and distribution	● 3.45×10^7 Pa (5,000 psi) operating pressure ● Composite materials	● Reduced system weight in lines and fluids
● Propellant feed and repressurization	● Composite materials	● Reduced system weight in lines and tanks
● Avionics	● LSI circuitry ● Laser radars ● Micro processors ● Solid state displays ● Bubble memories	● Reduced system weight in all areas
● Electrical power conversion and distribution	● Solid state power conditioning and switching equipment	● Reduced system weight
● RCS, OMS, prime power, ECS & crew provisioning	● Existing technology	● No impact

Figure 23 Subsystems Technology Growth Summary

Figures 24 and 25 show significant projected weight reductions on the basis of "normal technology projections" for SSTO vehicle - HTO structures and subsystems respectively, as compared with a vehicle using current technology. It is important to understand that generally when considering potential weight reductions these may reflect the impact of two factors. These are changes in requirements and improvement in technology (See figure 26).

Weight reductions range from 0 to 45% for structures and from 0 to 27.3% for subsystems. The P/L doors and crew compartment reflect existing shuttle technology. The total structural reduction is 17.1%. In figure 25 the RCS and OMS system weights reflect the existing technology of the Space Shuttle and RL-10 engine, respectively. The total overall subsystems reduction is 12.5%. Since the ratio of structural to subsystem weight is approximately 70:30, this combined with the structures reduction is a projected weight improvement of 15.8%.

The requirement differences alone can have a significant impact in several areas. Examples are the lower entry temperatures which affect materials usage and the 12-hour mission duration which reduces the overall subsystem loads. The weights reductions illustrated in figures 24 and 25 show only the impact

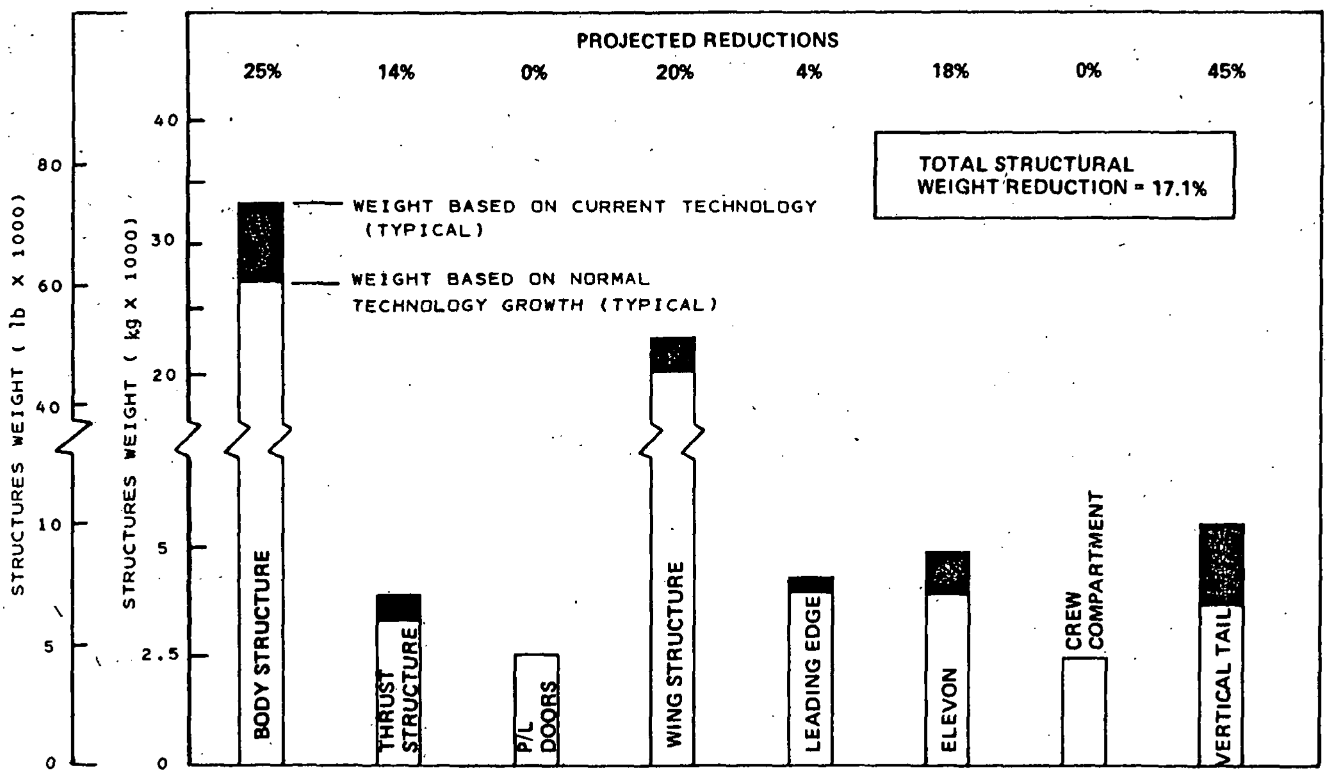


Figure 24 Structures Weight Reduction Summary

of the normal technology improvements because of the lack of a confirmed data base from the Space Shuttle program.

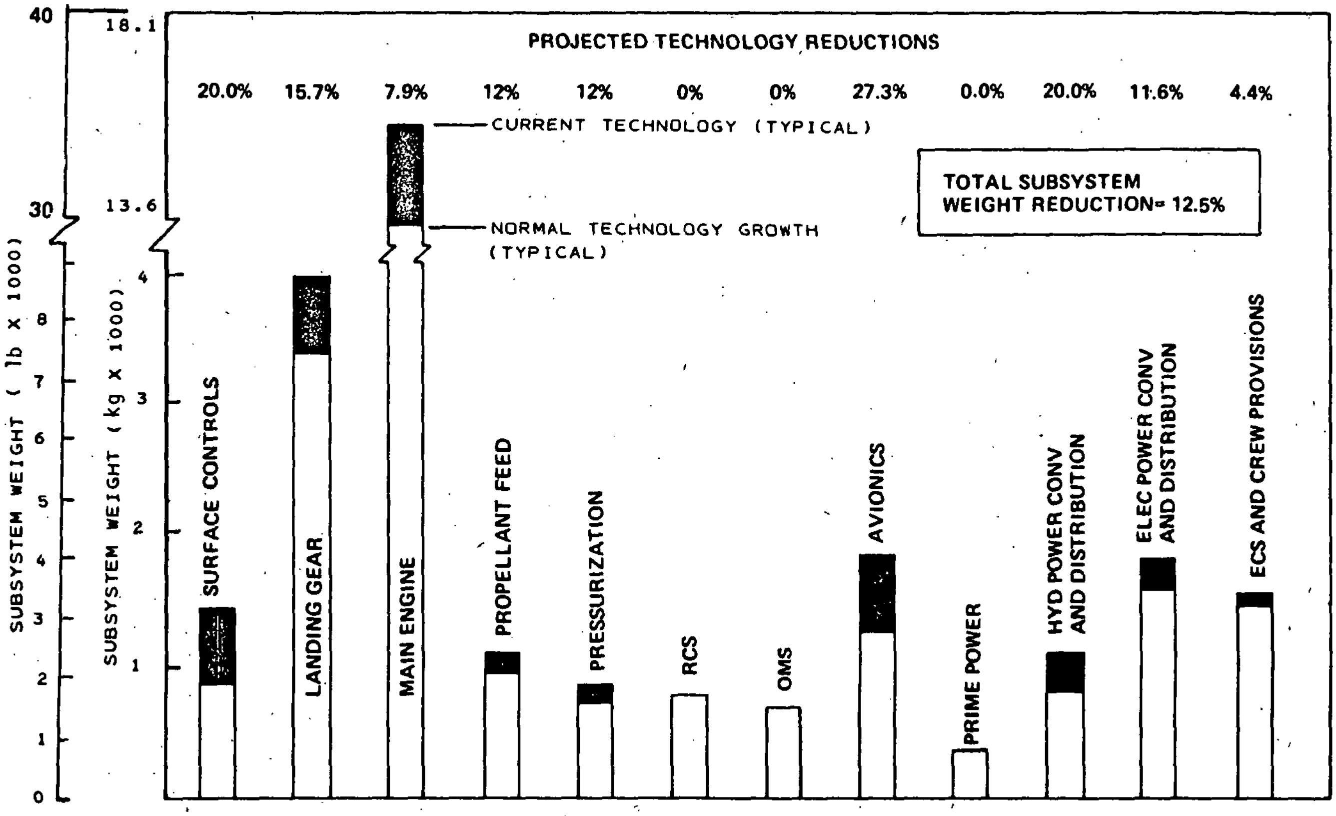


Figure 25 Subsystems Weight Reduction Summary

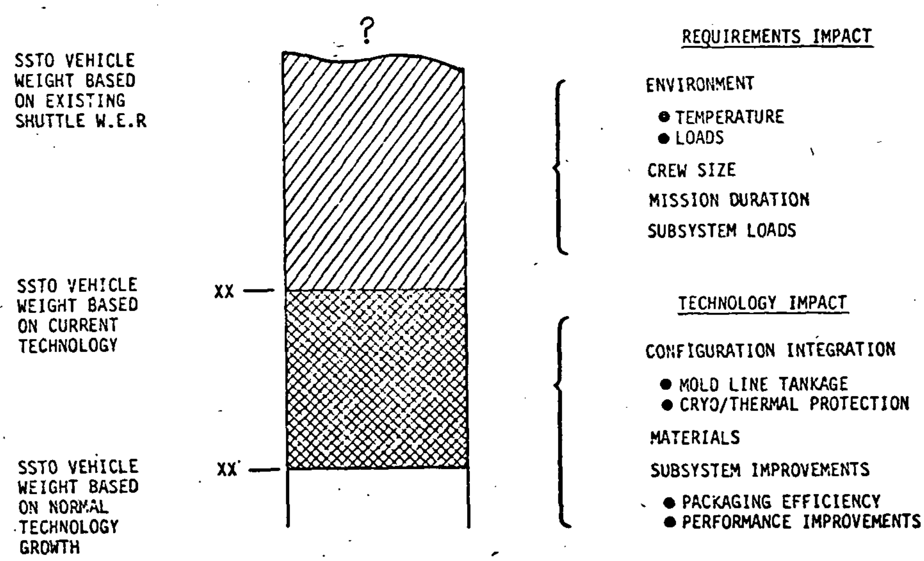


Figure 26 Weight Reduction Definition

DESIGN AND DEFINITION OF PERFORMANCE POTENTIAL OF VEHICLE SYSTEMS - TASK II

This task consists of defining four Single Stage to Orbit (SSTO) vehicle configurations, obtaining subsystem design data from Task I, defining subsystem performance requirements and environments, selecting subsystem concepts, analyzing and sizing subsystems and calculating total configuration weights. The vehicle model ALRS 205 was chosen as a baseline configuration from which other configurations were developed using a generic approach.

The four configurations are:

1. Sled Assist - Horizontal Take-Off (HTO) - Baseline
2. Sled Assist - Horizontal Take-Off-Aerial Refuel
3. Air Carry - Horizontal Take-Off-Air Launch
4. Vertical Take-Off (VTO)

These vehicles would have a first operational flight in 1995. Due to the generic vehicle concept, design differences between vehicle configurations reflect consistent design approaches, philosophy and technology levels. Thus it was possible to avoid repetition in the analysis of the various configurations and to apply analysis results to more than one configuration.

The selection of the baseline is rooted in Boeing's background experience and familiarity with the SSTO vehicle HTO concept built upon a substantial data base established over several years of study.

Requirements and Criteria

The purpose of this section is to summarize the requirements which pertain to the four above vehicle configurations. In addition, it is intended to establish guidelines which would provide a consistent set of ground rules to permit a valid comparison of the four vehicle system concepts developed in this study. These requirements can be grouped into mission (Figure 27) subsystems (Figure 28) and performance requirements. Performance requirements include aerodynamics (Figure 29), loads (Figure 30), thermal (Figure 31) and structural (Figure 32) analysis and design criteria.

Both NASA directed and Boeing proposed requirements are included.

- LIFETIME: 500 MISSIONS (LOW COST REFURBISHMENT)
- MISSION DURATION: 12 HOURS OF SELF-SUSTAINING LIFETIME FROM LIFT-OFF TO LANDING
- EASTERN LAUNCH FROM KSC @ 28.5^{o} INCLINATION
  (REFERENCE ENERGY ORBIT 92.6 X 185 km (50 X 100 nm)
- PAYLOAD: 29,483 kg (65,000 lb) (PAYLOAD VOLUME 18.3 m (60 ft) LONG 4.57 m (15 ft DIAMETER)
- ORBITAL MANEUVERING SYSTEM: = 198 m/s (650 fps)
- REACTION CONTROL SYSTEM: = 30.5 m/s (100 fps)
- TPS DESIGN MISSION (REENTRY): ENTRY FROM DUE EAST 28.5^{o} INCLINATION
  371 km (200 nm) ALTITUDE ORBIT
  PAYLOAD (29,483 kg (65,000 lb)
  2,038 km (1,100 nm) CROSS RANGE CAPABILITY
- FUEL: LO /LH
- CREW OF THREE (WITH ACCOMMODATIONS FOR ONE ADDITIONAL PASSENGER)
- DOCKING MECHANISM

Figure 27 Study Mission Requirements

AIRFRAME SUBSYSTEM

CREW COMPARTMENT = 5.14 X 10^{4} M/M^{2} (7.45 psi) O2/N2 MIXTURE

WHEEL WELL - UNPRESSURIZED -53^{o}C (-65^{o}) TO 148.9^{o}C (+300^{o}F)

LANDING SUBSYSTEM

EXTENDABLE GEAR WITH GSE RETRACTION

BRAKES AND STEERING

MAIN PROPULSION

HI PRESSURE LH2/LO2 BELL NOZZLE OR LINEAR ROCKET ENGINE

REACTION CONTROL

ΔV = 30.5M (100 fps)

OMS

ΔV = 198 M (650 fps) IN EXCESS OF REFERENCE ENERGY ORBIT

AVIONICS

AUTONOMOUS OPERATION

ALL WEATHER LANDING

Figure 28 Study Subsystem Requirements

- MINIMUM SUBSONIC FLYING SPEED (SEA LEVEL CONDITIONS AND MAXIMUM LANDED WEIGHT): NOT TO EXCEED 315 km (170 KNOTS) ANGLE OF ATTACK FOR THIS CONDITION NOT TO EXCEED 15^{o}

- 2% STATIC MARGIN AT SUBSONIC SPEEDS OR DEMONSTRABLE CONTROL AUTHORITY TO HANDLE STABILITY AND CONTROL REQUIREMENTS

- STATIC DIRECTIONAL STABILITY ($-C_{n\beta}$) $\geq$.002 (FOR NON-CONTROL CONFIG DES.)

- TRIMMABLE ANGLE OF ATTACK RANGE AT HYPERSONIC SPEEDS FROM A MINIMUM OF 20^{o} OR LESS TO A MAXIMUM OF 40^{o} OR GREATER FOR BOTH PAYLOAD LOADING CONDITIONS

- STABLE DYNAMIC PROPERTIES UTILIZING RCS DURING PERIODS OF LOW DYNAMIC PRESSURE AND AERODYNAMIC CONTROL SURFACES WHEN DYNAMIC PRESSURES ARE SUFFICIENT

Figure 29 Study Aerodynamic Requirements

- ASCENT LOADS WIND SHEAR AND GUST $q\alpha$ AND $q\beta$
  - WIND SHEAR BASED ON MOST ADVERSE LAUNCH LOCATION AND WIND DIRECTION
  - GUST-DISCRETE 1-COSINE SHAPED BOTH VERTICAL AND LATERAL

- HYPERSONIC ENTRY LOADS

LOAD FACTOR INCREMENT*		VELOCITY
$\Delta n_{z_{max}}$ - .17	>	4877 m/s (16,000 fps)
$\Delta n_{z_{max}}$ = .28	<	4877 m/s (16,000 fps)

- DESCENT MANEUVER LOADS
  - BALANCED PITCH MANEUVER, n_z = +2.5 AND -1.0

- DESCENT GUST LOADS
  - MASS PARAMETER FORMULA
  - TRUE GUST VELOCITY = 15.2 m/s (50 fps)

- LANDING IMPACT

CONDITION	MAX SINK RATE
SYMMETRIC	3.05 m/s (10 fps)
ROLLED	2.13 m/s (7 fps)

- GROUND LOADS

CONDITION	LOAD FACTOR
BRAKED ROLL (μ = 0.8)	n_z = 1.2
TAXI	n_z = 1.67
GROUND TURN n_y = 0.4	n_z = 1.0

*DUE TO TRAJECTORY DISPERSIONS

Figure 30 Study Load Requirements

- TRAJECTORY DATA (NO TRAJECTORY DISPERSIONS)

ATMOSPHERIC PROPERTIES:	1962 STANDARD ATMOSPHERE
ALTITUDE:	NOMINAL VALUE
VELOCITY:	NOMINAL VALUE
ANGLE OF ATTACK:	NOMINAL VALUE
ANGLE OF YAW:	NOMINAL VALUE = 0
BANK ANGLE:	NOMINAL VALUE

- AERODYNAMIC HEATING RATES

LAMINAR HEATING THEORY:	Rho-mu
TURBULENT HEATING THEORY:	SPALDING-CHI
REYNOLDS ANALOGY:	COLBURN MODIFICATION $ST = 1/2\ C_f P_r^{2/3}$
BASIC HEATING UNCERTAINTY FACTORS:	1.10 LAMINAR, 1.25 TURBULENT
SURFACE ROUGHNESS HEATING FACTOR:	1.0
INTERFERENCE HEATING FACTOR:	1.0
BOUNDARY LAYER TRANSITION ONSET:	RI/SD CORRELATION
$\frac{Re_\theta}{Me} = N$	N = 225 @ BODY CENTERLINE = 160 @ WING MIDCHORD = 80 @ WING TIP
TRANSITION REGION LENGTH:	$Re_{END} = 2\ Re_{BEG}$

VIRTUAL ORIGIN OF TURBULENT FLOW AT Re_{BEG}

- STRUCTURAL TEMPERATURES

NOMINAL VALUES. NO FACTORS

Figure 31 Study Thermal Requirements

NORMAL OPERATING CONDITIONS

TENSION: $f_{th} + f_t \leq KF_{ty}$: f_{th} = INDUCED THERMAL STRESS f_t = APPLIED TENSION STRESS

F_{ty} = ALLOWABLE TENSION YIELD STRESS; K = .7 FOR RENE' 41; K = .6 for 6AL -4V - Ti

CONDITIONS

	FACTORS ON LIMIT LOAD		
	COMPRESSION ALLOWABLE	YIELD	TENSION ULTIMATE
APPLIED FORCES			
UNPRESSURIZED STRUCTURE	[1]	1.0	[1]
THRUST STRUCTURE	1.5	[2]	1.5
PRESSURIZED STRUCTURE			
COMBINED LOADS	[1]	1.0	[1]
PRESSURE ONLY	2.0	-	2.0
WINDOWS, DOORS, HATCHES	2.0	-	2.0
PRESSURIZED LINES, FITTINGS AND PRESSURE VESSELS OTHER THAN MAIN PROPELLANT TANKS	-	-	2.0

[1] THE MINIMUM SAFETY FACTOR FROM LIMIT TO ULTIMATE LOAD SHALL BE 1.4 FOR PRELAUNCH, LIFTOFF, ASCENT AND IN-ORBIT DESIGN CONDITIONS. IT SHALL BE 1.5 FOR ENTRY, SUBSONIC MANEUVER AND LANDING CONDITIONS.

[2] AT A FACTOR OF SAFETY OF 1.0 ON MAXIMUM ANTICIPATED THRUST, THERE SHALL BE ZERO PERMANENT STRUCTURAL DEFORMATION.

Figure 32 Study Structural Requirements **(continued on next page)**

Figure 32 (continued)

TENSION: PRESSURIZED STRUCTURE, TANKS, LINES AND FITTINGS AND WINDOWS, DOORS AND HATCHES

THERMAL + PRESSURE $1.25\ \varepsilon_{th} + 2.0\ \varepsilon_{PRESS} = \varepsilon_{ALLOW}$

PRESSURIZED STRUCTURE

ASCENT $1.25\ \varepsilon_{th} + 1.4\ \varepsilon_{PRESS} + 1.4\ \varepsilon_{STATIC\ LOADS} \leq \varepsilon_{ALLOW}$

DESCENT $1.25\ \varepsilon_{th} + 1.5\ \varepsilon_{PRESS} + 1.5\ \varepsilon_{STATIC\ LOADS} \leq \varepsilon_{ALLOW}$

UNPRESSURIZED STRUCTURE

ASCENT $1.25\ \varepsilon_{th} + 1.4\ \varepsilon_{STATIC\ LOADS} \leq \varepsilon_{ALLOW}$

DESCENT $1.25\ \varepsilon_{th} + 1.5\ \varepsilon_{STATIC\ LOADS} \leq \varepsilon_{ALLOW}$

ε_{th} = INDUCED THERMAL TENSION STRAIN

WHEN ε_{th} IS NEGATIVE USE $\varepsilon_{th} = 0$

ε_{PRESS} = NET TENSION STRAIN CAUSED BY LIMIT PRESSURE LOADING

$\varepsilon_{STATIC\ LOADS}$ = NET TENSION STRAIN CAUSED BY LIMIT STATIC LOADS

ε_{ALLOW} = STRAIN AT ALLOWABLE STRESS

$\varepsilon_{ALLOW} = \varepsilon F_{tu}$ WHEN $\varepsilon F_{tu} \leq .8$ TOTAL

$= .8\ \varepsilon F_{tu}$ WHEN $F_{tu} \geq .8\ \varepsilon_{TOTAL}$

εF_{tu} = STRAIN AT ALLOWABLE ULTIMATE TENSION STRESS

ε_{TOTAL} = TOTAL MATERIAL TENSION STRAIN (ELONGATION)

COMPRESSION

THERMAL + PRESSURE $\varepsilon_{th} + 2.0\ \varepsilon_{PRESS} \leq \varepsilon F_{c}$

UNPRESSURIZED STRUCTURE

ASCENT $1.25\ \varepsilon_{th} + 1.4\ \varepsilon_{STATIC\ LOADS}\ \varepsilon F_{c}$

DESCENT $1.25\ \varepsilon_{th} + 1.5\ \varepsilon_{STATIC\ LOADS}\ \varepsilon F_{c}$

PRESSURIZED STRUCTURE

ASCENT $\varepsilon_{th} + 1.4\ \varepsilon_{PRESS} + 1.4\ \varepsilon_{STATIC\ LOADS} \leq \varepsilon F_{c}$,

DESCENT $\varepsilon_{th} + 1.5\ \varepsilon_{PRESS} + 1.5\ \varepsilon_{STATIC\ LOADS} \leq \varepsilon F_{c}$

ε_{th} = INDUCED THERMAL COMPRESSION STRAIN

WHEN ε_{th} IS POSITIVE USE $\varepsilon_{th} = 0$,

ε_{PRESS} = NET COMPRESSION STRAIN CAUSED BY PRESSURE LOADING

$\varepsilon_{STATIC\ LOADS}$ = NET COMPRESSION STRAIN CAUSED BY STATIC LOADS

εF_{c} = STRAIN AT COMPRESSION ALLOWABLE STRESS

TPS PANEL DEFLECTION

ITEM	SPAN/DEFLECTION
OVERALL PANEL	100
LOCAL PANEL (SKIN)	15

VEHICLE SKIN JOINTS SHALL HAVE NO FORWARD FACING STEPS

SKIN PANEL FLUTTER:

THE PANEL SHALL BE FREE OF FLUTTER AT ALL DYNAMIC PRESSURES UP TO 1.44 TIMES THE LOCAL DYNAMIC PRESSURE EXPECTED TO BE ENCOUNTERED AT ANY MACH NUMBER DURING NORMAL FLIGHT.

Configuration 1 - Sled Assist - Horizontal Take-Off (HTO) - Model ALRS 205

This section contains a discussion of the major features of the baseline configuration, its mode of operation as well as the considerations leading to its selection including design rationale.

A typical mission profile for the SSTO-HTO vehicle is shown in Figure 33 It includes a ground accelerator assisted takeoff at 183 m/s (600 fps) followed by a climb limited to a 1.25 g normal load factor. The acceleration phase is a lifting type ascent trajectory to orbit injection. After delivery of payload the vehicle uses its OMS engine to deorbit, entering at a planform loading of 1245 Pa (26 psf). The vehicle glides back and performs its final maneuvers to a power off horizontal landing.

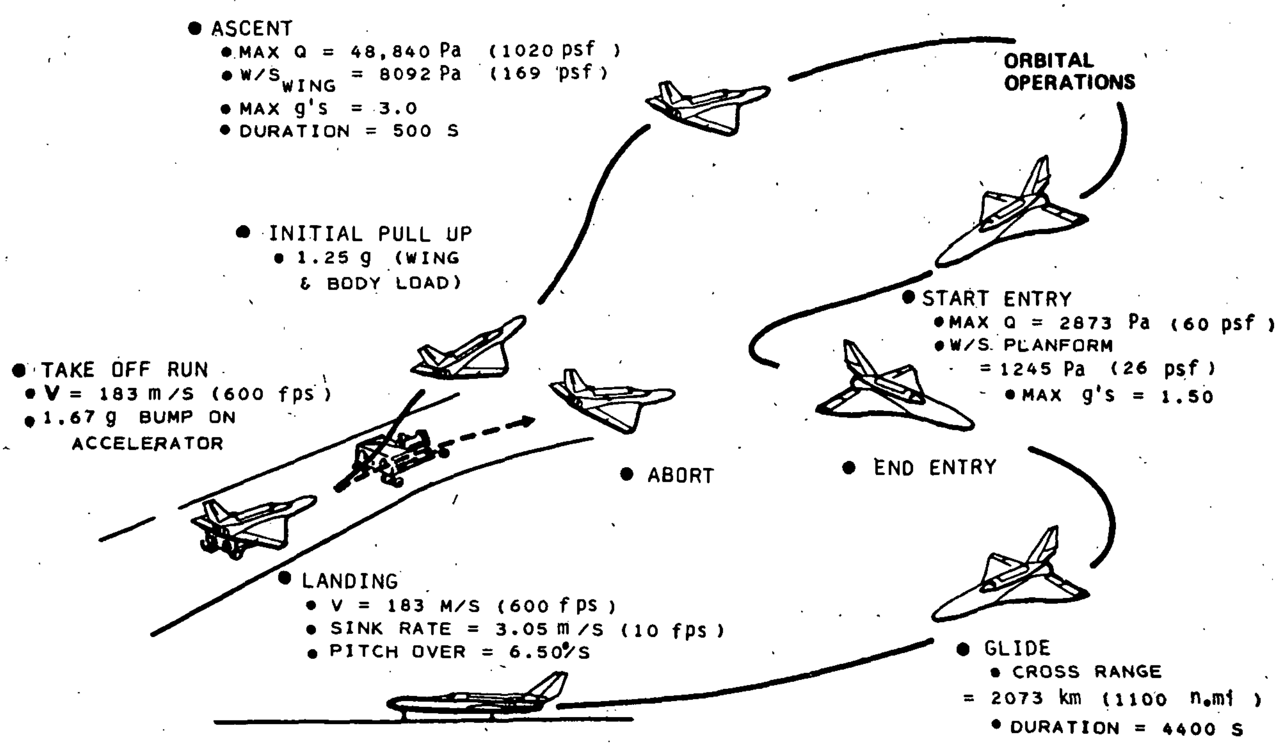


Figure 33 HTO-SSTO Mission Profile

Vehicle Design and Layout

The ALRS 205 shown in Figure 34, is a delta wing vehicle with integral wing and body tanks that takes off and lands horizontally. The liquid oxygen in the wings provides relief for the aerodynamic lifting loads during ascent. The oxygen's weight is sled supported during take off, then aerodynamically supported until expended. As a result, wing bending, landing gear punch and the resulting weight required by a conventional landing gear supported take-off is eliminated. At the same time, the weight savings associated with the low thrust-to-weight ratio of a landing-gear-supported take off is retained. A liquid hydrogen tank forms the major portion of the main body. The wing root bulkhead forms the interface between the body hydrogen tank and the liquid oxygen wing tanks. Like others of this series, the ALRS 205 has control surfaces in the wings and fin and reaction engines for control and manuevering in orbit. A 4.5 m diameter x 18.3 m long (15 foot diameter x 60 foot long) payload bay is partially submerged in the upper portion of the body LH_2 tank.

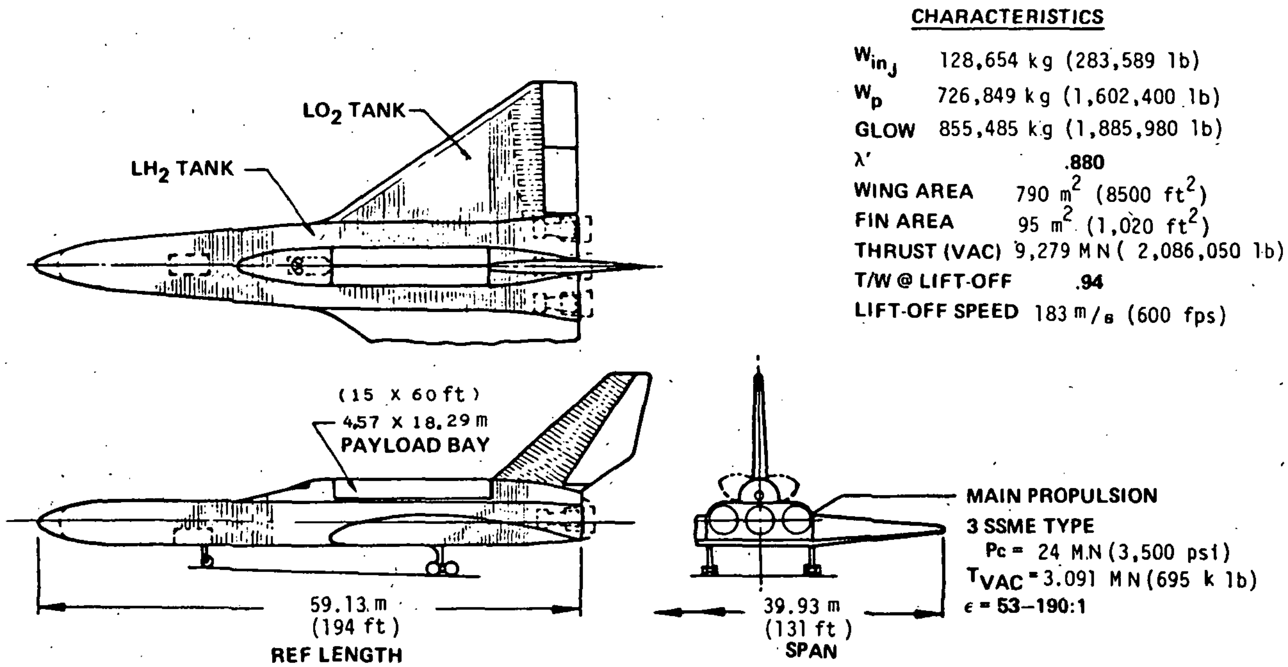


Figure 34 HTO-SSTO Vehicle Configuration

SSTO Configuration Synthesis. The SSTO configuration is worked in terms of structures in conjunction with vehicle flight performance to satisfy the specified dry weight figure of merit. Towards this end it is useful to recognize that the structure must provide housing and support for the other subsystems with shaping, materials and strength consistent with the environment encountered and the single stage to orbit function.

The most prominent feature of the SSTO is tankage, propulsion is the second, payload provisions the third. Because of the tankage size, the configuration size deltas (most of which are in Task III) are worked first in terms of effects on tankage as shown on Figure 35. Ripple effects on other subsystems are added on as seen from the extendable nozzle trade shown in Figure 36 as an example of a configuration delta.

The SSTO structures engineering, in common with other engineering, is constrained by physical laws and economics. The satisfaction of physical laws is a minimum constraint. The economic time frame and conditions are customer defined. Within these constraints, the configuration is evaluated in terms of (1) cost, (2) the dry weight figure of merit, (3) technology perturbation, and (4) valid exceptions to commonly accepted constraints (other than physical laws) whose conveniences are inconsistent with the pressure towards better cost performance.

Figure 36 shows a series of tankage-configuration structural features organized to indicate (1) their relationship to existing practice, (2) their cost performance characteristics, and (3) their relative success criteria. The integrated concept selection was reinforced by this rationale after the following structural concept synthesis and thermal protection system trades.

Structural Concept Synthesis. - Table 1 shows the vehicle structural concept synthesis in terms of materials, structural elements and structural assemblies referred to the baseline concept and three alternate structural concepts. The baseline concept was selected by the dry weight figure of merit.

Thermal Protection System Trade. -The critical common system at both the configuration level and the structural concept level is the thermal protection system. Assessment and selection of the Boeing baseline relates to four alternate approaches in terms of weight and response to the study objectives. A trade study summary including appropriate comments is shown in Table 2

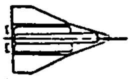
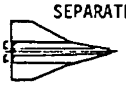
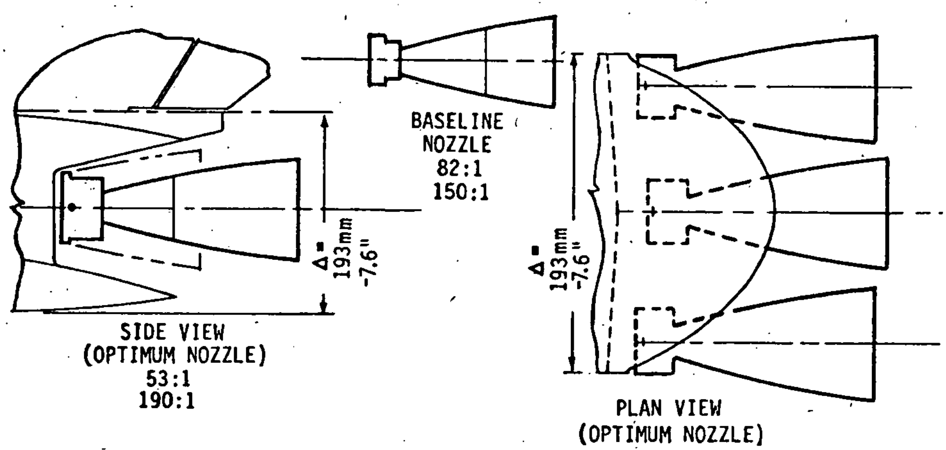
CONFIGURATION	RELATION TO EXISTING PRACTICE	COST-PERFORM. PRESSURE	RELATIVE SUCCESS CRITERIA
ALL CYLINDRICAL TANK SEPARATE AIR FRAME SEPARATE TPS 	MAX. LEAST CHANCE OF PROBLEMS FROM CONFLICTING FUNCTIONS LIMITED REQM'T FOR R&D & ADVANCING DESIGN METHODOLOGY	ESTIMATED RELATIVE DRY WEIGHT 1.38	AVAILABILITY OF OPERATING COST FUNDS
MODIFIED CYLINDRICAL TANKAGE PARTIALLY INTEGRATED AIRFRAME-TANKAGE SEPARATE TPS 	SUBSTANTIAL SOME CHANCE OF PROBLEMS FROM CONFLICTING FUNCTIONS LIMITED REQM'T FOR R&D & ADVANCING DESIGN METHODOLOGY	ESTIMATED RELATIVE DRY WEIGHT 1.24	AVAILABILITY OF OPERATING COST FUNDS
COMPLETELY INTEGRATED INSULATION, AIRFRAME TANKAGE 	CONSISTENT WITH REALIZING R&D & DESIGN IMPROVE. PAYOFF BY EVOLUTION ANALOGOUS TO THAT ALREADY ACCOMPLISHED ON JET AIRCRAFT	ESTIMATED RELATIVE DRY WEIGHT 1.0	AVAILABILITY OF R&D FUNDS AS INVESTMENT TOWARDS LWR OPERATING COSTS

Figure 35 Structures/Configuration Trade Summary



CONFIGURATION Δ's

PROPELLANT REDUCTION 57,334 kg (126,400 lb) INERT WEIGHT REDUCTION 5,413 kg (1,933 lb) WEIGHT REDUCTION FROM REDUCED DRAG & TANK SIZE (2,605 lb) LH_2 TANK VOLUME DECREASE 116 m^3 (4,095 ft^3) LO_2 TANK VOLUME DECREASE 39.79 m^3 (1,405 ft^3) LH_2 TANK (BODY) AREA DECREASE 37.16 m^2 (400 ft^2) LO_2 TANK (WING) THICK DECREASE .5%	REQUIRES THAT ENGINE GIMBALLING BE MECHANICALLY COUPLED OR PROGRAMMED TO PRECLUDE NOZZLE INTERFERENCE INCREASED NOZZLE WGHT 564 kg (1243 lb)

Figure 36 Extendable Nozzle Trade Study

STRUCTURAL CONCEPT/TPS SYSTEM (CONCEPTS ARE DEFINED IN FIGURE B & IN REF. C FIGURES 9 & 14)							
CONCEPT 1 (BASELINE)		CONCEPT 2 (ALTERNATE)		CONCEPT 3 (ALTERNATE)		CONCEPT 4 (ALTERNATE)	
AREA	TECHNOLOGY ASSESSMENT	AREA	TECHNOLOGY ASSESSMENT	AREA	TECHNOLOGY ASSESSMENT	AREA	TECHNOLOGY ASSESSMENT
MATERIALS • TITANIUM • RENE 41 • ADVANCED SUPER ALLOYS • COMPOSITES • REFRACTORY METALS • BRAZE ALLOYS	• STRENGTH IMPROVEMENT • INCREASE IN TEMP CAPABILITY • TOUGHNESS IMPROVEMENT • HIGH TEMPCOMP. STRENGTH & DENSITY • PRODUCIBILITY	MATERIALS • TITANIUM • COMPOSITES • REFRACTOR-IES (TPS-RSI) • BOND LINE MATERIAL • SURFACE COATINGS	• STRENGTH IMPROVEMENT • INCREASE IN TEMP CAPABILITY • TOUGHNESS IMPROVEMENT • HIGH TEMP COMP STRENGTH & DENSITY • PRODUCIBILITY	MATERIALS • TITANIUM • COMPOSITES • REFRACTORY METALS • TPS • SURFACE COATINGS	• STRENGTH IMPROVEMENT • INCREASE IN TEMP CAPABILITY • TOUGHNESS IMPROVEMENT • HIGH TEMP COMP STRENGTH & DENSITY • PRODUCIBILITY • SERVICE LIFE	MATERIALS • ALUMINUM • COMPOSITES • REFRACTOR-IES (TPS-RSI) • BOND LINE MATERIAL • SURFACE COATINGS	• STRENGTH IMPROVEMENT • INCREASE IN TEMP CAPABILITY • TOUGHNESS IMPROVEMENT • HIGH TEMP COMP STRENGTH & DENSITY • PRODUCIBILITY
STRUCTURAL ELEMENTS • HONEYCOMB • TUBES • FRAMES	• BRAZING PROCESSES • FABRICATION • STRENGTH	STRUCTURAL ELEMENTS • HONEYCOMB • TUBES • FRAMES • TPS • INSULATED PANEL	• BRAZE MATERIALS • FABRICATION • BONDING • ENVIRONMENT PROTECTION • SONIC LIFE • DAMAGE SENSITIVITY • CRYO PUMP	STRUCTURAL ELEMENTS • HONEYCOMB • TUBES • FRAMES • TPS	• BRAZE MATERIALS • FABRICATION • DAMAGE SENSITIVITY • ATTACHMENTS • CRYO PUMP	STRUCTURAL ELEMENTS • TUBES • FRAMES • TPS • INSULATED PANEL	• FABRICATION • BONDING • SONIC LIFE • DAMAGE SENSITIVITY • CRYO PUMP
STRUCTURAL ASSEMBLIES • VEHICLE COMPONENTS	• WELDING • BRAZING • MACHINING (ASSEMBLY) • ASSEMBLY	STRUCTURAL ASSEMBLIES • VEHICLE COMPONENTS	• WELDING • MACHNING • BONDING • ASSEMBLY	STRUCTURAL ASSEMBLIES • VEHICLE COMPONENTS	• WELDING • COATING • MACHINING (ASSEMBLY) • ASSEMBLY	STRUCTURAL ASSEMBLIES • VEHICLE COMPONENTS	• WELDING • BONDING • MACHINING (ASSEMBLY) • ASSEMBLY

Table 1 Vehicle Structural Concept Synthesis

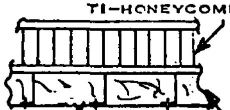
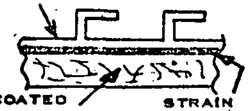
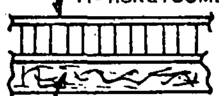
CONCEPT	STRUCTURAL-TPS SUBSYSTEM APPROACHES (LOWER SURFACE)	WEIGHT kg/m² (lb./ft²) PANEL+INS +FRAME	INSULATION ONLY	Δw kg/M² (lb/ft²) OVER BASELINE	MEETS INTENT OF PARAGRAPH 5.4.5 NORMAL GROWTH	ADVANCED TECH GROWTH	COMMENTS
1	BOEING BASELINE  BRAZED RENE 41 HONEYCOMB	12.35 (2.53)	—	0	YES	YES	• FULLY REUSABLE • LIGHTWEIGHT • REQUIRES DEVEL. SUPER ALLOY HONEYCOMB
2	AL-BRAZED TI-HONEYCOMB  COATED SILICA HRSI; STRAIN ISOLATOR SYSTEM	19.77 (4.05)	10.01 (2.05)	7.32 (1.5)	NO	POSSIBILY	• REDUCES RE-ENTRY STRUCTURAL THERMAL GRADIENTS • BRAZED TI-HONEYCOMB AVAILABLE • EROSION • REUSIBILITY REQUIRES DEVELOPMENT • HIGH IN WEIGHT • DAMAGE SENSITIVITY • CRYO PUMP • WATER INGESTION
3	AL-BRAZED TI-HONEYCOMB  COATED REFRACTORY ALLOY; Q-FELT INSULATION	22.85 (4.68)	10.74 (2.2)	10.74 (2.2)	YES	YES	• REDUCES RE-ENTRY STRUCTURAL THERMAL GRADIENTS • BRAZED TI-HONEYCOMB AVAILABLE • HIGH WEIGHT
4	ALUMINUM  COATED SILICA HRSI; STRAIN ISOLATION SYSTEM	22.61 (4.63)	11.72 (2.4)	10.25 (2.1)	NC	POSSIBILY	• STRUCTURAL SYSTEM AVAILABLE • HIGHWEIGHT • (SEE CONCEPT 2) • STRAIN ISOLATION DEVELOPMENT FOR CRYO TEMP.
5	BRAZED TI-HONEYCOMB  ABLATOR	18.8 (3.85)	9.15 (1.875)	6.35 (1.3)	NO	NO	• NOT REUSABLE WITHOUT COMPLETE REFURBISHMENT

Table 2 Structural/Thermal Concept Trade Study Summary

Structural Sizing for Weights. -Section cuts required for a bottoms-up weight analysis of the SSTO vehicle structure are shown in Figure 37 Weights of these representative sections are scaled to other areas of the vehicle to determine overall structures weights. Sections are designed to the level of detail shown by the figures on the following pages.

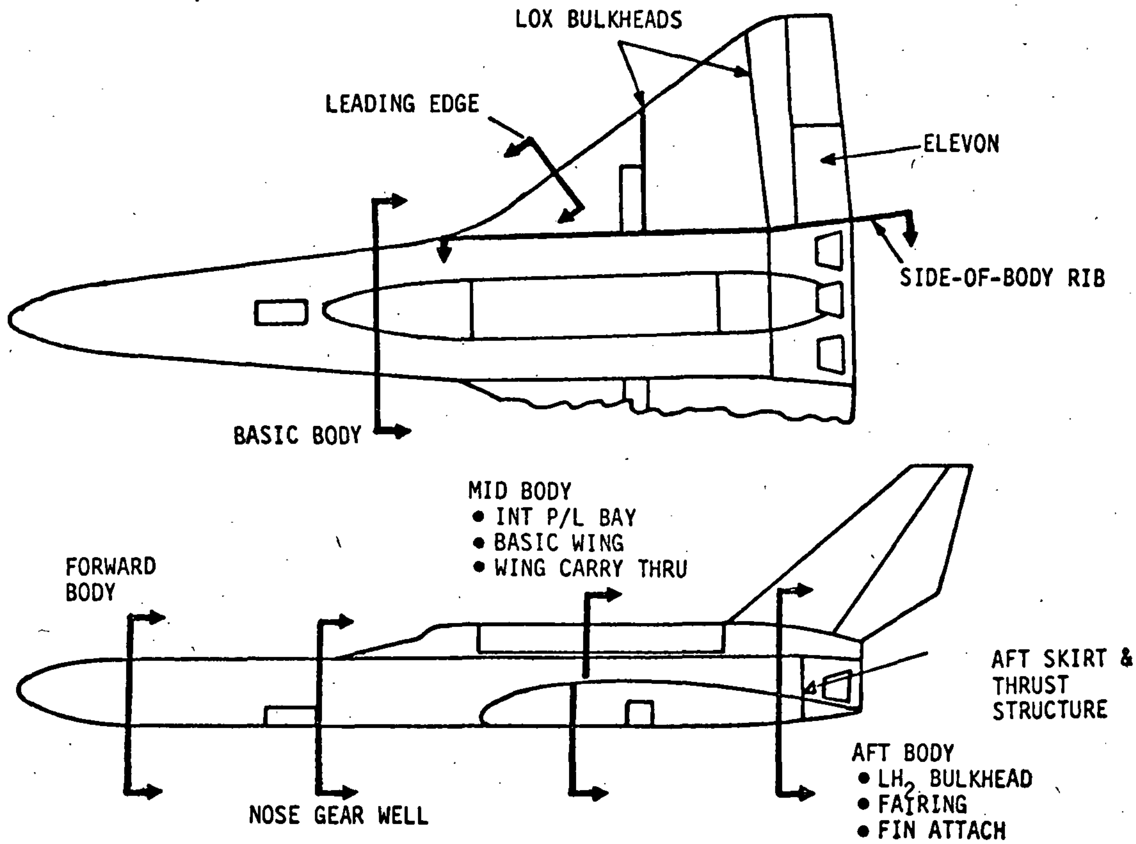


Figure 37 Structural Section Analysis

The body shell is made up of an aluminum brazed titanium (6AL-4V) honeycomb upper shell and a Rene'41 lower shell. The external lower surface contains longitudinal (fore-aft) slots on five inch centers to minimize radial and reduce longitudinal stresses resulting from thermal gradients. The slots which are shown in Figure 38 may be covered with an overlapping foil designed to not provide radial continuity. The upper frames and trusses are titanium. The lower frames are Rene'41. The inner chords of the frames are stabilized at each truss attach point.

The low thermal conductivity of the honeycomb prevents formation of liquid air on the outside of the body and prevents excessive boil off of the hydrogen.

The honeycomb panels beam the pressure loads between frames and carry body bending shear and axial loads. The frames beam pressure loads between truss points and carry axial load induced by surface pressure and truss action, see Figure 38.

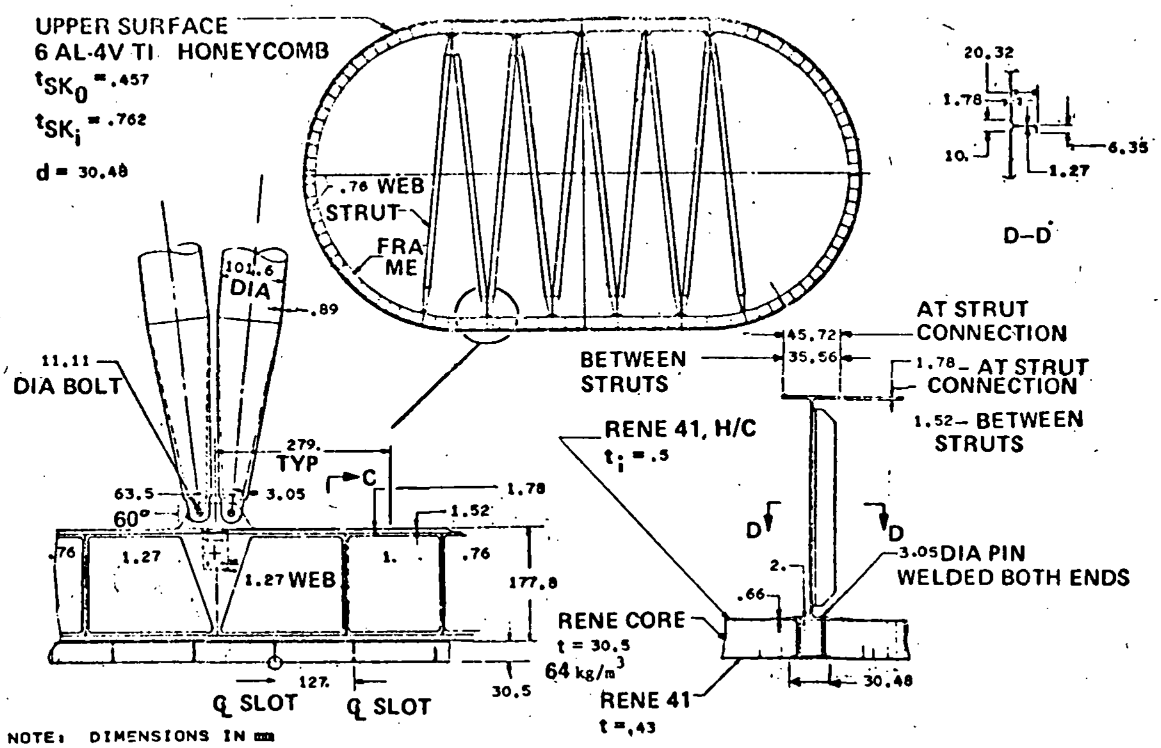


Figure 38 Typical SSTO Body Structural Section

Wing Structure. The wing shell shown in Figure 39, like the body, is made up of an aluminum brazed Titanium (6AL-4V) honeycomb upper surface and a brazed Rene'41 honeycomb lower surface. The upper spar chords and spar trusses are titanium and the lower chords are Rene'41. The inner side of the upper shell is gold coated to control heating from the lower surface.

The structural sizes were developed considering time dependent combinations of flight and pressurization loads.

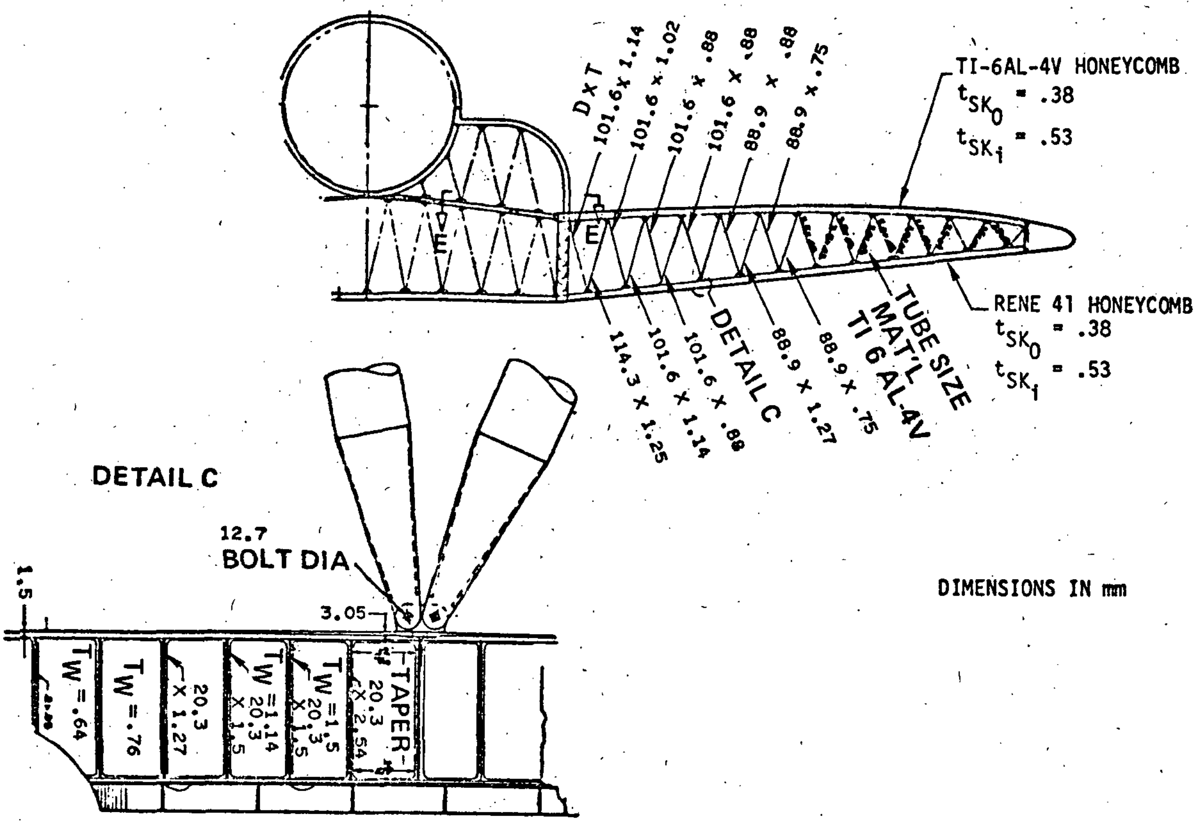


Figure 39 Typical SSTO Wing Structural Section

The surface panels beam pressure loads between spars, carry shear loads and both inboard-outboard and fore and aft axial loads. The surfaces are highly stabilized. The low conductivity of the honeycomb controls the LOX boil-off rate.

Payload Bay. The payload bay structure shown in Figure 40 uses the same construction as the wing and body. Upper shells, including the payload bay shell, are of titanium. The trusses are titanium. The lower shell is Rene'41. Chords on the shells are of the same material as the shells. The use of boron is also reflected in the sizing. Time-dependent combinations of flight and pressurization loads were imposed on the truss structure. The resulting sizing was for tension with some light compression loads. The honeycomb structure between LH_2 and LO_2 is sized to meet reliability requirements consistent with avoiding mixture of the fuel components.

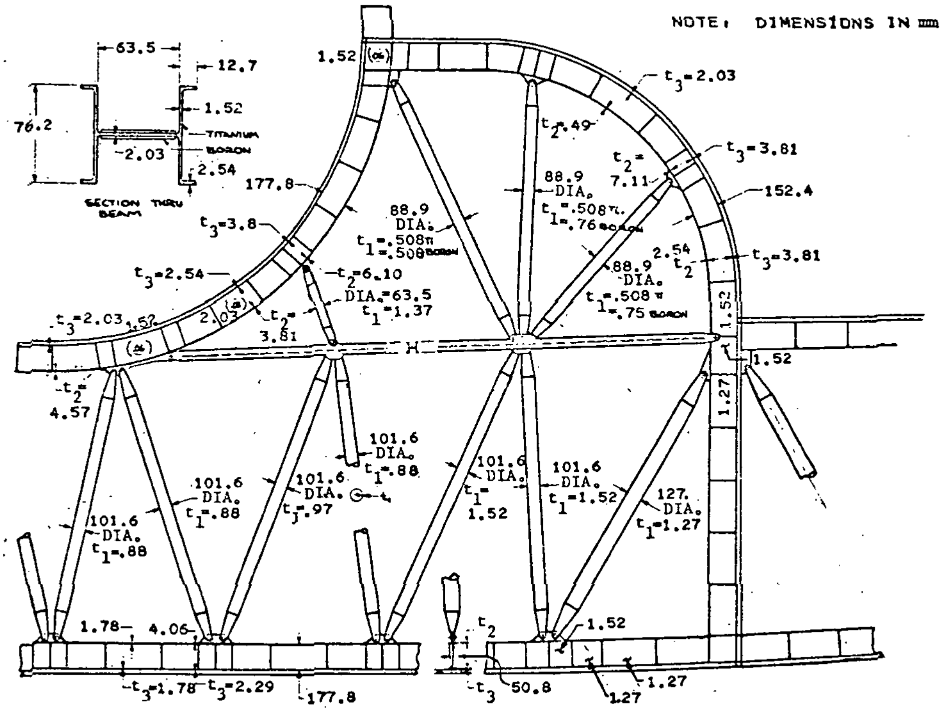


Figure 40 Typical SSTO Payload Bay Structural Section

Thrust Structure. The thrust structure shown in Figure 41 is integral with and just aft of the hydrogen tank bulkhead. As a consequence, it is designed by a combination of tank pressure and engine thrust loads. The engine thrust loads are made up of point loads applied at a single thrust post and two actuator attach points for each engine. In addition to the tank bulkhead, the thrust structure is made up of one horizontal and three vertical beams. These beams collect all the thrust loads and most of each pressure load and transmit them to the body shell where they are reacted. Two constructions are presently being considered. One is made up completely of titanium web and stiffener construction. The other uses titanium trusses rather than webs and stiffeners for the beams. Carbon epoxy constructions and design optimization utilizing built-up truss members have been examined and will be considered in terms of their effect on vehicle dry weight prior to selection of a single thrust structure design. The weight is based on sizing which considers time dependent combinations of engine thrust and tank pressurization loads.

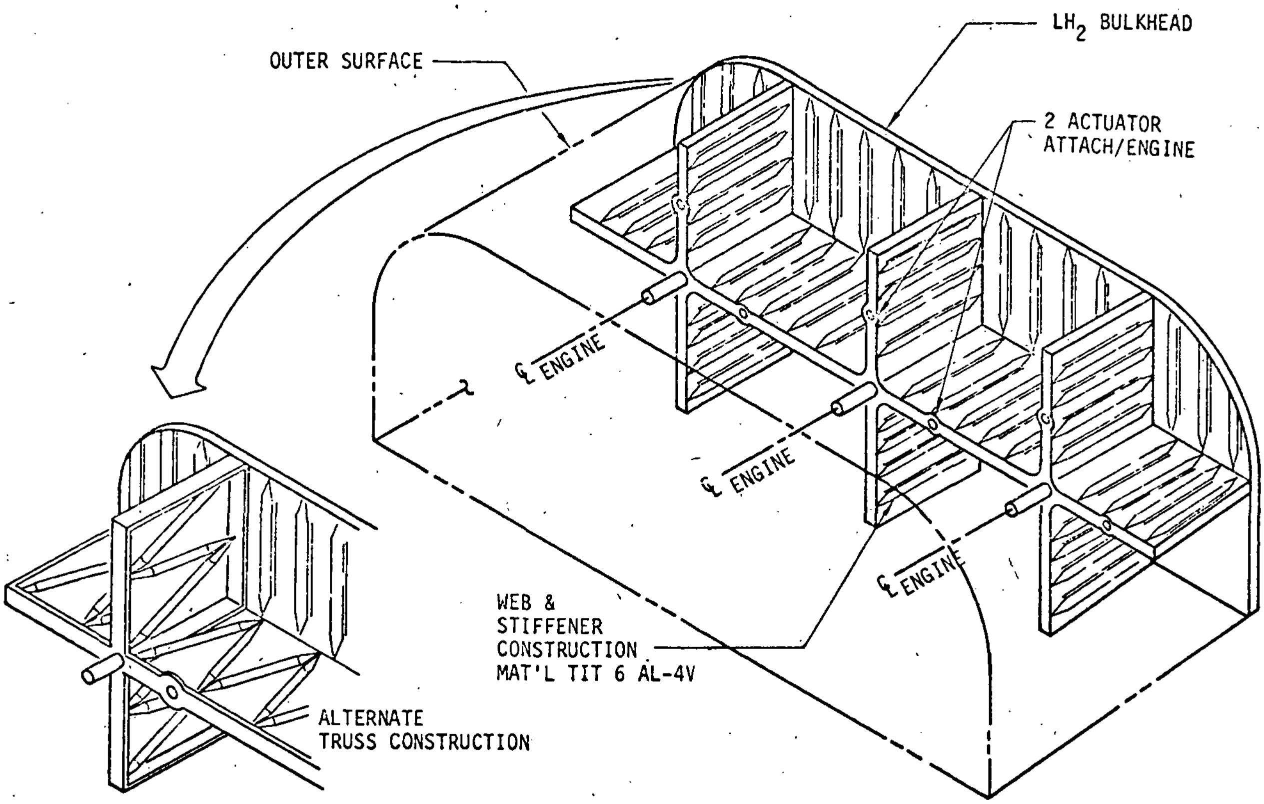


Figure 41 Typical SSTO Thrust Structure Details

Elevons. The elevon shown in Figure 42 is a single cell torque box consisting of honeycomb upper and lower surfaces and sine-wave welded front spar.

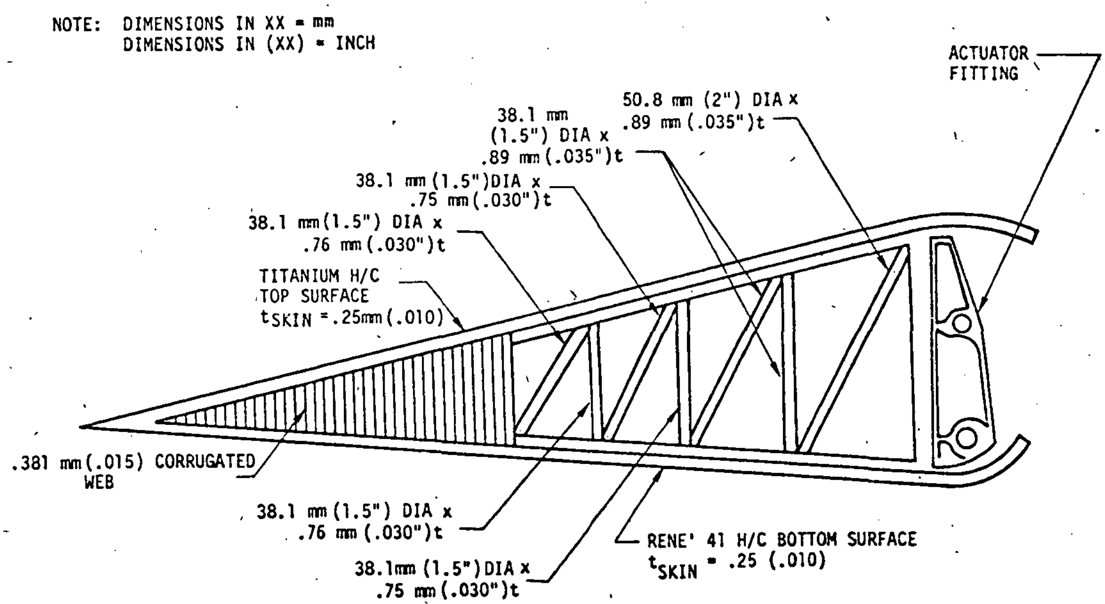


Figure 42 Typical SSTO Elevon Structural Section

The upper surface honeycomb is titanium and the lower surface honeycomb is Rene'41. They carry surface pressures to the ribs and in-plane shear and axial loads. The spar and ribs distribute surface loads and react actuator and hinge loads.

Each elevon is actuated by a dual tandem actuator. The actuator and elevon hinge centerlines coincide; the large inherent torsional capability of the elevon permits actuation from one location.

Wing Leading Edge. The wing leading edge shown in Figure 43 is made of 12-inch wide surface segments mounted on truss supported edge members. The edge members are segmented at each wing spar bay. The trusses provide a determinate support for the panel edge members.

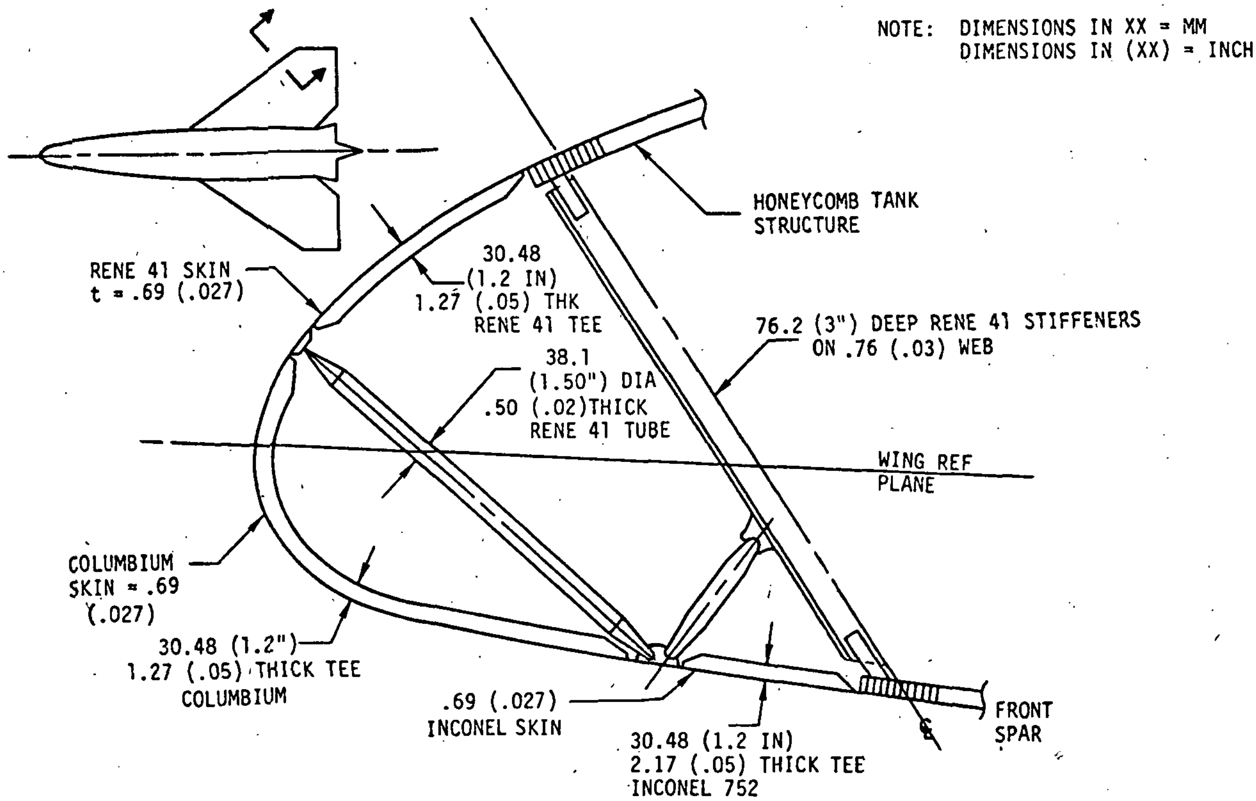


Figure 43 Typical SSTO Wing Leading Edge Structural Section

The slurry coated Columbium leading edge will have a long service with the maximum operating temperature being approximately 1421K (2100^{o}F). The other structural elements thermal environments are well within the material use temperatures.

The leading edge surface segments are overlapped to insure boundary layer inflow blockage. All materials are considered "State of the Art".

Vertical Tail. The majority of the vertical tail of the HTO is constructed of titanium. Construction of the rudder and speed brakes is similar to that of the elevons except that the material is all titanium rather than titanium on one side and Rene'41 on the other. The fin is of titanium from the rudder-speed brake to the front spar. Construction forward of the front spar is, like the wing leading edge of Inconel 753 and slurry coated columbium.

Crew Compartment. The crew compartment is made up of a pressurized cylindrical shell airlocked to the payload bay and within an aerodynamically shaped external fairing.

The cylindrical portion is essentially a 2-1/2 m diameter by 4 m long cylinder with a 1 m diameter airlock. It is a frame and skin aluminum construction cooled by a water wall and supported at thermal expansion accommodating attach points. The compartment fairing is of titanium honeycomb construction with primary attachment to the LH_2 tank honeycomb at frame stations.

Ground Accelerator

Ground Accelerator Benefits. A ground accelerator is utilized for take-off and is a very critical and important part of a horizontal take-off SSTO concept. The SSTO vehicle has a lift off wing loading of approximately 11012 Pa (230 psf.). This configuration has a buffet limit of approximately 11491 Pa (240 psf) at velocities above 158 m/s (520 fps) and a limit of $\alpha = 18^\circ$ below 153 m/s (520 fps). A more desirable α of 12° to 14° requires a lift off velocity of 182 m/s (600 fps) with a thrust to GLOW ratio of approximately .79; a take-off run in excess of 38 seconds would be required using in excess of 53070 kg (117,000 lb) of fuel. Another advantage is the weight savings in a landing gear designed for landing, with not full gross weight. Estimated savings are 35380 kg (78,000 lb), assuming a parametric relationship for landing gear of 4% gross weight. Additional weight is saved by the support capability of the accelerator struts which prevent excessive takeoff loads in the flight vehicle structure. This total savings in inert weight including tankage and structure is in excess of 42633 kg (94,000 lb) or only slightly less than 30% of the baseline injected weight. In addition, the powered accelerator permits take-off from several presently available runways because its T/W ratio is nearly twice that of the flight stage. Finally, it provides 6^{o} of vehicle rotation at no cost in terms of weight to the flight stage.

A significant effort has been expended in the past on various ground accelerator configurations and launch sites. A summary review shown in Table 3 of the elements of such a system was made to select the lowest cost, lowest flight vehicle impact, and most credible system. The selected system is identified as the baseline. Some of the factors influencing the baseline selection are also identified in Table 3.

SURFACE	• RUNWAY - STANDARD - DRY LAKE TRACK - RAIL - CONCRETE TROUGH - WATER FILLED - DRY LAKE OR OCEAN	PROPULSION	• ROCKET ENGINES - SOLID - LIQUID LINEAR INDUCTION CABLE - CATAPULT AIR BREATHER ENGINES WITH AFTER BURNERS STEAM, EXPULSION JET UNPOWERED (AIR VEHICLE POWER)
• VEHICLE SUPPORT	• WHEELS/TIRES ROLLERS SKIS/SKIDS/SLEDS/FOILS SURFACE EFFECT MAGNETIC LEVATATION	DECELERATION	• AERODYNAMIC • MECHANICAL • HYDRAULIC ROCKET CABLE • BARRIER RESTRAINT
• VEHICLE GUIDANCE	• ELECTRONIC - RADAR - LASER MECHANICAL - RAIL - CABLE PNEUMATIC	AERO INTERFERENCES	ROCKET ENGINE GROUND PLANE AIR VEHICLE - COLLISION

• BASELINE

Table 3 Ground Accelerator Configurations Summary

Surface. It was considered desirable that the vehicle system utilize, to the maximum extent possible, existing facilities. This would then encompass the large military air fields, runways in excess of 3658 m (12,000 ft.), large dry lake beds and the lakes or oceans. The lakes or oceans were eliminated due to probable adverse wave action and the related problem that any spray would immediately form ice on contact with the cryogen filled launch vehicle. The track, guideway or trough were eliminated due to the high cost of installation.

Vehicle Support. Magnetic levitation was eliminated due to high development and installation costs. Surface effects including ram cushion and pressurized air cushion were examined and eliminated for control and stability reasons. As dynamic pressure approaches cushion pressure, severe pitch instability occurs. Ski/skids/sleds/foils have all had significant investment in the past for various high speed sled systems. However all have shown limited life as well as an inability to accept normal surface irregularities which inhibit the system utility. Rollers, either fixed axle or translating axle, offer promise. However, development problems associated with the footprint shape and loading as well as alignment stability problems made the selection of the standard wheel/tire configuration the most desirable.

Vehicle Guidance. The flight crew work load and launch environment are such that automatic guidance was considered to be a very desirable feature of the ground accelerator system. For this guidance, minimum impact on existing facilities was a desired constraint. Both of the mechanical and pneumatic concepts would require significant revision to any existing facility to accept these systems. The electronic system, either laser or radar, could be vehicle contained with simple reflector antennas located at the end of the runway. This feature also permits increased launch site location flexibility.

Propulsion. The linear induction and cable systems were eliminated because of the large estimated development costs and the requirement that significant facility impacts would be required. Air breather engines were not considered a reasonable approach due to the large thrust levels necessary requiring a large number of engines with the consequent complexity. The steam expulsion jet provided an interesting option, however this system severely constrained launch time due to necessary pre-heat time. The unpowered sled or dolly is an acceptable option, however this does impose a significant penalty to the flight vehicle as noted earlier. The solid rocket engine is an acceptable approach with two adverse factors, adverse environmental effects and an inability to modulate thrust levels of a given motor. This latter factor can be accommodated by utilizing the flight vehicles' engines to provide the necessary thrust modulation to provide an acceleration level which does not significantly impact the flight vehicle structure due to acceleration loads imposed by the fuel. The liquid rocket engine appeared to offer the optimum solution requiring little if any development investment.

Deceleration. The desirability of a low cost reusable vehicle deceleration system providing a controlled and predictable deceleration schedule without excessive loadings dictated the system selected. The cable and the rocket systems were eliminated for reasons of site location flexibility, cost and controllability; as was the water trough system. The system selected uses a combination of aero surfaces acting as lift spoilers and drag brakes, a parachute system consisting of a metallic drogue and fabric main chutes sequentially deployed, a hydraulically actuated skid plate system, and finally a net barrier restraint system which functions as a fail safe reserve system.

Aero Interferences. Additional configuration concerns which have been examined and tested but which will require further investigation include the rocket engine plume impingement and possible deleterious effects on the runway surface, the ground plane effects on both the flight vehicle and the ground accelerator vehicle creating significant control problems providing the potential for collision immediately after lift-off, and the unpredictable interference effects associated with such a system in a high subsonic aero regime. Preliminary examinations and wind tunnel tests have indicated solutions to these problem areas.

Operations. The artist's concept shown on Figure 44 illustrates the major points of the ground accelerator launch profile and the operational concept. The initial acceleration is approximately 1.2 g resulting in reaching the separation velocity of 183 m/s (600 fps) in a little over 16 seconds approximately 149m (4900 feet) down the runway. The air vehicle is rotated to 13° by the ground accelerator and separated. The sled engines are shut off, the drogue chute deployed, aero drag flaps deployed, and skid plates extended to stop the accelerator approximately 4145 m (13,600 feet) from the start. The restraining barrier provides a backup.

An abbreviated description of the salient features and constraints of the operational concept is shown on Table 4 to describe the launch and recovery sequence. Preliminary analyses have been conducted in most of the areas to identify the major concerns.

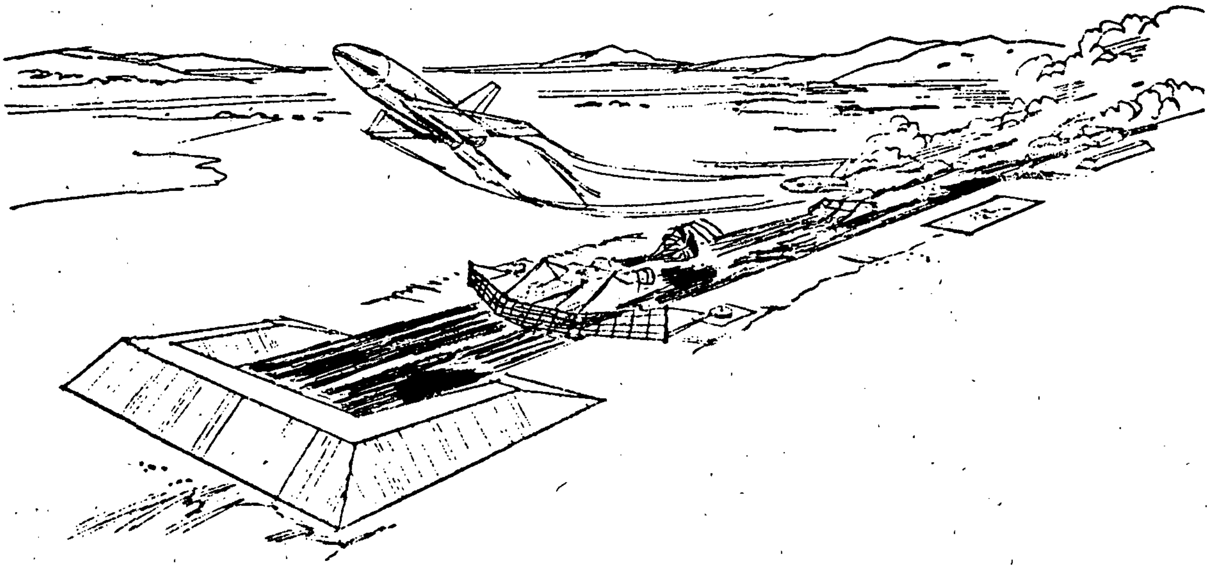


Figure 44 Ground Accelerator/SSTO Launch

• PREFLIGHT	• PERFORMED IN ASSEMBLY BUILDING
• ASSEMBLY	• VEHICLE MOUNTED ON GROUND ACCELERATOR
• TOW TO TAKE-OFF PAD & ATTACH TO RESTRAINT	• GROUND ACCELERATOR SUPPORTED ON SKID PLATES TO PREVENT FLAT SPOTS ON TIRES - RESTRAINT ATTACHED TO ACCELERATOR
• FUELS AND PROPELLANT LOADING	• CRYOGENIC STORAGE
	• MINIMUM CLEARANCES 2743.2m (9,000 FEET) 3658m (12,000 FEET)
• CREW/PAYLOAD BOARDING	• PRE-TAKEOFF CHECKLISTS - AUTOMATIC
• IGNITION - RELEASE	• COOLING WATER TO PAD
	• UMBILICAL DISCONNECTS
	• ARM RUNWAY COOLING SYSTEM
	• SKID PLATE SUPPORTS RETRACTED
	• RELEASE
• TAKE-OFF ABORT	• DISTANCE TO STOP - 710m (2,330 ft)
	• VELOCITY - 30.5 m/s (100 fps)
	• SHUT-DOWN ALL ENGINES - SKID PLATES ACTUATED - COOLING FOG ACTUATED - REMOTE CREW/PAYLOAD REMOVAL - BOILOFF
	• UMBILICAL ATTACHMENT - FUEL/PROPELLANT PUMP DOWN - INERTING
• TAKE-OFF & CLIMB	• MAINTAIN HEADING FOR MINIMUM CLEARANCE OF 365 m (12,000 ft) TERRAIN OR POPULATED AREAS - CLIMB TURN TO ORBIT HEADING
• GROUND ACCELERATOR	• DROUGHE CHUTE STABILIZATION
	• DRAG FLAPS DEPLOYED
	• SKID PLATES ACTUATED AS OVERRUN BARRIER CONTRACTED
	• COOLING FOG INERTING IN REVETMENT
	• REMOTELY OPERATED TUGS & MANIPULATORS COMPLETE INERTING
• TOW BACK TO ASSEMBLY BUILDING	• REFURBISHMENT
• SSTO RECOVERY	• TOW TO CREW/PAYLOAD DOCK
	REMOVE CREW/PAYLOAD
	TOW TO PREFLIGHT DOCK

Table 4 HTO-SSTO Launch Sequence

Configuration. The general configuration is as shown in Figure 45 and 46. Two F-1 engines locked in the position shown provide takeoff thrust. (Later studies resulted in the selection of three SSME engines as a more desirable option, based on cost and performance). The engines are pointed up and out to reduce plume interference effects on each engine, the vehicle, and the runway. Metallic drogue chutes and drag flaps provide aerobraking for deceleration with final braking provided by hydraulically actuated skid plates. The accelerator incorporates an integral guidance system which steers the front wheels to maintain the proper runway track. Four support pads support the flight vehicle. Two struts support the vehicle body; one located under the forward LO_2 tank and one at the aft end of the hydrogen tank. Two outrigger pads provide support in the wings during the take-off run.

Also shown is a cutaway view of the ground accelerator tire. The unconventional design is proposed because of the high loads at high speeds. It utilizes radial/axial glass ply cores with a solid core of silastic

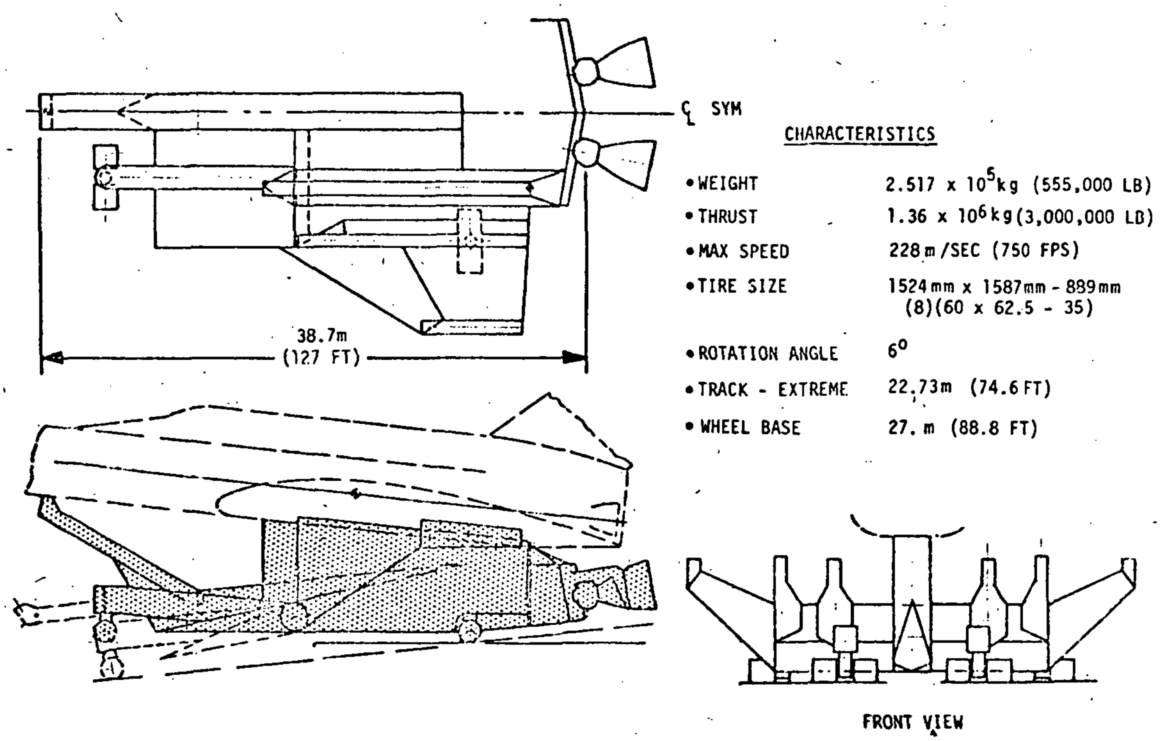


Figure 45 HTO-SSTO Ground Accelerator/Sled Configuration

foam. The outer plies are locked to a cast epoxy core at intermediate points inside the outer walls. The wheel/tire design for the ground accelerator represents an advanced technology approach. However, all of the features illustrated are typical of high-speed high wheel loading tires. The low aspect ratio serves to reduce the localized footprint loadings limiting maximum runway loadings.

The integral wheel/tire design approach permits simpler, lighter and stronger combinations in that assembly/disassembly provisions are eliminated as well as the additional life provisions normally designed into wheels. The internal drop thread webs provide additional load carrying capability without additional sidewall thickness. The foam core provides a more predictable loading pattern somewhat reducing the design task.

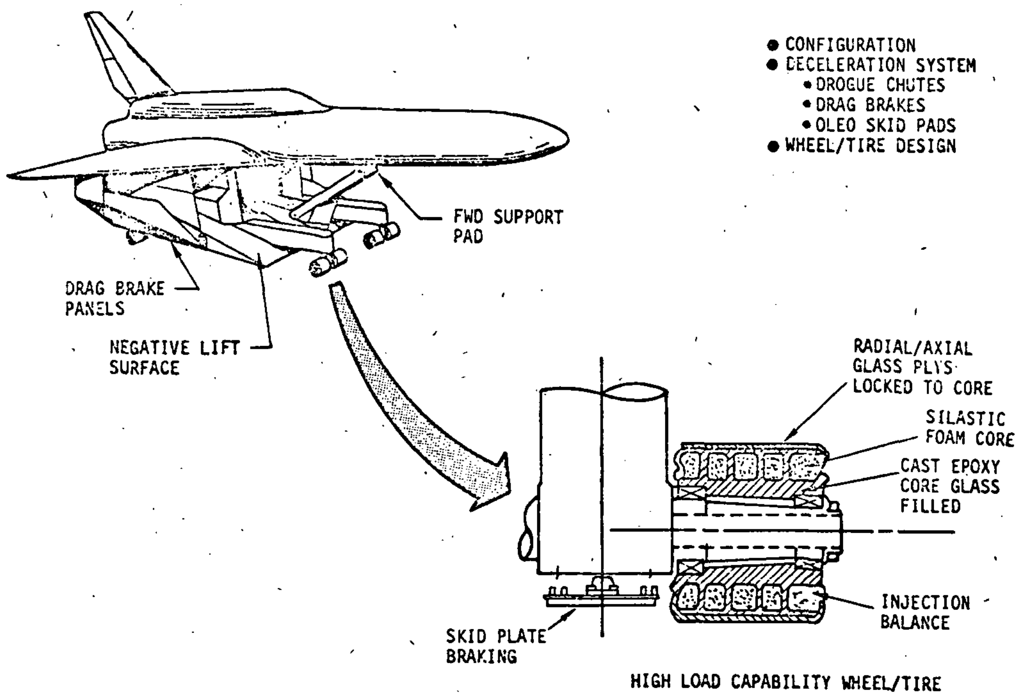


Figure 46 Ground Accelerator Development Requirements

Subsystems Installation Arrangement

The general arrangement and location of the major elements of the SSTO HTOHL vehicle are shown on Figure 47. The SSTO VTOHL subsystems arrangement is similar with the principal difference being that of size. The profile illustrates the sizing of the particular element as well as significant features or developments for each. Specific discussions of each element will follow.

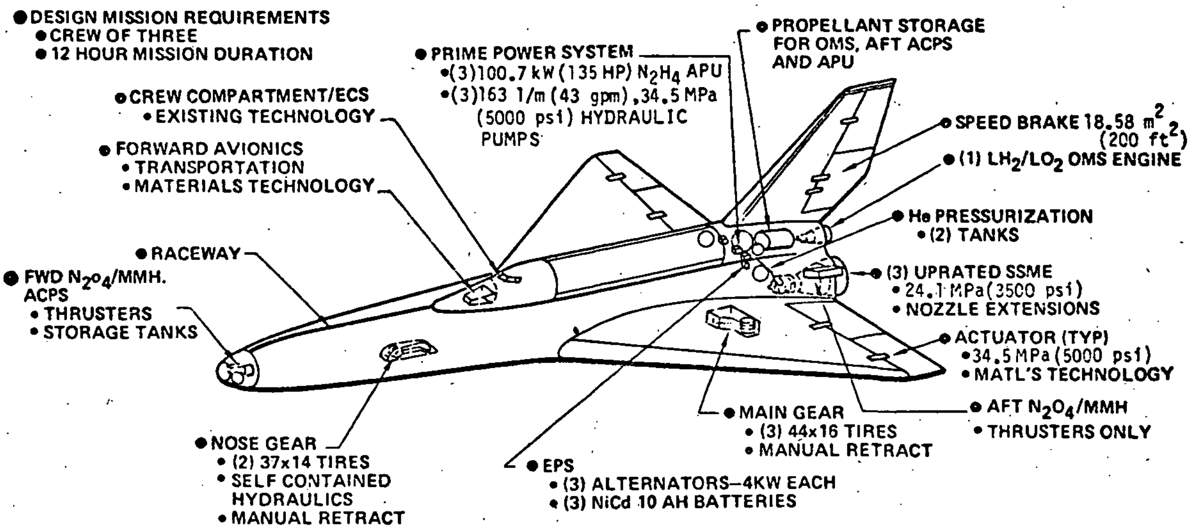


Figure 47 SSTO Subsystems Arrangement

Secondary Power Generation. The baseline secondary power generation is based on utilizing the hydrazine APU developed for the shuttle shown on Figure 48. The low weight-to-power ratio, .36 kg/kW (.59 lb/hp) and the low average specific fuel consumption, approximately .85 mg/W-sec (5 lb/hp hr) make this unit a very desirable option. Three of these units located in the equipment bay aft of the payload bay provide the prime power for the electrical and hydraulic system. A single hydrazine tank, approximately 1.37 m^3 (48.4 ft^3) and weighing 69 kg (152 pounds), contains 132 kg (292 pounds) of hydrazine to be supplied to the three APU's as well as the aft RCS at 2.76×10^6 Pa (400 psi) pressure. APU cooling is provided by propellant bleed flow on ascent and by high-pressure water boilers on orbit and during reentry. The exhaust is directed out either side of the body to eliminate any thrust moments on the vehicle. The APU schematic Figure 49 reflects the general arrangement of each APU as well as the weights associated with each APU. Power demands on each APU range from a peak of 97.98 kW (131.4 hp) during ascent and reentry down to less than 17.15 kW (23 hp) on-orbit.

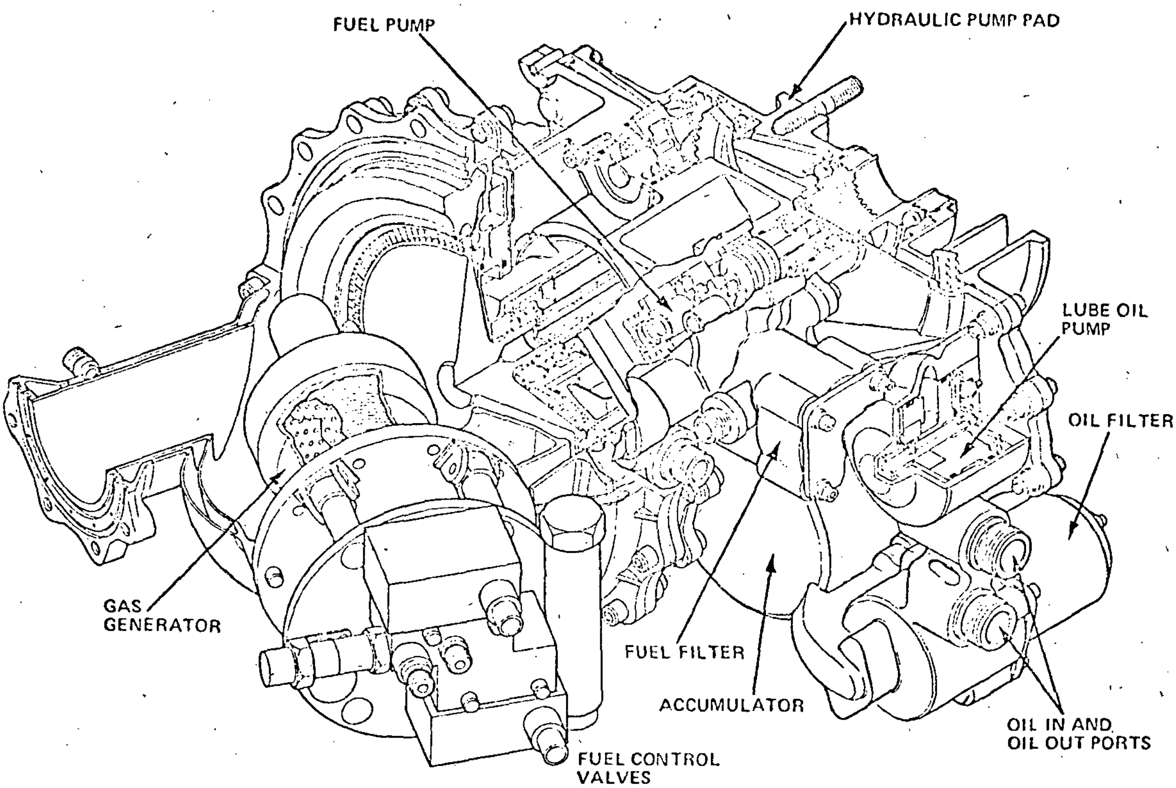
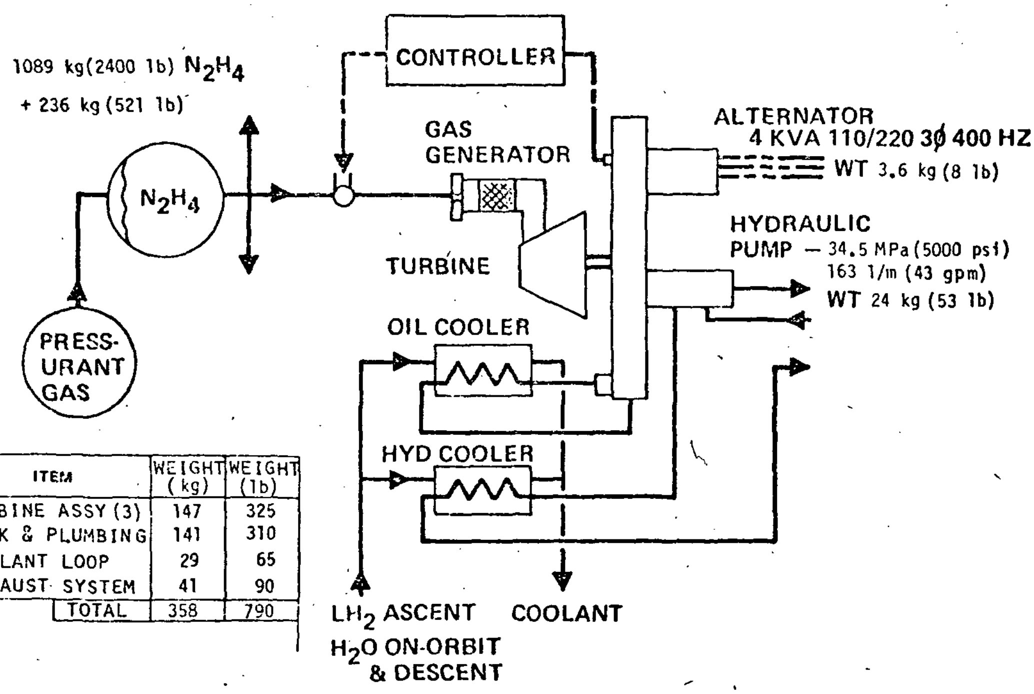


Figure 48 SSTO Auxiliary Power Unit



ITEM	WEIGHT (kg)	WEIGHT (lb)
TURBINE ASSY (3)	147	325
TANK & PLUMBING	141	310
COOLANT LOOP	29	65
EXHAUST SYSTEM	41	90
TOTAL	358	790

 Figure 49 Auxiliary Power System Schematic

Electrical Power Generation and Distribution. The baseline electrical power generation and distribution system is three 110/220 VAC 400 cycle - 3-phase systems with one 4 kW alternator on each APU. Solid state AC to DC converters are used as required at the component. Signaling and control systems are 28 VDC and 8 VDC. Power relays, circuit breakers, and overload protection are either solid state devices or solid state controlled. This approach has been selected to achieve the minimum weight fractions possible; AC power generation and distribution normally shows weight ratios of kg/kW of 10% to 30% that of DC.

The primary power demand is that of the avionics, which is almost a constant 5 kW demand. Another 2.8 kW is required for SSME and TVC control power. An additional small amount of power is required for miscellaneous services such as lighting and subsystems control.

Power and signaling cables are routed forward and aft through separate cableways through the payload bay between the aft equipment bay and the cockpit and forward avionics bay. An external raceway connects the nose equipment bay and the forward avionics bay. An insulated double walled and vented conduit through the hydrogen tank connects the nose gear well and the forward avionics bay. Routing to the main gear well from the aft equipment bay is along the upper surface wing root inside fairings to the gear well. Routing outboard is through the trailing edge area immediately aft of the wing rear spar. Routing up the vertical fin is similarly located aft of the fin rear spar.

An emergency system is provided by the three Ni Cd 10 AH batteries. This DC power is intended to provide minimum lighting, control stabilization, and emergency egress control. This can also be used for APU starting. Prelaunch power and signaling is provided through an electrical umbilical on the right aft upper fuselage surface adjacent to the aft equipment bay immediately aft of the payload bay. Power is provided through this umbilical until the APU's are started and brought on line.

Hydraulic Power Generation and Distribution. The baseline hydraulic power generation and distribution system shown on Figure 50 is three 3.45×10^7 Pa (5000 psi) systems powered by three variable displacement pumps, one on each APU. The hydraulic system reservoirs, thermal conditioning and valving associated with system conditioning are located in the aft equipment bay. The system supplies power to the flight control surface actuators, main landing gear, payload bay doors, and the environmental control fans and pumps. TVC actuator power is supplied by the engines with centering and locking power supplied by the main hydraulic system. The nose gear extension and steering is provided by a self contained monopropellant emergency power unit (MEPU) located in the nose gear well. This MEPU drives a hydraulic pump which is part of an integral hydraulic system to provide the necessary power and control.

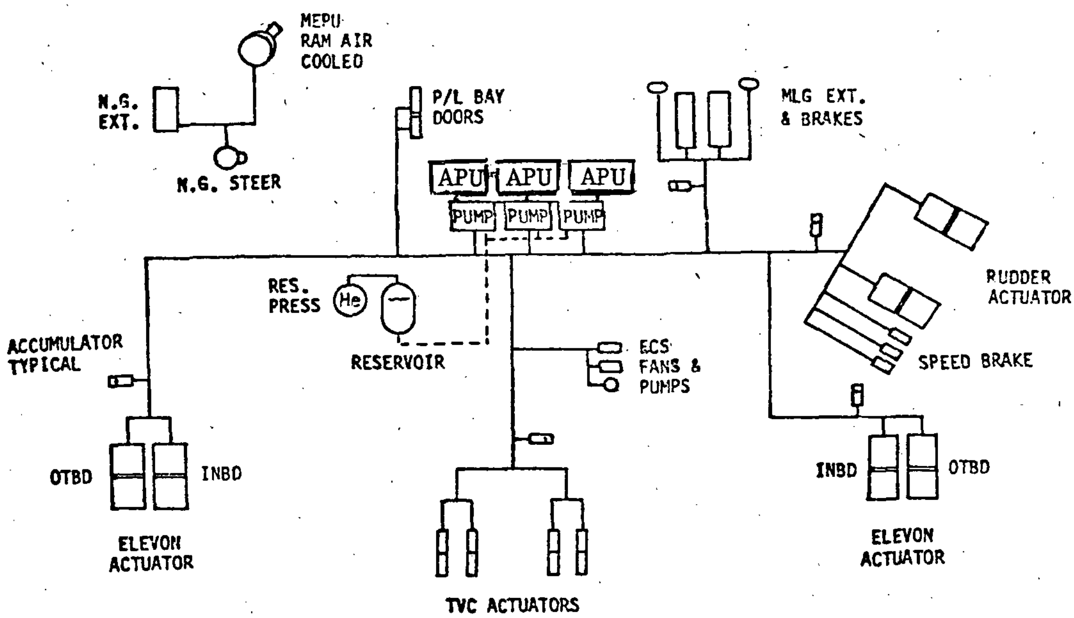


Figure 50 Hydraulic Power System Schematic

The elements of the system appear to be normal state of the art development of existing off the shelf components requiring only the development normally associated with the integration of any hydraulic system.

The demand load profiles are accommodated by accumulators sized to provide rapid response with the minimum prime power. These accumulators will be located immediately adjacent to the individual flight control surface actuators which are the principal demand.

Not shown on the schematic is the prelaunch umbilical attachment which provides power for operational checkout of the system and components as well as providing circulating flow for thermal conditioning. Also not shown are the heat exchangers and water boilers of the thermal control system.

Flight Control System - Aerodynamic. The SSTO vehicle is a computer flown vehicle which in the early phase of ascent is controlled with a combination of TVC and the elevon/rudder surfaces. In the latter phase of ascent, control is provided by TVC and the reaction control system. On-orbit control is by the RCS and orbital maneuvering system engine with de-orbit impulse provided by the OMS. Initial entry control is by RCS with the latter stage of entry including approach and landing flare controlled by aerodynamic surfaces. The Flight Control System Schematic Figure 51 shows a simplified schematic of the system which illustrates the inter-relationship of the various elements of control system. The estimated weight of the surface actuators and their support and installation is also shown.

ITEM	WEIGHT (kg)	WEIGHT (LB)
INBOARD ELEVON (2)	471	214
OUTBOARD ELEVONS (2)	471	214
RUDDER (2)	267	121
SPEED BRAKE (4)	583	264
SUPPORTS/INSTALLATION	179	81
TOTAL	894	1971

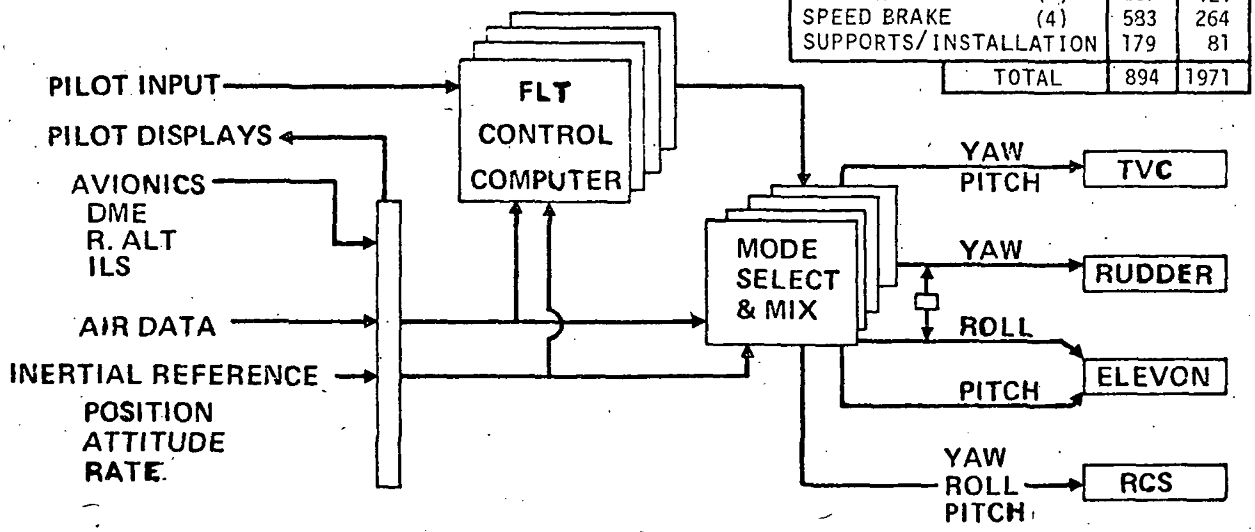


Figure 51 Flight Control System Schematic

The baseline actuator is a dual-tandem actuator powered by two of the three hydraulic systems. Hydraulic control is provided by four electro-hydraulic transfer valves, two of which are always operating in a monitor mode. Four electrical signal channels thus command three hydraulic control channels, with switching, and in turn, hydraulic power channels, with switching. Thus one actuator per surface with multiple surfaces provides the minimum level of redundancy necessary with the minimum weight.

Aero-Surface Actuator. The schematic Figure 52 illustrates the salient control features typical of the aero-surface actuators. All three hydraulic systems are plumbed to each of the actuators. The system selector valve connects two of the four electro -hydraulic valves to one of the systems and one each to the other systems. The four electro-hydraulic valves control the main servo valve which is connected through the servo selector to cylinders I and II. Mechanical linkage provides feed-back to the main servo valve and to the electro-hydraulic control valve where it is summed with the mechanical feedback from the main servo valve to the electro-hydraulic valve. Electrical feed-back from the actuator is directed to the servo amplifier summing which sums, balances, and matches the four channel input and output.

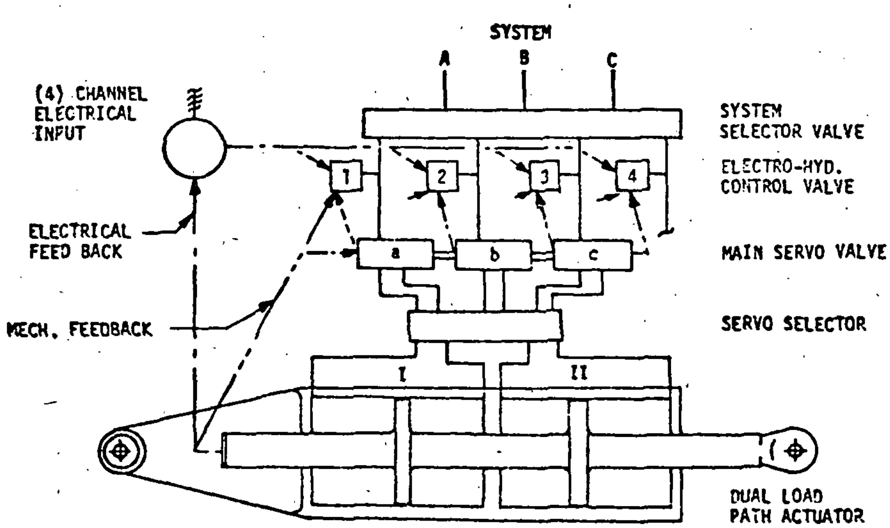


Figure 52 Aero-Surface Actuator Schematic

Other features not illustrated include integral dynamic load dampening, an integral cylinder block and circulation circuit for on-orbit conditioning, over-sized filters with a large particle separator, and a capability through multiple circuitry and linkage deflection to accommodate major control valve malfunctions.

The approach of utilizing three hydraulic systems and four electrical control systems with appropriate switching and selecting assures that full fail operational, fail safe criteria will be met. The switching-selecting logic as well as the sensing and cues to be sensed will be identified as more in-depth failure modes and effects analyses are completed.

Avionics. The avionics baseline shown on Tables 5, 6, 7, and 8 is the result of a very careful review of the minimum required for a space transportation system. A minimum level of redundancy is provided with any additional redundancy or mission peculiar requirements to be met by adding to the basic equipment list and charging this additional equipment as payload. In addition, a conservative estimate of the development of each of these elements has been made to establish a reasonable weight/performance level for the 1986 time frame.

G & N

(1) IMU
(3) TACAN
(2) RADAR ALT
(3) MSBLS RCVR
(1) STAP TRACKER

DATA PROCESSING

(3) COMPUTER
(18) MDM UNITS
(2) MASS MEMORY
(20) EVENT CONTROLLERS
(2) ME/RCS/TVC MONITORS

COMMUNICATION & TRACKING

(1) DOPPLER EXTR
(2) UHF RCVR
(2) 'S'BAND XPNDR
(2) N/W SIG PROCSR
(1) LLLTV

ANTENNAS

RADAR ALT
MSBLS
UHF
(5) 'S' BAND

DISPLAYS & CONTROLS

(1) FLUIDS/GAS DISPLAY & MONITOR
(1) KEYBOARD/PANEL
(1) DISPLAY ELEC/UNIT
(1) DISPLAY DRIVER
COCKPIT CONTROLS & XDCRS
(1) CONTROLS MUX
(3) ENGINE INTERFACE UNITS
(2) AUDIO SYSTEM
(2) AIR DATA XDCRS
(20) SERVO AMPS
(12) RCS DRIVERS

INSTRUMENTATION

(200) PICKUPS & PROBES

Table 5 Avionics System Baseline Equipment List

DISPLAYS & CONTROLS

(1) FLUIDS/GAS DISPLAY & MONITOR	3.6 kg (8 LBS)
(1) KEYBOARD PANEL	8.1 kg (18 LBS)
(1) DISPLAY ELECTRONIC UNIT INTEGRATED CATHODE RAY TUBE DISPLAY	29.9 kg (66 LBS)
(1) DISPLAY DRIVER	9.1 kg (20 LBS)
COCKPIT CONTROLS & XDCRS	8.6 kg (19 LBS)
(1) CONTROLS MULTIPLEXERS	4.1 kg (9 LBS)
(3) ENGINE INTERFACE UNITS	6.4 kg (14 LBS)
(2) AUDIO SYSTEM	6.8 kg (15 LBS)
(2) AIR DATA TRANSDUCERS	1.8 kg (4 LBS)
(20) SERVO AMPLIFIERS	0.9 kg (2 LBS)
(12 RCS DRIVER AMPLIFIERS	0.9 kg (2 LBS)

ANTENNAS

RADAR ALTIMETER	1.4 kg (3 LBS)
MSBLS	6.8 kg (15 LBS)
UHF	4.5 kg (10 LBS)
(5) 'S' BAND	2.3 kg (5 LBS)

INSTRUMENTATION

(200) PICKUPS & PROBES
COAXIAL CABLE AND INSTALLATION PROVISIONS WEIGHT ESTIMATED AT 10% EACH
ELECTRICAL POWER 5 KW WEIGHT 545.4 kg (1,202.4 LBS)

Table 6 Avionics System Weight Statement

GUIDANCS AND NAVIGATION

(1) INERTIAL MEASUREMENT UNIT	34.0 kg (75 LBS)
(3) TACTICAL AIR NAVIGATION	9.4 kg (20.7 LBS) EA)
(2) RADAR ALTIMETERS	3.2 kg (7 LBS) EA
(3) MICROWAVE SCANNING BEAM LANDING SYSTEM	2.3 kg (5 LBS) EA
(1) STAR TRACKER	4.5 kg (10 LBS)

DATA PROCESSING

(3) COMPUTERS	9.4 kg(20.7 LBS) EA
(18) MULTIPLEXER UNITS	2.7 kg(5.9 LBS) EA
(2) MASS MEMORY	5.4 kg(12.0 LBS) EA
(1) EVENT CONTROLLERS	13.6 kg(30 LBS
(2) MAIN ENGINE, REACTION CONTROL SYSTEM. THRUST VECTOR CONTROL MONITORS	6.8 kg(15 LBS) EA

Table 7 Avionics System Weight Statement

COMMUNICATION & TRACKING

(1) DOPPLER EXTRACTOR	2.3 kg (5 LBS)
(2) ULTRA HIGH FREQUENCY RECEIVERS	2.7 kg (6 LBS) EA
(2) 'S' BAND TRANSPONDERS	4.1 kg (9 LBS) EA
(2) NARROW BAND SIGNAL PROCESSORS	4.5 kg (10 LBS)EA

Table 8 Avionics System Weight Statement

An example is the single IMU for the guidance, navigation and control. This unit is a dodecahedron arrangement of six (6) single degree of freedom strapdown gyros providing instrument level redundancy for a significant reduction in weight. This unit is in development for the tug. Similar efforts have been made in each of the areas. The summary sheet identifies element quantities by system with a summary power and weight. The subsequent charts identify element weights within each subsystem.

<u>HTO Main and Nose Gear Configuration.</u> The main and nose gear configurations are shown on Figures 53 and 54 to illustrate the simple, direct approach utilized. The weight savings, which have been accomplished, have been accomplished in three main areas; lowered requirements, improved structural materials, and a simplified design. The actuation system is designed for extension only with gravity and air loads aiding. Maximum advantage is taken of composites and 2.41×10^9 Pa (350 ksi) steel. The landing gear subsystem technology projection chart summarizes the current state of the art for the various elements, as well as an estimate of what improvement can reasonably be expected. The requirements chart Figure 15 illustrates the reduction in requirements which has permitted a significant portion of the overall weight savings. The landing gear weight trends, Figure 21, while nearly chronological, illustrate the general improvement for landing gear systems associated with materials developments. The 747 illustrates the weight penalty imposed by the requirement to limit footprint pressures to existing runways and taxiways necessitating a multiple gear design. The improvement of steels from 1.24×10^9 Pa (180 ksi) to 2.07×10^9 Pa (300 ksi) over the past 25 years has contributed to the weight reduction in both the running gear and structure. Tire design improvements and particularly the current low profile designs are significantly more efficient also contributing to the improved weight fractions. Brake system improvements such as beryllium and carbon/carbon heat sink materials have permitted smaller more compact lower weighted systems. A reasonable weight estimating approach is illustrated on Figure 18 to show the impact on weight of permitting higher brake temperatures. The rejected take off (RTO) condition more closely matches the design condition envisioned for the SSTO. Using the developed relationships, the next chart, Figure 19, illustrates a landing gear weight for the SSTO vehicle as well as the impact of various materials technology improvements. Thus it can be seen that within the relationships and constraints developed, a 2.8% landing gear system is achievable. An

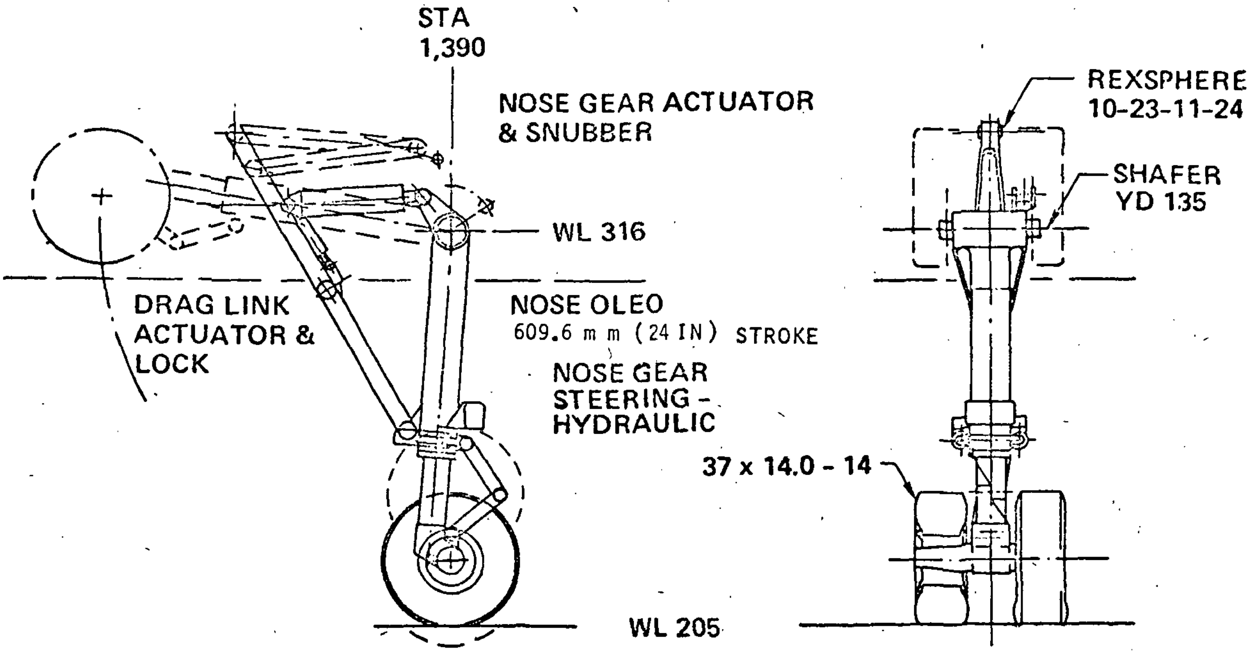


Figure 53 SSTO Nose Gear Configuration

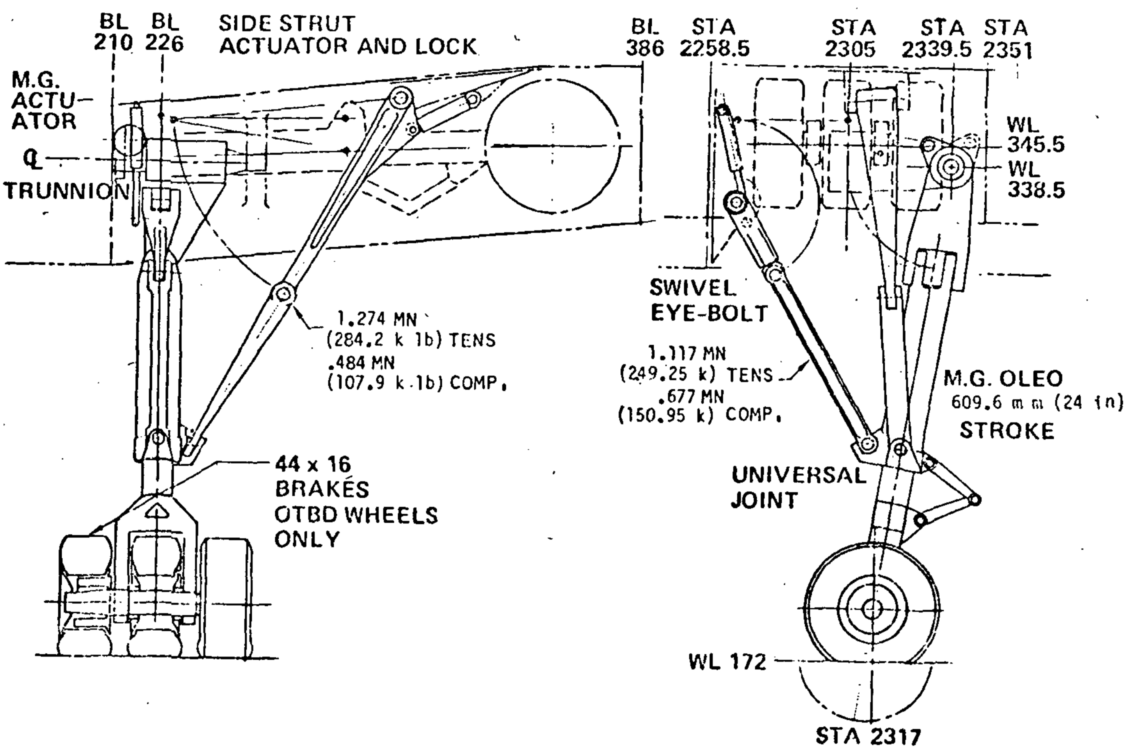


Figure 54 SSTO Main Gear Configuration

additional weight reduction can be achieved through the use of a servo actuated strut replacing the current air/oil oleo. This is discussed in Task III.

Crew Accommodations. Crew accommodations are based on the Space Shuttle definitions with exception that for the SSTO vehicle a significant reduction is accomplished by the reduction in crew size and by the limited mission duration, less than twelve hours. Significant effort has been expended in the development of basic crew accommodations for the shuttle and it is not envisioned that any further development would be required for the SSTO beyond the normal effort required for integration.

Environmental Control System. The environmental control system requires a complex integration of several systems which at this point have not been completely identified. The active system includes thermal conditioning for the landing gear wells, the crew compartment, the avionics bay, the aft equipment bay, the hydraulic system including surface actuators, and the APU systems including gear box, pump, alternator, and fuel. The passive system includes the insulation around the wheel wells, the water walls strategically located to protect the crew compartment, and the insulation locally provided for equipment protection. In addition, heater tapes will be utilized for local problem areas, e.g. wheel wells for the prelaunch cold soak. Preliminary thermal balance assessments of the critical control areas indicate a heating problem requiring the equivalent of approximately 413 kg (910 pounds) of microquartz insulation plus 189 kg (416 pounds) of water to be boiled away. This will accommodate the heating due to external surface heating and subsequent conduction, and re-radiation. In addition, the heat load as a consequence of power dissipation will require approximately an additional 544 kg (1200 Pounds) of water. However, without an in depth analysis and heat balance study, the best approximation can only be made as an extrapolation of the Space Shuttle with a reduction for the reduced time on orbit.

Propulsion

For the purpose of this study growth/uprated versions of the SSME engine were assumed as a baseline. Linear engine analysis and technology projection was conducted in order to assess effects of possible technology perturbation on propulsion system design.

For increased authenticity, a subcontract was given to Rocketdyne in order to provide parametric engine performance and weight data for such growth engines. This encompasses possible trades and extrapolation to advanced technology, including increased chamber pressures, higher vacuum thrust, two-position nozzles, increased expansion ratios and linear engine development and performance.

Main Engines. The main engines are representative of the optimum engine for the HTO. Engine characteristics are shown in Table 9. Detailed performance trades have not been completed to define the best engine for the vehicle. The engine results from an uprating of the SSME, which should be well within the state of the art in technology in the 1985 to 1990 time frame. The chamber pressure of 24.13 M Pa (3500 psia) should not require new technology for the turbomachinery, nor should the growth.

VACUUM THRUST (EPL)	$3.03 \times 10^6/3.09 \times 10^6$ NEWTONS (680,403/695,350 LB)
SEA LEVEL THUUST	2.51×10^6 NEWTONS (564,993 LB)
I_{sp} VAC	455.2/465.2 SEC
I_{sp} SL	377.8 SEC
EXPANSION RATIO	80/150
CHAMBER PRESSURE (NPL)	20.7 (3,500) - N/M^2/OSI PSI
EXIT DIAMETER	2.54/3.50 (101/138) - M/IN
ENGINE LENGTH	4.88/7.82 (192/308) - M/IN
WEIGHT DRY	4,218 (9,300) - kg/LB
FLUID WEIGHT	268 (591) - kg/LB
ACTUATOR WEIGHT	278 (613) - kg/LB

Table 9 HTO-SSTO Main Engine Characteristics

Specific features incorporated which require continued development are zero NPSH pumps and two-position nozzles. Technology exists for these features but not for their specific applications.

Feed System. A feed system schematic is shown in Figure 55 The feedlines are sized to provide sufficient flow area to satisfy engine inlet requirements and yet provide minimum residuals. The hydrogen feedlines are basically straight ducts from the sumps to the engines. Anti-dropout plates are provided at the sumps to minimize residuals. An LO_2 manifold connects the two tanks with the feedlines branching off at each engine. Turning vanes are assumed in each elbow to minimize the straight section inlet length requirements at the engines. The LO_2 sump and anti-dropout plates are also configured to minimize residuals. Fill, drain and vent provisions are included in the feed system weight statement.

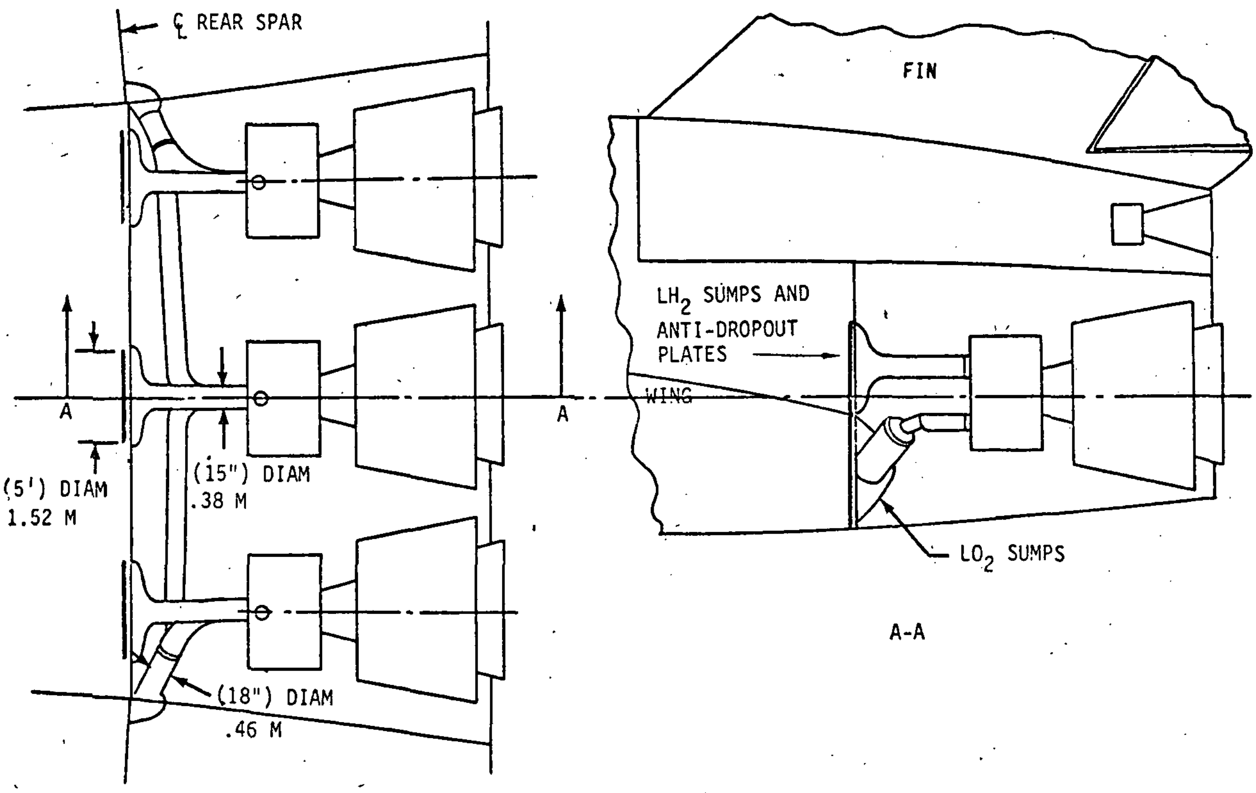


Figure 55 Main Engine Feed System Installation

Pressurization System. A pressurization system schematic is shown in Figure 56 . The pressurization system is used on reentry, not ascent. The ascent pressurization is a flash-boiling, self-pressurization concept. During reentry, it is necessary to have low tank pressures during peak heating periods; hence, the tanks are vented on orbit. After the peak heating period, when the ambient pressures begin to increase, it is necessary to repressurize the tanks to provide a positive pressure. Helium, stored in the LH_2 tank, is used for this purpose. The helium is heated by APU exhaust and then run through the helium tank to assist in expulsion prior to being injected into the main propellant tanks.

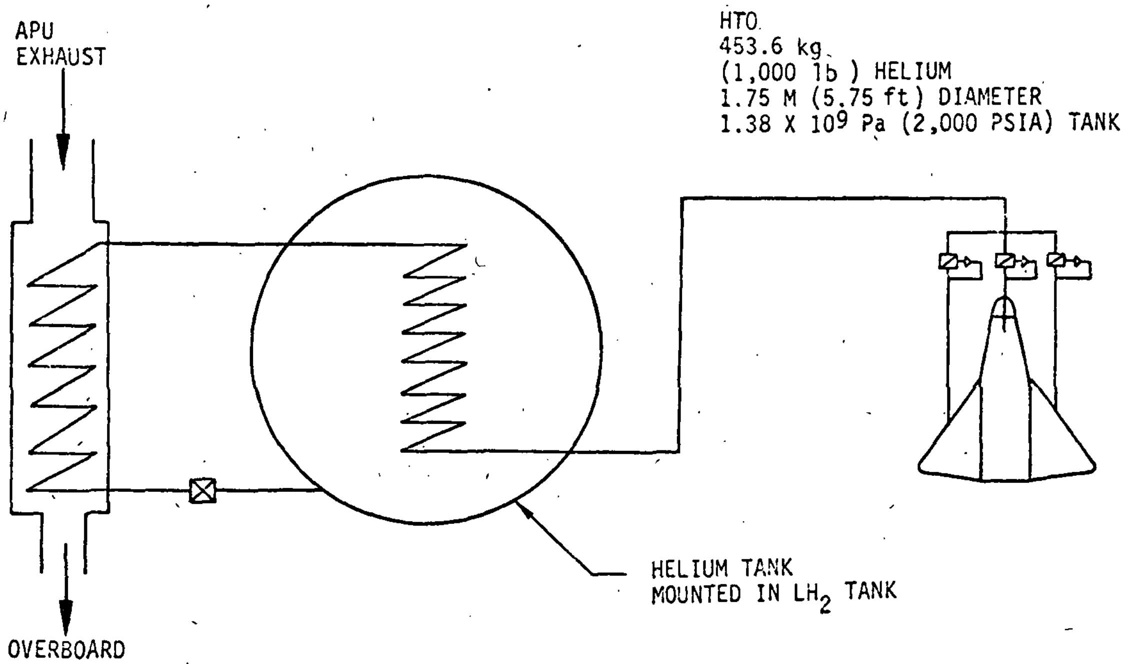


Figure 56 Reentry Pressurization Schematic

Reaction Control System (RCS). A system schematic is shown in Figure 57. The reaction control system selected for the HTO is a N_2H_4 MMH system. This system was selected because it is state of the art and only represents a weight penalty of less than 45.4 kg (100 pounds) relative to an LO_2/LH_2 system which still requires considerable development. The system weight differences are about 317 kg (700 pounds) for the VTO, hence the LO_2/LH_2 system is considered for the VTO.

The significant factor for the low subsystem weights, and why the LO_2/LH_2 system does not offer significant advantage on the HTO, is the low requirement used for sizing. Basically, this Δ V requirement stems from the shorter stay time on orbit compared to the Space Shuttle, hence less maneuvering. Second, it is assumed the forward thrusters are used on reentry, providing significant advantage over using the aft thrusters only. The Δ V of 30.5 m/s (100 fps), including reserves, is comparable to shuttle, considering these differences.

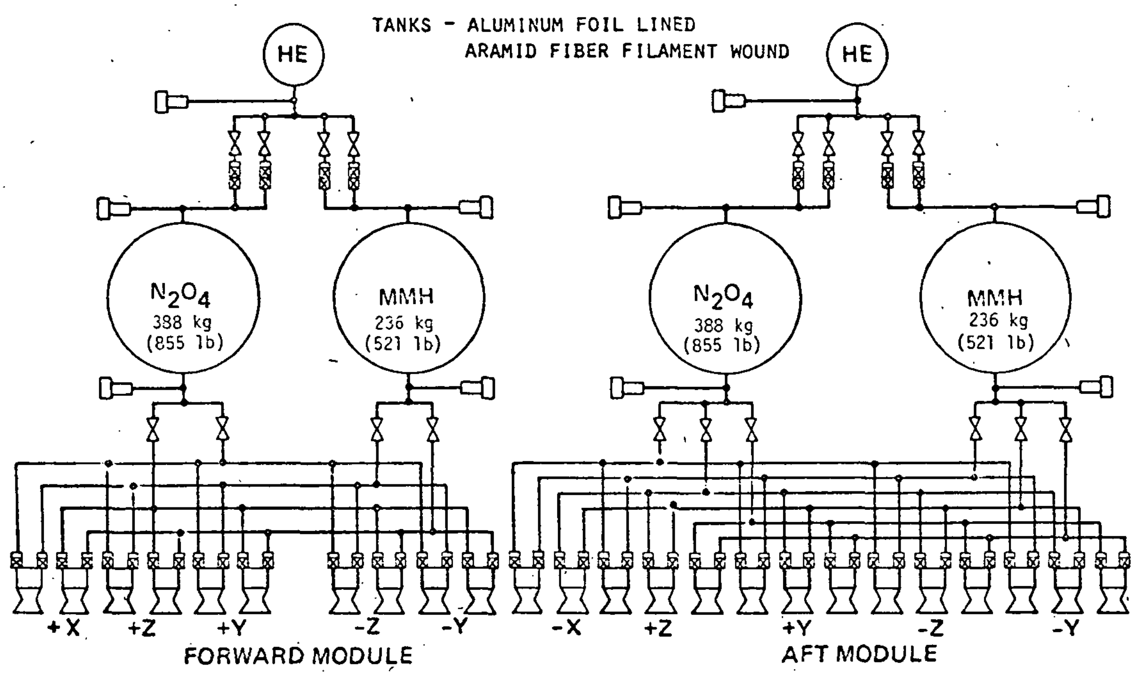


Figure 57 Reaction Control System Schematic

<u>Orbit Maneuvering System (OMS).</u> A system schematic is shown in Figure 58 The orbital maneuvering system is an LO_2/LH_2 system. The engine is a staged combustion cycle, high expansion ratio engine in the thrust range that would be developed for the reusable tug. The tankage is of a type used for long-term cryogenic storage, utilizing a soft-shell multilayer insulation blanket for on-orbit performance and a mylar honeycomb substrate which cryopumps to provide good insulation performance during ascent.

<u>Cryogenic Propellant Boil-Off.</u> Excessive boil-off of the cryogenic propellants during loading on the ground accelerator prior to take-off can result in a large and expensive resupply system, increased cost of propellants, and increased loading errors. An analysis was performed to determine the maximum and normal LH_2 and LO_2 boil-off rates for a SSTO on the ground accelerator, excluding fill transients. The boil-off rates are dictated by the insulative properties of the walls, surface area exposed, and frost/ice formation.

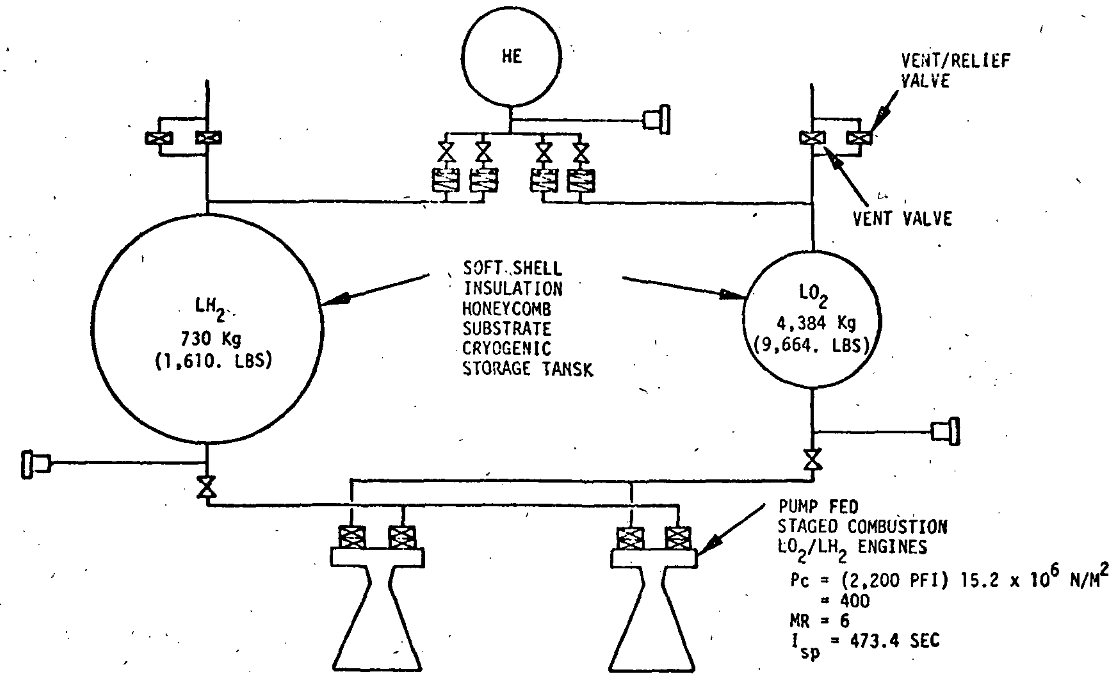


Figure 58 Orbit Maneuvering System Schematic

Figures 38, 39, and 40 present the SSTO tank arrangement and the structural configuration in each area. The boil-off, attributed to heat transfer through each surface as well as the total boil-off for each tank were calculated. The resulting boil-off rates were then compared to those of previous Saturn V stages to assess the impact. Each of the tanks is discussed below.

<u>Liquid Hydrogen Tank</u>. Heat will enter the LH_2 tank from the forward bulkhead, the aft engine bulkhead, the outer body wall, the LH_2/LO_2 side of body rib, common bulkhead, payload bay and forward wheel wall. Heat transferred through each of these structures was calculated and the net heat entering the LH_2 tank determined.

The payload bay wall and forward wheel well wall are insulated and provide negligible heat addition to the propellant. The side of body rib, between the LO_2 and LH_2, is a honeycomb structure which, when accounting for flanges and attachments, permits about 1576 W/m^2 (500 $BTU/HR\text{-}FT^2$) between the LO_2 and LH_2. The rest of the surfaces will be covered with frost or ice, depending on the ambient conditions. Test and actual flight vehicle data have shown that vehicle wall construction is not significant with respect to propellant heating if frost and ice form on the walls. That is, the heating rates for a honeycomb tank are comparable to a single wall tank if the honeycomb thermal performance will allow frost and ice to form. The primary benefit afforded by the honeycomb is that it prohibits air liquefaction on the LH_2 tank. Typically for a nominal day, the heat transfer rate on a frosted wall to a cryogenic tank is about 1261 W/m^2 (400 $BTU/HR\text{-}FT^2$). A maximum hot, windy day would be 2.5 times that value, and with heavy rain and wind the value could be six times the nominal. The rain conditions are not considered for the HTO. Since ice buildup on the wings would result in such prohibitive weights, the HTO will require shelter from the rain or operational constraints. In conclusion, a representative heat rate to the hydrogen, for the nominal conditions is assumed to be 1261 W/m^2 and the maximum value of 3152 W/m^2 (1000 $BTU/HR\text{-}FT^2$).

The exposed surface area of the hydrogen tank is approximately 1300m^2 (14000 ft^2). The boil-off rates for the hydrogen are therefore 3.653 kg/s (29,000 pounds/hour) nominal and 9.07 kg/s (72,000 pounds/hour) maximum.

Expressing the boil-off rate (per hour) as a percentage of the propellant load, the hydrogen boil-off for the HTO is about 10.7% nominal. The S-IVB typically experienced 6.2%, hence the HTO is higher. This is to be expected due to the less optimal configuration (oval cross-section vs circular) but the 10.7% rate should pose no problems.

Liquid Oxygen Tank. The heat entering the oxygen tanks is through the skin panels, tank bulkheads and side of body rib. The heating rates for these areas are comparable to those of the hydrogen tank. The exposed surface area of both tanks is approximately 1022 m^2 (11,000 ft^2). The resultant boil-off rates are then 6.05 kg/s (48,000 lb/hr) nominal and 15.12 kg/s (120,000 lb/hr) maximum. This is approximately three percent of the LO_2 load, and again, should pose no problems.

Environment and Mass Properties

The mass properties to be determined are dependent upon the design loading and the environmental conditions. This necessitates loads, dynamics, thermal and structural analyses which result in structural sizing of the vehicle components leading to the determination of vehicle weight.

Loads and Dynamics. Body bending and shear loads for all conditions making up the design envelope are shown in Figures 59 and 60. Loads for the 1.8 g hypersonic entry condition are also shown, since entry loads are applied to a hot structure. Ascent $q\alpha$, descent maneuvers and gusts, nose and gear impact, braked roll, ground turn and 1.67 g taxi bump conditions were also analyzed and did not produce critical body loads.

The 1.75 g pullup condition at takeoff gross weight produces the maximum positive body bending loads. Maximum negative bending loads are produced by the 1.67 g bump condition on the ground accelerator and the 3.05 m/s (10 ft./sec.) sink rate landing condition.

Axial loads for the ground acceleration and liftoff conditions are shown in Figure 61. A maximum axial acceleration of 1.0 g is used during ground acceleration. Drag loads are included in the liftoff condition but not the ground acceleration condition. LO_2 tank loads are assumed in along the wing root. LH_2 inertia loads are introduced at the LH_2 tank aft bulkhead. Thrust loads from the engines and ground accelerator are both applied at BS 71.68m (2822 in.).

Wing spanwise bending loads and shear loads for all conditions making up the design envelope are shown in Figures 62 and 63. Loads for the hypersonic 1.8 g entry condition are also shown, since entry loads are applied to a hot structure. Other conditions analyzed which did not produce critical wing loads include positive 2.5 g supersonic and subsonic maneuvers, 15.24 m/s (50 fps) gusts during descent, landing and taxi.

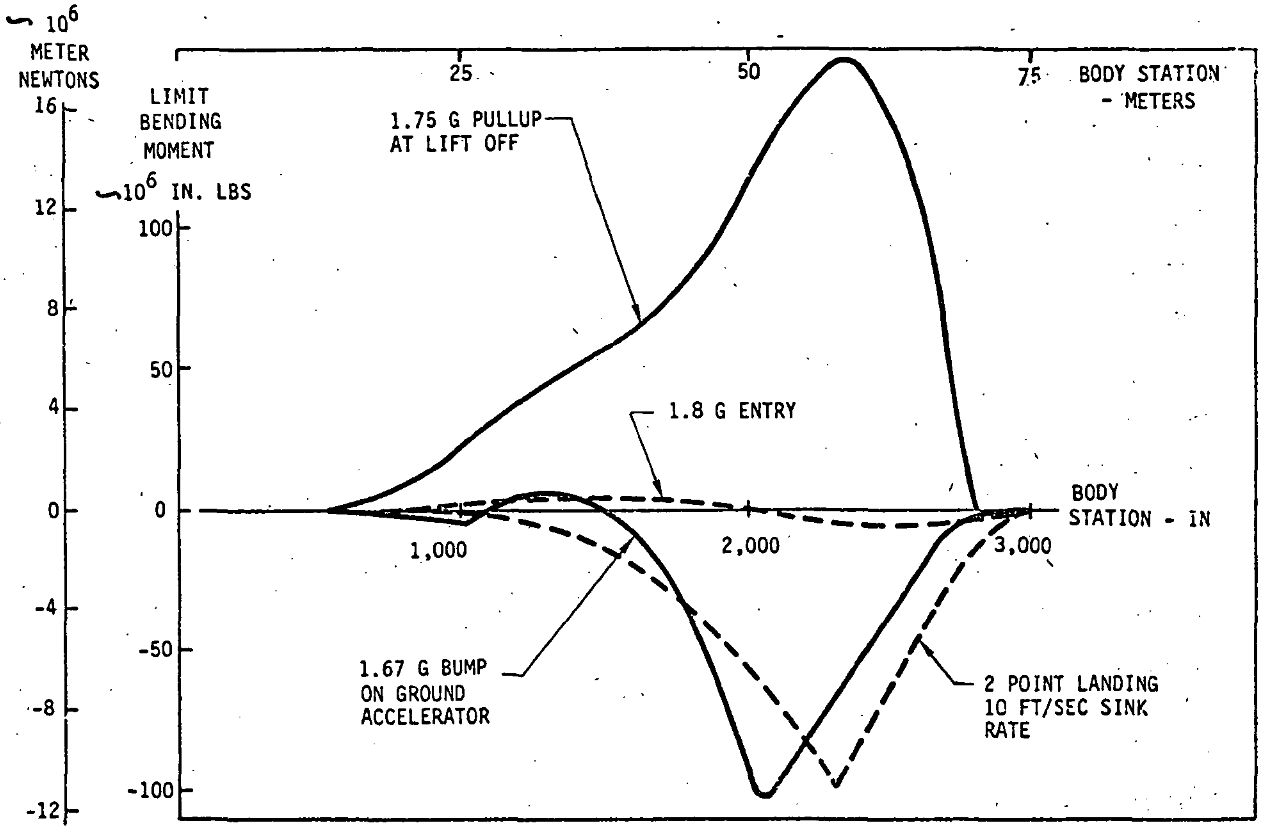


Figure 59 HTO SSTO Body Bending Loads

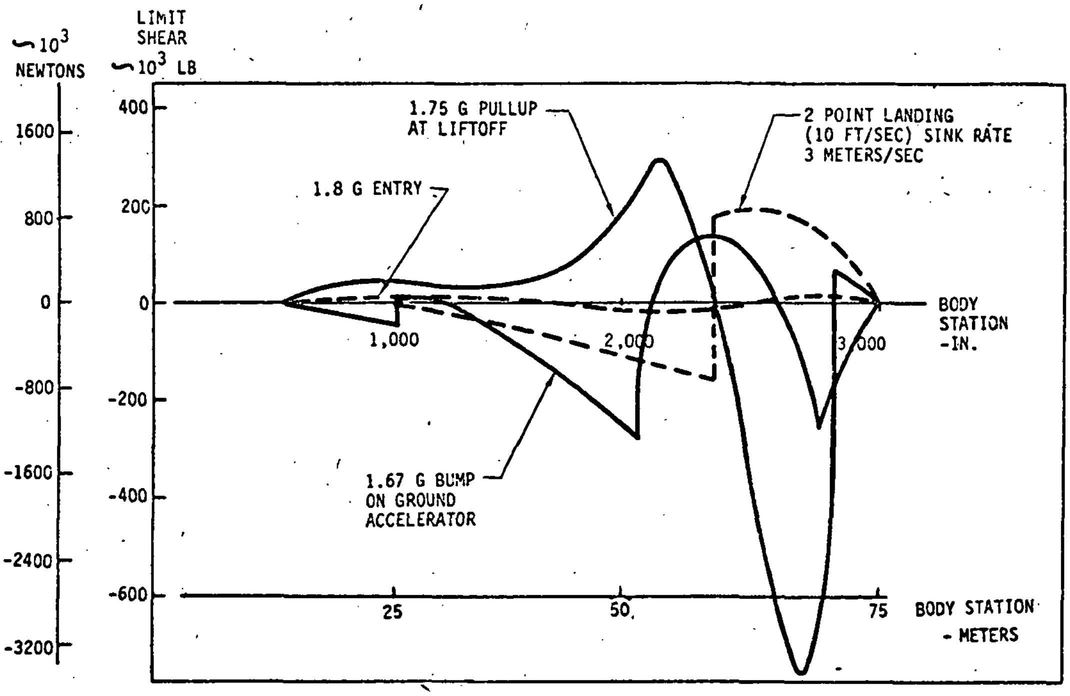


Figure 60 HTO SSTO Body Shear Loads

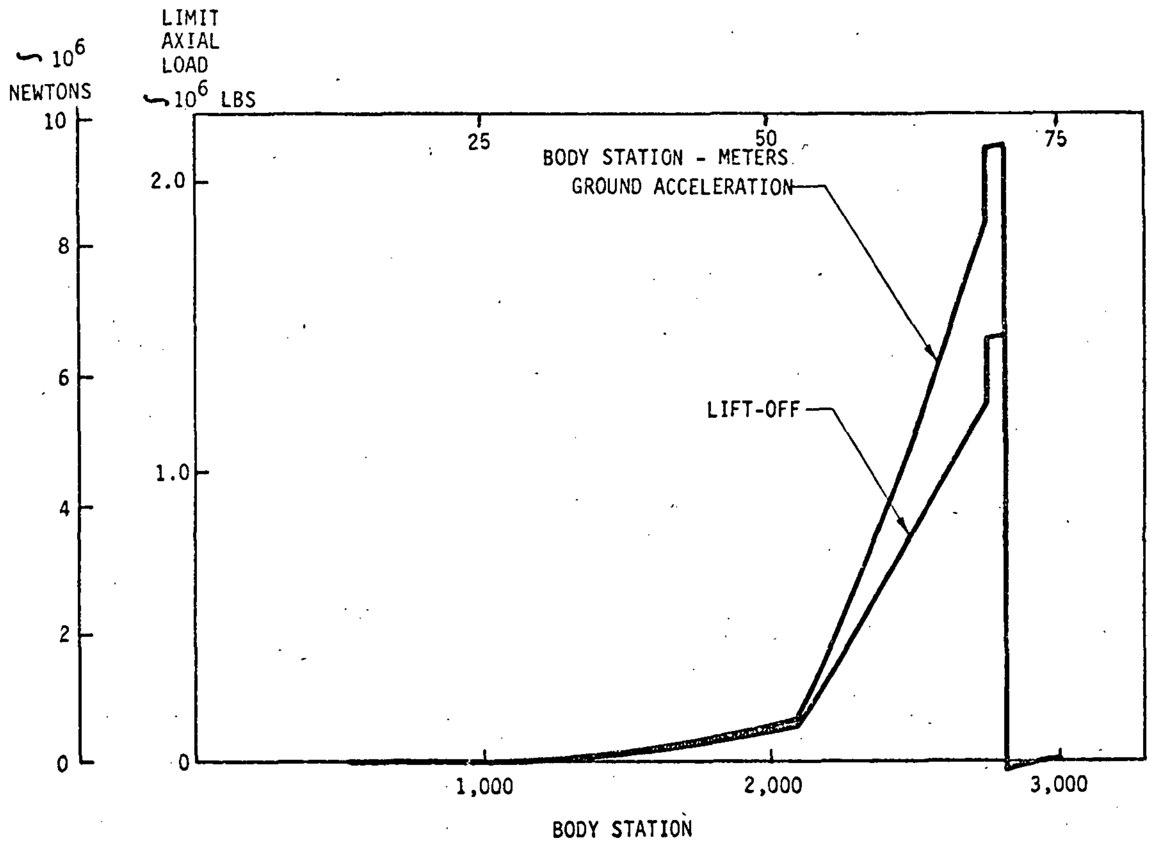


Figure 61 HTO SSTO Body Axial Load

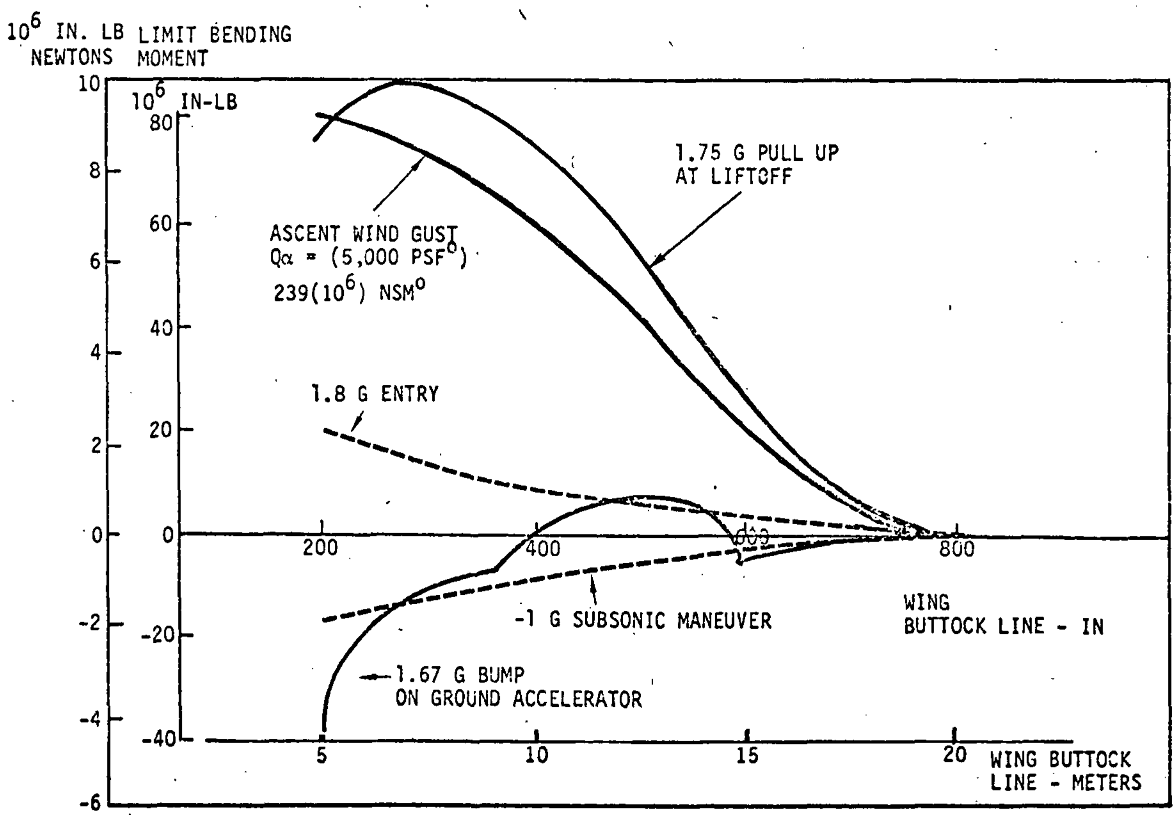


 Figure 62 HTO SSTO Wing Bending Loads

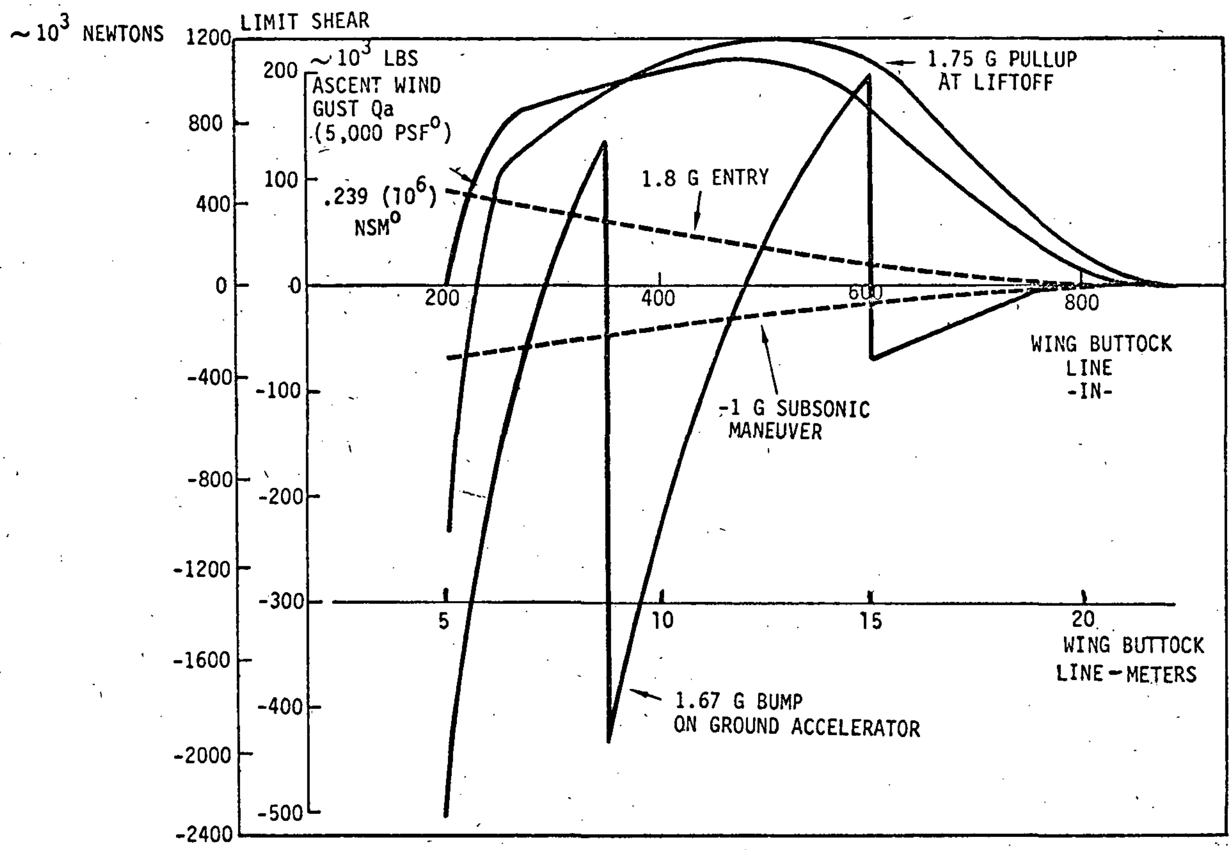


Figure 63 HTO SSTO Wing Shear Loads

Wing aerodynamic loads were determined in a computer analysis based on the aerodynamic influence coefficient method using 67 panels for the half airplane planform. Newtonian theory was used for the entry condition. The design ascent $q\alpha$ of .239 x 10^6 Pa (5000 psf) degrees was determined from 3-degree-of-freedom computer simulations of flights through 99% synthetic wind/gust profiles based on the same wind criteria as the Space Shuttle.

The 1.7 g pullup at liftoff produces maximum upbending loads except near the wing root. The LO_2 tanks are located in the wings in order to provide maximum inertia relief of wing loads. The net bending moment of 7.49 (10^6) meter newtons (66.3 (10^6) in. lb) at the root results from aerodynamic and inertia bending moments of 31.3(10^6) n.m. (277.3(10^6) in. lb.) and -23.8(10^6) meter newton (-211.0(10^6) in. lb) respectively.

Negative bending loads due to a 1.67 g bump on the ground accelerator are kept small by supporting the wing along WBL 9.14 m (360 in) and WBL 15.24 m (600 in). Maximum reactions are .152 (10^6) N/m (870 lb/in) at WBL 9.14 and .175(10^6) N/m (1000 lb/in) at WBL 15.24.

Limit landing gear loads are given in Table 10. The impact loads are based on a .61 m (24 in) actual stroke assuming 85% efficient oleo strut energy absorption; i.e., average vertical reaction 85% of the maximum. (the body bending loads due to landing impact shown in a previous figure were based on .457 m (18 in) oleo stroke, which produced 33% higher vertical gear reactions.) Nose gear impact loads are based on a maximum nose down pitch rate of 6 1/2°/sec. Elevon pitch control is adequate to hold the nose up after touchdown. Typical pitch rates in 707, 727 and 737 performance landing tests were in the 3 to 6°/sec. range. Spinup and springback loads are based on a coefficient of friction of .55 with dynamic factors of 1.28 and -.9 for spinup and springback, respectively.

LOAD CONDITION	LIMIT GEAR LOADS (KIPS)					
	NOSE GEAR			RIGHT MAIN GEAR		
	DRAG	LAT	VERT	DRAG	LAT	VERT
10 FT/SEC SYM 2 POINT LANDING	0	0	0	139.6 kg (31.4 K.LBS)	0	558.7 kg (125.6 K.LBS)
SPIN UP	0	0	0	393.2 kg (88.4 K.LBS)	0	530.6 kg (119.3 K.LBS)
SPRINGBACK	0	0	0	-276.7 kg (-62.2 K.LBS)	0	530.6 kg (119.3 K.LBS)
NOSE GEAR IMPACT 6 1/2°/SEC PITCH RATE	83.6 kg (18.8 K.LBS)	0	333.6 kg (75.0 K.LBS)	0	0	470.1 kg (105.7 K.LBS)
SPIN UP	234.9 kg (52.8 K.LBS)	0	317.4 kg (71.3 K.LBS)	0	0	470.1 kg (105.7 K.LBS)
SPRINGBACK	-156.8 kg -37.1 K.LBS	0	317.1 kg (71.3 K.LBS)	0	0	470.1 kg (105.7 K.LBS)
1/2-g GROUND TURN (RIGHT)	0	33.8 kg (9.5 K.LBS)	84.5 kg (19.0 K.LBS)	0		318.5 kg (57.4 K.LBS)
(LEFT)	0	-33.8 kg (-9.5 K.LBS)	84.5 kg (19.0 K.LBS)	0	328.1 kg (INBD) (-99.3 K.LBS)	820.2 kg (198.6 K.LBS)
BRAKED ROLL CF = .8 n_z = 1.2	0	0	326.0 kg (73.3 K.LBS)	456.8 kg (102.7 K.LBS)	0	571.1 kg (128.4 K.LBS)
DRIFT LANDING (RING (RIGHT)	0	0	0	0	223.3 kg (INBD) (-50.2 K.LBS)	279.3 kg (62.8 K.LBS)
	0	0	0	139.6 kg (31.4 K.LBS)	212.2 kg (INBD) (-47.7 K.LBS)	265.5 kg (59.7 K.LBS)
(LEFT)	0	0	0	0	167.7 kg (OTBD) (37.7 K.LBS)	279.3 kg (62.8 K.LBS)
	0	0	0	139.6 kg (31.4 K.LBS)	159.3 kg (OTBD) (35.8 K.LBS)	265.5 kg (59.7 K.LBS)
TAXI 1.67-g BUMP	0	0	141.0 kg (31.7 K.LBS)	0	0	951.0 kg (213.8 K.LBS)

NOTES: (1) 1.223(10^6) NEWTONS (275,000 LS) g.w., (Iy = 19.6 SLUG FT^2 x 10^6), IY = 26.5(10^6) Kg - METER^2
(2) 6.9% WEIGHT ON NOSE GEAR STATICALLY
(3) (24-IN) STROKE DURING IMPACT .61 M

Table 10 HTO-SSTO Landing Gear Loads

Elevon and rudder hinge moments, fin bending moments, gimbal angle, and maximum $q\alpha$ and $q\beta$ are shown for six ascent control modes in Table 11.

ASCENT CONTROL MODE	ELEVON HM/ SIDE METER NEWTONS	RUDDER H.M. METER NEWTONS	FIN ROOT MOMENT METER NEWTONS	MAX GIMBAL ANGLE	
				PITCH	YAW
1. NO AERODYNAMIC CONTROLS USED $S_r = 0^o$, $S_e = 0^o$	$-.77(10^6)$ $(-6.82(10^6))$	$.36(10^6)$ $(3.16(10^6))$	$6.62(10^6)$ $(58.6\ (10^6))$	-9.9^o+63^o CAN NOT CONTROL ROLL	—
2. USE ELEVONS FOR ROLL CONTROL $S_r = 0^o$, $S_e = \pm 2.80^2$	$-1.48(10^6)$ $(-13.08(10^6))$	$.36(10^6)$ $(3.16(10^6))$	$6.62(10^6)$ $(58.6\ (10^6))$	-8.9^o	—
3. USE ELEVONS FOR ROLL CONTROL, RUDDER FOR LOAD ALLEVIATION ($S_r = \beta$) BASELINE	$-.94(10^6)$ $(-8.4\ (10^6))$	$-.11(10^6)$ $(-1.0\ (10^6))$	$3.62(10^6)$ $(58.6\ (10^6))$	-8.9^o	—
4. USE ELEVONS FOR PITCH TRIM & ROLL CONTROL, RUDDER FOR LOAD ALLEVIATION ($S_r = \beta$) MOD 1	$-.77(10^6)$ $(-6.8\ (10^6))$	$-.11(10^6)$ $(-.98(10^6))$	$3.67(10^6)$ $(29.8\ (10^6))$	-2.9^o	—
5. SAME AS 4 EXCEPT LIMIT MAX ELEVON H.M. = .56 x (10^6) METER NEWTONS (5 x (10^6) IN-LB) MOD 1B	$.56(10^6)$ $(5.0\ (10^6))$	$-.11(10^6)$ $(-.97(10^6))$	$3.67(10^6)$ $(29.8\ (10^6))$	3.5^o	—
6. SAME AS 4 EXCEPT POSITION ELEVON AT ZERO H.M. FOR PITCH MOD 2	$.34(10^6)$ $(3.0\ (10^6))$	$-.11(10^6)$ $(-1.0\ (10^6))$	$3.67(10^6)$ $(29.8\ (10^6))$	6.7^o	2.4^o

NOTE: LOADS AND GIMBAL ANGLES IN THIS EXAMPLE ARE FOR THE 99% WIND SHEAR PROFILE FOR A QUARTERING WIND, $q\alpha = .239\ (10^6)$ NSM^o (5000 PSF^o) AND $q\beta = .122\ (10^6)$ NSM^o (2560 PSF^o)

Table 11 HTO-SSTO Ascent Control Mode Summary

The first of these, using no aerodynamic controls, is unsatisfactory because of insufficient roll control. In the second control mode, elevons are used for roll control only. This mode is capable of providing control, but large hinge moments are produced on the down elevon in a combined high $q\alpha$ and $q\beta$ condition. The third control mode was the baseline for the load alleviation control modes studies. This control mode employs elevons for roll control and the rudder is feathered ($S_r = \beta$) to alleviate fin loads, rudder hinge moments and rolling moment due to $q\beta$. The reduction in rolling moment due to $q\beta$ reduces the elevon deflection for roll control and hence reduces the maximum elevon hinge moment. The fourth control mode called MOD 1 uses elevons for roll control and pitch trim in addition to the feathered rudder. Elevon hinge moments and engine gimbal angle are reduced for this mode. The fifth control mode called MOD 1B is the same as the fourth except the elevon hinge moment is limited to $.56(10^6)$ meter newtons $(5(10^6)$ in. lb) which is the maximum entry elevon hinge moment. This control mode is effective in reducing the elevon hinge moment but may be difficult to design. The sixth control mode called MOD 2 was the same as the fourth except the elevon deflection is varied to produce zero hinge moment except for that required for roll control. This system results in lower elevon hinge moments than the others and may be easier to implement.

Thermal Analysis. The thermal analysis was carried out in accordance with the criteria specified in Figure 31. Both ascent and reentry trajectories were analyzed. The isotherms shown in Figure 64 are based on peak equilibrium radiation temperatures and do not account for internal radiation or material heat sink effects. Both ascent and reentry critical regions are shown. The reentry trajectory is corresponding to an equilibrium glide trajectory with W/SC_L = .273 kg/m^2 (56 lb/ft^2). The reentry angle of attack of 50° is reduced to 30 degrees at 91490m (300,000 ft.) altitude.

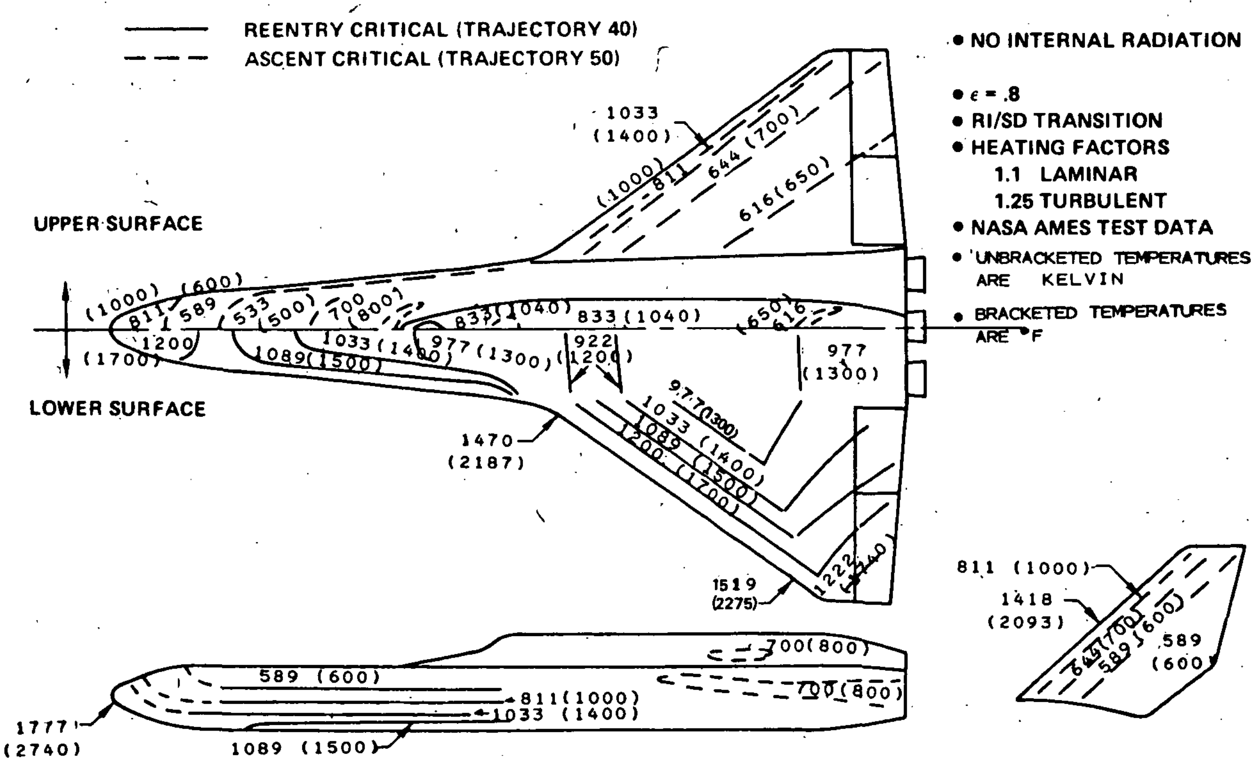


Figure 64 HTO SSTO Peak Equilibrium Temperature Distribution

Computed heating rates include uncertainty factors of 1.1 for laminar and 1.25 for turbulent flow. Turbulent flow heating is predicted using the Spalding-Chi method in conjunction with a Reynolds analogy. Transition is determined using the Rockwell International/Space Division transition $RE_\theta/Me = N$, where N = 225 at the body centerline, 160 at wing midchord, and 80 at the wing tip. The body nose radius is .5m (20"). The leading edge radii are .4m (16") on the inboard and .33m (13") at the outboard wing and .29M (11.5") minimum on the vertical fin.

Interference heating was accounted for using data obtained from hypersonic tests of a representative SSTO configuration in the NASA-Ames 1.07m (3.5 foot) hypersonic tunnel.

For structural sizing actual temperatures and temperature distributions are required including the effects of heat sinks and internal radiation exchange as shown in Figure 65 for a typical body cross section.

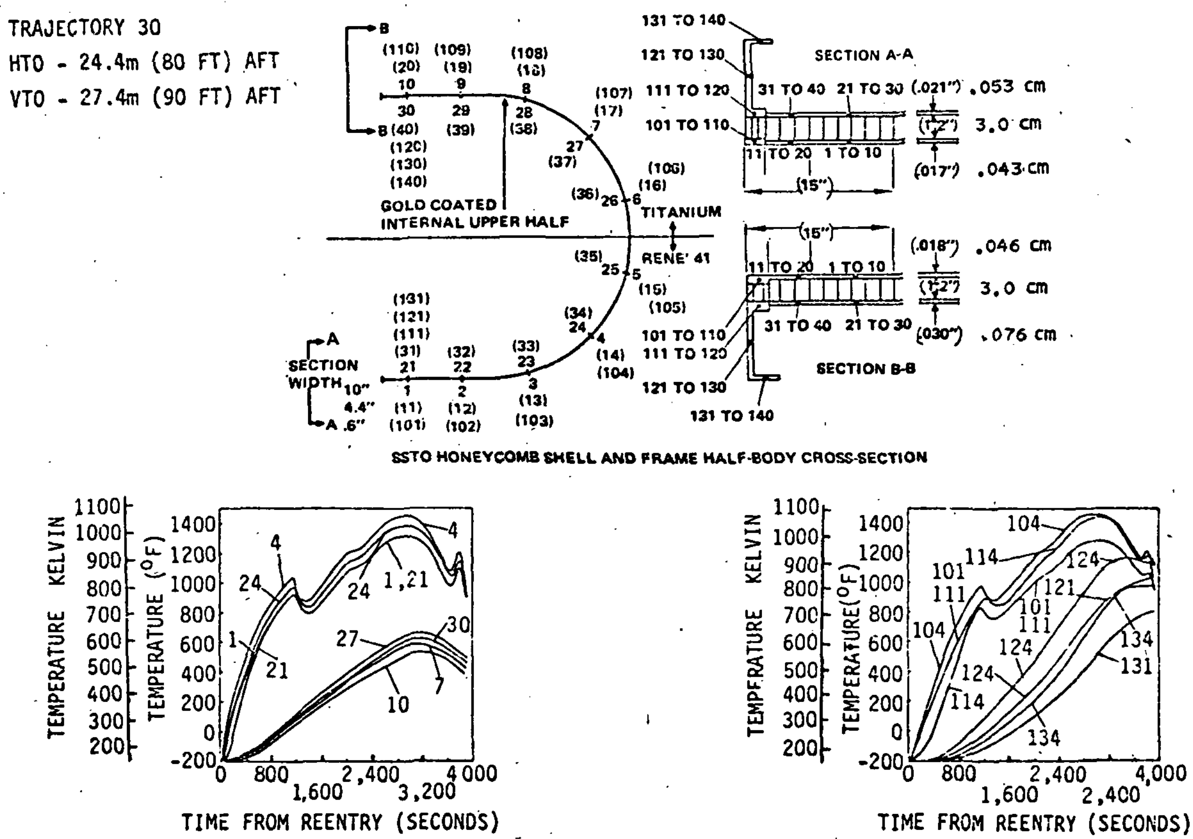


Figure 65 HTO SSTO Fwd Body Reentry Temperatures

The temperature distributions during reentry were obtained using the Boeing Engineering Thermal Analysis (BETA) program which accounts for internal radiation, conduction and heat storage.

The body cross section taken approximately 24 m (80 ft.) aft of the nose consists of Rene'41 frames, face sheets and honeycomb core on the lower, and titanium frames, face sheets and honeycomb core on the upper half. The internal face sheet and the frame of the upper body half are gold coated.

Figures 66 and 67 show temperature distributions during ascent on a simplified one inch body cross section at a distance of 24 m (80 ft) aft of the nose for the lower and upper surface respectively. The lower half of the body is made of Rene'41 whereas the upper half is made of titanium. The structure consists of face sheets and honeycomb respectively. Internal cooling and heat transfer due to LH_2 and GH_2 are accounted for. It is assumed here that the section becomes dry at 200 seconds after launch. With this

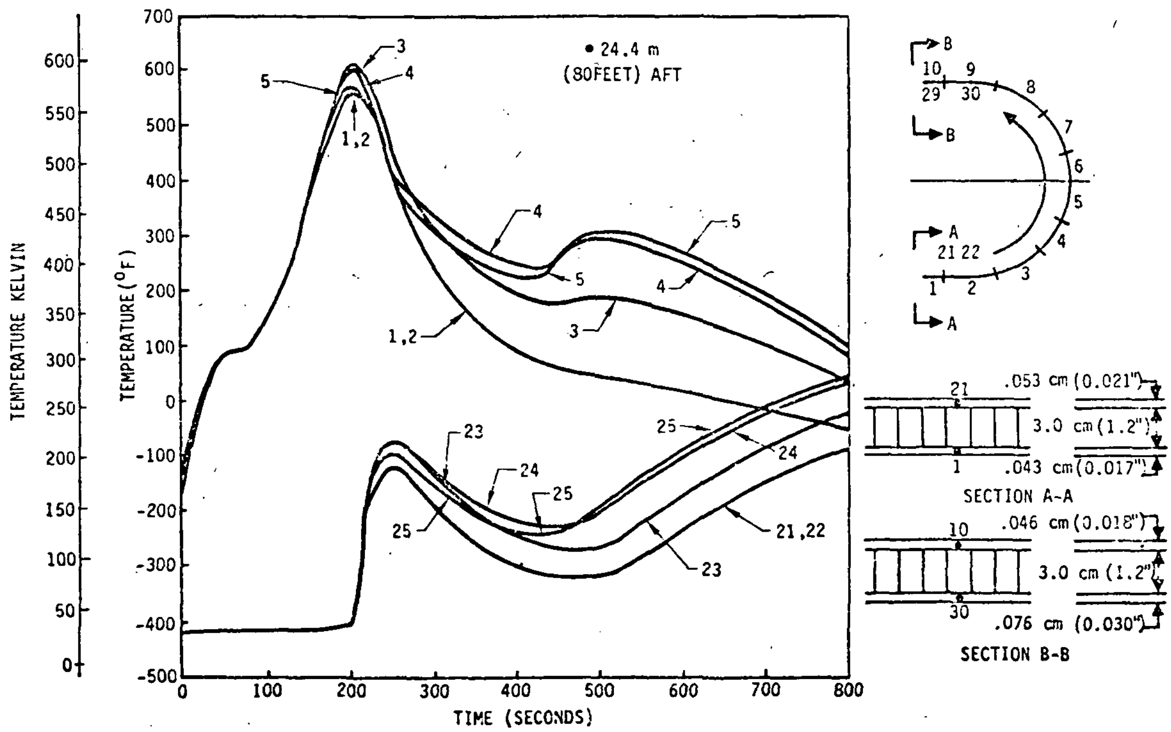


Figure 66 HTO SSTO Lower Body Ascent Temperatures

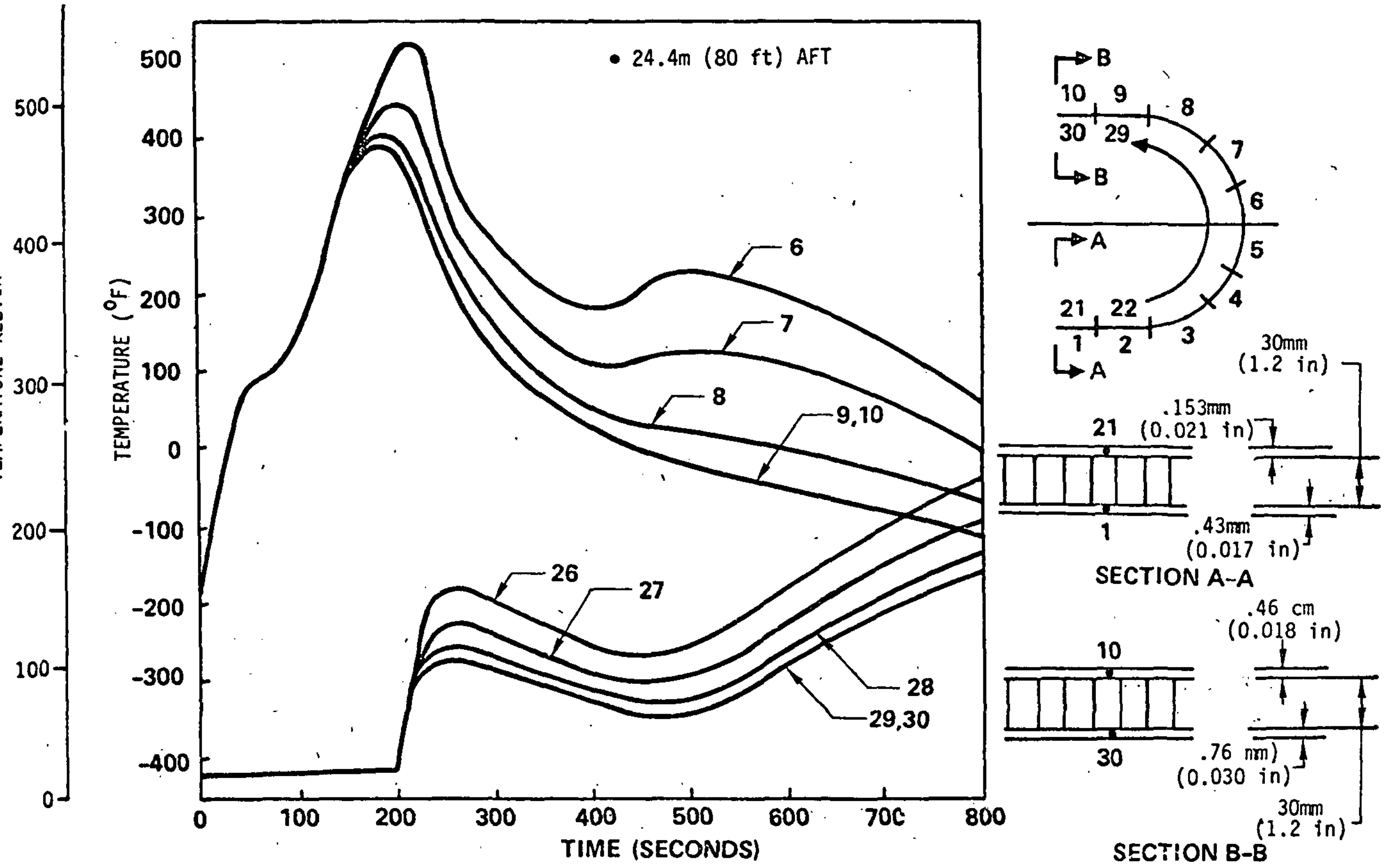


Figure 67 HTO SSTO Upper Body Ascent Temperatures

assumption, it can be seen that ascent produces the most severe gradient condition for the honeycomb panels with maximum ΔT of up to 550K (1000°F). Typical leading edge temperature distributions during reentry are shown in Figure 68. The leading edge has a .3m (12") nose radius with dimensions and material distributions as shown. The internal spar wall and the truss struts are gold coated for temperature control.

Structural Analysis. A finite element analysis was conducted on a model of a typical SSTO forward body frame bay. Conditions imposed on the model included: maximum entry temperatures, maximum entry thermal gradients and maximum tank pressures and entry aerodynamic pressures. The model is shown in Figure 69. Inner and outer sandwich surfaces were modeled as continuous plates. The core was modeled as longitudinal shear plates. The inner and outer frame chords and frame support struts were modeled as beams. The frame webs were modeled as shear plates. The model represents a section of the SSTO that is 5.1m (200 inches) deep, 9m (354 inches) maximum width and with the typical .76m (30 inch) frame spacing.

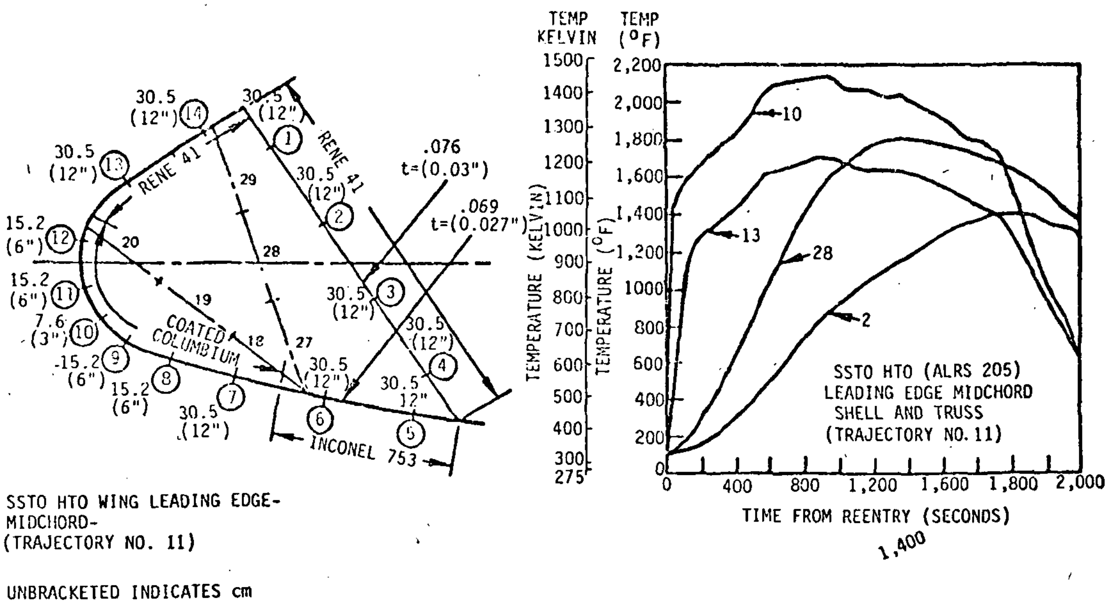


Figure 68 HTO SSTO Leading Edge Reentry Temperatures

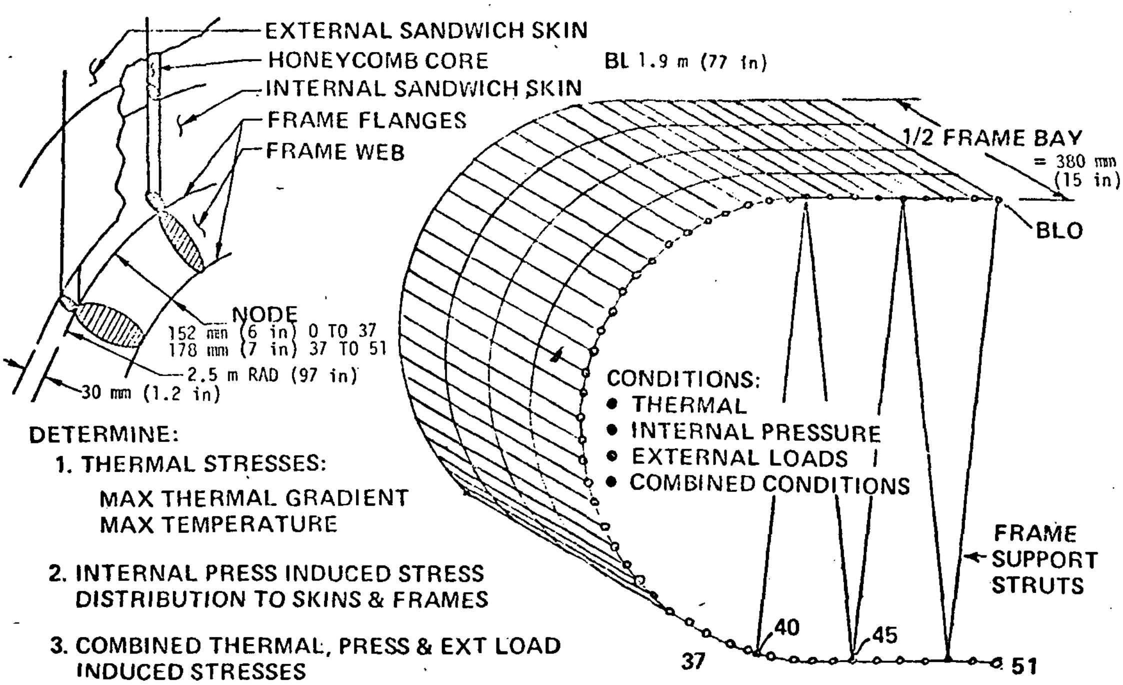


Figure 69 Finite Element Analysis Model

Maximum thermal stresses on the model were caused by the maximum thermal gradient occuring during entry. The vehicle is very lightly loaded by aerodynamic loads and is subjected to an internal pressure of only 13790 Pa (2 psi) during the hypersonic segment of entry. Hypersonic entry pressure and aerodynamic loads are relatively insignificant compared to the thermal stresses shown in Figure 70 . The entry tension thermal stress levels in Rene'41 inner skins were below the .7 x Tension Yield Strength (F_{t_y}) noted in Figure 32 .

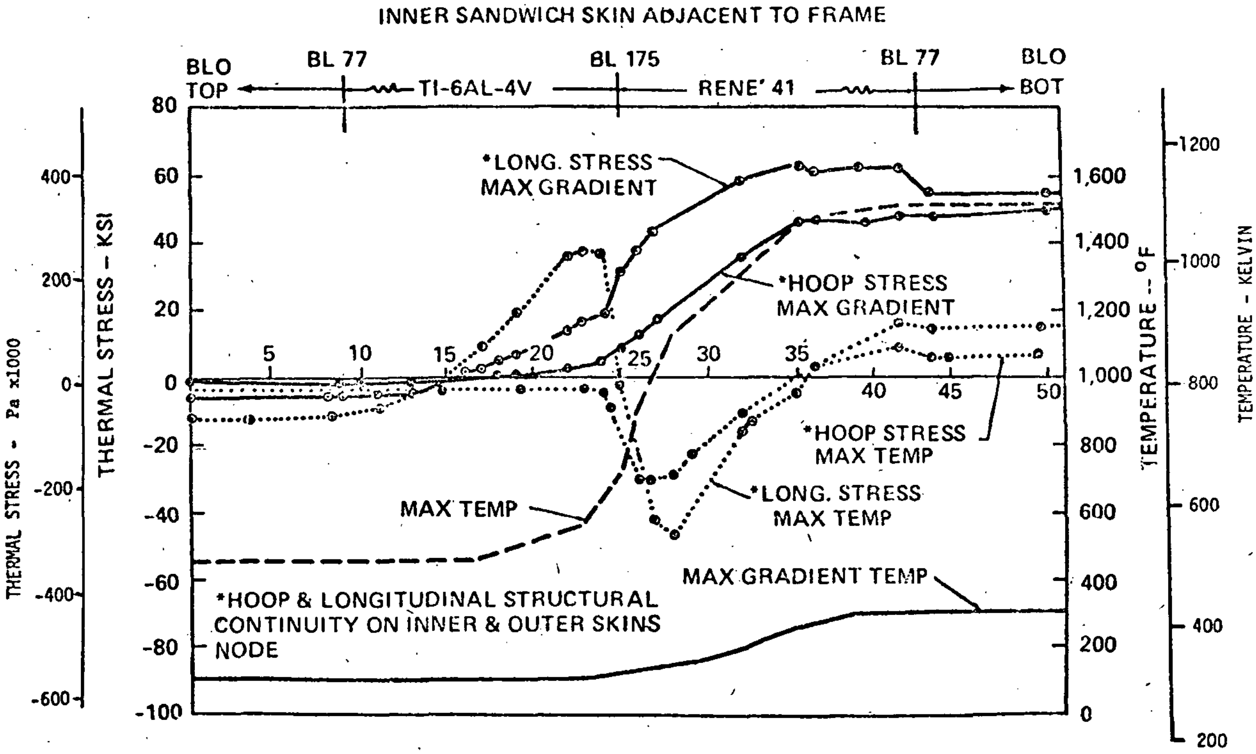


Figure 70 Typical Body Reentry Thermal Stresses

The finite element analysis was conducted prior to obtaining a boost thermal profile. The boost profile indicates that the maximum boost temperature differential (ΔT) between inner and outer skins is 583 K ($1050^{o}F$) as compared to 256 K ($460^{o}F$) ΔT during entry. The higher boost ΔT combined with the body LH_2 and wing LO_2 tank pressures of 1.1×10^5 and 1.12×10^5 Pa (15.3 and 16.3 psi) respectively indicate a requirement to partially relieve thermal stresses on the Rene'41 lower surfaces by slotting the lower surface Rene'41 skin. It is not necessary to slot the upper surface titanium skin. The slotted skin detail will allow free expansion across the slot. The outer slotted lower surface skins are effective in carrying loads from frame to frame in the body and spar to spar in the wing. Both inner and outer skins carry body bending pressure bending and longitudinal loads and the inner skins carry circumferential pressure loads. In the wing, inner and outer skins carry pressure bending and chordwise pressure loads. The inner skin and the lower spar caps carry the wing bending loads on the lower surface and both inner and outer skins are effective in carrying wing bending.

Structural analysis criteria are shown in Figure 32. Sandwich surface skin gages in the LH_2 and LO_2 tanks at frames and spars are sized primarily as follows:

Outer surface: compression strains

$$\varepsilon_{th} + 2.0\varepsilon_{PRESS} \leq \varepsilon_{F_c}$$ See Figure 32

Inner surface: tension stresses

$$F_{th} + F_t \leq KF_{ty}$$ See Figure 32

Inner surface tension thermal stresses are primarily responsible for prompting the decision to slot the lower surface skins for thermal stress relief. The center bay skin sizing in the body and the wing is based on the following criteria:

Outer surface - (tension stresses)

$F_t \leq K\ F_{ty}$ thermal stresses are neglected because they reduce pressure stresses

Inner surface - (compression strains, fore and aft direction)

$2.0\ \varepsilon_c \leq \varepsilon_F$ thermal strains are neglected because they reduce pressure strains

Inner lower surface - (tension stresses - circumferential in body and inboard-outboard in the wing)

$$F_t \leq KF_{ty}$$

Skin gages in a typical body frame bay are shown in Figure 71. The upper part of the curve shows the sizing at the frame where thermal and pressure stresses have the same sign and require a thicker gage. The lower curve shows the sizing required at center bay where pressure and thermal stresses have opposite signs and there is considerable use of .3 mm (.012 in.) minimum gage. Figure 71 also shows the effect of curvature and the joining of dissimilar materials on skin gage sizing. Curvature reduces pressure bending stresses rapidly after curvature is initiated at BL77 moving outboard. The joining of dissimilar materials causes a local increase in stress at temperature.

Figure 72 shows a typical wing skin sizing for the critical boost thermal-pressure condition at 200 seconds after lift off. Locally increased skin gages are required near the mid-wing baffle spar and at the rear spar because of head pressure effects.

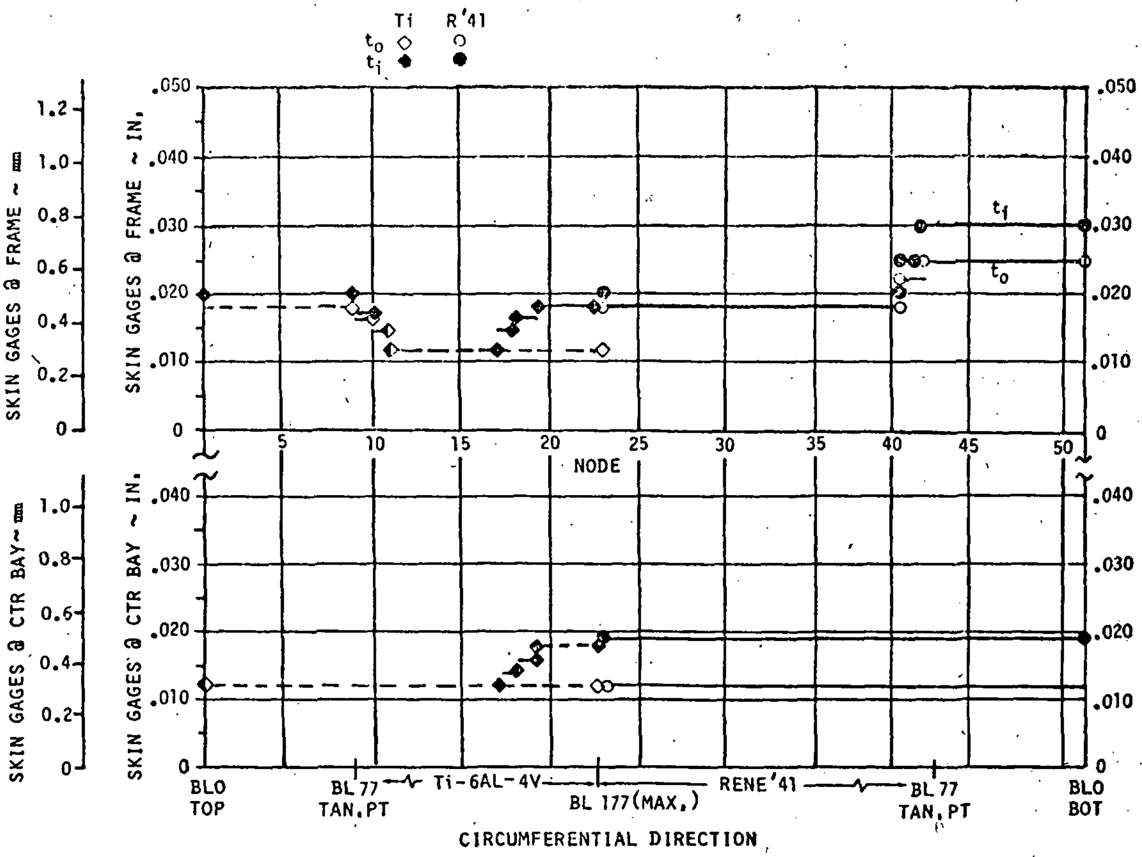


Figure 71 Typical Body Skin Gages

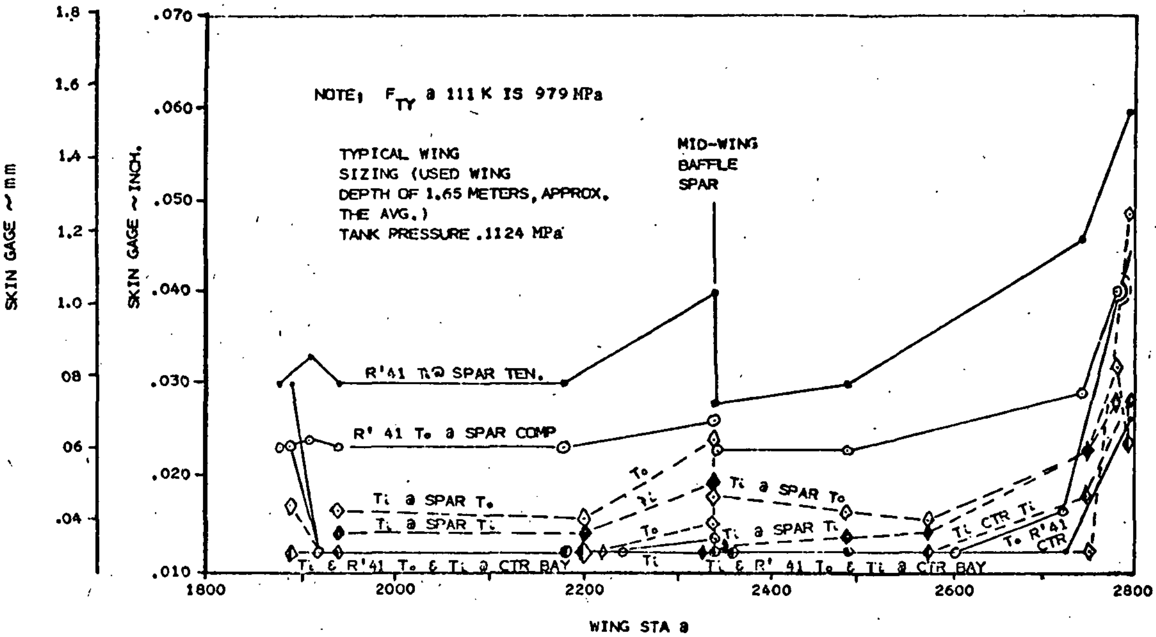


Figure 72 Typical Wing Skin Gages

Weights

Horizontal Take-Off Weight Statement. Structures, subsystem and fluid weights are shown in Table 12 for the HTO vehicle. Definition of the majority of the systems is sufficient to provide reasonable confidence in calculated weights. Both the structures and subsystem include a 10% margin for weight growth. The flight performance reserves are .85% of the total ΔV. Reaction control and OMS propellant are 30.5 m/s (100 fps) and 198 m/s (650 fps) ΔV respectively. The residuals/unusable and subsystem fluid weights are based on a detailed analysis and design. Due east payload is 29,077 kg (64103 lb.).

STRUCTURE	71,178 kg (156,922 lb)	
BODY STRUCTURE		35,265 (77,746)
WING STRUCTURE		26,172 (57,700)
TAIL STRUCTURE		3,270 (7,210)
STRUCTURES MARGIN		6,471 (14,266)
SUBSYSTEMS	28,103 kg (61,958 lb)	
SURFACE CONTROLS		998 (2,200)
LANDING GEAR		3,342 (7,368)
ROCKET ENGINES		13,458 (29,669)
PROPELLANT FEED		983 (2,169)
PRESSURIZATION		726 (1,600)
RCS SYSTEM		782 (1,724)
OMS SYSTEM		718 (1,583)
AVIONICS		1,306 (2,880)
PRIME POWER		358 (790)
ELECT CONV & DIST		1,619 (3,570)
HYD CONV & DIST		1,619 (3,570)
ENVIRONMENTAL CONTROL		986 (2,173)
PERSONNEL PROVISIONS		1,134 (2,500)
SUBSYSTEMS MARGIN		362 (797)
		1,331 (2,935)
VEHICLE DRY WEIGHT	99,282 kg (218,880 lb)	
PERSONNEL		263 (580)
PAYLOAD		29,076 (64,103)
FLUIDS	16,300 kg (35,937 lb)	
FLIGHT PERFORMANCE RESERVES		2,218 (4,890)
REACTION CONTROL PROPELLANT		1,249 (2,753)
ORBIT MANEUVER PROPELLANT		5,114 (11,275)
RESIDUALS/UNUSABLE		6,158 (13,576)
SUBSYSTEM FLUIDS		1,562 (3,443)
VEHICLE INJECTED WEIGHT		144,923 (319,500 lb)
ASCENT PROPELLANT		854,568 (1,884,000 lb)
PRELAUNCH WEIGHT		999,491 kg (2,203,500 lb)

Table 12 HTO-SSTO Weight Statement

Flight Performance

Ascent. HTO vehicle trajectory and selection rationale are presented in this section along with the analysis of optimization parameters and constraints consistent with design loading conditions and mission requirements. The data results provided an injection weight capability of 141,757 kg (319,131 lb) to be used for structural and subsystem analysis to finally determine orbital payload capability. Vehicle loads and equilibrium temperatures were also developed from the baseline trajectories presented in Figures 73 and 74

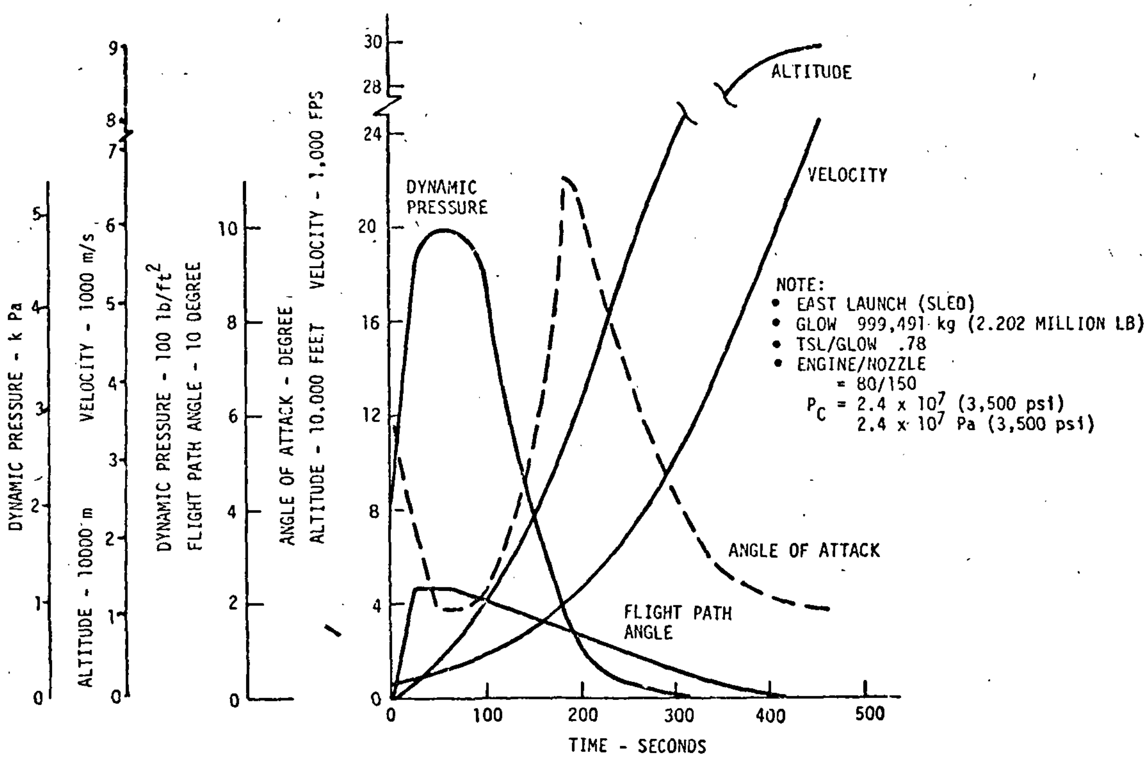


Figure 73 HTO SSTO Ascent Trajectory

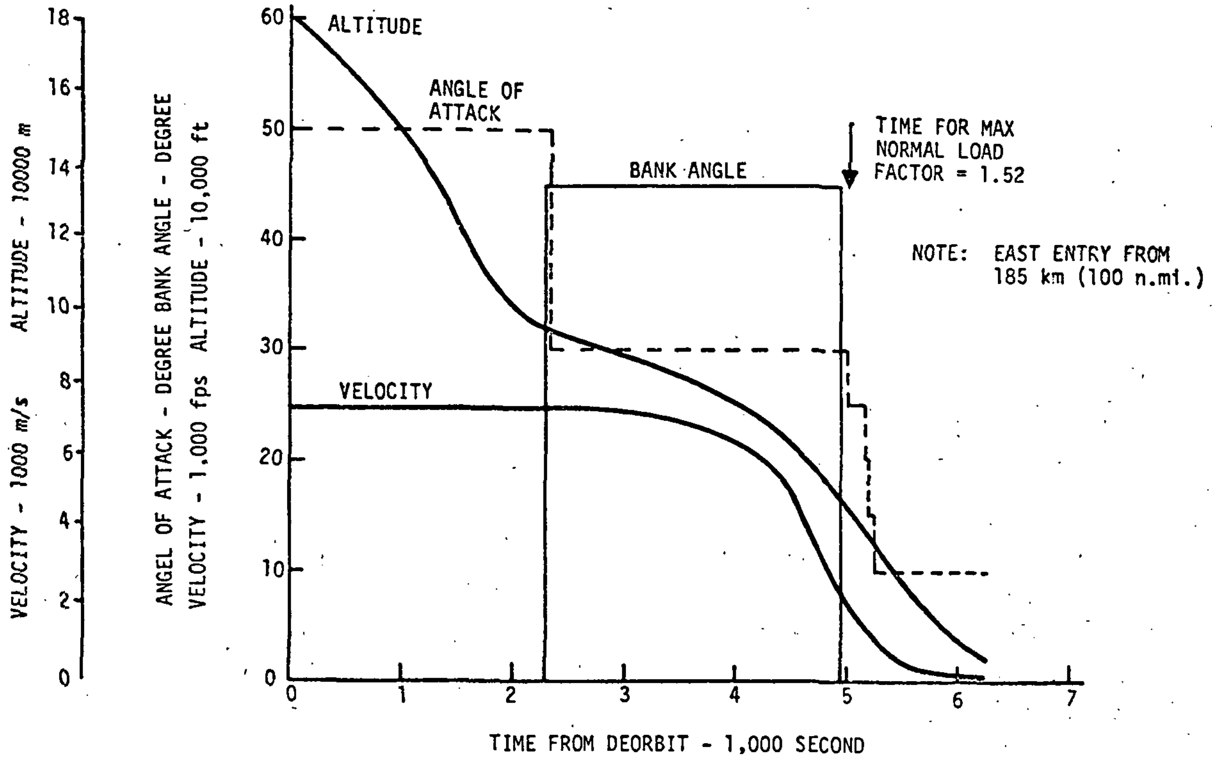


Figure 74 HTO SSTO Entry Trajectory

Engine/nozzle trade studies, see Vehicle Performance Trades, have indicated significant performance gains of using two-position extendable rocket nozzles for the HTO vehicles. For the baseline vehicle expansion ratios, ε, of 80/150 were initially selected with a liftoff thrust loading T/GLOW of 0.79 (subsequent engine/nozzle trades, revealed alternate selections to increase payload performance). The first position had a nozzle expansion ratio of 80:1 which was extended to 150:1 at an altitude of approximately 15240 m (50,000 ft) to increase rocket engine specific impulse and thrust. The engine vacuum specific impulse for the 80/150 expansion ratios with a chamber pressure of 24 x 10^6 Pa (3500 psia) were 455.2 and 465.2 seconds, respectively. For these man-rated systems the tangential load factor was limited to 3 g's.

The HTO vehicle is east launched from ETR. A Boeing trajectory program AS2530 was used to determine the trajectory characteristics for these studies, see Figure 73 for ascent. Although these are not true optimized trajectories, past studies have indicated that increases in payload should not exceed about 1360 kg (3000 lb), and practically can be even less than this when dynamic pressure, q, and angle of attack, α, constraints on trajectory and structural weights are taken into account. The flight sequence of the selected ascent trajectory for the HTO vehicle is described as follows:

With a horizontal takeoff from a sled ground accelerator of 182.88 m/s (600 fps), a pull-up was made to a flight path angle of 23 degrees with an angle of attack not exceeding 13 degrees and with a normal load factor of 1.25. This flight path angle was held constant until after passing the maximum dynamic pressure region of 46922 Pa (980 psf), then gradually reduced at the rate of 0.08 degrees per second until the inertial velocity increased to 1524 m/s (5000 fps), where an iterative guidance mode was activated to steer the vehicle to the terminal injection points of 7891.3 m/s (25,890 fps) inertial velocity, 92354 m (303,000 ft) altitude and zero flight path angle.

The total velocity losses were 1496 m/s (4910 fps) for ascent to an injection orbit of 92.6 km (50 n.mi.) by 185.2 km (100 n.mi.). The largest contributor was gravity with 55 percent followed by drag with 29 percent of the total losses. The remainder was composed of the rocket engine thrust vector and back pressure losses. Payload sensitivity factor due to drag for the HTO vehicle is 56.7 kg (125 lb) of payload change per one percent change in minimum drag coefficient.

Descent. The descent trajectory (Figure 74) was initiated with a deorbit ΔV of 33.5 m/s (110 fps) from a 185.2 km (100 n.mi.) circular orbit with an east entry and with 28.5^{o} orbit inclination. An initial angle of attack of 50 degrees was maintained until the flight profile first leveled off (i.e. flight path angle = 0 degrees) followed by a decrease in angle of attack to 30 degrees to provide a high cross range; bank angle of 45 degrees was also initiated at this time. These control angles were held fixed until the velocity had decreased to about 1524 m/s (5000 fps), at which point, the bank was removed and a transition from 30 to 10 degrees angle of attack was accomplished. It was estimated that aerodynamic directional control was restored at these flight conditions (RCS not required beyond this point). This trajectory achieved a cross-range slightly in excess of 2222 km (1200 n.mi.). The preliminary thermal analysis was based upon this entry trajectory. Entry wing loading based upon reference area was about 1388.5 Pa (29 psf) and at 30 degree angle of attack equilibrium glide, $W/(SC_L)$ is 2681 Pa (56 psf). The above wing loading corresponds to a planform loading of approximately 1245 Pa (22 psf).

Vehicle Performance Trade Studies. Rocket engine/nozzle trades on payload and Gross Lift-Off Weight (GLOW) were made for variations in thrust level, number of engines, chamber pressure and expansion ratio (ε) of 2-position nozzles. Ballast weight was included in the analysis when engine/nozzle weights exceeded the baseline configuration in order to maintain comparable aero stability characteristics due to C.G. movement. The trades with Glow are presented in Figures 75,76, and 82 and with payload in Figures 77 to 81.

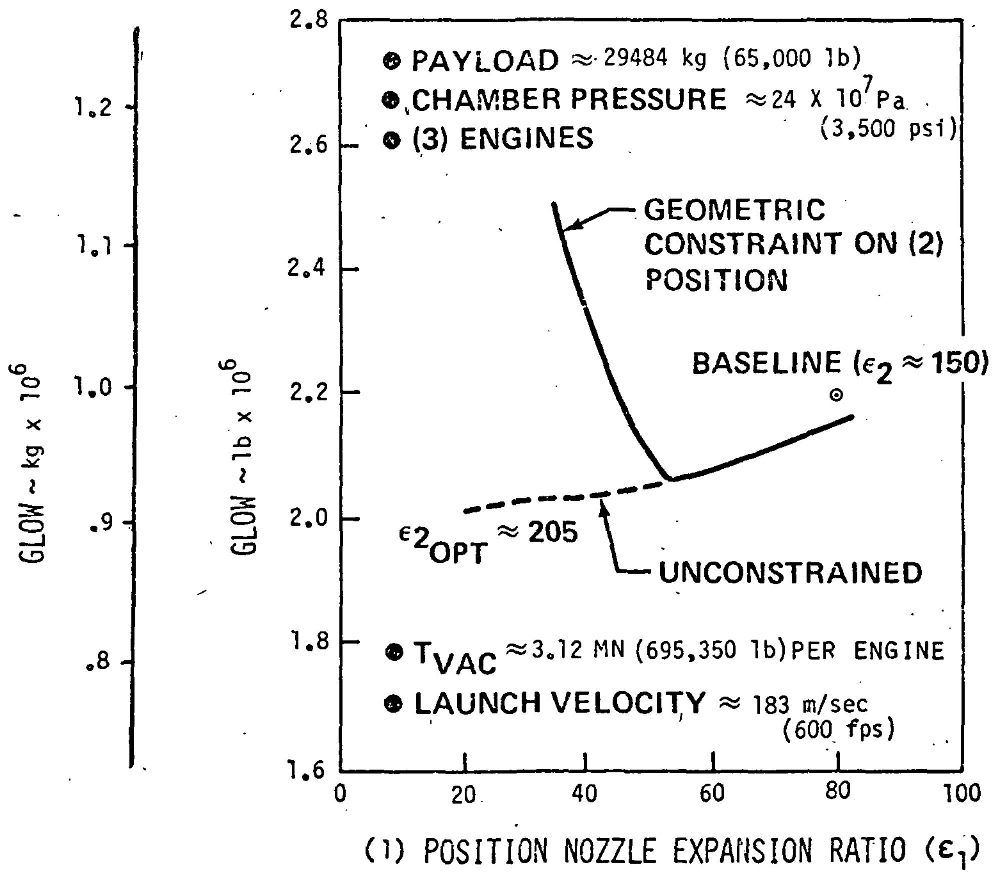


Figure 75 GLOW Versus Expansion Ratio Trade Study

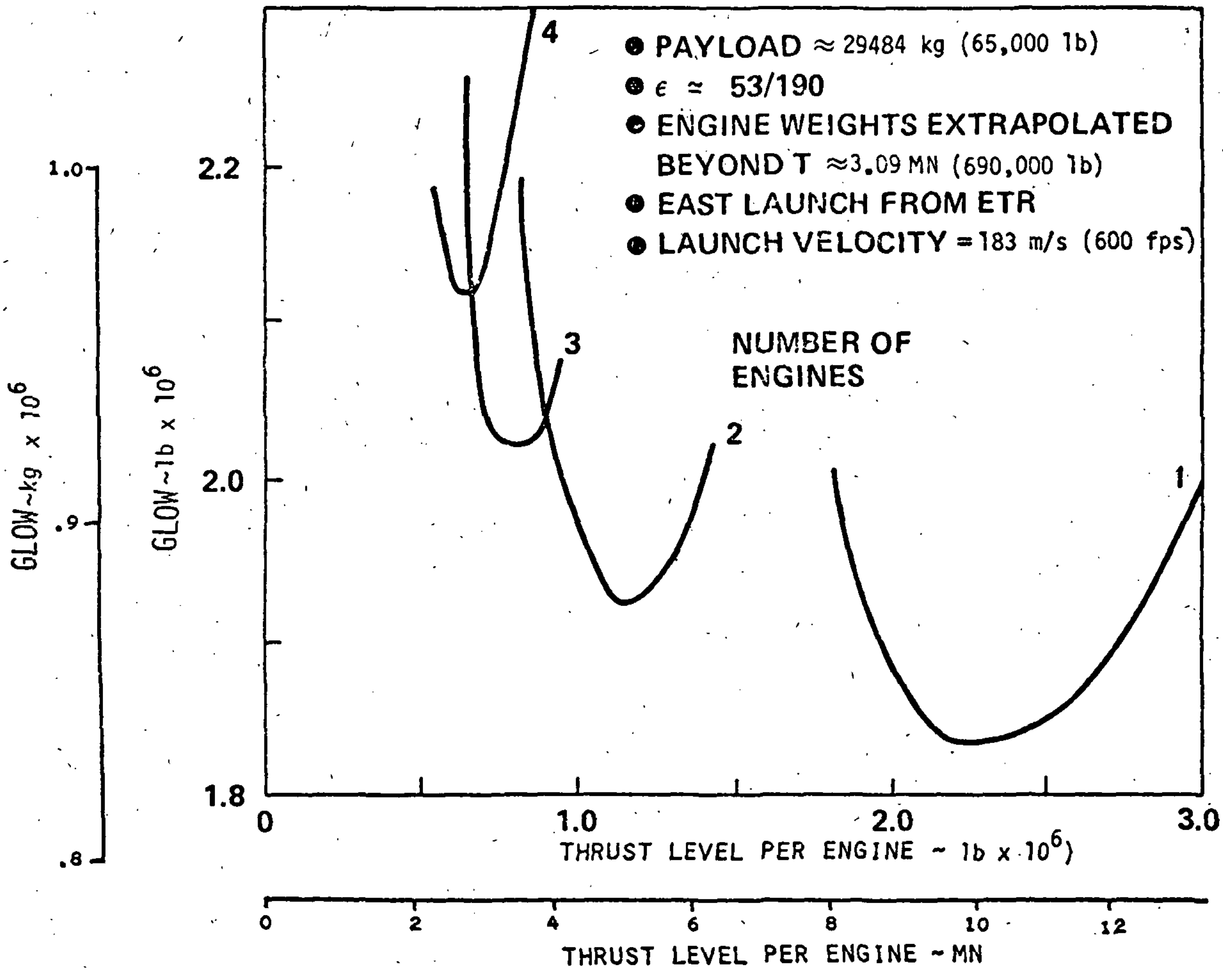


Figure 76 GLOW Versus Thrust Level Trade Study

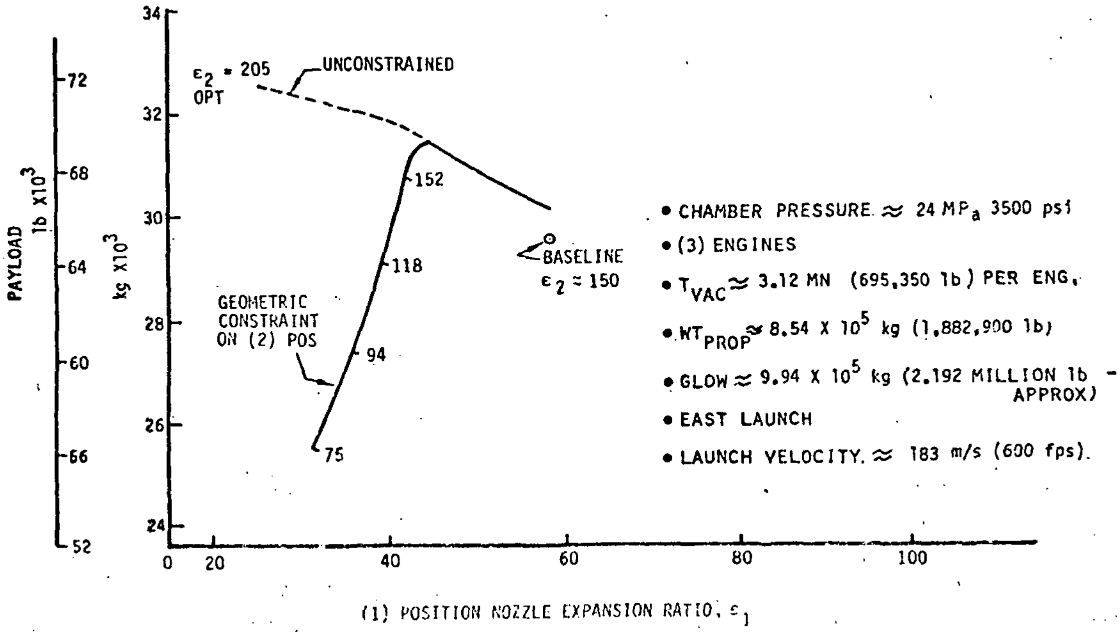


Figure 77 Nozzle Expansion Versus Payload Trade Study

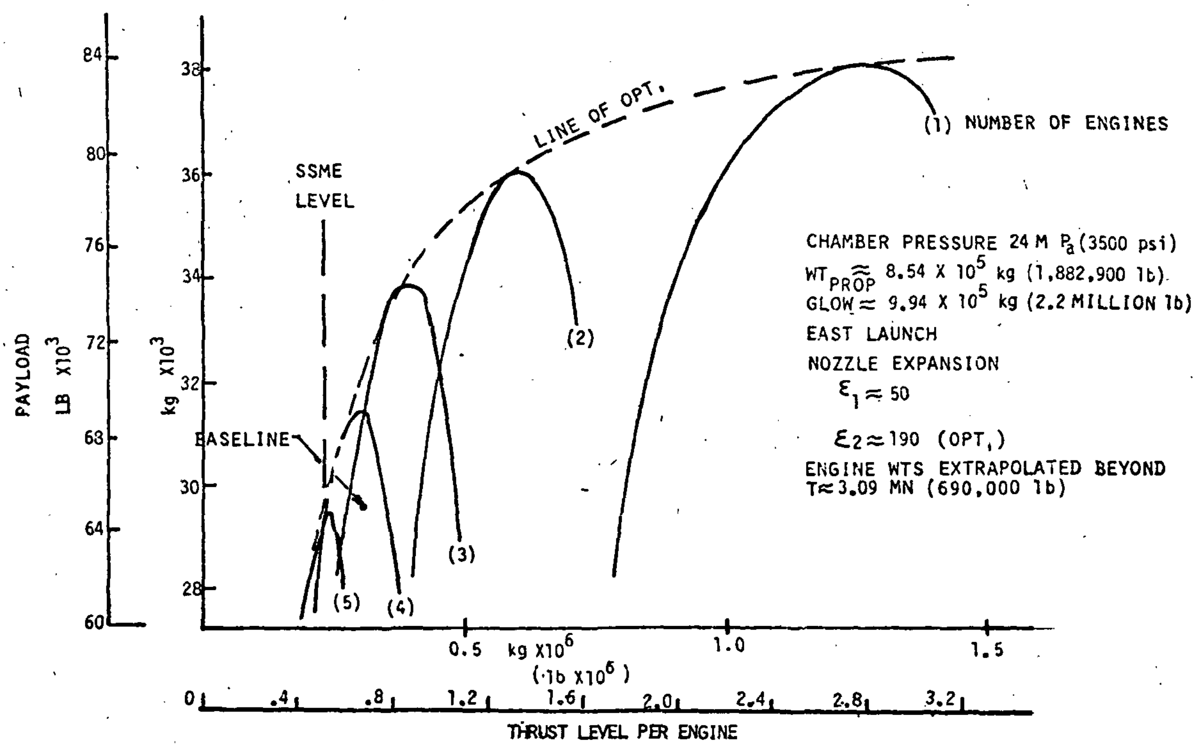


Figure 78 Thrust Level Versus Payload Trade Study

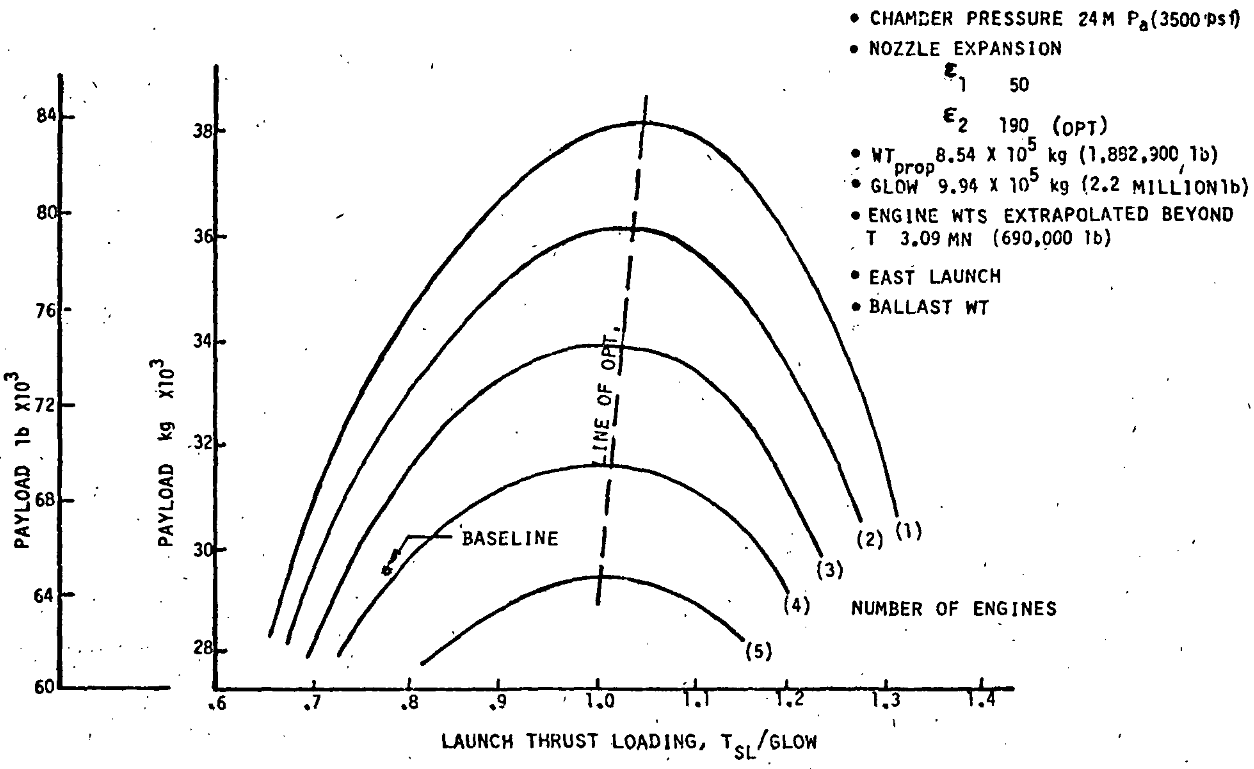


Figure 79 Launch Thrust Loading Versus Payload Trade Study

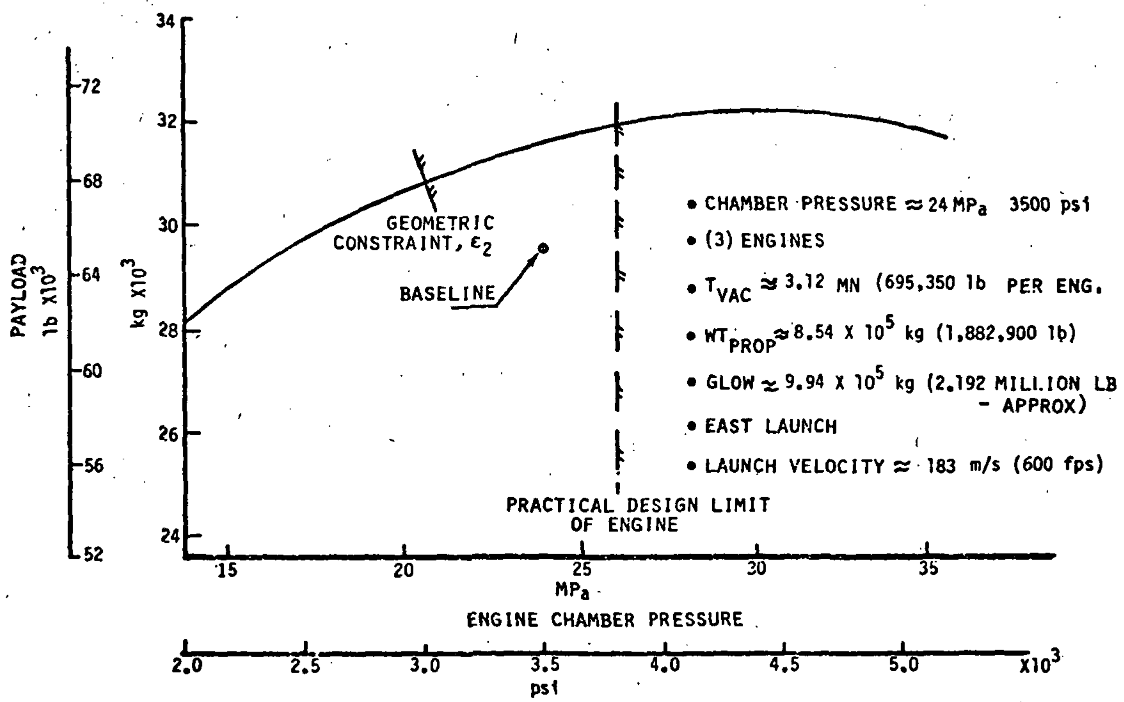


Figure 80 Engine Chamber Pressure Versus Payload Trade Study

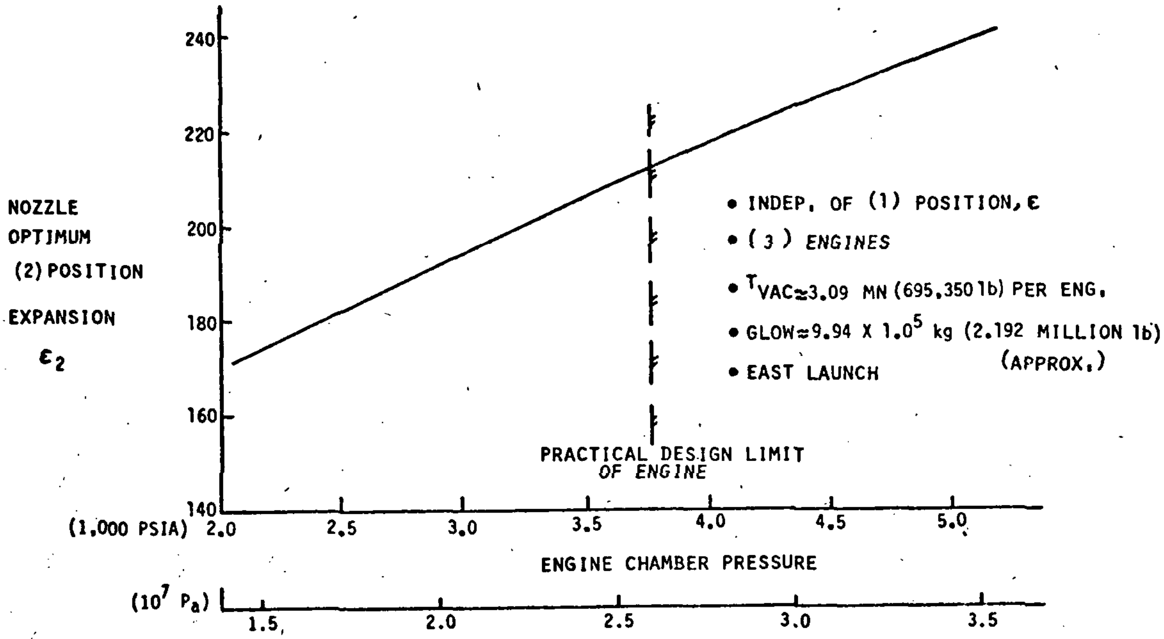


Figure 81 Engine Chamber Pressure Versus Nozzle Expansion Trade Study

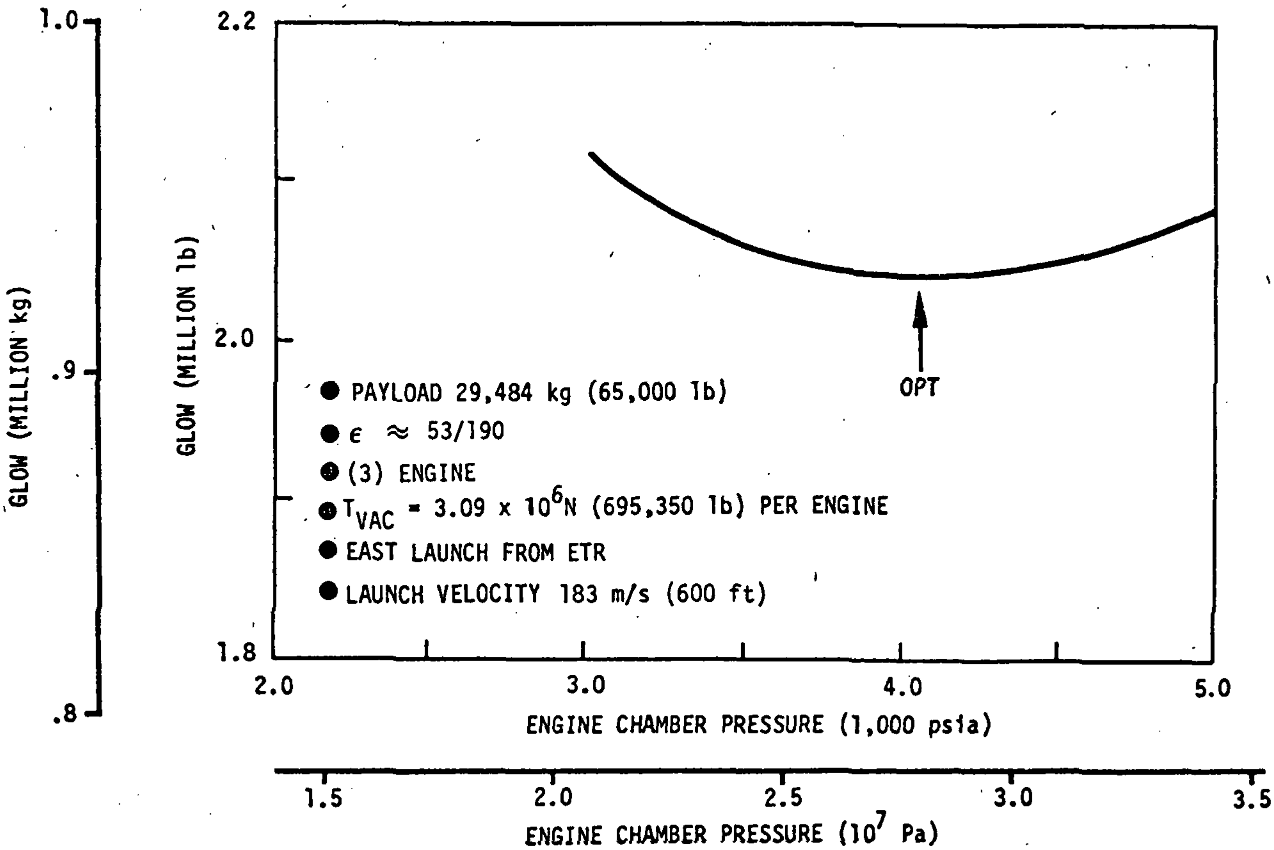


Figure 82 GLOW Versus Chamber Pressure Trade Study

With no constraints on possible nozzle expansion ratios a 2-position nozzle attains best performance in terms of payload or GLOW when the first position (i.e. nozzle retracted) has a low expansion ratio of about 20 along with a second position (extended) of about 200 for a chamber pressure of 24 M Pa (3500 psia). When geometric constraints are included the second position ϵ is reduced at low first positions and an optimum is obtained with the first position increased to a ratio of 54. Shown in Figures 76 and 82 are the original baseline and a fixed nozzle configuration to illustrate the relative performance gains of two position nozzles.

Increasing chamber pressure up to about 28.9 M Pa (4200 psia) improved vehicle performance, see Figure 82. Practical design considerations may limit the chamber pressure to lower values than this optimum. These study results used a first position ϵ = 50 and an unconstrained optimum of the extended position of the nozzle.

When engine thrust level trades are undertaken, care must be exercised in defining the ground rules in order to understand the particular study results. For the results presented in Figure 75 and 76 it is assumed that integral number of engines are used and the engine performance and weight trends Are similar to those currently supplied by Rocketdyne. These results show vehicle performance improves significantly with increasing thrust level per engine and reduced number of engines. Optimum values occurred at sea level thrust loadings close to one. If the effects of thrust loading on the vehicle structural (higher dynamic pressure and tangential acceleration) weight are included this optimum would be reduced a little. The original baseline vehicle has a thrust loading of 0.78.

In the extended performance studies, the final baseline configurated has taken advantage of the gains shown in these engine/nozzle trade study results.

Vehicle Aerodynamics. Drag: Drag coefficients, C_D, have been estimated over the entire speed range from subsonic to hypersonic conditions. Analysis methods are based upon DATCOM and well established in-house techniques. Minimum drag coefficient, C_{DO}, along the ascent trajectory is shown in Figure 83 and a component buildup of C_{DO} at a Mach number of two, see Table 13, indicated that the largest drag component was the wing wave drag ($\sim$ 50 percent power on) due to its high thickness ratio of about 10 percent. Velocity losses due to drag during the ascent boost in the supersonic range ($M \approx 1$ to 4) account for about 90 percent of drag losses. These velocity losses due to drag were 434 m/sec (1424 fps).

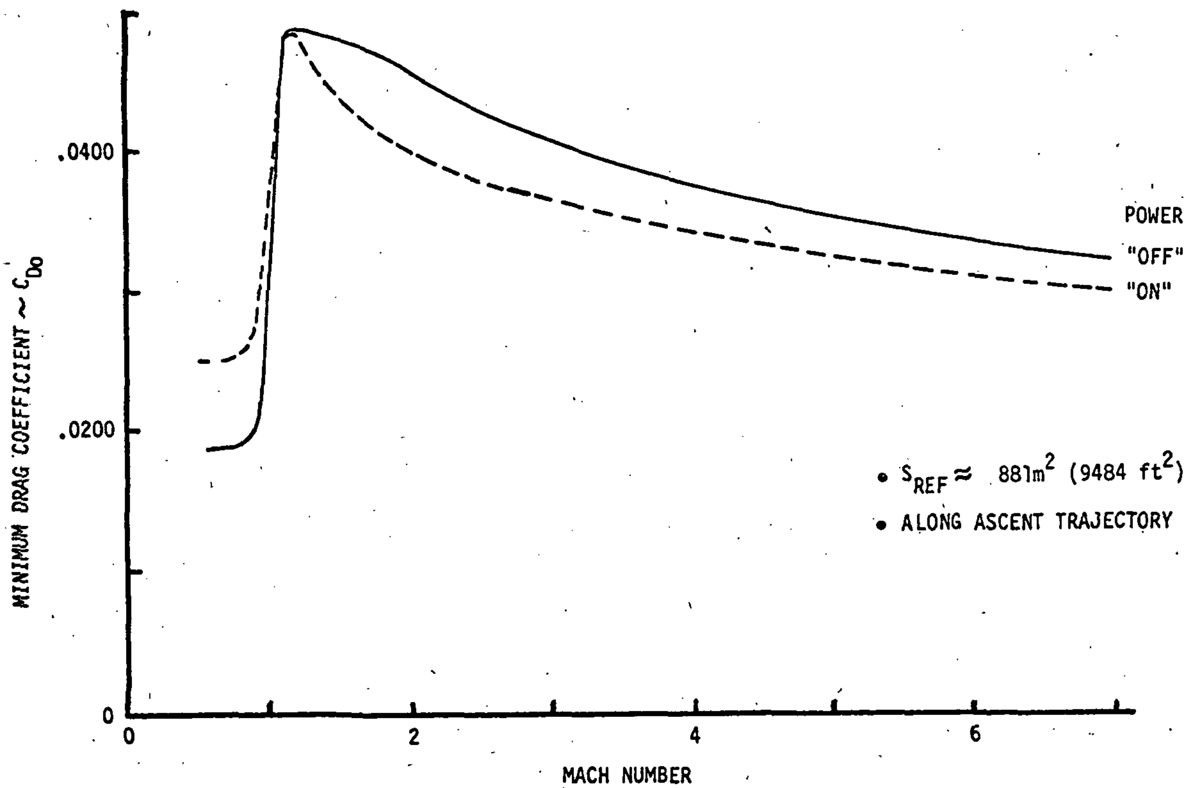


Figure 83 HTO SSTO Drag Coefficient

CONFIG. COMPONENT	DRAG COMPONENT, C_{D_0}			
	FRICTION	WAVE	BASE	SUBTOTL.
WING	.0020	.0198	0	.0218
BODY	.0020	.0062	.0093/.0031	.0175/.0113
TAIL	.0005	.0038	0	.0043

SUBTOTAL .0436/.0375

POWER "OFF"/"ON"

$M \approx 2.0$

ADD 5% FOR MISCEL ITEMS AND CONSERVATISM

TOTAL $C_{D_0} \approx .0456/.0395$

POWER "OFF"/"ON"

Table 13 Drag Coefficient Buildup

Lift: The estimated normal force coefficient slope with angle of attack, CN/α , was estimated over the ascent speed range. Wing tunnel test data at subsonic speeds were also obtained for the HTO configuration. Between Mach number 2 and 4 the drag due to lift (i.e. dc_D/dC_L2) is inversely proportional to the CN/α values for angles of attack up to about 10 degrees. At hypersonic speeds, modified Newtonian Theory was used to determine normal force coefficients and drag due to lift.

Moment: Aerodynamic moment coefficients were estimated in order to determine the vehicle's static stability and control characteristics. Of most interest is the attainment of stable static stability margins (i.e. $dC_M/dC_L \geq -.02$) at subsonic speeds and, at least, a neutral margin at hypersonic speeds. The vehicle must also be aerodynamically trimmable over the design angle of attack range at subsonic and supersonic speeds along with very high angle of attack trim capability at hypersonic speeds (perferably with neutral or up elevons). Regions where the dynamic pressure is too low for aerodynamic control, RCS provides the necessary control characteristics. During ascent, aerodynamic control also assists the power on rocket engine control through its nozzle gimballing capability. All these considerations impact a very narrow range of permissible center of gravity positions for the vehicle configuration. The HTO vehicle meets all of these preliminary design criteria. The HTO subsonic and hypersonic aerodynamic stability and trim characteristics are presented in the following section.

Subsonic Aero: This configuration is very similar to a Boeing configuration which was tested in the NASA/Ames 14-ft wind tunnel (Test 032-1-14) during July of 1974. The present configuration has increased the leading edge sweep of the wing one degree to a value of 56 degrees, Figure 84.

The aerodynamic center from the wind tunnel model was located at 0.700 in terms of body length. Using DATCOM methods, the estimated location was 0.713. This provided a good basis for employing the DATCOM methods. The increased L. E. sweep had the effect of moving the aerodynamic center aft about 0.010 reference to body length. This configuration does not experience any pitch up (unstable) characteristics for angles of attack as high as 25 degrees (limit of available test data). At takeoff and landing angles of attack (from 13° to 7°)

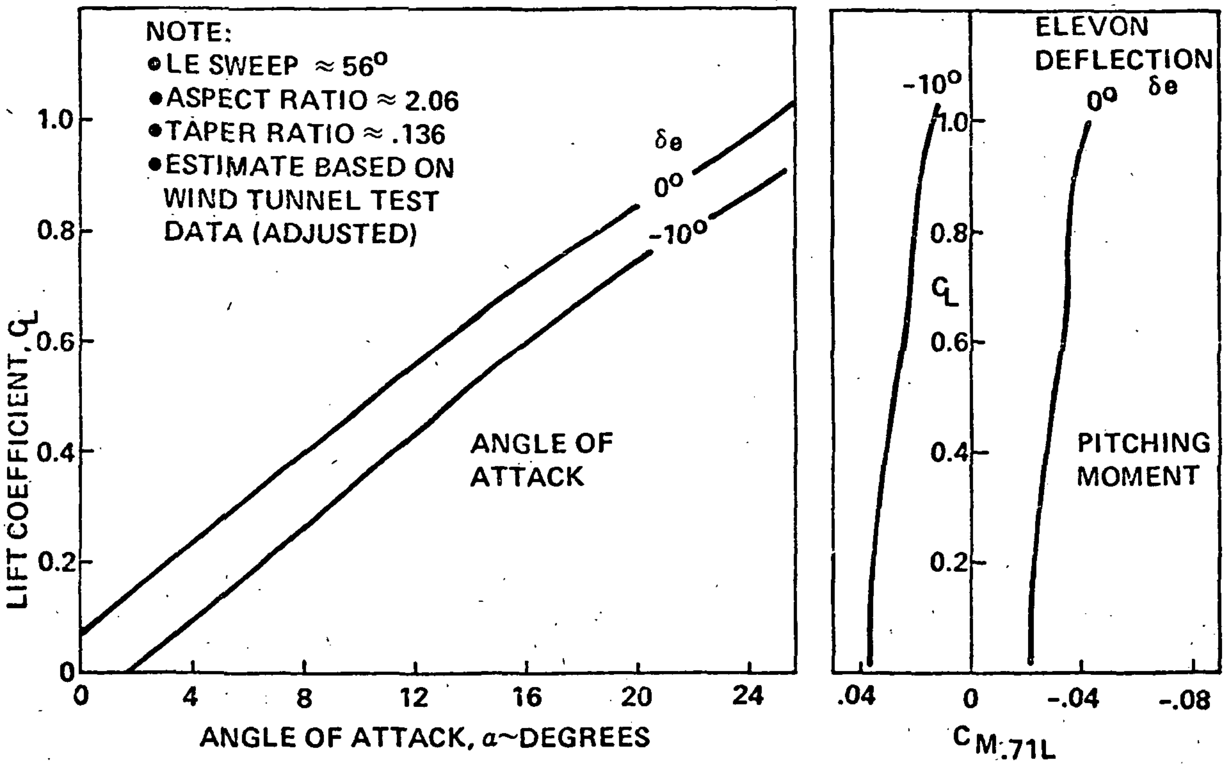


Figure 84 HTO SSTO Subsonic Aero Characteristics

the aerodynamic center is located approximately 0.715 of body length. Landing speeds at these trimmed angles range from 69.5 m/s (135 kt) to 84.9 m/s (165 kt), respectively.

Hypersonic Aero: For a moment center located at 0.71 of body length the estimated variations of pitching moment with normal force coefficients at various elevon deflections at hypersonic speeds are presented in Figure 85 These estimates are based upon using modified Newtonian Theory and comparison with test data. At this C.G. location, the configuration is stable and trimmable with ±5 degrees of elevon deflection. For angle of attack entry requirements from 50 to 30 degrees the aft C.G. limits range from 0.73 to 0.715 and the corresponding trim limits require from 8 to 5 degrees of down elevons. For elevons to not be deflected downward the C.G. should not exceed 0.705 of body length. The estimated available aft C.G. for entry and landing is 0.715 resulting in a stable and trimmable vehicle at both hypersonic and subsonic speeds.

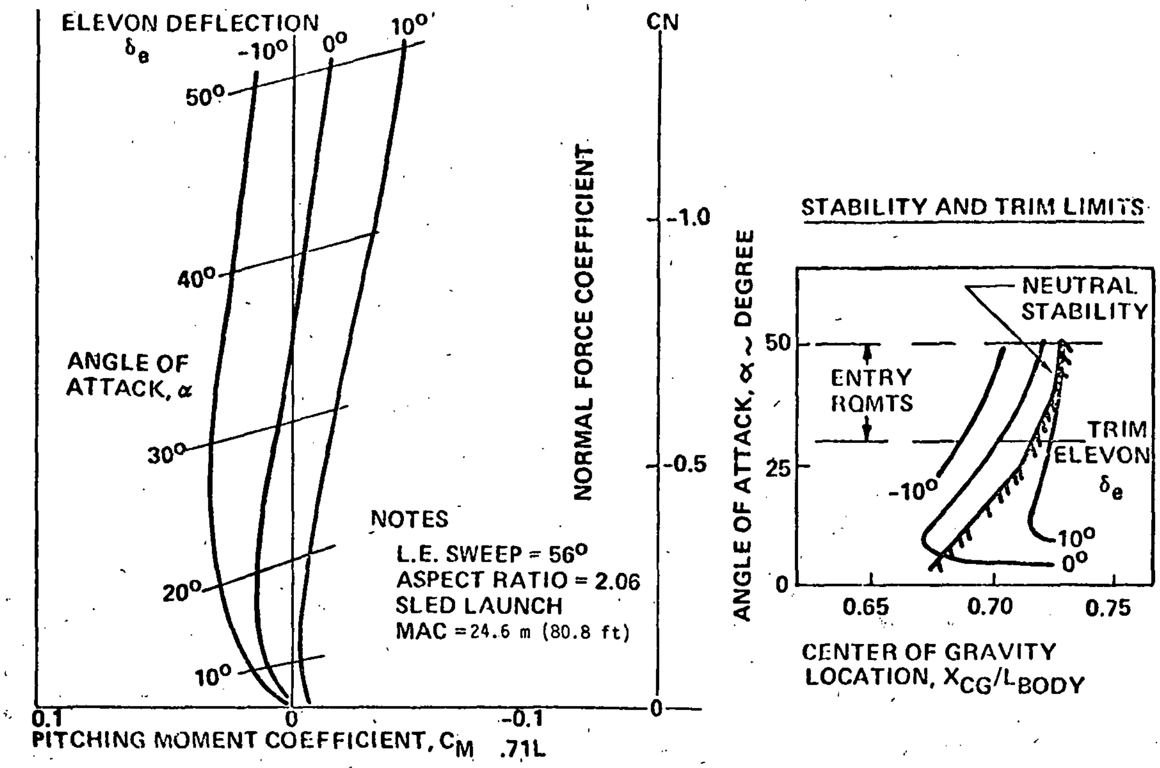


Figure 85 HTO SSTO Hypersonic Aero Characteristics

Configuration 2 - Aerial Refueling

System Configuration. The ALRS 207 Air Launch Configuration as shown in Figure 86 was developed to support both the aerial refueling and the air launch study candidates. The major reason for the substantially smaller size of the aerial refueling concept is that it is refueled in the air at 9144m (30,000 feet).

The summary chart, Table 14, identifies some of the major benefits and penalties associated with aerial refueling as an operational approach to SSTO. As originally conceived, it was assumed that a ground accelerator launched vehicle would be configured. However, further investigation indicated that the lightly loaded vehicle realized very little benefit from a ground accelerator. Thus of the operational candidate concepts only Options C, D3 and F of Table 15 utilized a ground accelerator vehicle. All of the options with exception of Option C require the development of a new 1.8×10^6 to 2.3×10^6 kg (4 to 5 million lb) gross takeoff weight tanker aircraft. Option C

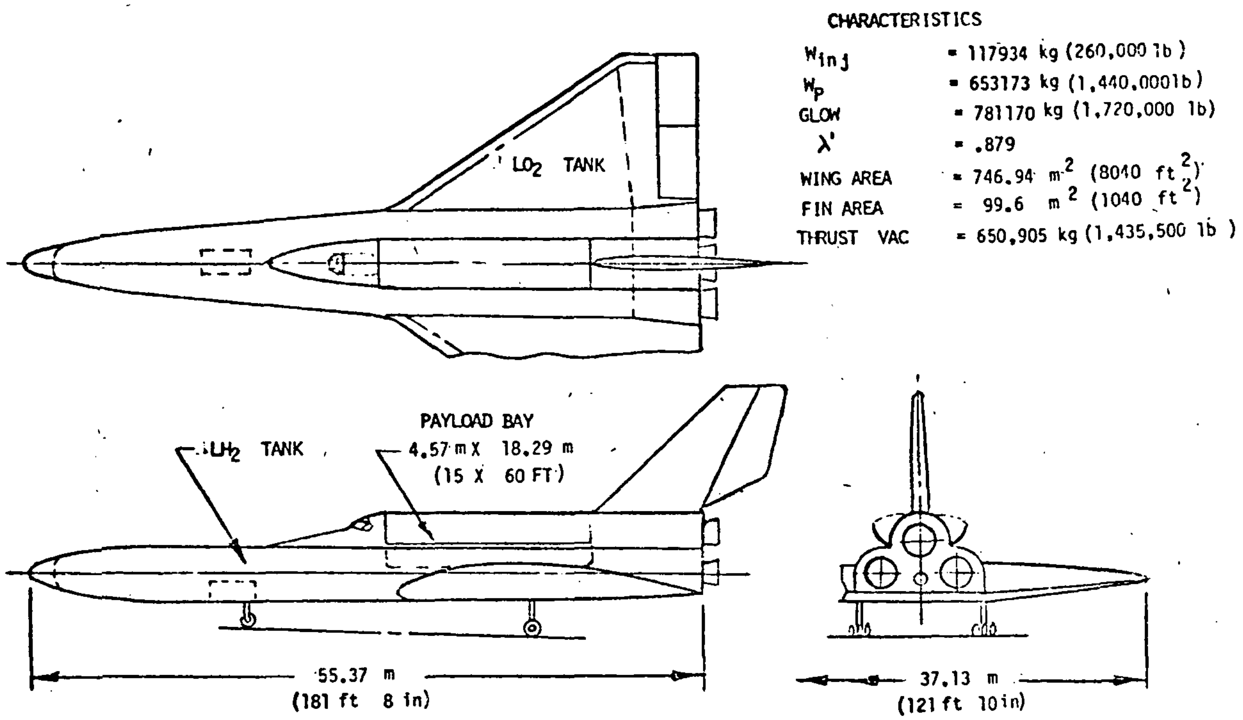


Figure 86 Aerial Refueling Vehicle Configuration

PRO	CON
REDUCES SIZE OF SSTOV FOR A GIVEN PAYLOAD - VEHICLE SIZED FOR 9144 m (30,000 ft) ALTITUDE & M = .5 LAUNCH	ADDED LOX & HYDROGEN LINES - TANK BAFFLES & UMBILICAL CONNECTIONS
PROVIDES HIGHER λ' FOR A GIVEN PAYLOAD	ADDITIONAL HAZARD EXPOSURE
MINIMUM WEIGHT TAKEOFF PERMITS ELIMINATION OF GROUND ACCELERATOR VEHICLE	FLIGHT INTERRUPTION REQUIRES REVISED GN & C UPDATE ADDITION ADDITIONAL SAFEING & INERTING EQUIPMENT & PROCEDURES
GROUND SUPPORT EQUIPMENT MINIMIZED	RECONNECTABLE CRYOGEN UMBILICAL IS MAJOR DEVELOPMENT
LAUNCH POINT, AZIMUTH-LONGITUDE-LATITUDE LIMITED BY TANKER-TANKER TOWS SSTO TO LAUNCH POINT	WEIGHT & BALANCE FOR CG CONTROL MAJOR PROBLEM AREA FOR BOTH VEHICLES

Table 14 Aerial Refueling Benefit Analysis

OPERATING CONCEPTS	TANKER T.O. WT	SSTO O.W.E	SSTO T.O. WT	SSTO GROSS	FUEL / LOX X-FR
A SSTO TAKEOFF & CLIMB TO RENDEZVOUS WITH TANKER - M = .6 9144 M (30,000 FT) X = FR LO_2 & LH_2	2.145×10^6 kg (4.73×10^6 lb)	$.79 \times 10^5$ kg (1.75×10^5 lb)	$.94 \times 10^5$ kg (2.07×10^5 lb)	$.75 \times 10^6$ kg (1.65×10^6 lb)	92,014 kg (202,857 lb) / 562,086 (1,217,142)
B SSTO TAKE-OFF WITH FULL LOAD OF LH_2 RENDEZVOUS WITH TANKER M = .6 9144 M (30,000 FT) X-FR LH_2 LO_2	1.84×10^6 (4.057×10^5)	82,361 (181,575)	206,242 (454,687)	$.75 \times 10^6$ (1.656×10^6)	552086 (1217142)
C SSTO TAKE-OFF WITH FULL LOAD OF LO_2 RENDEZVOUS WITH TANKER M = .5 9144 M (30,000 FT) X-FR LH_2	3.07×10^5 (6.76×10^5) . *747 POTENTIAL MODIFIED +3 ENGINES	$.799 \times 10^5$ (1.75×10^5) *GRND ACCL	$.799 \times 10^5$ (1.762×10^5)	$.75 \times 10^6$ (1.65×10^6)	92014 (202857) / — —
D 1 - SAME AS "A" XFR SLUSH HYD; 2 - SAME AS "B"; 3 - SAME AS "C"	WEIGHT CHANGES MINM; WEIGHT CHANGES MINIMAL - ASSOCIATED WITH 15% REDUCTION IN LH_2 TANK VOLUME				
E SSTO TOWED EMPTY XFR LO_2 & LH_2 M = .6 & 9144 M (30,000 FT)	2.145×10^6 (4.73×10^6)	$.79 \times 10^5$ (1.75×10^5)	$.79 \times 10^5$ (1.75×10^5)	$.76 \times 10^6$ (1.65×10^6)	92,014 (202,857) / 552,086 (1,217,142)
F SSTO TOWED - BALANCE FUEL LOAD BETWEEN TANKER & SSTO XFR LO_2 & LH_2 M = .6 & 9144 M (30,000 FT)	1.459×10^6 (3,217,480)	87,856 (193,690)	3.82×10^5 (842,530)	819,992 (1,807,773)	62,547 (137,892) / 375,280 (827,352)

Table 15 Aerial Refueling Candidate Concepts

could possibly be a modified 747 vehicle. Additional engines would be required to accommodate the towing requirement during fuel transfer. This requirement developed from the fuel flow transfer capability and the minimum flow rate requirement for the three SSME engines. Examination indicated that the transfer rate only slightly exceeded the flow requirements of the rocket engines, necessitating that for the transfer period the engines be shut down. Thus, in effect the aerial refueled concept also becomes the air launched concept.

The mission launch profile is illustrated in Figure 87. The major elements associated with the launch site are identified, as well as the significant points on the launch profile. As previously noted, the various configuration concept options would slightly modify the profile, however, the significant features would be identical.

The major elements of the fuel transfer system are shown on the aerial refueling schematic, Figure 88; those associated with the tanker aircraft to the left with those of the SSTO vehicle to the right. The LO_2 would be

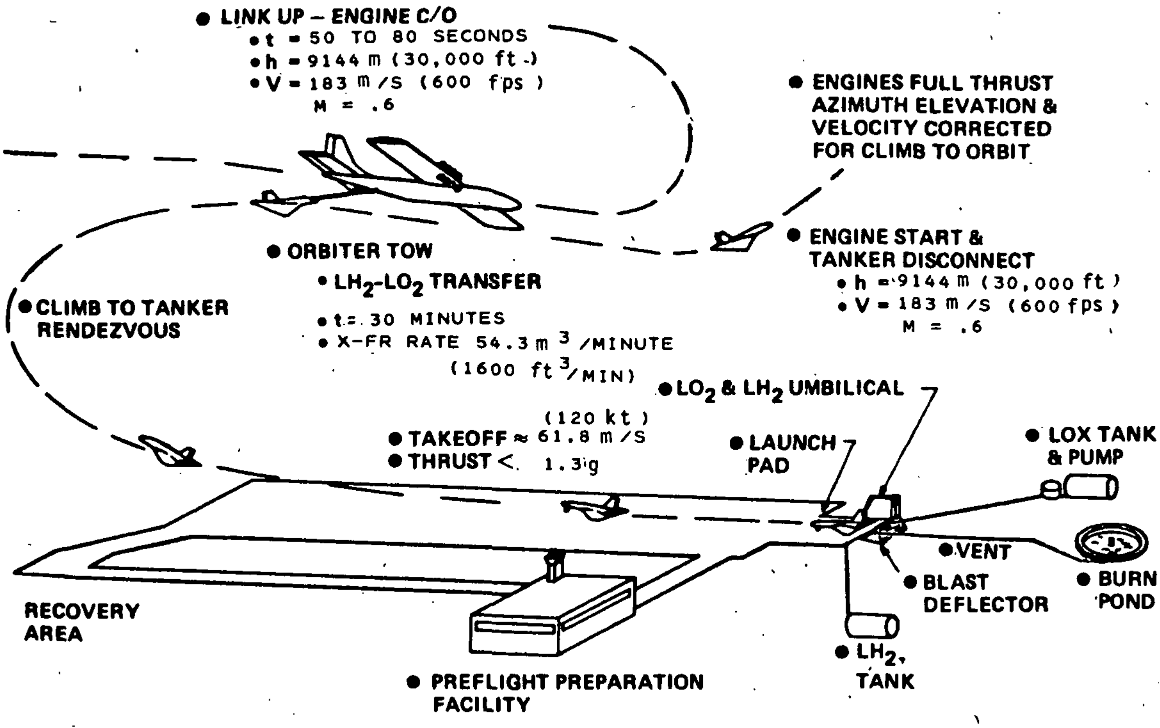


Figure 87 Aerial Refueling Mission Profile

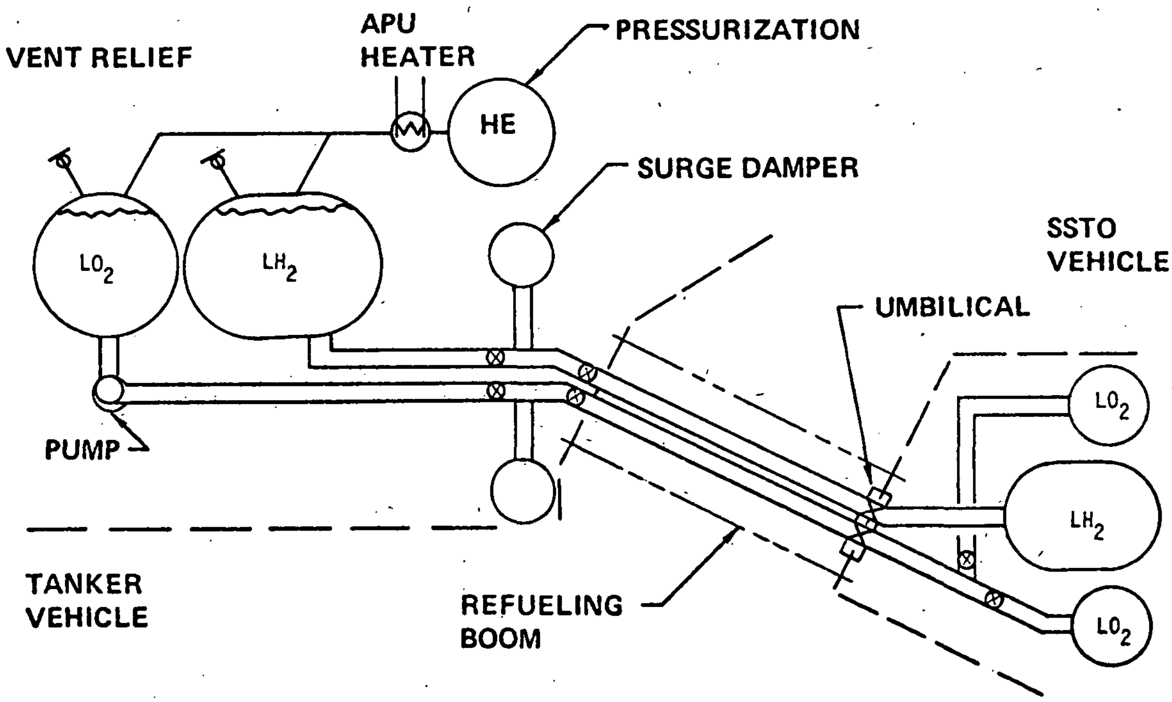


Figure 88 Aerial Refueling System Schematic

pumped by a high capacity hydraulic driven pump. The hydrogen would be pumped by the regulated pressure head in the hydrogen tank. Pressurization for both tanks is provided by a helium pressure tank exhausted through a heat exchanger on the APU exhaust through pressure regulator valves to the fuel and oxidizer tanks. Vent relief valves provide additional regulation and control. The fuel and oxidizer are routed through separate circuits to the refueling boom. Upstream of the boom shutoff valves control delivery. Surge dampers are provided in both lines to smooth delivery pressures and to provide a dump accumulator for inadvertant shutoffs or disconnects. Parallel circuits in the boom provide elevation and azimuth accommodation for the boom through individual dual bellows. Length adjustments are accepted by a compound bellows arrangement. The boom outer structure accommodates all loads transmitted from the umbilical to the tanker aircraft, e.g., flow and pressure loads as well as SSTO towing loads. Automatic vent and purge valves are accommodated in the umbilical. The umbilical receptacle on the SSTO connects to the hydrogen and LO_2 tanks through control valving to regulate the flow to the various tanks and in that manner control loading to unstable c.g. locations.

The aerial refueling tanker configuration shown on Figure 89 is

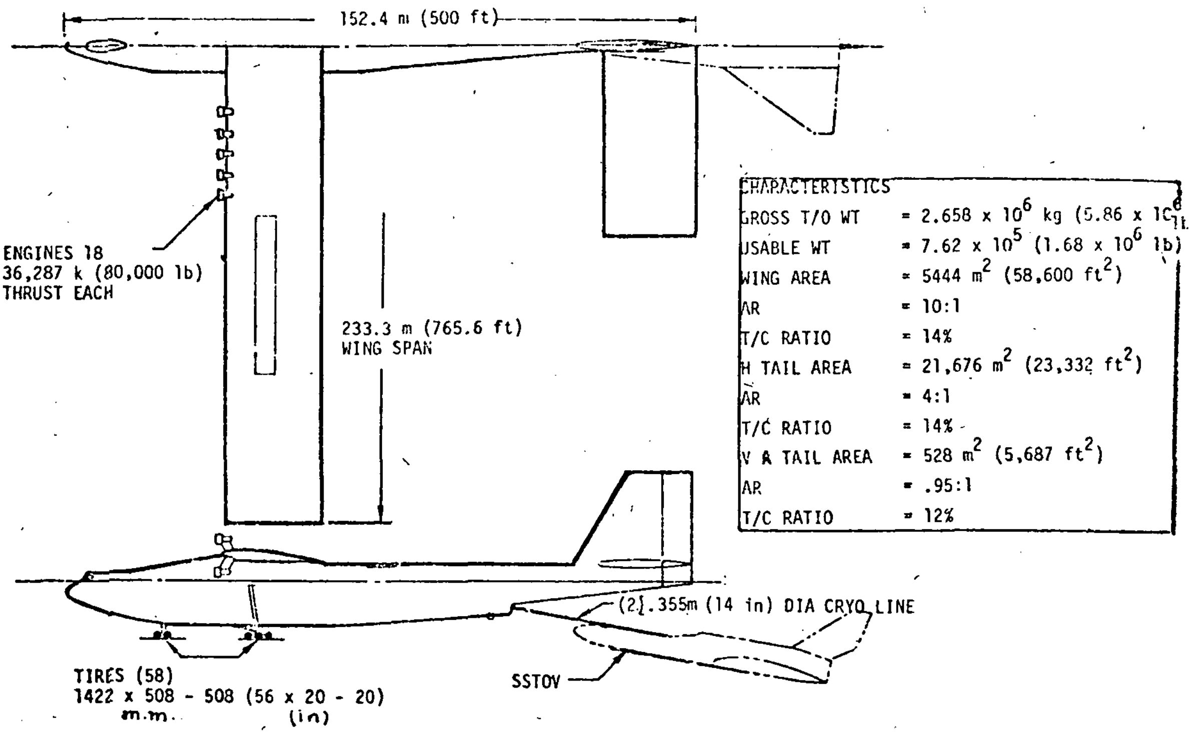


Figure 89 Aerial Refueling Tanker Configuration

shown to illustrate the size of the vehicles under consideration. The SSTO vehicle is shown in phantom for comparison. The large horizontal tail area provides pitch stability for large c.g. location transfers which occur during refueling. The straight wing and fixed landing gear are considered compatible with the mission considered for the vehicle. The internal tankage is contained in externally insulated tanks suspended with thermal isolators from the aircraft fuselage structure. The engines shown are considered to be normal growth versions of existing large high bypass engines.

The concept feasibility of the SSTO vehicle aerial refueling appears to be acceptable within the time frame specified for the SSTO system. However significant development items are required. The major items are identified in Table 16 with comments as to their probable availability as well as some estimation of the success probability of a development program.

ITEM	COMMENTS
TANKER VEHICLE 2.5 x 10^6 kg T.O. WT (4.73 x 10^6 lb)	NUMEROUS STUDIES OF LARGE RESOURCES TRANSPORT AIRCRAFT CORROBORATE FEASIBILITY
3.5 x 10^5 NEWTONS (80,000 lbs) THRUST AIR BREATHERS & ENGINES	CURRENT ENGINE DEVELOPMENTS WITH HIGH BYPASS RATIO ENGINES INDICATE AVAILABILITY
RECONNECTABLE CRYOGENIC UMBILICALS	APOLLO PROGRAM DEVELOPMENT THOUGH NOT DEMONSTRATED
FUEL/LOX TRANSFER SYSTEM COMPONENTS • BOOM • SURGE – GEYSER DAMPERS • VALVES • PUMPS • WEIGHT & BALANCE MANAGEMENT SYSTEM	• TELESCOPING BOOM WITH SLIDING SEALS REPLACED BY BELLOWS WITH COMBINED LOX-HYD PATHS & INTEGRAL SURGE DAMPERS IS A MAJOR DESIGN DEVELOPMENT • HIGH RESPONSE VALVES & RELATED SURGE DAMPERS WHILE NOT DEVELOPED ARE WITHIN DESIGN TECHNOLOGY • HIGH OUTPUT PUMPS LARGER THAN CURRENTLY AVAILABLE WOULD REQUIRE DEVELOPMENT • WEIGHT & BALANCE MEASUREMENT AND CONTROL SYSTEM WOULD BE SIGNIFICANT DEVELOPMENT

Table 16 Aerial Refueling Development Requirements

Flight Performance

Ascent. The aerial refuel vehicle mission profile is presented in Figure 87. After refueling is completed, the SSTO vehicle rocket engines are ignited at an altitude of 9144 m (30,000 ft) and a velocity of 183 m/sec (600 fps) to boost it to orbital injection conditions. The total velocity losses are 1282.9 m/s (4209 fps), which is about 213.3 m/s (700 fps) less than the ground launched HTO vehicle. The GLOW has been reduced from about 9.979×10^5 kg (2.2 million lb) for the ground HTO to 7.711×10^5 kg (1.7 million lb) for the aerial refuel and launch HTO.

Descent. The entry (W/SC_L) parameter for this vehicle is very close to that for the ground launched vehicle, Configuration I, and it was assumed the same entry trajectory applied to Configuration 2, see Figure 74.

Aerodynamics. It was assumed that for Configuration 2 the same aerodynamics at that for Configuration I applied. Since this vehicle concept was not selected for follow-on studies, no additional analysis of Configuration 2 was made.

Configuration 3 - Air Launch

System Configuration. The ALRS 207, Air Launch Configuration, is identical to that developed for Configuration 2 (Figure 86). The major reason for the substantially smaller size of the air launched vehicle is that its launch is initiated at 9144m (30,000 ft). A version in which the air launch vehicle is ground mounted on top of a carrier aircraft for transport to launch altitude has been investigated.

Flight Performance.

Ascent. Configuration 3 has the same boost trajectory as Configuration 2 (i.e., after this vehicle is refueled). Thus, the GLOW and injected weights are also the same. The air launch altitude was selected to provide increased injected payload and a practical size limit on the air carrier. Figure 90 shows how the payload increases with launch altitude up to about 15,240m (50,000 ft) while the size, i.e., wing area, of the carrier grows beyond practical limits.

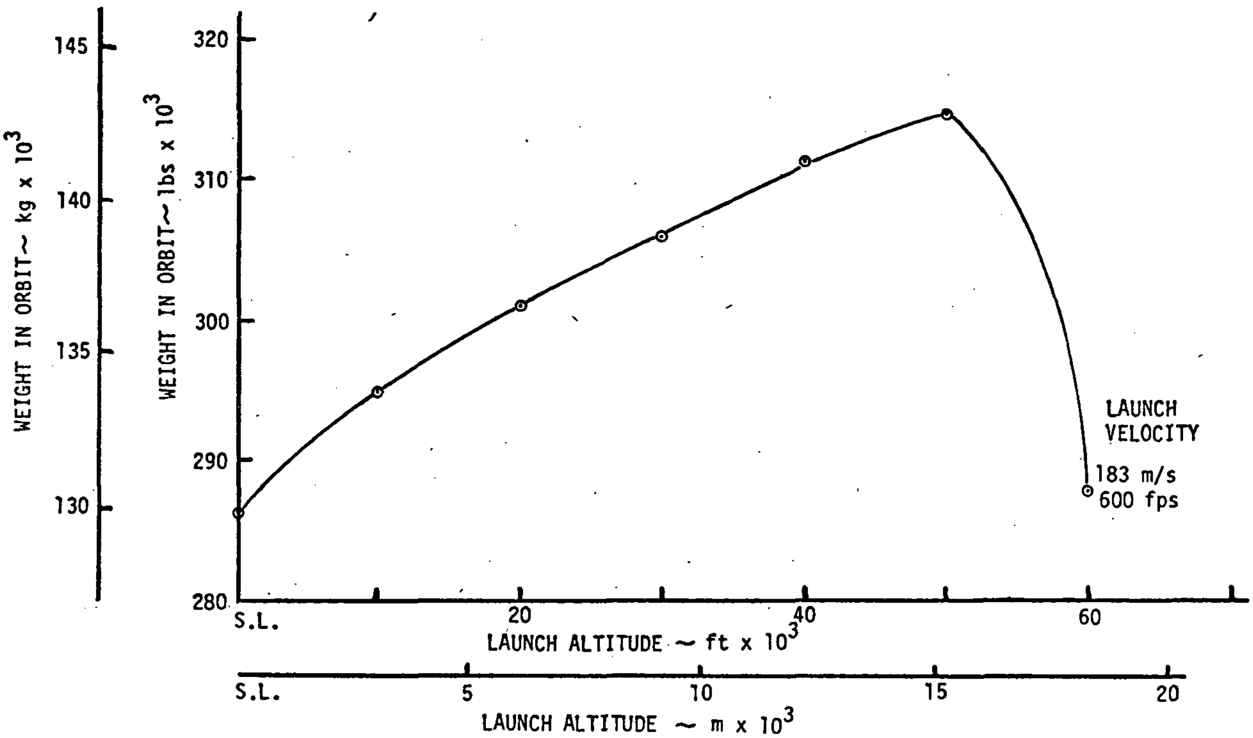


Figure 90 Payload Capability Versus Launch Altitude

Descent. Configuration 3 has the same entry trajectory characteristics as Configuration 2 and Configuration I, see Figure 74.

Aerodynamics. For Configuration 3 all aerodynamic characteristics were assumed to be very similar to those of Configuration I. This vehicle concept was not selected for follow-on studies, and no additional analysis of Configuration 3 was made.

Configuration 4 - Vertical Takeoff (VTO)

Design Configuration. The ALRS 206 is a delta winged, vertical take-off horizontal landing vehicle with the same integral wing and body tanks, aerodynamic control surfaces, reaction engines, etc., typical of this series of configurations. Its most significant characteristic is its size and dry weight relative to the ALRS 205 and 207. The predominant origin of this dry weight is the requirement for a thrust to weight ratio adequate to provide vertical liftoff ($T/W \approx 1.3$).

The ALRS has three gimballed engines and a gross lift-off weight of 2,005,366 (4,421,000 lb). It has a 4.5m dia. x 18.3m (15 ft. x 60 ft) long payload bay which is identical to the other configurations as shown in Figure 91. See Figures 92 to 97 for VTO-HTO comparisons.

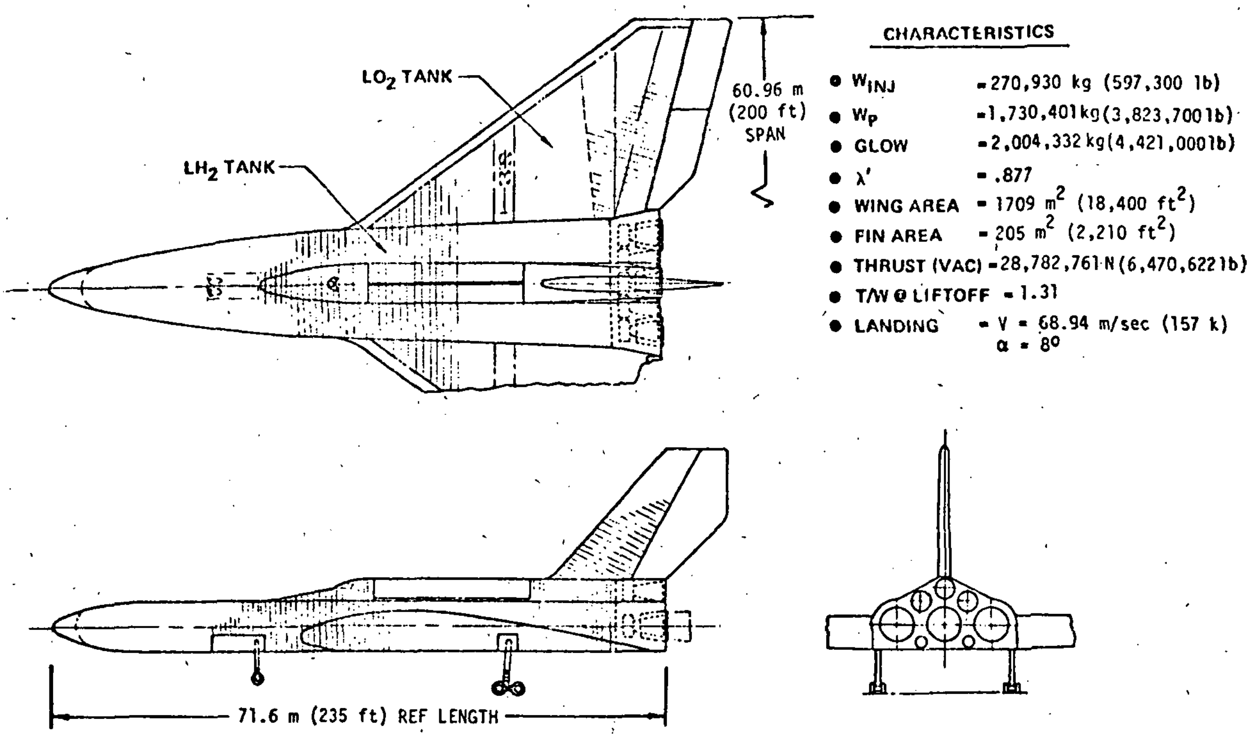


Figure 91 VTO SSTO Vehicle Configuration

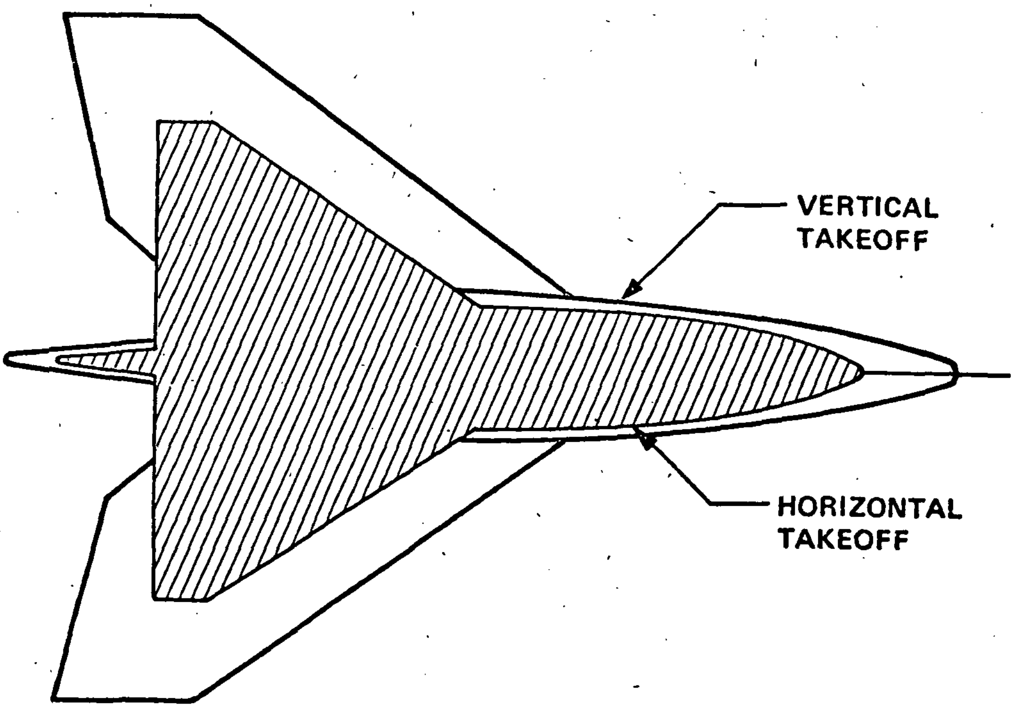


Figure 92 HTO-VTO Planform Comparison

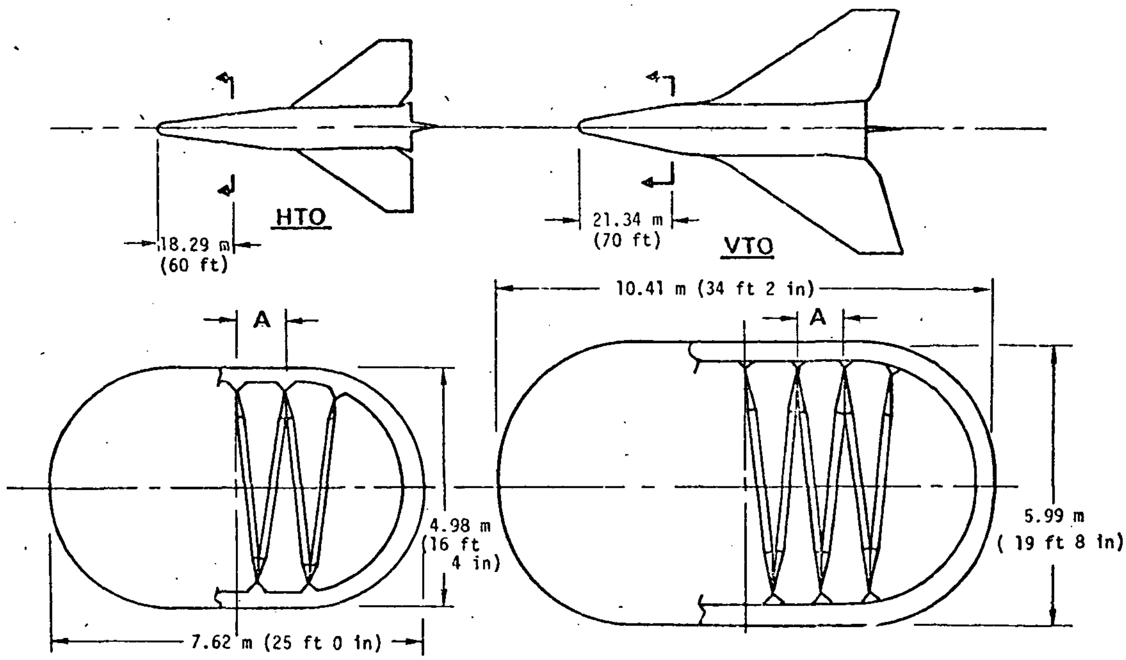


Figure 93 HTO-VTO Body Section Comparison

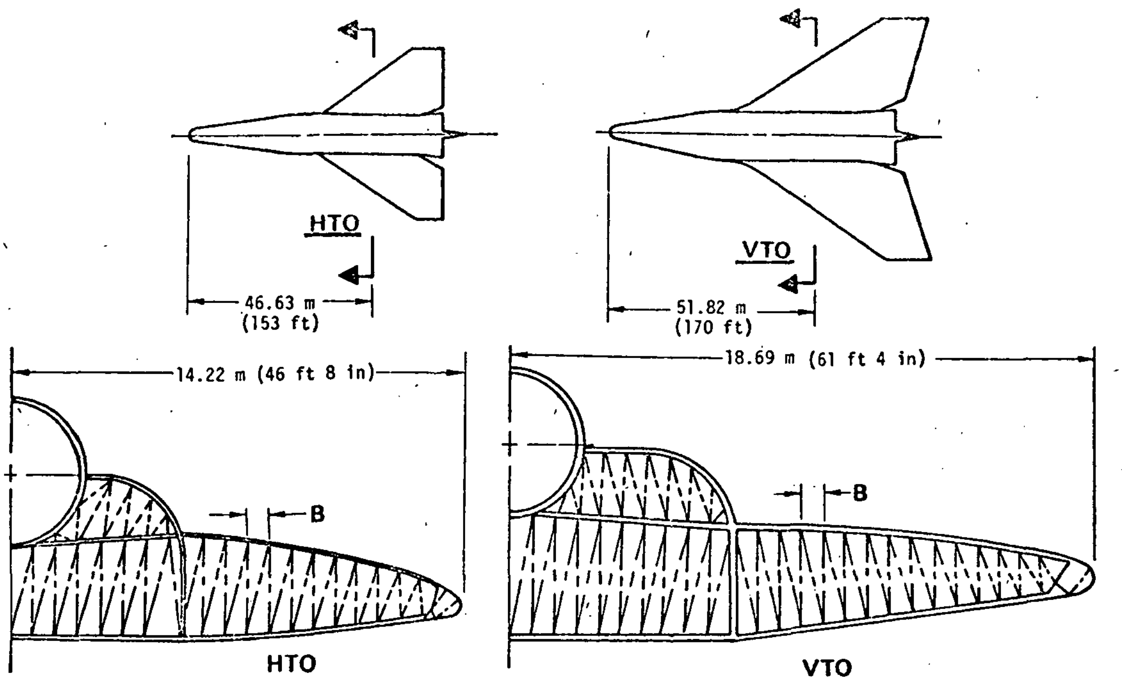


Figure 94 HTO-VTO Wing Section Comparison

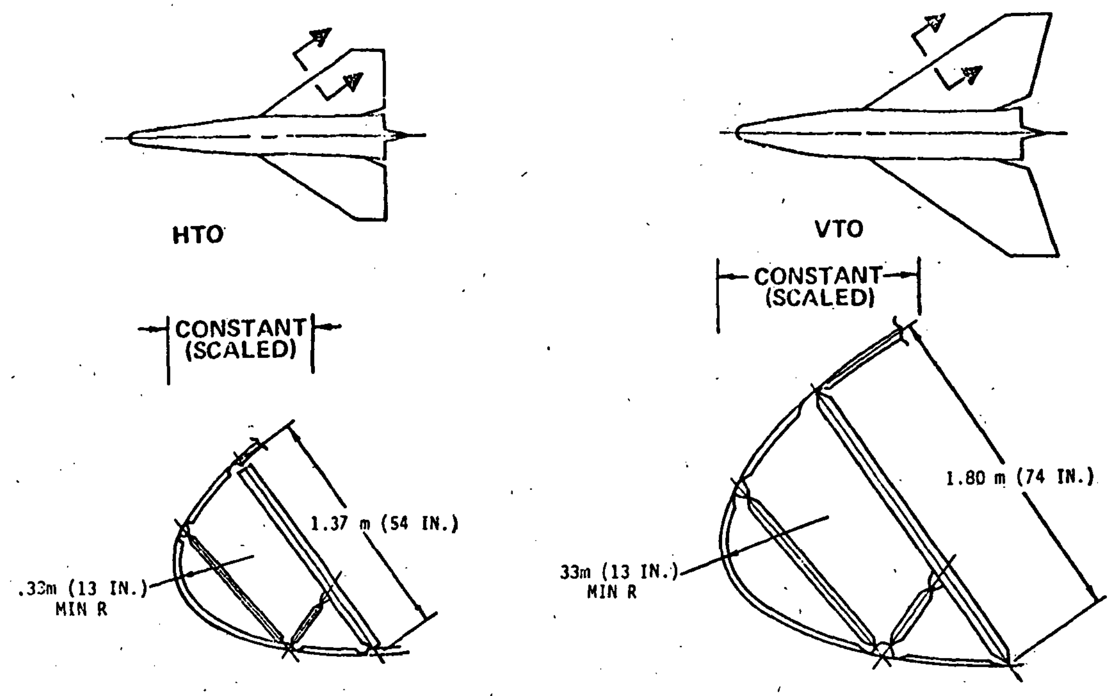


Figure 95 HTO-VTO Leading Edge Section Comparison

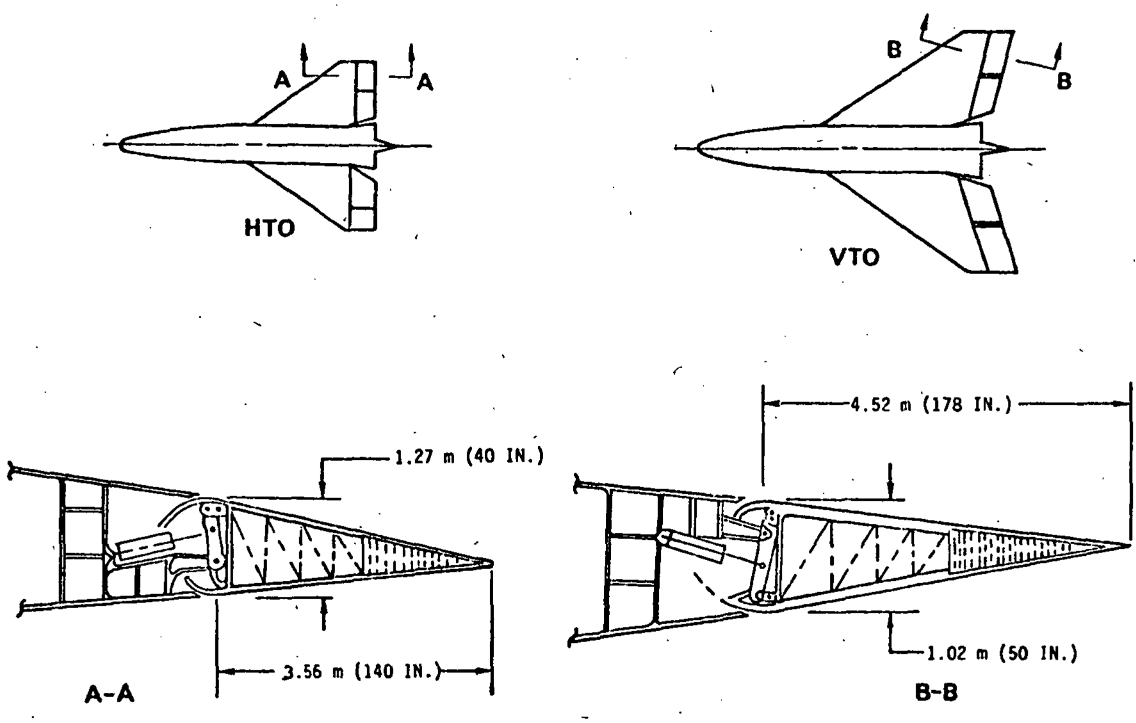


Figure 96 HTO-VTO Elevon Section Comparison

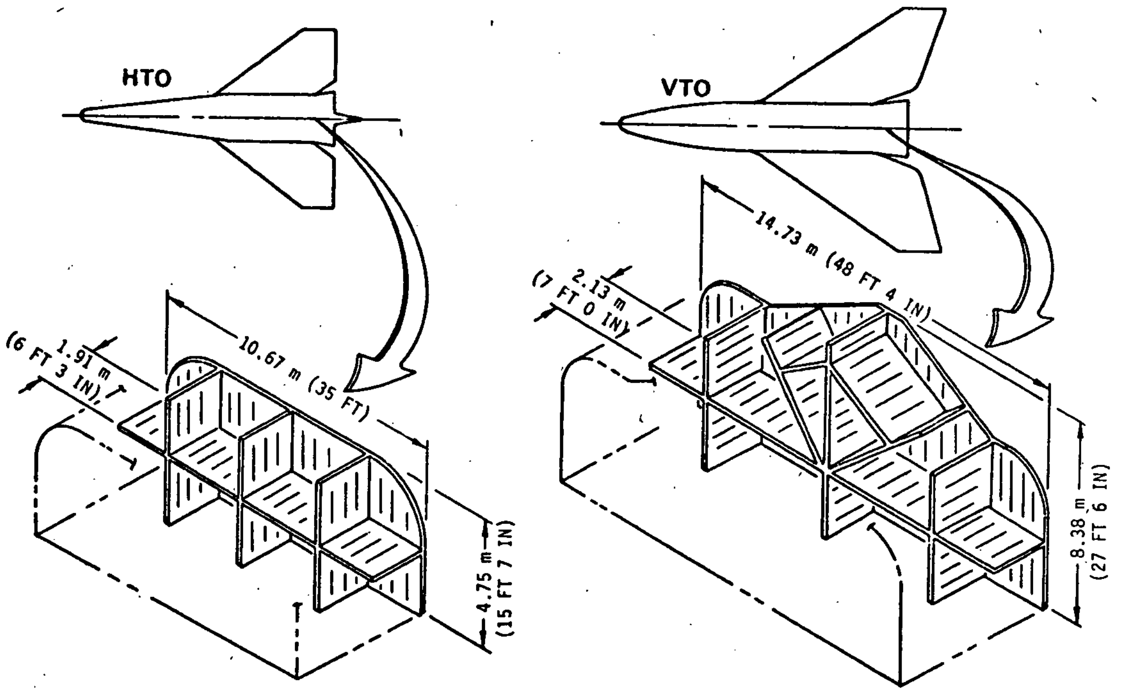


Figure 97 HTO-VTO Thrust Structure Section Comparison

Vertical Take-Off Structures Weight Rationale. Table 17 summarizes the weight rationale for the total vehicle structural system for both the HTO and VTO vehicles. The first column shows the HTO vehicle size, expressed as a unit weight and resultant structures weight for each element. A $\frac{VTO}{HTO}$ ratio was then determined, as illustrated, dependent upon element size or load. The resultant factor is used in combination with size to determine VTO weight. Several elements such as the crew compartment, payload bay bulkheads, and payload bay doors are not impacted by size of load and, consequently, show no weight delta between HTO and VTO. The final VTO weight is shown in Table 18

	HTO WT	$\frac{VTO}{HTO}$ RATIO	VTO WT
BODY			
NOSE COMP	X	CONSTANT	Y
FWD BODY	X	+5.56 kg/m^2 (1.14 lb/ft^2)	Z
MID BODY	X	+5.37 kg/m^2 (1.10 lb/ft^2)	Z
AFT BODY	X	+11.62 kg/m^2 (2.38 lb/ft^2)	Z
AFT SKIRT (ENG)	X	236/200 x 2.07	Y
EQUIP COVER	X	X1.00	Y
NOSE GEAR WELL	X	X 1.88	Y
THRUST STRUCTURE	X	X 3.4 + 2,268 kg (3.4 + 5,000 lb)	Y
AFT BULKHEAD	X	X 1,200/500	Y
HEAT SHIELD	X	X 1,200/500	Y
P/L BAY BULKHEAD	X	X 1.00	Y
SOB REINFORCE	X	+1,225 kg (2,700 lb)	Z
P/L DOORS	X	X 1.00	Y
WING			
SURFACE PANELS	X	+1.8 kg/m^2 (.37 lb/ft^2)	Z
FRAME & SPARS	X		Z
SOB RIBS	X	X 1,250/650	Y
LEADING EDGE	X	X 1,600/1,000	Y
MAIN GEAR WELL	X	X 1.88	Y
FWD & AFT BULKHEAD	X	X 3,200/2,000	Y
ELEVONS	X	X 1,950/1,010	Y
LAUNCH SUPPORT	X		Y
CREW COMPARTMENT	X	X 1.00	Y
VERTICAL TAIL	X	600/480 x 2.88	Y

$$HTO \times \frac{VTO}{HTO} = VTO - Y$$

$$HTO + \frac{VTO}{HTO} = VTO - Z$$

Table 17 HTO-VTO Weight Scaling Rationale

Propulsion

Main Engines. The VTO engines represent an updated version of SSME which is at the limit of technology. Engine characteristics are shown in Table 19. The 26 x M Pa (3800 psia) nominal chamber pressure, 28.6 x Pa (4150 psia) at EPL, is the maximum allowable with current turbo-machinery temperature limits and heat and bearing characteristics. The 4.89 MN 1.1 million pound) thrust also represents a significant scaling up in size. Full capability of the projected technology is required, however, for the optimum engines for the VTO.

Structure	148,054 kg (326,397 lb)	
Body structure		73,010 Kg (160,956 lb)
Wing structure		54,367 Kg (119,856 lb)
Tail structure		7,218 Kg (15,912 lb)
Structures margin		13,460 Kg (29,673 lb)
Subsystems	58,356 kg (128,653 lb)	
Surface controls		1,876 Kg (4,136 lb)
Landing gear		5,805 Kg (12,798 lb)
Rocket engines		34,351 Kg (75,732 lb)
Propellant feed		2,237 Kg (5,131 lb)
Pressurization		1,473 Kg (3,248 lb)
RCS system		1,358 Kg (2,993 lb)
OMS system		1,350 Kg (2,976 lb)
Avionics		1,306 Kg (2,880 lb)
Prime power		574 Kb (1,266 lb)
Elect conv & dist		1,619 Kg (3,570 lb)
Hyd conv & dist		1,853 Kg (4,085 lb)
Environmental control		1,919 Kg (4,230 lb)
Personnel provisions		362 Kg (797 lb)
Subsystems margin		2,182 Kg (4,811 lb)
Vehicle dry weight	206,407 Kg (455,050 lb)	
Personnel		263 Kg (580 lb)
Payload		31,404 Kg (69,235 lb)
Fluids	32,856 Kg (72,435 lb)	
Flight performance reserves		4,112 Kg (9,067 lb)
Reaction control propellant		1,664 Kg (3,668 lb)
Orbit maneuver propellant		9,491 Kg (20,923 lb)
Residuals/unusable		14,552 Kg (32,081 lb)
Subsystem fluids		3,307 Kg (6,696 lb)
Vehicle injected weight		270,931 Kg (597,300 lb)
Ascent propellant		1,734,401 Kg (3,823,700 lb)
Prelaunch weight		2,005,332 Kg (4,421,000 lb)

Table 18 VTO-SSTO Weight Statement

	FIXED NOZZLE	TWO POSITION NOZZLE
VACUUM THRUST (EPL) - N (lb)	4.70×10^6 (1,056,874)	4.77×10^6 (1,072,054)/ 4.89×10^6 (1,000,000) (1,100,000)
SEA LEVEL THRUST - N (lb)	4.34×10^6 (875,076)	4.24×10^6 (953,293)
$I_{SP\,VAC}$ - sec	442.8	449.2/460.9
I_{SP} SL - sec	408.5	399.4
EXPANSION RATIO	39.9	57.94/110
CHAMBER PRESSURE (NPL) - n^2m^2/psi	26.2×10^6 (3,800)	26.2×10^6 (3,800)
EXIT DIAMETER - m/(in)	2.14 (84.2)	2.57 (101.4)/3.53 (138.8)
ENGINE LENGTH - m/(in)	4.73 (186.2)	5.28 (207.8)/7.85 (309.2)
WEIGHT DRY - kg/(lb)	5,257 (11,590)	5,743 (12,662)
FLUID WEIGHT - kg/(lb)	425 (938)	425 (938)
ACTUATOR WEIGHT - kg/(lb)		450 (992)

Table 19 VTO-SSTO Main Engine Characteristics

Three of the engines are fixed nozzle, low expansion ratio engines which are only burned in the first part of ascent. The other three engines burn during the total flight and therefore have two-position nozzles. The nozzle expansion ratio split on the two-position nozzle and with the fixed nozzle engine result from an analysis to assure no direct flame impingement on the extendable portion prior to its deployment. Further, the fixed nozzle engines must be shut down before the extendable nozzle is deployed. The powerheads for both types of engines are identical.

In addition to the two-position nozzles, these engines also assume zero NPSH pumps.

Aerodynamics

Drag. Drag coefficients, C_D, for the vehicle and variations in C_D along the ascent trajectory were determined including minimum drag and total drag coefficient. The largest contributor to the drag was wing wave drag with about 57 percent of the total. Drag coefficients at angle of attack were also developed. However, for VTO vehicles, velocity losses due to drag during ascent were very much smaller than the losses for a HTO vehicle. These drag velocity losses were only 92.35 m/s (303 fps) out of a total of 1472.79 m/s (4832 fps).

Environment and Mass Properties

Loads and Dynamics.

Loads for this configuration were factored from Configuration I by the ratios of weight, length, $q\alpha$ and $q\beta$. Even though maximum q increased, the maximum $q\alpha$ decreased from .239 (10^6) Pa^o (5000 psf^o) to .153 (10^6) PA^o (3200 psf^o) because of reduced angle of attack required for the VTO. Maximum $q\beta$ increased from .122 (10^6) Pa^o (2560 psf^o) to .167 (10^6) Pa^o (3500 psf^o) because of the increased velocity through the wind shear spike.

Bending moments were factored from Configuration I by the products of the ratio of weights and lengths except for the gust condition where the factor was the product of area, $q\alpha$ and length. Shear factors were the same, except the length ratio term was not included. The same logic was used for wing loads.

Body bending loads for all conditions making up the design envelope are shown in Figure 98 . Loads for the 1.8 hypersonic entry condition are also shown, since entry loads are applied to a hot structure.

The ascent q α condition produces the maximum positive body bending loads. Maximum negative bending loads are produced by the 3 m/s (100 fps) sink rate landing condition. Body shear loads that go with the bending loads shown in the previous figure are shown in Figure 99

Wing spanwise bending loads for the conditions which dictate the design envelope are shown on Figure 100. Lesser loads for the 1.8 g entry condition are also shown, since entry loads are applied to a hot structure.

The ascent wind gust condition produces maximum upbending loads. The negative 1 g subsonic maneuver produces the maximum downbending loads.

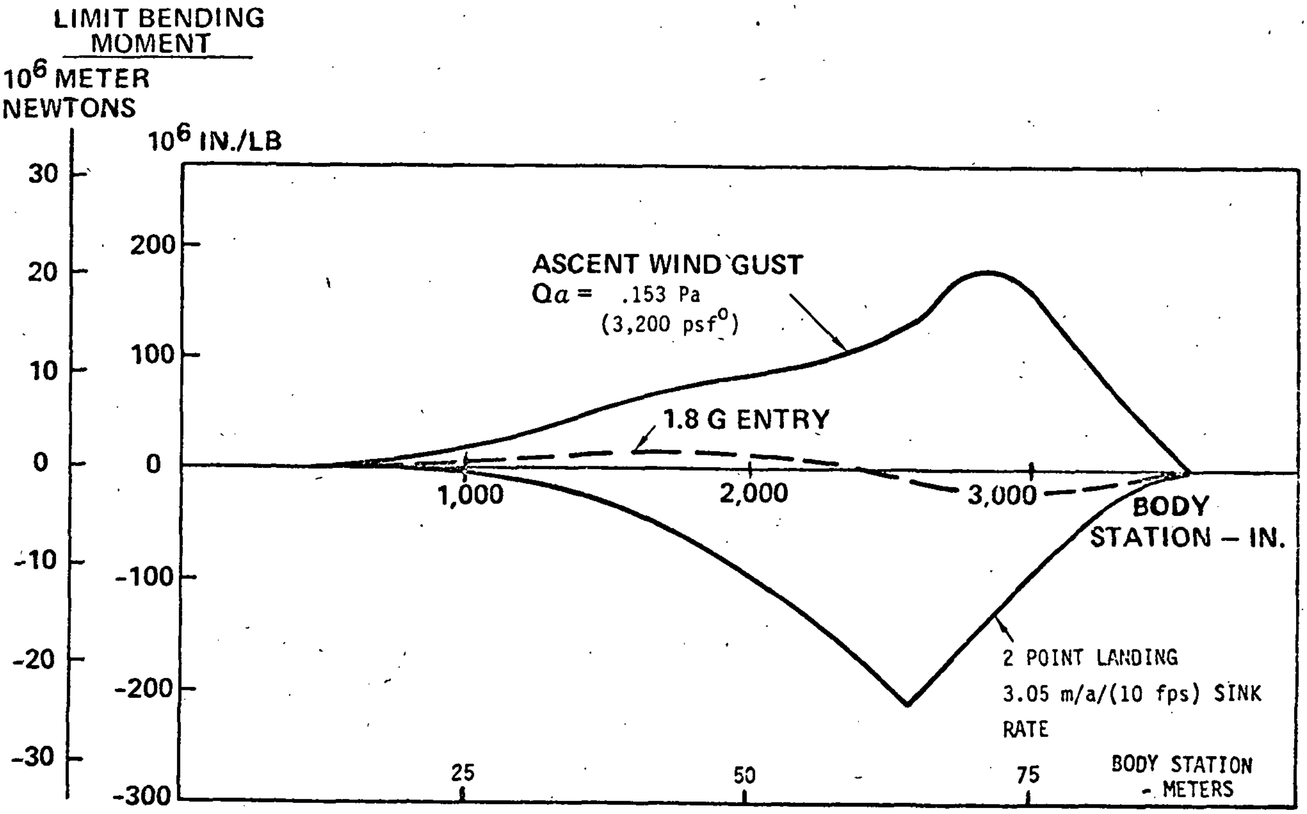


Figure 98 VTO SSTO Body Bending Loads

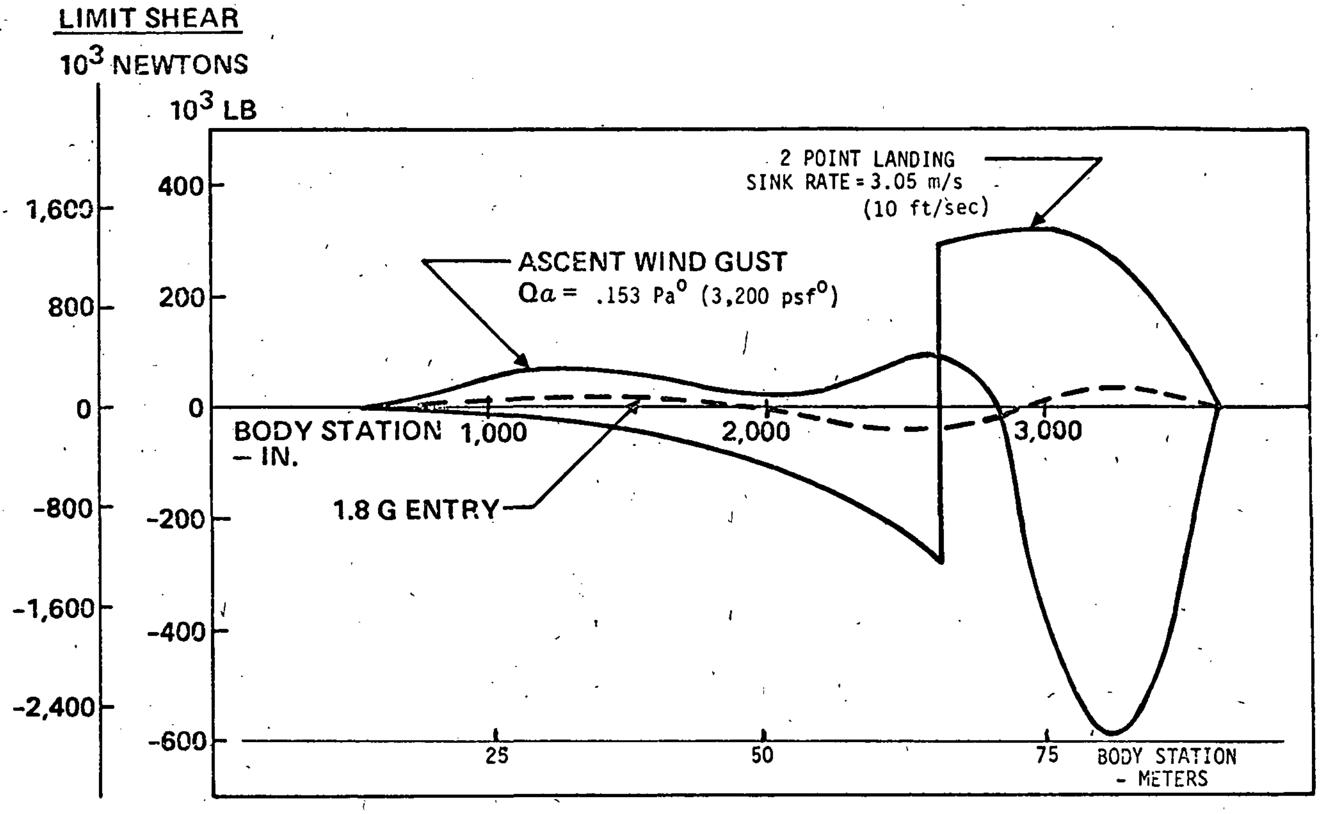


Figure 99 VTO SSTO Body Shear Loads

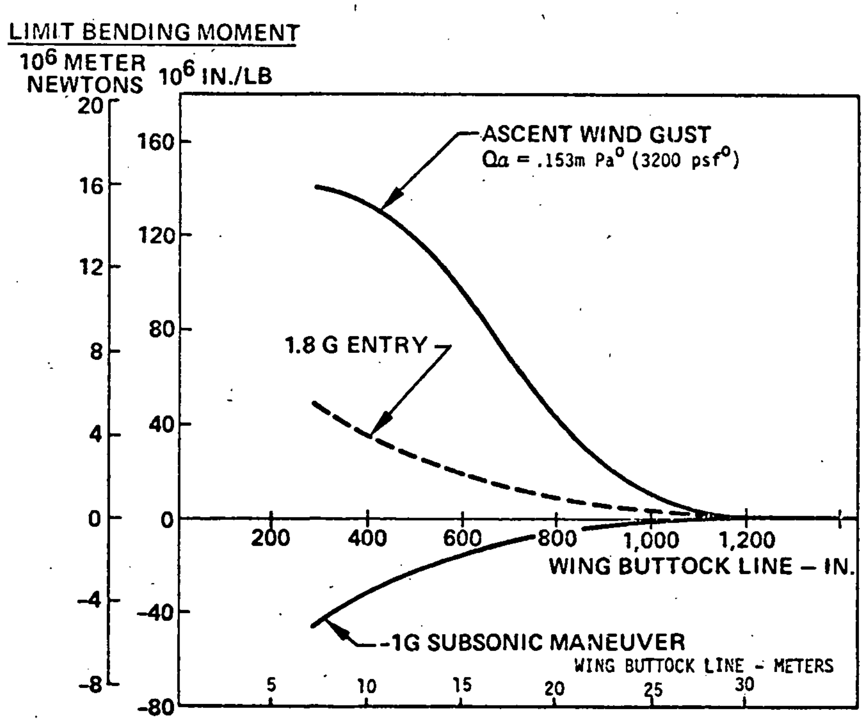


Figure 100 VTO SSTO Wing Bending Loads

Wing spanwise shear loads that go with the bending loads shown in the previous figure are shown in Figure 101.

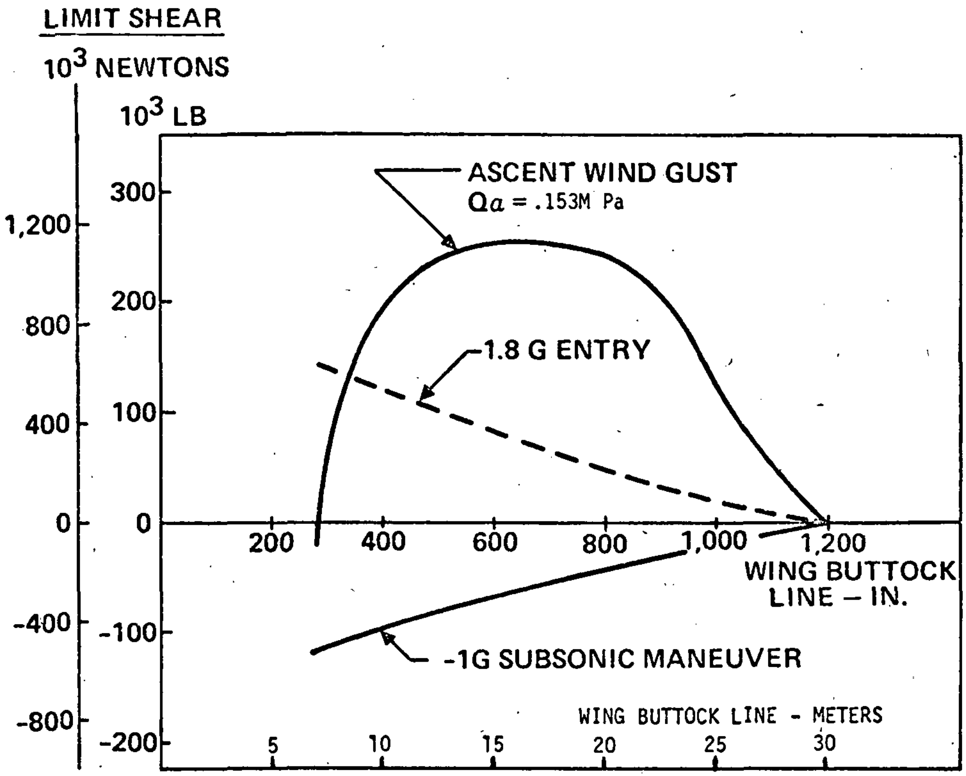


Figure 101 VTO SSTO Wing Shear Loads

Maximum root bending moments on the fin were 19.2 meter newtons for the control system where $\delta = 0^o$ and 10.6 meter newtons for the control system where the rudder was feathered ($\delta = \beta$). These moments were 2.9 times larger than Configuration I fin moments. This factor is the product of the fin area, $q\beta$ and fin length ratios.

Thermal Analysis. The results of the thermal analysis for this configuration are similar to those of Configuration 1, as shown in Figure 102. The entry peak equilibrium temperatures are slightly higher because of the somewhat higher wing loading and due to the vehicle size, resulting in larger areas subjected to turbulent flow.

The isotherms shown in Figure 102 are based on peak equilibrium radiation temperatures, not accounting for internal radiation or material heat sink effects. Both ascent and reentry critical regions are shown.

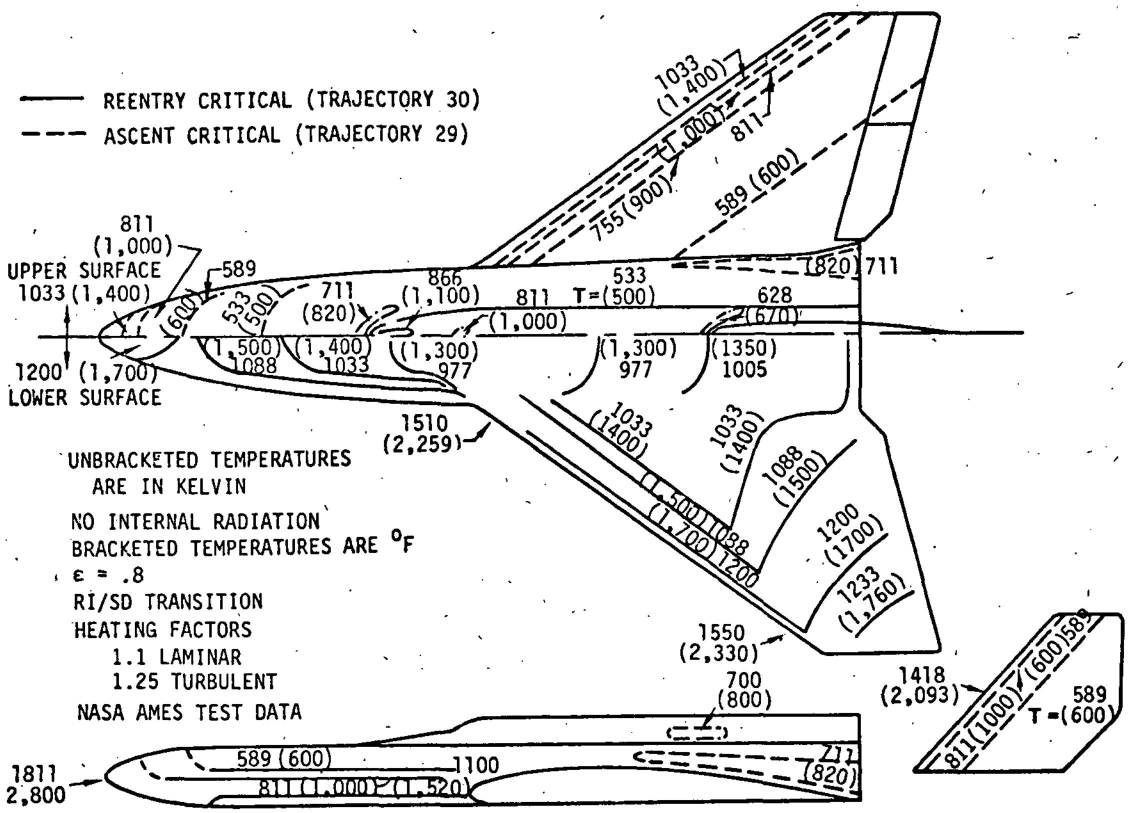


Figure 102 VTO SSTO Peak Equilibrium Temperature Distribution

The reentry trajectory is corresponding to an equilibrium glide trajectory with $\frac{W}{SC_L}$ = 3160 Pa (66 lb/ft^2). The reentry angle of attack of 50^o is reduced to 30 degrees at 91.44 km (300,000) ft. altitude.

Computed heating rates include the same uncertainty factors and are based on the same turbulent flow prediction techniques as those used on Configuration 1.

The body nose radius is .5m (20"). The leading edge radii are .4m (16") on the inboard wing, .33m (13") at the outboard wing and .29m (11.5") minimum on the vertical fin.

Interference heating was accounted for using data obtained fom hypersonic tests of a representative SSTO configuration in the NASA-Ames 3.5 foot hypersonic tunnel.

Figures 103, 104 and 105 are identical with Figures 65, 66, and 67, except that they show temperature distributions 29.4m (90 ft) aft of the nose for the present configuration, and it is assumed here that the section becomes dry at 100 seconds after launch.

TRAJECTORY 30
HTO - 24.4m (80 FT) AFT
VTO - 27.4m (90 FT) AFT

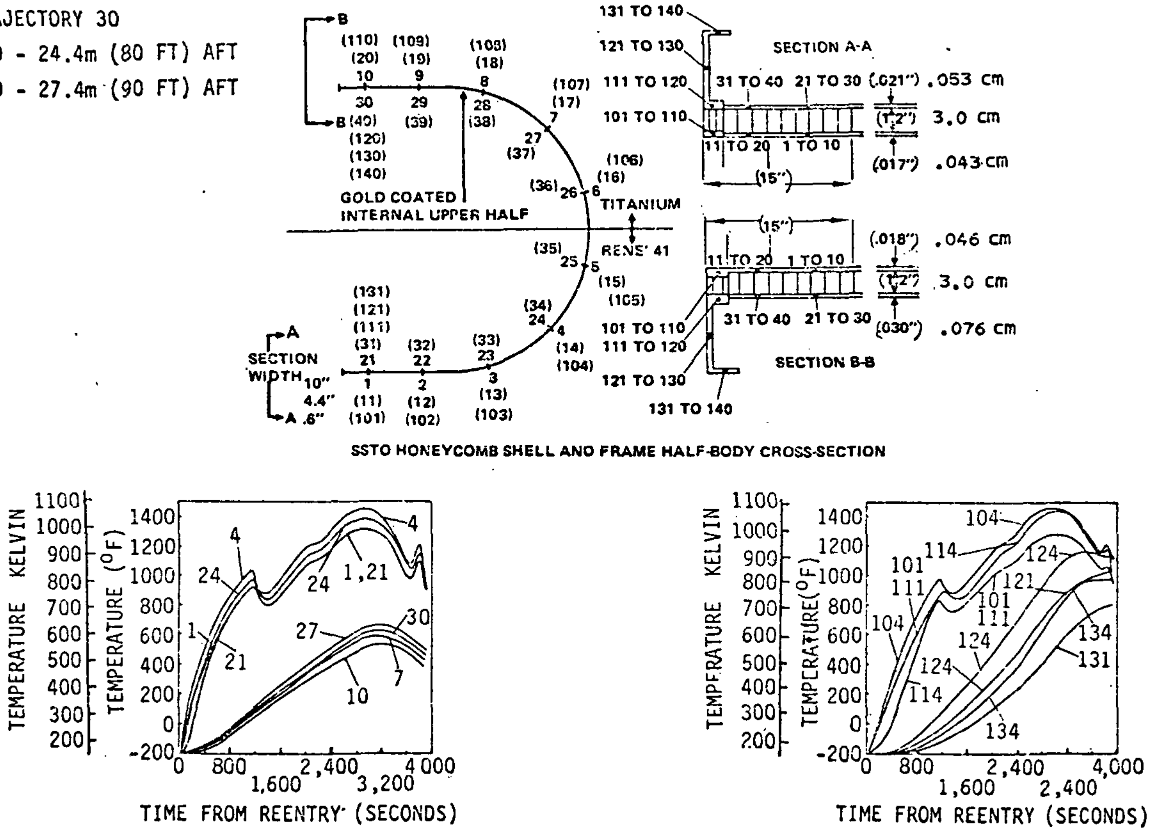


Figure 103 VTO SSTO Fwd Body Reentry Temperatures

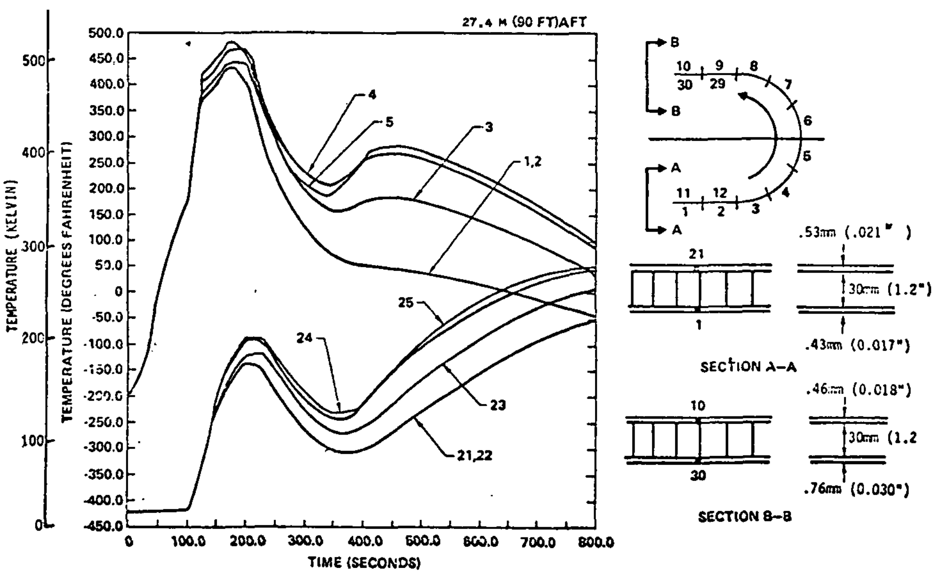


Figure 104 VTO SSTO Lower Body Ascent Temperatures

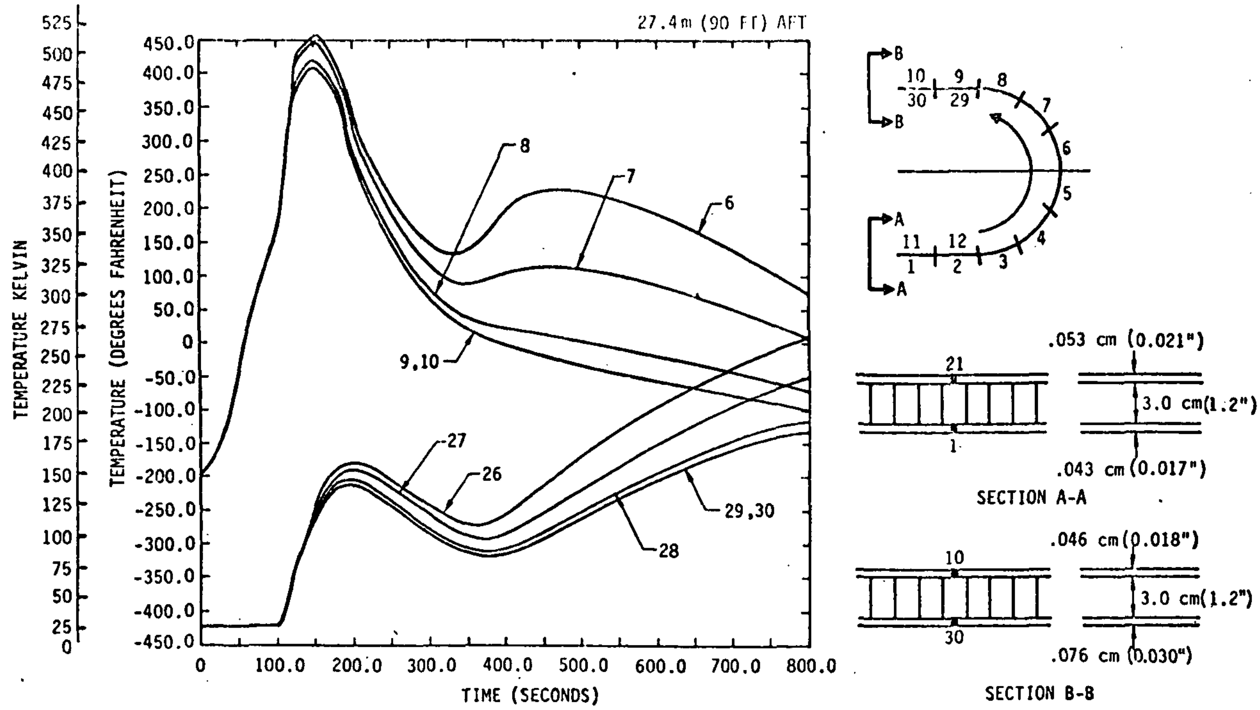


Figure 105 VTO SSTO Upper Body Ascent Temperatures

Flight Performance

Ascent. The VTO vehicle is east launched from ETR. Boeing trajectory program AS2530 was also used to determine the trajectory characteristics for the VTO vehicle. As previously discussed on HTO results, these nonoptimized trajectories should not significantly reduce injected payload from optimized trajectories when realistic $q\alpha$ constraints are accounted for in the vehicle structural weight analysis. The injected weight is 271,096 kg (597,666 lb) for the trajectory presented on Figure 106.

The flight sequence is described as follows: A vertical rise to 121.9 m/s (400 fps) followed by a rapid tilt in the flight path angle to 84.1°. The vehicle then proceeds on a gravity turn until the velocity increases to 2,438 m/s (8000 fps), where the iterative guidance mode is activated to steer the vehicle to the terminal injection points which are the same as for the HTO vehicle. Because of possible plume impingement effects, the 2-position nozzles are not extended until the speed reaches 2,438 m/sec (8000 fps), at which point the engines with the fixed nozzles are shut down. This shutdown speed was determined from trade studies to maximize injected weight. The tangential load factor is also limited to 3 g's. With a lift-off thrust loading, Tsl/GLOW, of 1.31 (see Trade Studies for optimum value) the total velocity losses were 1477 m/s (4847 fps).

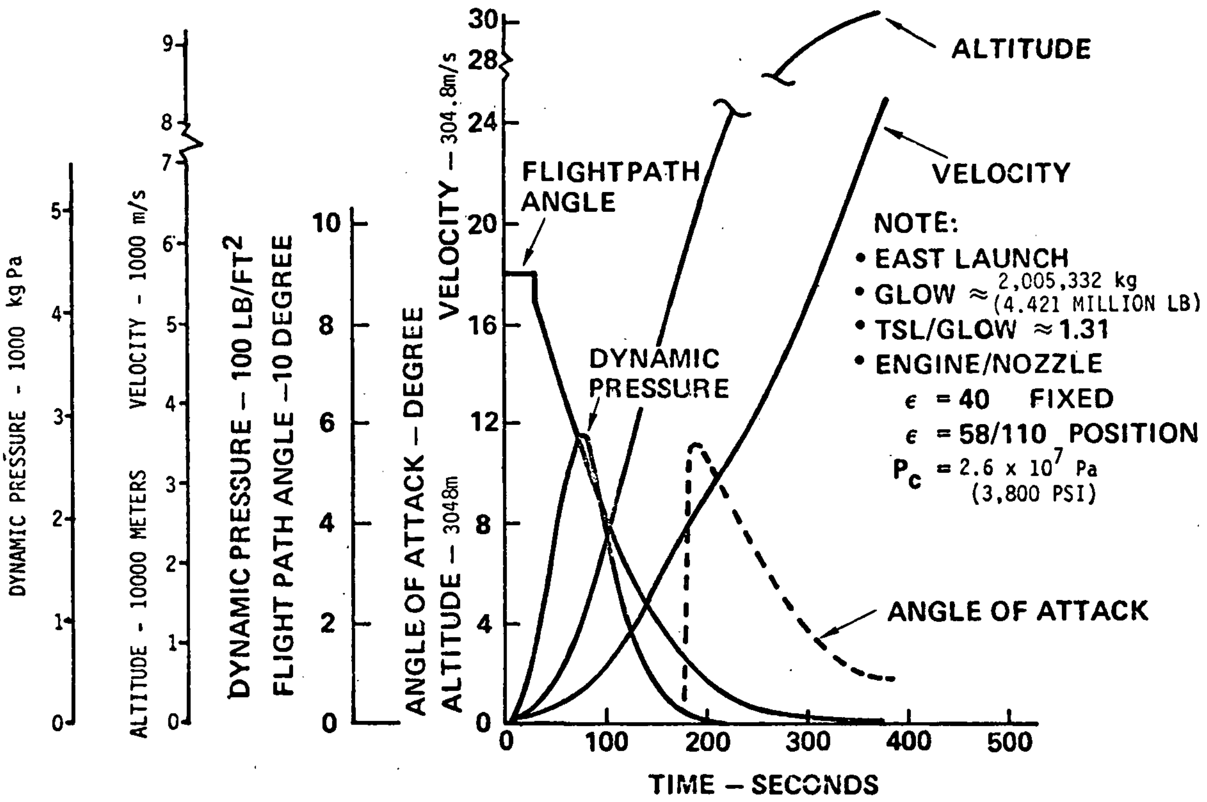


Figure 106 VTO SSTO Ascent Trajectory

Descent. This descent trajectory, see Figure 107, is very similar to that previously shown for the HTO vehicle, with the flight profile controlled by the same bank and angle of attack schedules. Entry wing loading is 1388 Pa (29 psf) and at 30^{o} angle of attack equilibrium glide, $W/(SC_L)$ is 3160 Pa (66 psf) compared to 2861 Pa (56 psf) for HTO, resulting in a slight reduction in equilibrium glide altitudes.

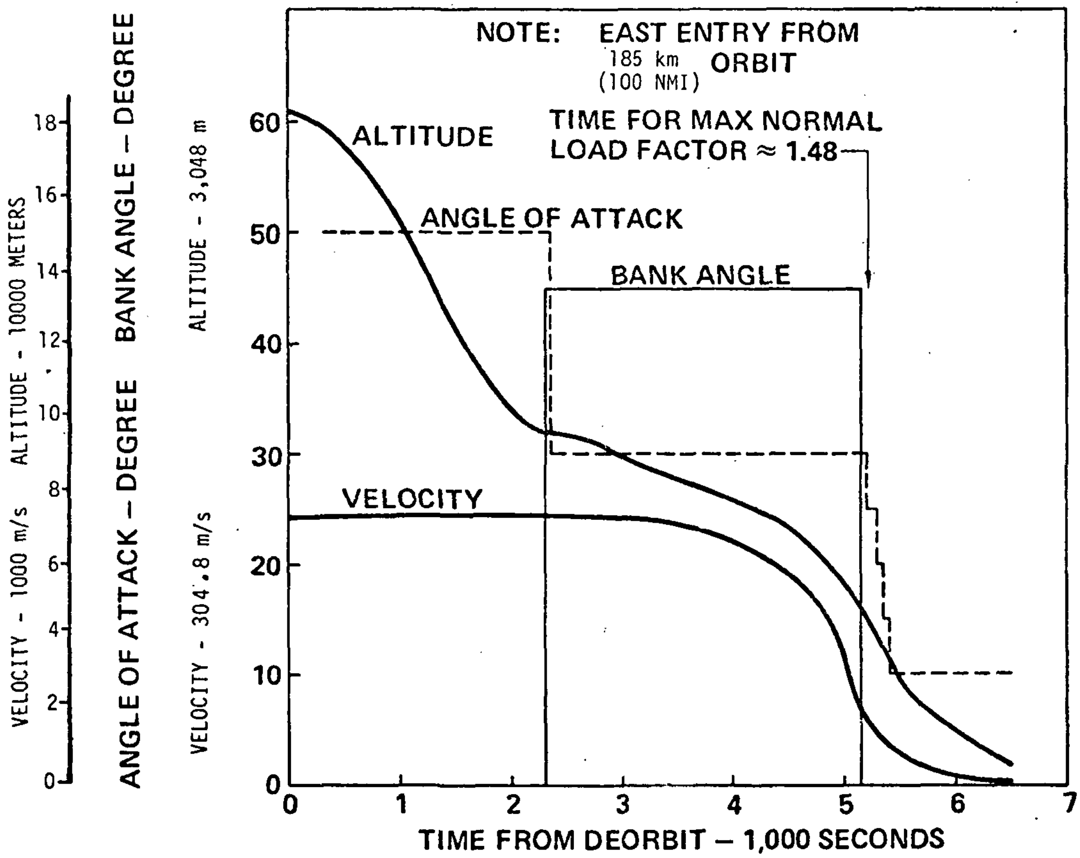


Figure 107 VTO SSTO Entry Trajectory

Lift. For the VTO vehicle, estimated normal force coefficient slope with angle of attack, C_{N_α}, and the drag due to lift and dC_D/dC_L^2 characteristics are similar to those for the HTO vehicle.

Moment. Aerodynamic moment, stability and control characteristics were determined by the same methods as those used for the HTO vehicle.

Although the VTO vehicle as initially configured was stable at hypersonic speeds, it was very unstable at subsonic speeds. Stability fixes are proposed in the following discussions.

Subsonic Aero Characteristics (See Figure 108). This configuration has a wing planform somewhat similar to the HTO configuration. It has 2° less L. E. sweep, a slightly increased aspect ratio, and an increased taper ratio. Using the same estimating methods (i.e., Datcom) as that used for the HTO the aerodynamic center, A.C. (for α from 13 to 7°) is located at approximately 0.725 of body length. This planform was determined from the results of trade studies involving both subsonic and hypersonic stability and trim considerations. Since the C.G. turned out to be about 0.033 of body length aft of the subsonic A.C., the configuration is unstable at subsonic speeds after entry. Possible fixes to this problem area are treated in follow-on discussions.

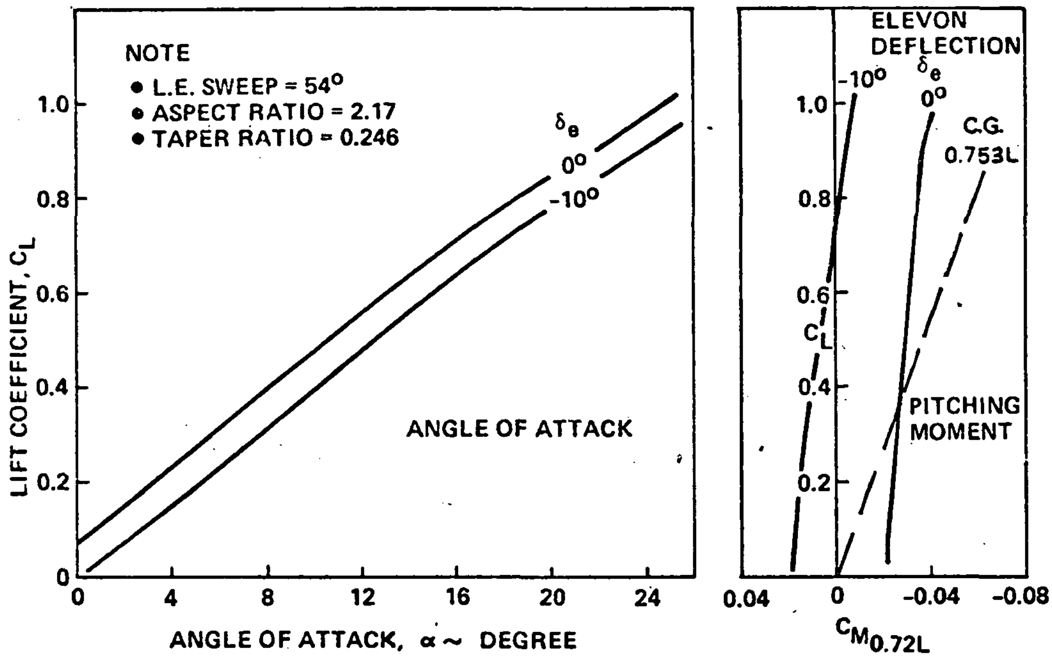


Figure 108 VTO SSTO Subsonic Aero Characteristics

<u>Hypersonic Aero Characteristics</u> (See Figure 109). For a moment center located at 0.74 of body length, the estimated variations of pitching moment with normal force coefficients at various elevon deflections are presented. At this C.G. location the configuration is stable and trimmable with elevon deflections from -20 to +5°. For angle of attack entry requirements from 50 to 30° the aft C.G. limits range from 0.755 to 0.740 and the corresponding trim limits vary from -5 to 0°, respectively.

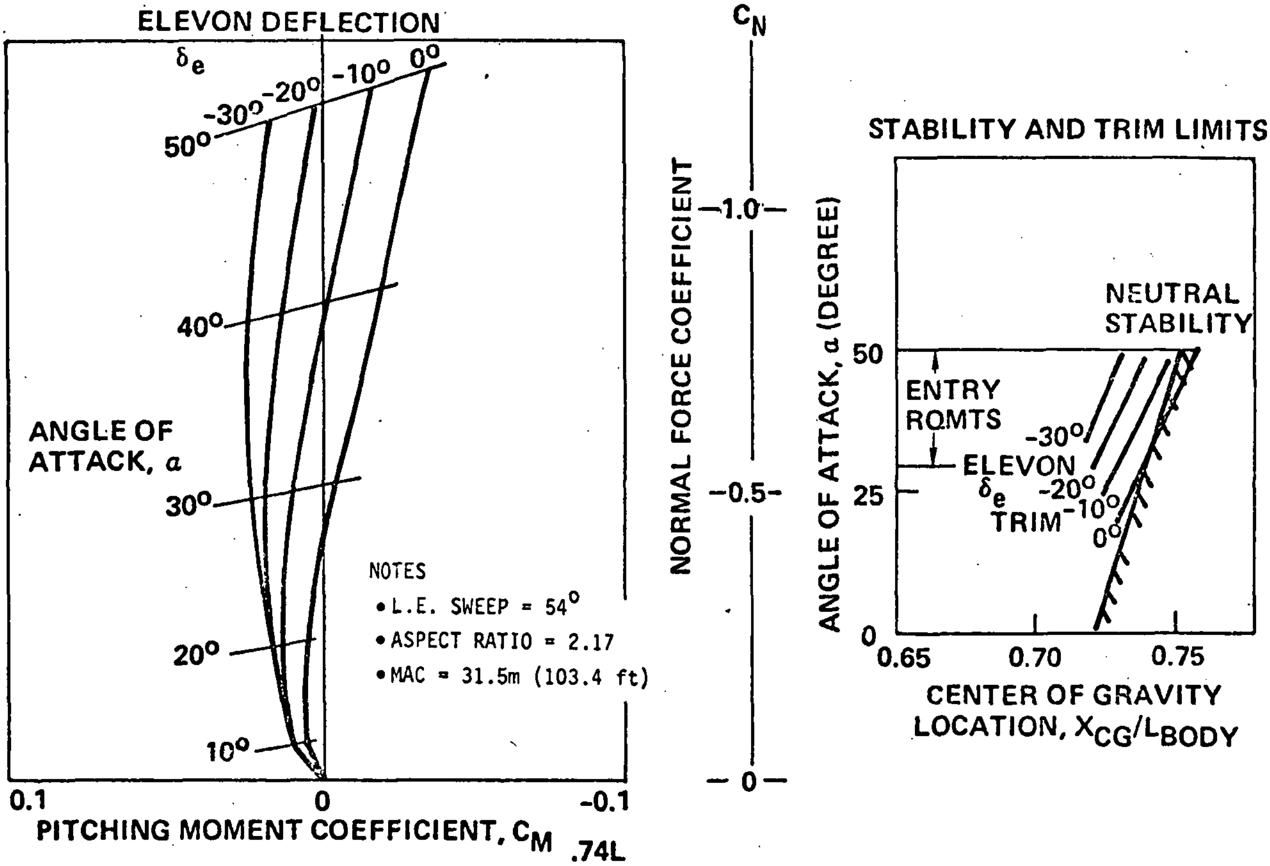


Figure 109 VTO SSTO Hypersonic Aero Characteristics

The estimated available aft C.G. for entry and landing is 0.758 resulting in a slightly unstable but trimmable vehicle at hypersonic speeds. No down elevon deflections are required. Thus, this VTO configuration has satisfactory hypersonic aero characteristics, but suffers at subsonic speed.

Configuration Alternatives. The VTO vehicle as configured is unstable at subsonic speeds with a static margin of -3.3% of body length. It is estimated that a control-configured vehicle would have the capability to accept a -1% static margin at landing. Table 20 illustrates the various alternates which could be configured to improve the stability margin to acceptable limits. Moving the complete wing aft 2.74m (108 inches) improves the stability margin by 2.3%. The additional weight is required for larger wing spars and LO_2 boost pumps. The boost pumps are required as the LO_2 bulkhead is also moved aft to maintain the wing t/c at a maximum of 12.5%. An alternative of installing 7257 kg (16,000 lb) of ballast in the nose would require a vehicle GLOW increase to about 5.4 million lb to provide a 29,483 kg (65,000 lb) payload.

Configuration	Delta inert weight kg(lbs)	Subsonic stability margin (Body length)	GLOW kg (LBS)	P/L kg (LBS)
Baseline (As drawn)	–	-3.3%	2.0×10^6 (4.421M)	31,435 (69.3K)
Wing moved aft 2.74m	* 1814.4 (4,000 LB)	-1.0%	2.0×10^6 (4.421M) 1.99×10^6 (4.400M)	29,620 (65.3K) 29,484 (65.0K)
Ballast in nose	7,257.6 (16,000 LB)	-1.0%	2.0×10^6 (4.421M) 2.45×10^6 (5.400M)	24,177 (53.3K) 29,484 (65.0K)
Crew compartment and associated equipment moved forward 22.86m	** 2,268 (5,000 LB)	-1.0%	2.0×10^6 (4.421M) 1.99×10^6 (4.500M)	29,166 (64.3K) 29,484 (65.0K)
Body flaps, folding wing tips, tee tails, all movable tail	Requires detailed analysis/study			

*Includes added structural weight and boost pumps

**Includes new P/L fairing, additional TPS, and added subsystem weight for increased distance

Table 20 VTO Stability Management Alternatives

Another alternative of separating the crew compartment from the P/L bay by moving it forward 22.86 m (900 inches) would increase the GLOW slightly due to the structural and subsystem weight increases. Other stability improvements could come from various control surface devices, but the total impact on the vehicle and the actual aerodynamic benefits would require detailed study and possible wind tunnel verification.

The weight configurations and a comparison of the three vehicle concepts are shown in Figure 110.

Cost Analysis

Cost Ground Rules and Guidelines. The following ground rules and guidelines were provided by NASA:

1. Launch rate = 114/yr baseline. This rate will be perturbed for rates on both sides of the baseline (± 30%) to determine launch rate sensitivity.
2. Program length = 15 years
   1710 flight total for baseline.

3. Flight vehicle requirement = 5 (reduced to 4 by later analysis).
4. Two operational sites (KSC and Vandenberg).
5. Costs in 1976 dollars and present value analysis (discounted at 10% yr)
6. LO_2/LH_2 propellant costs = $.35/kg ($.16/lb) for O/F mixture ratio of 6:1 (LH_2 = $2.2/kg ($1.00/lb), LO_2 = $.044/kg ($.02/lb).

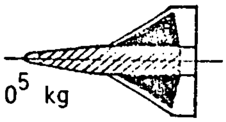
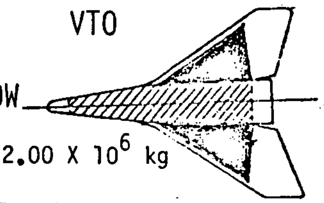
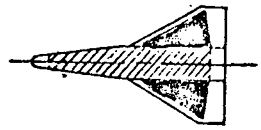
AIR LAUNCH  GLOW = 7.80×10^5 kg	VTO  GLOW = 2.00×10^6 kg	GLOW = 9.97×10^5 kg (2.2×10^6 lb) HTO 
BODY TANK (INCLUDES WHEEL WELL & INSTRUCTURE)		
19205 kg (42,339 lb)	57050 kg (125773 lb)	24948 kg (55,000 lb)
WING TANK (INCLUDES WHEEL WELLS & INSTRUCT.)		
12463 kg (27476 lb)	37399 kg (82450 lb)	16171 kg (35,650 lb)
REMAINING STRUCTURE		
26989 kg (59500 lb)	53603 kg (118174 lb)	30060 kg (66272 lb)
EQUIPMENT		
23169 kg (51078 lb)	58356 kg (128,653 lb)	28104 kg (31,958 lb)
TOTAL		
81825 kg (180,393 lb)	206407 kg (455,050 lb)	99282 kg (218,880 lb)

Figure 110 Configuration Comparison

The following ground rules and guidelines were developed by Boeing. Vehicle and facility numbers were developed from turnaround and service life requirements discussed later in the operations section.

1. The working units of the cost model are manhours; resulting costs are displayed in 1976 dollars.
2. Manhours are converted to dollars using current Boeing direct and indirect labor and material rates and factors.
3. Model is based upon a detailed breakout of all functional organization effort contributing to space and airplane programs in which Boeing has participated plus Space Shuttle.

4. Program management and SE and I are factors.
5. Facilities requirements:
   a. Assumes minimum use of existing KSC and WTR facilities;
   b. Requires a two-launch position at each launch site; and
   c. Discrete manufacturing, test and launch facilities identified.
6. Vehicle quantities:
   a. Ground test SSTO's (PTA and STA);
   b. One flight test SSTO and 1/2 unit flight spares;
   c. Four production SSTO's; and
   d. Four production sleds
7. Propulsion system costs furnished by Rocketdyne Division of Rockwell International.
8. Program management includes the contractors effort only. NASA program management is not included.
9. Spares are valued as a percentage of production hardware.
10. No fee is included.

Cost Model Comparisons. The Boeing cost model predicts the cost of aerospace programs from a set of preliminary physical or performance inputs. The model's working units are manhours. They are converted to dollars using rates and factors for any time period desired.

Table 21 compares "PCM" to three other estimating models to highlight its features. The capability to handle the cost effects of "off-the-shelf hardware" and the cost effect of using existing designs with various levels of modification are particularly noteworthy. These features reflect the drive to apply the maximum amount of off-the-shelf hardware (or mods of existing designs) to new programs to minimize the costs of space hardware.

DDT&E Methodology. This flow diagram (Figure 111) illustrates the build-up of DDT&E costs from the constituent functional categories. These functional relationships are based upon strong statistical correlations occurring in all Boeing space programs and aircraft programs.

Feature/parameter	Boeing PCM	Aero-space	Econo-metrics	KOELLE
Working units	Manhours	Dollars	Dollars	Manhours
Level of hardware manhour/cost visibility	Subsystem	Subsystem	Subsystem	System*
Level of manhour/cost element visibility				
Total DDT&E	Yes	Yes	No	Yes
First unit	Yes	Yes	Yes	Yes
System engr	Yes	Yes	No	No
System test	Yes	Yes	No	No
Software engr	Yes	No	No	No
Quality control	Yes	No	No	No
Assembly & C/O	Yes	Yes	No	No
Factory labor	Yes	No	No	No
Tooling	Yes	Yes	No	No
Design engr	Yes	No	No	No
Developmental shop	Yes	No	No	No
Management	Yes	Yes	No	No
Support equipment	Yes	Yes	No	No
Facility work load	No	No	No	Yes
Length of prog effects	No	No	No	Yes
Off-the-shelf hardware effect	Yes	Limited	No	No
Existing design modification effect	Yes	Limited	No	No

* With the exception of one subsystem area; i.e., liquid rocket engines.

Table 21 Program Cost Model Comparison

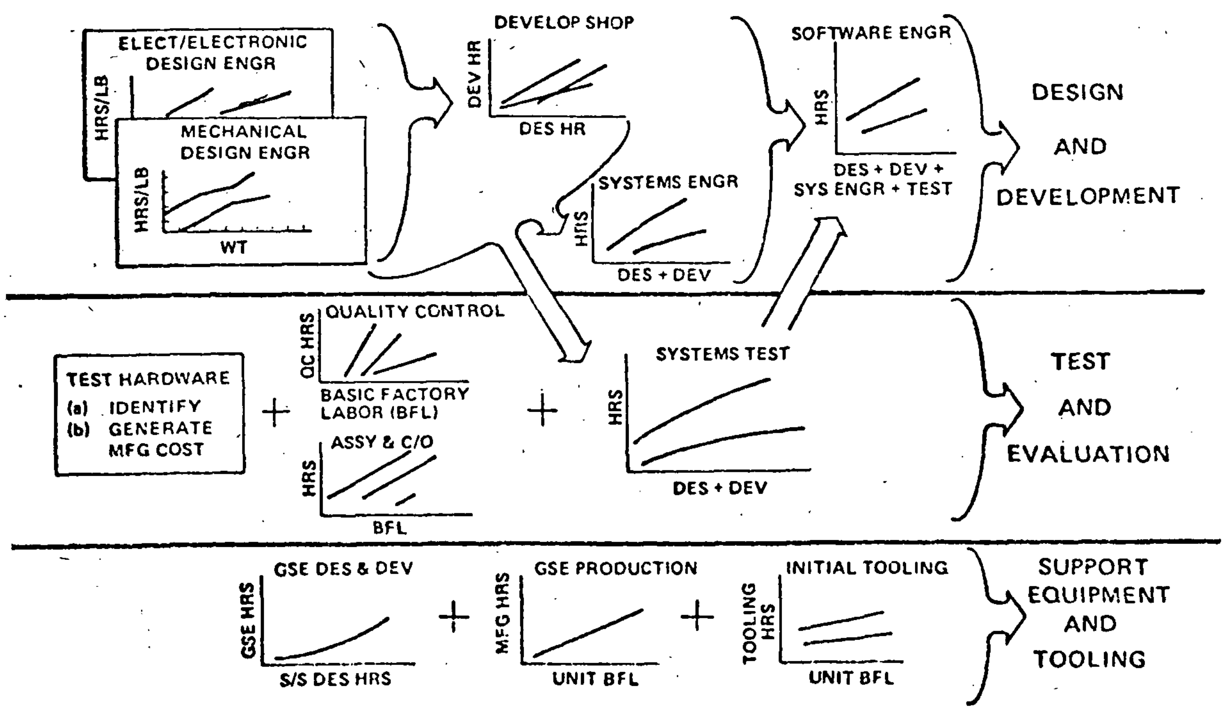
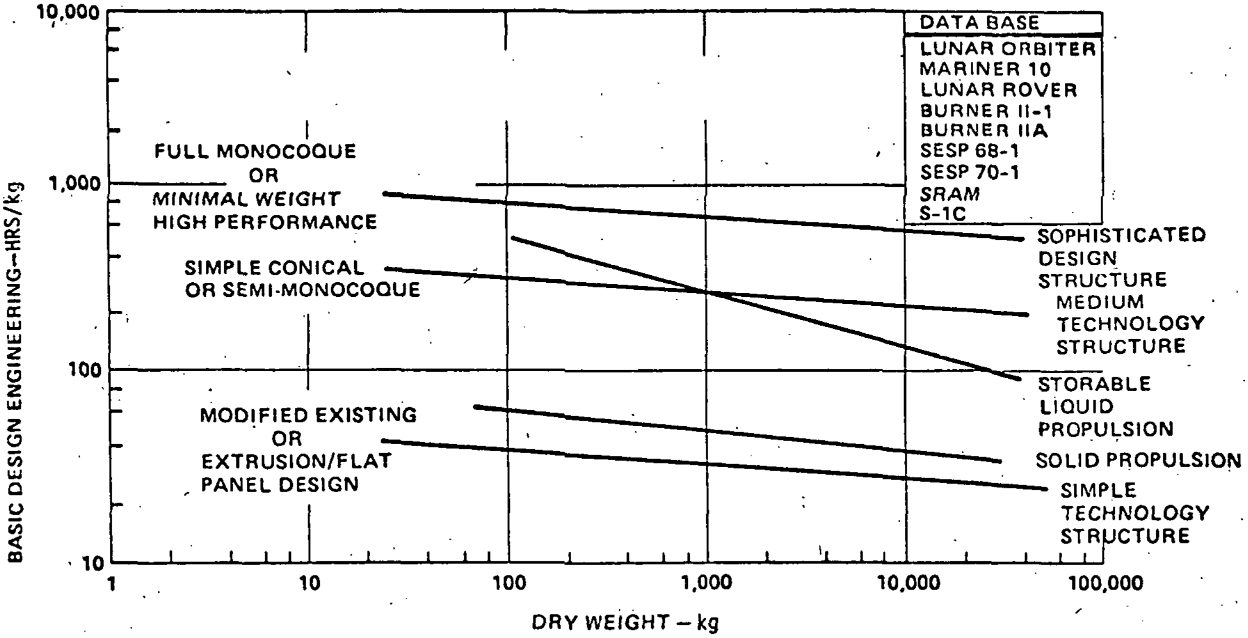


Figure 111 DDT&E Cost Model Methodology

Subsystems Design. The cost estimating relationships shown on Figure 112 differentiate between categories and composites of mechanical hardware. By using the adjustment factor, the benefit of using off-the-shelf designs, or modifications of existing designs, can be accounted for as a reduction in necessary design effort.



•ADJUSTMENT FACTOR TO ACCOUNT FOR OFF-THE-SHELF HARDWARE AND MODIFICATION OF EXISTING DESIGNS

Figure 112 Subsystem Design Cost Relationship

First Unit Cost. Unit cost build-up is basically a function of Manufacturing, Quality Control and assembly and checkout effort. Figure 113 illustrates how the inputs for DDT&E are selectively distributed to the first unit cost category by subsystem element and related with support elements.

DDT&E Background. The manufacturing technology organization provided inputs to the finance organization based on vehicle structural drawings and experience gained on the SST program with aluminum brazed titanium honeycomb. The guidelines provided to the estimators are shown in Figure 114. In addition, the producibility of the Rene'41 was compared to that of aluminum and titanium for milling, drilling, tapping, and turning operations. The average ratio was used as a complexity factor in the cost model. (See Table 22.)

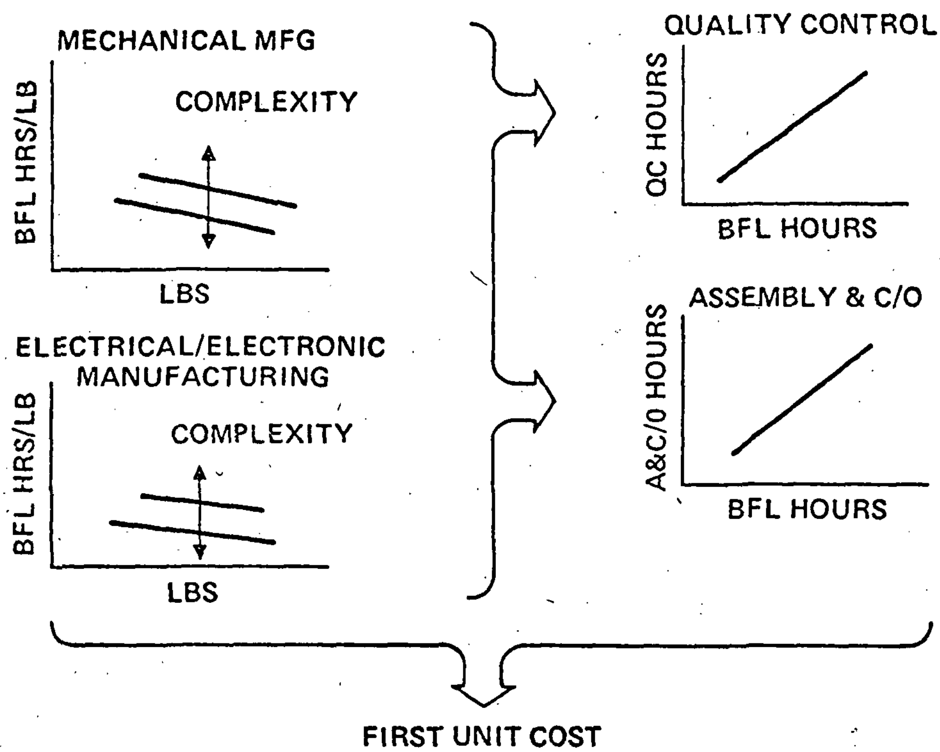


Figure 113 Unit Cost Model Methodology

Technology

Metal removal { Chem-milling, Machining, EDM, ECM }

Forming

Joining–Fusion
–Fastening
–Brazing

Metallurgy (heat treat)

Cleaning

Sealing

Guidelines

- Basis of estimate are drawings typically represented by drawing 254-20518
- Assume successful processing development
- Define facilitization requirements
- Define required development areas
- Define test structure fabrication requirements to verify producibility

Figure 114 Manufacturing Technology Inputs

Machining time ratio*

	7075-T6 aluminum	C120 av. ann. titanium	Rene' 41 (280 BHN)
Face milling	1:1	33.3:1	100:1
End milling	1:1	10:1	16.7:1
Drilling	1:1	10:1	25:1 **
Tapping	1:1	2.5:1	5:1 ***
Turning	1:1	10:1	20:1
Average		13.2:1	33.3:1

* USING CARBIDE TOOLS & HIGH SPEED STEEL (HSS) DRILLS & TAPS

** INCREASE BY FACTOR OF 4.0 FOR RENE'41 WHICH HAS BEEN HEAT TREATED & AGED (HT&A)

*** INCREASE BY FACTOR OF 2.0 FOR RENE'41 (HT&A).

Table 22 Material Complexity Factors

Hardware development costs were based on inputs from the designers (see Figure 115) as to which complexity/availability classification the subsystems were categorized. These inputs were a result of the Task I technology study and range from catalogue order to new development. This input determined the DDT&E and first unit costs explained previously and illustrated by Figure 112 .

ITEM	Kg WEIGHT	AVAILABILITY/COMPLEXITY						
		CAT. ORDER	PROG. SPARE	QUAL. HDWRE	DES. MOD.	NEW DEV.	COMP. DEV.	OTHER
SURFACE CONTROLS	1098					X		
LANDING GEAR	3676					X		
ROCKET ENGINES	13458				Thruput			
PROPELLANT FEED	1082				X			
PRESSURIZATION	798					X		
RCS	860			X				
OMS	790				X			
AVIONICS	1437			X	X			
AUXILLARY POWER	394		X					
ELECTRICAL	1781			X	X			
HYDRAULICS	1084					X		
ENVIRONMENTAL CONTROL	553		X					
THERMAL CONTROL	794			X				
PERSONNEL PROVISIONS	398	X						

Figure 115 Subsystems Development Status

Operations Cost Analysis. To achieve space transportation costs that approach under $220/kg ($100/lb) of payload is not possible if current launch and flight costs are representative of future program operating costs. Therefore, new approaches and concepts must be developed. This is not to imply that past launch practices have been wasteful or inefficient. Past and current launch practices have been the result of the R&D nature of space operations and the results of meeting a national goal within a tight time frame. Approaches for obtaining low costs for launch and flight operations are discussed below.

Turn-Around/Launch Operations. Analysis of early and current manned space programs (including Saturn IB/Apollo and Saturn V/Apollo programs) indicates that the following items are major contributors to their relatively high costs of prelaunch and launch operations.

A national goal of achieving a manned lunar landing within a tight time schedule required that research and development be accomplished concurrently with hardware production and operations. As a result, governing criteria for space vehicle (booster stages and spacecraft) design emphasized maximum vehicle performance and mission and crew safety.

To be compatible with these requirements, prelaunch operations were required to provide maximum flexibility for incorporating vehicle systems and hardware modifications during the prelaunch processing. Also, prelaunch operations were designed to provide maximum assurance of vehicle reliability, crew safety and mission success by successive testing at the component, subsystem, stage, stacked launch vehicle, spacecraft and integrated space vehicle levels. Consequently, relatively high prelaunch costs were incurred because of the lengthy processing time involving large numbers of personnel and support material required to accomplish the detailed and extensive testing, modification, prelaunch checkout and launch activities. Also, because of the research and development nature of these programs and the detailed attention given to each vehicle during prelaunch processing, only relatively low launch rates (three to four launches per year) were experienced and significant cost reduction associated with higher launch rates was not realized.

To minimize turn-around/launch operations costs of future SSTO (Space Transportation) programs, it is apparent that the space logistics vehicle should be designed for processing from recovery through the next succeeding launch with a minimum of vehicle-to-ground interfaces, ground operations and ground processing time. During the remainder of this discussion, an approach to minimizing the number of vehicle-to-ground interfaces, ground operations and processing time will be presented. Also, expected turn-around/launch operations costs for representative vehicles concepts utilizing this approach are developed in SSTO preliminary form.

The basic assumptions for this approach are:

1. The vehicle and mission will be standardized:

   a. Standardized vehicle design will allow turn-around/launch operations for each vehicle to be essentially the same as for the previous vehicle and will allow maximum learning benefits to be realized. Designing the vehicle independent of the cargo with the cargo prepackaged and self-sustaining will minimize the effect of cargo loading and unloading operations on ground operations. Prelaunch payload integration procedures similar to commercial air cargo carriers will be developed and employed.

   b. Each flight within the program requirements envelopes will be repetitive in type. The vehicle will serve only as the carrier of the cargo and will deliver the cargo to or recover the cargo from some destination in earth orbit.

2. The vehicle will be designed for minimum checkout requirements at the turn-around/launch facility:

   a. Design for maximum on-board autonomy and maximum use of an on-board checkout computer system for preflight and postflight operations will be realized. The computer also will be capable of inflight system status checks.

   b. The vehicle will be designed for easy access to on-board systems and components for preflight and postflight checkout activities. Components will be modularized so that items requiring repair or refurbishment can be replaced with a minimum of repair accomplished on board the vehicle.

   c. The vehicle will be designed so that airplane techniques of turn-around operations can be applied. The vehicle and facility will be designed to be mutually compatible and with a minimum number of interfaces and cost generating functions (operations) involved.

3. Vehicle modifications will be limited:

   a. Modifications, if they become necessary, will be limited to planned block type changes so that mixed vehicle and mission configurations will not exist simultaneously to compound turn-around/launch operations from one launch to the next.

Present checkout methods for space vehicles utilize extensive support equipment to determine vehicle condition, with access through numerous umbilical connections and telemetry links. The equipment being used varies in type and configuration between the various test locations making test data correlation difficult. With the advances being made in electronics functional density, size, and reduced power consumption, it is feasible to perform this new scope of testing with a large share of the equipment located on-board the vehicle. Also, the test equipment would be available during the mission to perform in-flight checkout. With the checkout equipment on-board the integral vehicle, checkout system/vehicle system interconnects are permanently established at the factory, minimizing the chance of human error at checkout and launch sites. Confidence levels and test depths will be increased over present methods. Requirements for ground checkout complexes will be reduced and systems will be less costly.

Vehicle Facility Requirements. Launch processing requirements for a launch rate of 114 flights per year for a vehicle were forecast. For this launch rate, it was assumed that launches could be made on any day of the year. This results in a launch on an average of every 3.20 days.

In order to determine the system vehicle and facility requirements, a launch operations processing schedule was prepared for each vehicle. The schedule covers the operations from vehicle approach and landing after the mission through launch of the next vehicle and launch facility refurbishment. The schedule was developed by reviewing operations analysis of the Space Shuttle and commercial aircraft. A typical flow for the HTO/SSTO vehicle concept is shown in Figure 116.

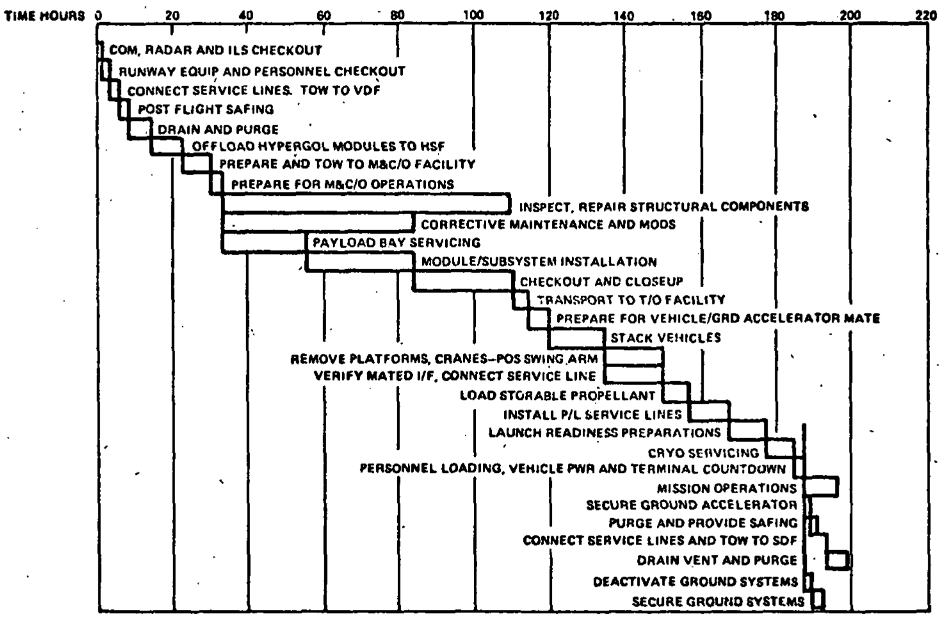


Figure 116 HTO SSTO Operational Flow

The SSTO vehicle and ground accelerator requirements are either dictated by the turnaround or service life requirement for each launch rate.

The number of vehicles required based on turnaround can be determined from the following:

$$n = \frac{\text{Launch Rate/Yr x Turnaround (days)}}{\text{Days/Yr}}$$

The number of vehicles required based on service life can be determined from the following:

$$n = \frac{\text{Launch Rate/Yr x Program Life (yrs)}}{\text{No. of Reuses}}$$

SSTO Vehicle Requirement.

Based on turnaround:

$$\frac{80.66 \times 8.25}{365} = 1.823 \qquad \frac{114 \times 8.25}{365} = 2.58 \qquad \frac{147.3 \times 8.25}{365} = 3.33$$

Based on service life:

$$\frac{80.66 \times 15}{500} = 2.42 \qquad \frac{114 \times 15}{500} = 3.42 \qquad \frac{147.3 \times 15}{500} = 4.419$$

Ground Accelerator.

Based on turnaround:

$$\frac{80.66 \times 4 \text{ days}}{365} = .884 \qquad \frac{114 \times 4 \text{ days}}{365} = 1.25 \qquad \frac{147.3 \times 4 \text{ days}}{365} = 1.6$$

Based on service life:

$$\frac{80.66 \times 15}{500} = 2.42 \qquad \frac{114 \times 15}{500} = 3.42 \qquad \frac{147.3 \times 15}{500} = 4.4$$

Facilities.

It was concluded that the most conservative approach was to use the results of the turnaround analysis for equipment quantities. Service life was not a firm requirement due to the many unknowns at this time. It was decided to limit the analysis to the 114/yr flight operations baseline with the concurrence of NASA/LaRc.

Facilities requirements are determined by establishing the use and refurbishment tent poles within the overall vehicle turnaround schedule. Once the tent poles are established, the facilities need/costs are derived by determining how many of each facility is required to support the vehicle launch rate.

Runway.

$$\frac{80.66 \times 4}{365} = <1 \qquad \frac{114 \times 4}{365} = 1.25 \qquad \frac{147.3 \times 4 \text{ days}}{365} = 1.61$$

Maintenance and Operations.

$$\frac{80.66 \times 3.21}{365} = <1 \qquad \frac{114 \times 3.21}{365} = 1.00 \qquad \frac{147.3 \times 3.21}{365} = 1.29$$

System Desafing Facility.

$$= <1 \qquad = <1 \qquad = .454$$

So long as the launch frequency does not require a launch more often than one every three days, a launch rate of 114 launches/year may be achieved with four active flight vehicles and require two launch positions. If runway refurbishment time can be reduced to as low as 1/2 day or the launch facility constructed to require no refurbishment for, say, 100 launches, this launch rate can be achieved from a single launch position.

Ground Operations Costs. The direct hands-on-vehicle contractor costs were estimated based on a review of the space shuttle operations and ratioing the manhour estimates to horizontal take-off launch. Table 23 shows an example of this ratioing analysis. The first column is an existing estimate of ground operations manhours on the Space Shuttle orbiter, SRM and external tank. This column is based on very low launch rates in comparison with those projected as the SSTO baseline (114 launches per year). As a result, the actual manhour utilization/manhours available in a full shift period are very low. The estimate of efficiency as a function of launch rate is shown in Figure 117. The second column in Table 23 reflects the differences in efficiency between launch rates of a shuttle estimate and the SSTO vehicle. The third column is the estimated changes due to differences in configuration between the vehicles. This estimate is used to cost the vehicle contractor portion of the ground operations cost. Not included above are the propulsion engine normal servicing costs and the spares labor costs. The development of these costs is found in the next two sections.

Operation	Orbiter base	Increased utilization	HTO base	Reason for delta
Runway landing	32	20	20	No reduction
Safing/deservicint	490	304	279	-30 reduction in hypergols
				+65 addition of LH2/LO2 tank
				-60 efficiency
Maintenance and checkout	3,775	2,340	1,790	-360 metallic heat shield
				-125 reduction in payload support
				-375 efficiency
				+310 addition of sled
Launch pad	6,738	4,178	1,783	-890 no SRB stacking
				-620 no vertical installation
				-150 reduction in payload support
				-280 no E/T mating
				-455 efficiency
Hypergol service facility	726	450	200	-100 no hypergol OMS
				-100 no ΔV-kits
				-50 efficiency
E/T C/O facility	916	568	268	-125 no E/T tank insulation
				-125 no mating or I/F checkout
				-50 efficiency
E/T tank demate	214	–	–	No E/T
SRB disassembly	2,781	–	–	No SRB
SRB reliability and subassy	2,486	–	–	No SRB
manhours	18,158	7,860	4,340	

Table 23 Ground Operations Manpower Estimate

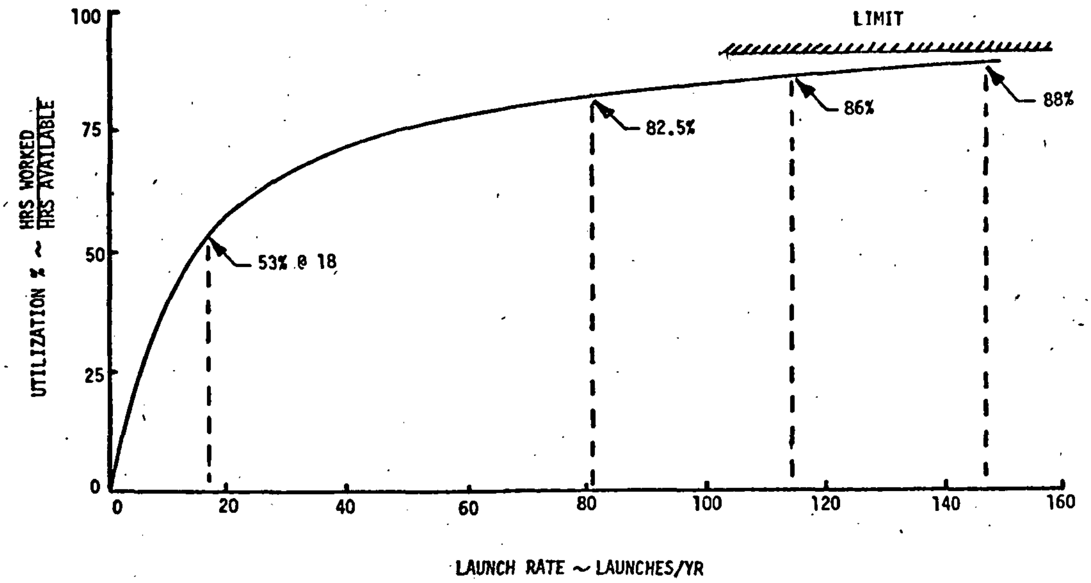


Figure 117 Ground Operations Utilization Estimate

Main Engine Support Cost Analysis. As a part of Rocketdyne's subcontract (G.O. 07822) to the Boeing "Advanced Earth Orbital Transportation Systems" study for NASA/Langley (NAS1-13944), Rocketdyne provided cost data for advanced propulsion systems. Development cost, unit procurement cost, engine overhaul cost amortized as a cost per flight over the number of flights before overhaul, the operational support cost per flight and spare (replacement) parts costs were the cost items provided.

Three vehicle concepts/propulsion systems were defined for cost estimating:

1. A 3.3 MN (740K lb) SSME modified to incorporate a reusable two-position nozzle for use in the "in-flight refueled" vehicle concept.
2. A 3.02 MN (680K lb) uprated version of the SSME (at Pc = 24.13 MPa (3500 psia) with a reusable two-position nozzle ($\epsilon = 80/150$) for use in the "horizontal take-off" vehicle concept.
3. A 4.89 MN (1,100K lb) growth version of the SSME (at Pc = 26.2 MPa (3800 psia) with a reusable two-position nozzle ($\epsilon = 58/110$) for use in the "vertical take-off" vehicle concept.

A fleet of four vehicles is required to perform a baseline 1710 missions program over a 15-year period. Concepts (1) and (2) utilize three engines in each vehicle and concept (3) has six engines. In addition, concept (2) uses five boiler plate SSME engines on a sled that launches the vehicle.

The baseline program would require an average of 342 flights per engine. This quantity is based on a set of spare engines (3) being included in the overall usage analysis. This set of spare engines is required to maintain the short turnaround cycle during periods of major overhaul to the engine. Based on this requirement, it has been assumed that the propulsion system can perform 70 flights before overhaul. This assumption is based on the Space Shuttle program providing the experimental data needed to define the means of increasing engine life from 55 flights to 70 flights. An overhaul cost equivalent to 30 percent of the initial cost of a new engine has been used for the engines of the SSTO. This compares to 28 percent for the SSME. A typical overhaul cost/flight analysis for the baseline HTO 3.02 MN (680K lb) vacuum thrust engine is shown below:

Engine

1710 flights x 3 engines/vehicle = 5130 engine flights

5130 engine flights/15 engines = 342 flights/engine

$$\frac{342 \text{ flts/eng}}{70 \text{ flts/oh}} = 4.88 \text{ cycles } -1.0 \text{ (new engine)} = 3.88 \text{ or } 4 \text{ required overhauls}$$

$$\frac{4.0 \text{ oh} \times 30\% \times 13.8\text{M} \times 3 \text{ eng}}{342 \text{ flights}} = 145.2\text{K/flight}$$

Nozzle

$$\frac{4.0 \text{ oh} \times .30\% \times 1.2\text{M} \times 3}{342} = 12.7\text{K/flt}$$

Total Overhaul Cost = 157.9K/flt

Replacement parts requirements for performing unscheduled corrective maintenance between overhauls are based on the SSME program requirements of approximately 2.5 equivalent engines in hardware to support 15 engines flying on five vehicles. The SSTO program has 12 engines on four vehicles but carries three spare engines because of turnaround operations. The three spare engines are included in the replacement/rotation cost per flight. The 2.5 engine ratio is projected to a 710 flight program over that of the Space Shuttle which is 445. The total replacement parts cost is subdivided: 60 percent labor and 40 percent hardware. Labor costs are for repairing the removed component or subassembly.

Operational costs for the study are based on data for the SSME program. However, it was assumed that the number of operations and time required to check the engines can be reduced in half, based on experience gained in the Space Shuttle program.

Dedicated test stands and crew to resolve flight anomalies would not be anticipated for the SSTO except for the 4.89 MN (1,100K lb) engine which would not have a prior flight program to resolve anomalies. The 3.02 MN (680K lb) engine is assumed to be so close in size and thrust to the SSME that flight anomalies would not be expected since the initial design of the engine would incorporate any fixes to the SSME design resulting from flight anomaly resolution. For the 4.89 MN (1,100K lb) engine, the cost of maintaining a test stand and crew to resolve flight anomalies has been based on comparable SSME data.

The unit procurement cost for these engines has been based on an update of a trend curve of engine cost vs. thrust established by the Aerospace Corporation. The update was an adjustment to reflect the SSME cost in 1974 dollars.

The development costs are based on the SSME development cost. The 3.02 MN (680K lb) engine should benefit significantly from the SSME program and reduced engineering, development hardware and development testing are assumed. The 4.89 MN (1,100K lb) engine is a new engine since its pressure and size are significantly different from the SSME.

Costs for the ground accelerator (concept 2), using five boiler plate SSME engines (except with a 35:1 nozzle), on each of the four accelerators, are based on being able to achieve 89 firings of 16 seconds duration before overhaul. Since the accumulated time for 89 starts is less than 0.5 hours, the overhaul cost was assumed to be one-half that of the SSME. Replacement parts are also assumed to be one-half of SSME requirements.

The cost data are summarized in Table 24 with the SSME data shown for reference. The total cost/flight column reflects two separate costs. The total engine cost per flight are shown above the line and include everything associated with the main engine. The value below the line are the costs associated with the main engine cost per flight element. The differences are labor costs associated with replacement and are included in the ground operations cost per flight element discussed previously.

Spares Cost Analysis. The vehicle spares analysis include the materials/hardware associated with replenishment and refurbishment of the vehicles.

The X-15 vehicle provides refurbishment data as a point for comparison with the SSTO vehicle and is shown in Table 25. Also shown, for contrast, is the typical refurbishment cost of large airplanes as exemplified by the Boeing 720. Refurbishment is expressed as a percent of first unit cost. The data show that the X-15 refurbishment history is consistent with the projected SSTO vehicle with the exception of the propulsion system. This difference is explained by the fact that there were additional items in the X-15 refurbishment associated with the R&D nature of the flights. In an operational program,

Engine \ Cost element	Develop cost ($M)	First unit cost ($M)	Overhaul cost/flt ($K) [1]	Flight servicing cost/flt ($K) [2]	Replacement rotation cost/flt ($K) [3]	Flight support cost/flt ($K)	Propellant and trans. cost/flight ($K)	Total cost/flight ($K)
Space shuttle SSME (ref.)	520	11.3	106.6	52.5	158.6	136.5	52.3	506.5
2.09 MN (470K lb) THRUST		11.3	118.9	33.5				
2 POSITION NOZZLE	50	1.0	10.5	4.0	122.8			292.0
3 ENGINE CLUSTER	50	12.3	129.4	37.5	57.8	2.3		227.0
3.02 MN (680K lb) THRUST	350	13.8	145.2	33.5				
2 POSITION NOZZLE	50	1.2	12.7	4.0	149.6			347.3
3 ENGINE CLUSTER	400	15.0	157.9	37.5	70.4	2.3		268.3
4.89 MN (1100K lb) THRUST	550	17.0	357.9	54.5				
2 POSITION NOZZLE	50	1.5	15.9	4.0	349.0			859.7
3 FIXED/3-2 POS. CLUSTER		18.5	373.8	58.5	162.5	56.7	21.7	673.2
GROUND ACCELERATOR								
BOILER PLATE SSME	–				82.9			176.2
5 ENGINE CLUSTER $\epsilon = 35:1$		11.3	79.3	14.0	33.1	–	–	126.4
GROUND ACCELERATOR								
BOILER PLATE SSME					66.3			159.6
4 ENGINE CLUSTER $\epsilon = 35:1$	–	11.3	79.3	14.0	26.5	–	–	119.8

[1] SCHEDULED MAINTENANCE
30% of new engine cost/70 flights

[2] 1/2 shuttle cost

[3] UNSCHEDULED MAINTENANCE
Same factor as shuttle 2.5/5.0 equivalent engines for 710 flights for hardware which is 40% of total cost (i.e., 60% labor) + x unit spares for turnaround

Table 24 Main Engine Cost Summary

System	X-15 (%)	Space shuttle (%)	720 airplane (3 hr/flt) (%)	New SSTO Subsystem cost Present $	Present %	Potential $	Potential %
Main engine	6.26	0.921	0.02	268K/45M	0.595	112K/45M	0.247
Structure (airframe)	2.50		0.003	77K/257M	0.03	7.7K/257M	0.003
Guidance and navigation (avionics)	5.43		0.02	2.4K/12M	0.02	2.4K/12M	0.02
Thermal protection	N/A	0.67	N/A	12K/4M	0.30	1.2K/4M	0.03
RCS/OMS	N/A		N/A	71K/12M	0.595	36K/12M	0.247
Others (hydraulic electrical power landing gear, etc.)	0.012		0.0001	1.2K/10M	0.012	1.2K/10M	0.012
Percent of first unit cost	3	0.701	0.006	432K/340M	0.127	160.5K/340M	0.047

Table 25 Refurbishment and Maintenance Cost Analysis

these costs would be reduced and become consistent with the expected SSTO vehicle costs. The 720 airplane data are true operational refurbishment costs prorated over several thousand flights and indicate what could eventually be achieved by a SSTO vehicle.

The main engine costs are already included in other cost per flight elements. Deletion of these costs results in a refurbishment maintenance cost of $\sim$.048% of the first unit cost. This cost from historical commercial aircraft data is subdivided: $\sim$.033% hardware and .022% labor. The spares cost reflect the hardware costs only for the vehicle. The labor costs are included under ground operations.

<u>Fuels and Propellant.</u> As previously discussed, the LH_2/LO_2 propellant costs are estimated at $.35/kg ($.16/lb) and make up the majority of fuel costs. Propellant boiloff, cooldown losses and periodic testing propellants, as well as subsystem propellants and facility gases are also estimated.

<u>Program Support</u>. This cost per flight element includes the ground support contractor, sustaining engineering for ground systems, roads and grounds janitorial, fire, security, printing and reproduction, plant maintenance, etc. The manpower estimate is based on two times the flight support (hands on) estimate provided in the ground operations section. This estimate is probably high when considering sharing of some of these facilities with the Space Shuttle. Also included in this cost element are the flight operations.

The cost of flight operations for past space programs has represented a significant per flight cost. During this study, flight operations costs, like launch operations costs, have proved to be difficult to ascertain, except for the highest level cost elements.

Mission control costs include the costs for maintenance and modification from mission to mission, for reconfiguration of the mission control center (excluding cost of the computer) and for development of mission programs for the real time computer complex (including maintenance and operation of the computer system).

Space operations costs include the costs of preflight and flight operations and flight crew operations. Preflight and flight operations costs include mission planning, computer support of trajectory studies, and recovery training.

Flight crew operations costs include program development, maintenance and operation of mission simulators, crew procedure simulation, and other communications, data processing and equipment.

Criteria used for establishment of the flight operations system has been influenced by the R&D nature of the manned flight program, and the desire for complete knowledge of the vehicle's position, attitude, trajectory and condition of its occupants. As manned space flight matures, and moves from the R&D phase into more routine operational activities, it appears that certain changes can be made that will result in lower recurring flight operations costs. This reduction in flight operations cost must be achieved if space transportation costs are to be lowered to an acceptable level.

Since past space flights have been different in mission requirements and flight profiles, this has resulted in new trajectories, new computer programs, and documentation changes for each flight, all resulting in higher recurring costs. A potential solution to this problem is standardized missions, i.e., all missions the same, or nearly so.

Standardized missions will result in fewer changes to flight profile computer programs, hardware and documentation. A great deal of the costs associated with present programs are involved with software changes. If the program can be designed to offer flexibility for small changes (mission sequence) but maintain the software configuration intact, costs can be reduced. Standardized missions are compatible with automated data collection and therefore permit a reduction in the number of ground personnel required to monitor flight progress and to receive and distribute incoming flight performance data.

Standardized missions also result in near negligible costs, on a per flight basis, for recovery training and flight crew mission simulations. Once initial flight crew training is completed, recurrent training for repetitive missions should be a negligible recurring expense.

Communications, tracking and data acquisition recurring costs may be reduced to consist solely of satellite rental costs. Compatible, government-owned satellites may become available that will negate rental of commercial satellites. The availability of government-owned relay satellites should result in further reductions in flight operations recurring costs.

This approach to reducing flight operations recurring costs is summarized in Table 26 where costs are compared between existing Apollo flight operations, anticipated flight operations of a large commercial jet transport, and potential flight operations for a SSTO vehicle. The conclusion that can be drawn from this figure is that, if flight operations for the SSTO vehicles are similar to current airline operations concepts, reduced flight operations recurring costs consistent with requirements for achieving an order of magnitude reduction in space transportation costs to orbit can be achieved.

	SATURN IB/APOLLO COSTS (4/YEAR RATE)*	747 AIRPLANE (3HOURS PER FLIGHT)	POTENTIAL NEW SYSTEM COSTS (PER LAUNCH)
MISSION CONTROL	$8.8M	LESS THAN $.0001 M	SMALL
MAINTENANCE AND RECONFIGURATION OF MISSION CONTROL CENTER		1	SMALL ALL LAUNCHES THE SAME
REAL TIME MISSION PROGRAM & COMPUTER SYSTEM OPERATIONS			ON-BOARD SYSTEM
SPACE OPERATIONS	$7.7M	LESS THAN $.0001 M	
PREFLIGHT & FLIGHT OPERATIONS INCLUDING RECOVERY TRAINING & COMPUTER STUDIES			SMALL - ALL FLIGHTS THE SAME
FLIGHT CREW OPERATIONS INCLUDING MISSION SIMULATORS			INITIAL COST ONLY. AMORTIZED OVER PROGRAM WILL BE LESS THAN $.0001M PER FLIGHT
COMMUNICATIONS, TRACKING AND DATA ACQUISITION	$11.5M	$.0015 M	$.015M
MANNED SPACE FLIGHT NETWORK/ FLIGHT COMMUNICATIONS AND CONTROL			COMMUNICATIONS VIA SATELLITES, MINIMUM GROUND CREW
(NASA PERSONNEL NOT INCLUDED IN COSTS)			
TOTAL RECURRING COST PER FLIGHT	$28.0M	$.0015 M (EST)	$.015M

* 4 SATURN IB/APOLLO (EARTH ORBIT) AND 4 SATURN V/APOLLO (LUNAR TRAJECTORY) FLIGHTS PER YEAR

1 Governmental support is covered in part by local and general taxes on products and services.

Table 26 Flight Operations Cost Analysis

Costs for flight operations, on a per flight basis therefore can be reduced to the cost of maintaining a small ground crew for monitoring flight progress and for data collection and distribution and the cost of rental of satellites for communications and relay of spacecraft data. Since only recurring costs are being considered for purposes of these analysis, facility costs, i.e., mission control center and equipment, are not included. If it is required that these costs be amortized over the life of the program, this will result in increase in the per flight, flight operations cost.

In summary, it is possible to achieve flight operations recurring costs in the order of $.015M per flight by utilizing a minimum of ground crew personnel and by use of communications satellites.

Cost Results. Life cycle costs are presented in Table 27 for the three different configurations. The major cost brackets are design, development, test and engineering (DDT&E), production, and operations.

DDT&E costs are presented in Table 28 for the three study configurations. The major cost brackets are shown under the cost elements. The costs include 3/4 unit each for the propulsion and structural test units under system test hardware, and one production unit plus 40% spares for the flight test program.

Design development and first unit costs are presented in Table 29 for the three study configurations by major subsystem cost elements. Propulsion development cost elements were furnished by Rockwell International's Rocketdyne Division and were a "through put" to the PCM. The one billion dollar design development and 400 million dollar unit cost for the tanker were furnished by the Boeing Commercial Airplane Company.

$1976 millions

Cost element	HTO	IFF	VTO
DDT&E*	3,395	4,142	4,887
Production**	2,327	2,731	3,568
Operations***	2,440	2,505	4,168
Total	8,162	9,378	12,623

*2.9 test units
**4 vehicles
***1,710 flights

Table 27 Program Life Cycle Cost Comparison

$1876 millions

Cost element	HTO	IFF	VTO
Program management	192	120	276
Systems engineering and integration	389	261	488
Vehicle design			
Sled/airplane	199	1,526	–
Orbiter	1,020	608	1,683
Systems test engineering and software	187	255	294
System test hardware*	550	610	881
Flight test hardware**	610	597	875
Ground support equipment	136	96	203
Tooling	29	21	47
Facilities			
Manufacturing	–	–	–
Test	20	16	28
Launch	63	32	112
Mission control	–	–	–
Total	3,395	4,142	4,887

*1.5 vehicles (structural and propulsion test)
**1.4 vehicles (production unit and spares)

Table 28 DDT&E Cost Comparisons

COST ELEMENT	HTO/SLED				IN FLIGHT FUELED				VTO	
	SLED		ORBITER		TANKER		ORBITER		ORBITER	
	DESIGN DEV.	UNIT	DESIGN DEV.	UNIT	DESIGN DEV.	UNIT	DESIGN DEV.	UNIT	DESIGN DEV.	UNIT
STRUCTURE/MECHANISMS	198	58	516	253		65	262	228	942	503
PROPULSION		33	400	41			50	21	600	91
AVIONICS	1	1	51	11			43	10	51	11
THERMAL CONTROL			24	4			20	3	37	5
ELECTRICAL POWER			14	4		335	12	3	14	4
AUXILLARY PROPULSION			24	11			21	9	39	19
ASSEMBLY AND CHECKOUT		2		23				20		45
TOTAL	199	94	1029	347	1000	400	408	294	1683	678

Table 29 Vehicle Design and Unit Cost Comparisons

Production costs are presented in Table 30 for the three study configurations by major cost elements. The orbiter costs are representative of four flight vehicles which are required due to the assumed maximum service life of 500 reuses. The HTO program requires four ground accelerators, and the "in-flight fueled" has two production tanker aircraft. The orbiter vehicles are on a 90% learning curve. Propulsion system components are on a 95% learning curve.

$1976 millions Cost element	HTO	IFF	VTO
Program management	132	92	202
Systems engineering and integration	43	29	54
Vehicle hardware			
Sled/airplane	319	*1,098	—
Orbiter	**1,190	**1,055	**2,316
Vehicle GSE	92	65	138
Tooling	377	278	613
Engineering support/liaison	19	14	31
Initial spares	155	100	214
	2,327	2,731	3,568

*Total airplane production costs (2 required)
**4 vehicles

Table 30 Production Cost Comparisons

Operations costs are presented in Table 31 for the three study configurations by major cost element. Ground operations costs include vehicle contractor launch personnel, propulsion, labor for the vehicle and sled (if applicable), and vehicle spare labor. Main engine support includes flight servicing, overhaul (parts and labor, spares material, flight support and propellants and transportation for all engines - sled and vehicle). Spares include replenishment items other than the main engine. Fuel and propellants include the main ascent propellant, subsystems fluids, as well as facility fluids and gases. Program support includes the flight and mission operation costs as well as the facility operations personnel (i.e. GSE contractor, facilities, maintenance, fire, security, etc.). All tanker operations are included in the one cost element.

$1976 millions	HTO/sled	IFF	VTO
Expendable hardware	0	0	0
Ground operations	513	360	775
Main engine support	675	388	1,151
Spares	195	145	309
Fuels and propellants	670	496	1,330
Program support	249	233	367
Subtotal	2,302	1,622	3,932
Tanker operations	–	741	–
Total	2,302	2,363	3,932
CPF	1.35	1.38	2.29
Transportation cost $/kg	45.64	46.96	78.04
($/lb)	(20.7)	(21.3)	(35.4)

Table 31 – Operations Cost Comparisons

The tanker airplane operations costs are very similar to the HTO/Sled (2363 vs 2302 million) even though the flight vehicle is smaller. The 741 million dollars for tanker operations over the 1710 flights reflect aircraft operational philosophy except for the cryogens and size. However, the tanker requires a completely different logistics program as it has no real commonality with the flight orbiter.

The breakdown of these dollars is as follows:

Three hundred thirty-five million dollars for ground operations. This value is based on an estimate of the "hands on" and "hands off" manhours required for post and preflight serving of the tanker airplane as well as routine support operations between flights.

One hundred twenty-two million dollars for engine support. This includes refurbishment at 6% per 100 flights and replenishment at 0.5%/100 flights. Estimated value of the airbreathing engines is 65 million dollars.

Two hundred twenty-nine million dollars for aircraft spares (less engines). This includes refurbishment at 6% per 100 flights and replenishment at .18%/100 flights. Estimated value of the tanker aircraft is 335 million dollars.

Seventeen million dollars for fuel and propellants. This estimate is based on the 747 airplane requirement of $425/flight hour for fuel. A factor of 7.2 was used to account for the additional engines at higher thrust levels.

Thirty-eight million dollars was estimated for program support. The value is a historical percentage number based on previous program experience.

The SSTO vehicles are compared with the Space Shuttle in cost per flight elements (see Table 32) The shuttle costs are the quoted 1971 dollars escalated to 1976 dollars. Besides elimination of the expendable external tank and solid rocket motors, significant reductions occur in the program support and spares/refurbishment cost elements.

$1976 millions

	S/S 72	S/S 76	HTO	IFF	VTO
Program support (ground operations, flight operations, and program reserve)	3.92	5.17	0.445	0.565	0.668
Spares	0.91	1.2	0.114	0.219	0.181
SRM	3.33	4.40	—	—	—
External tank	1.75	2.31	—	—	—
Engines	0.23	0.30	0.394	0.297	0.673
Fuel and propellant	0.31	0.41	0.392	0.300	0.777
	10.45	13.79	1.35	1.38	2.29

Table 32 Space Shuttle Operations Cost Comparison

To provide a discounted dollar analysis, it is necessary to distribute the program dollars based on the program schedule.

Figure 118 is representative of a typical SSTO type schedule. Based on an IOC of 1995, a program start was required in 1987 for an eight year design and test activity. This provided approximately 10 years of R&D funding (1977 - 1987) prior to ATP. Flight operations occurred between 1995 and 2010 at the rate of 114 flights per year for a total mission model of 1710 flights.

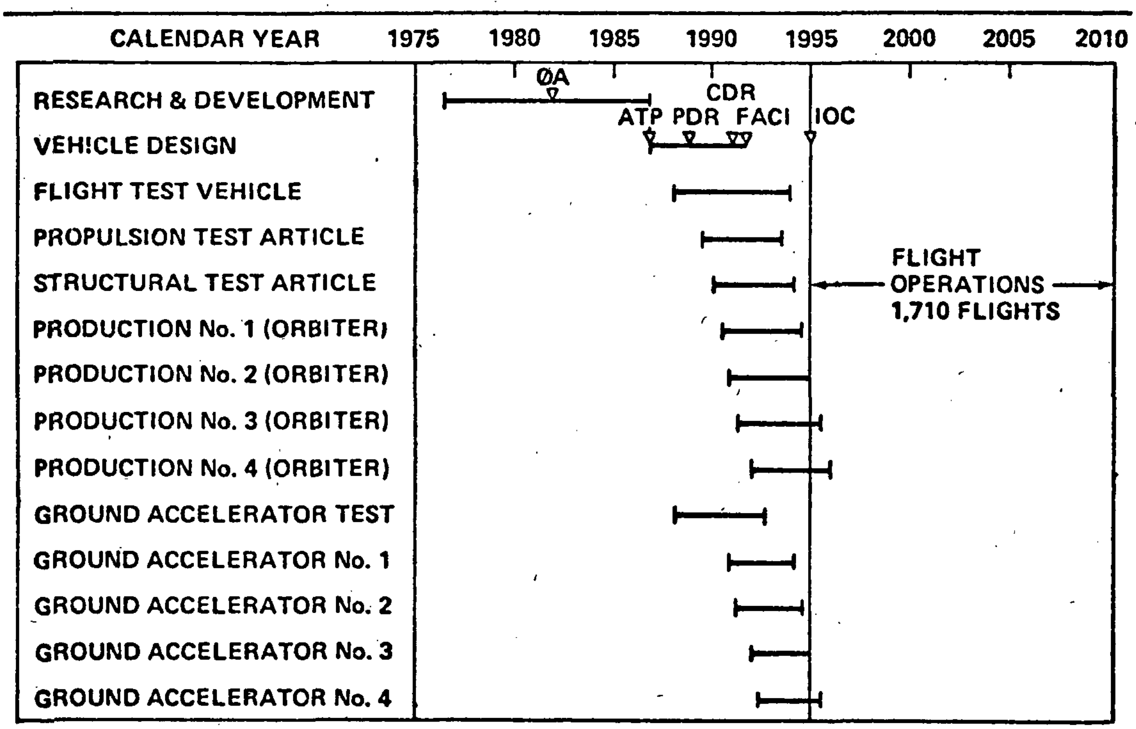


Figure 118 Typical SSTO Program Schedule

Vehicle design, test, production and operations costs are distributed by historical relationships (Figure 119) to provide an SSTO funding profile. The distribution of dollars is similar for all programs, only the totals are scaled to match the cost element differences.

The total program costs are presented in Figure 120 for the three study configurations. The shaded area represents the costs discounted at 10% per year. The costs are nearly proportional to the size of the SSTO vehicles.

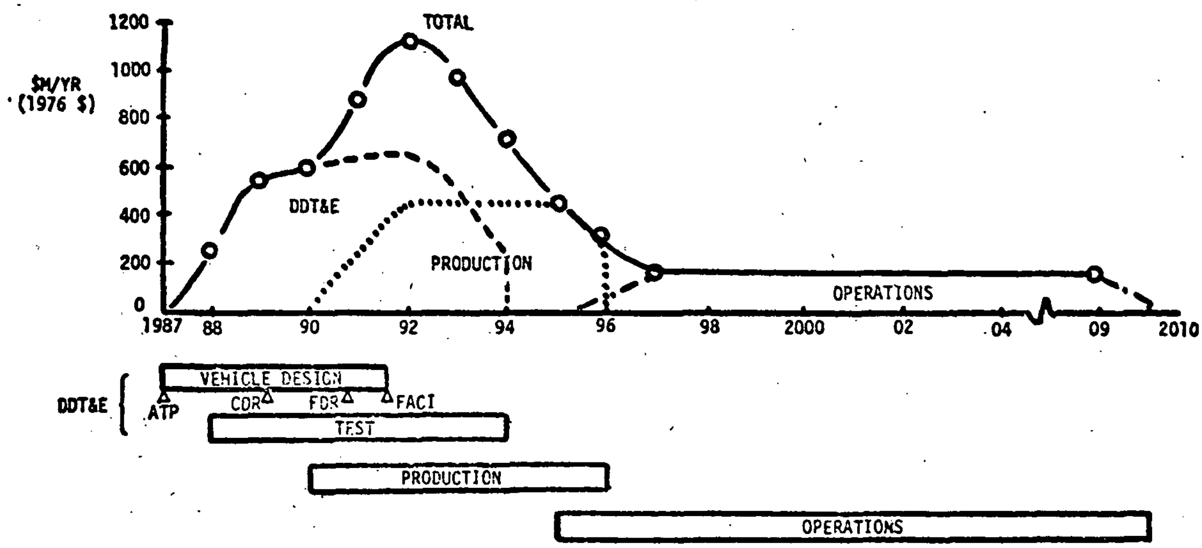


Figure 119 Typical SSTO Funding Profile

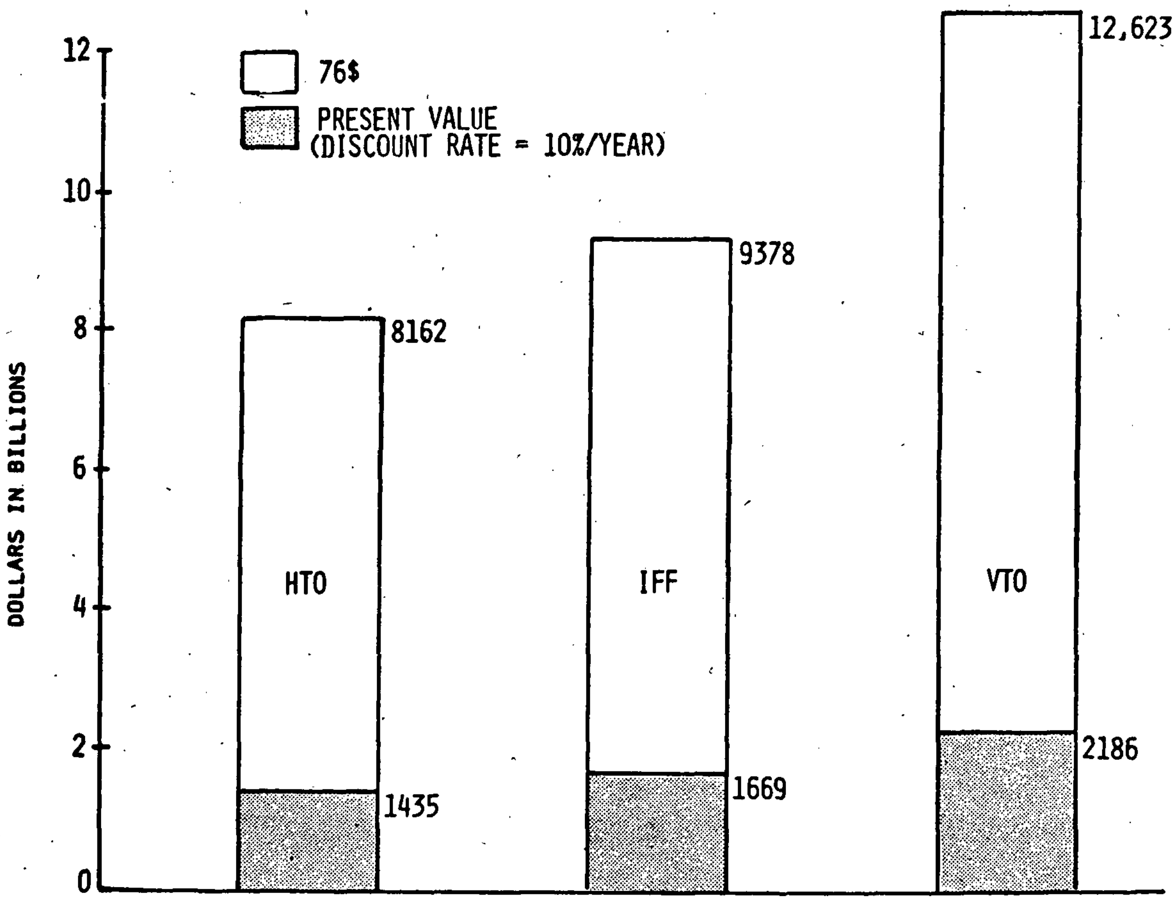


Figure 120 Discounted Program Cost Comparison

ADVANCED TECHNOLOGY ASSESSMENT - TASK III

For the single operational concept selected by NASA in Task II, subsystem weight and performance sensitivities relative to vehicle payload, weight and GLOW were determined. This process defined those subsystems or technology areas where performance advancements had the greatest payoff. For those subsystems selected for technology improvement, a dollar estimate to produce it was made. This estimate was based on in-house experience where applicable, in addition to discussions with outside vendors when appropriate. The Rocketdyne Division of Rockwell International provided the majority of estimates associated with the Main Propulsion System (see Figure 121). The dollar estimate to "produce" is defined as the technology program cost estimate to bring the program to demonstration of feasibility. This does not include the normal DDT&E cost associated with that technology during the regular vehicle program startup. However, in most cases, the DDT&E program is reduced somewhat by

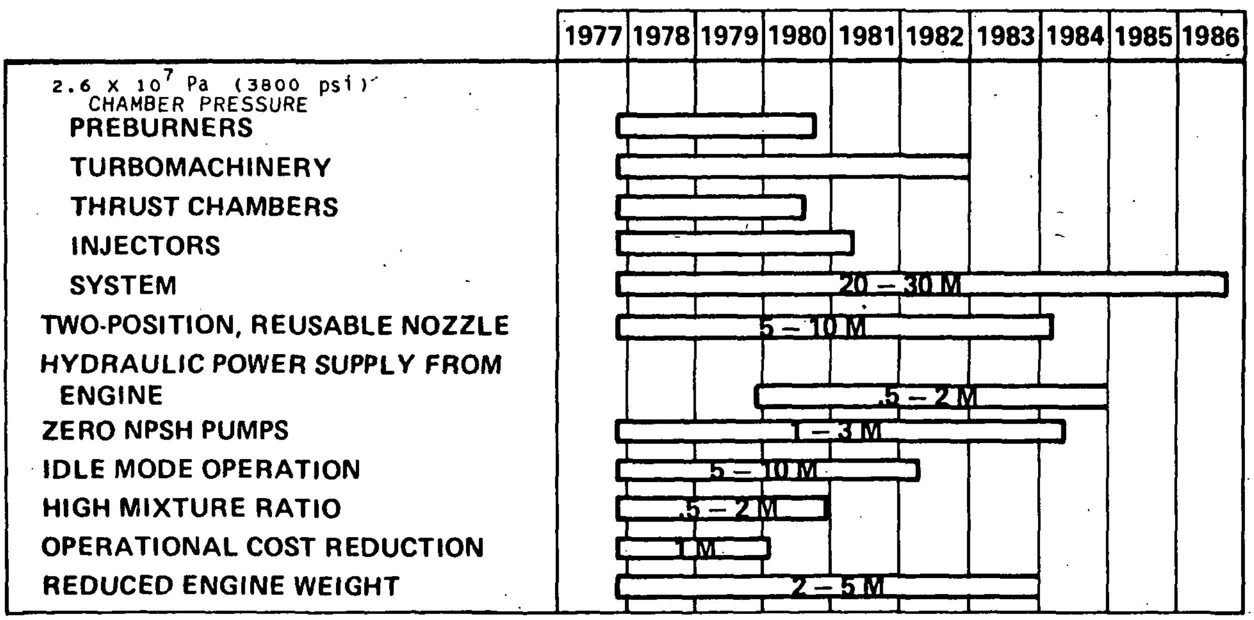


Figure 121 Main Propulsion R&D Schedule

the early R&D funding. For example, the cost to demonstrate technical feasibility of the two-position nozzle is estimated by Rocketdyne to be 10 million dollars. The total two-position nozzle development dollars (DDT&E) based on go-ahead of an SSTO program are 50 million dollars. If the R&D program were to precede the vehicle program, the development bill (DDT&E) reduction would be a 7.5 million dollar or a 75% recovery of R&D funding.

Certain technology areas were selected by Boeing and approved by the Government to define the impact of various levels of R&D funding support. These projections were to be made in the technology areas offering the greatest potential payoff. This assessment would define a level of technology available in the time frame of interest (through 1987).

In all cases the necessary technology programs were to be identified and estimated funding levels indicated. The various technology parameters were to be ranked in terms of a cost/performance/benefit figure of merit (see figure of merit discussion).

In reviewing the projections for normal technology, it became apparent that although certain technology items were considered in this category, if for some reason the projection was too optimistic, the technology program might not get the consideration it warrants. Examples of this are the Rene'41 honeycomb development and the SSME two-position nozzle development. In each case these areas were classified as normal technology growth. The rationale was based on the application potential of both technology benefits to other programs such as Space Shuttle growth, hypersonic research vehicle, Space Shuttle booster derivatives, heavy lift, etc. A typical technology advancement benefits analysis for the main propulsion system is shown in Table 33.

As a result, some of the normal technology items which were felt critical to development of an all metallic reusable thermal structural concept and some technology developments which could be high yield investments to an advanced Space Shuttle derivative were placed in a category call "focused" technology, and evaluated based on "figure of merit" (see Table 34).

The advanced technology programs which would require additional funding and, in some cases, new starts to support an SSTO type program were categorized separately under perturbed technology.

TECHNOLOGY AREA	SPACE SHUTTLE	SPACE SHUTTLE PRODUCT IMPROVEMENT	SPACE SHUTTLE SRB REPLACEMENT	HEAVY LAUNCH VEHICLE	ORBITAL TRANSFER VEHICLE (OTV)	SINGLE STAGE TO ORBIT (SSTO)	DUAL FUEL ENGINE	LINEAR ENGINES
2.6×10^7 Pa (3800 psi) CHAMBER PRESSURE								
PREBURNERS		X	X	X		X	X	X
TURBOMACHINERY		X	X	X		X	X	X
THRUST CHAMBERS		X	X	X		X	X	X
INJECTORS		X	X	X		X	X	X
SYSTEM		X	X	X		X	X	X
TWO-POSITION NOZZLE (REUSABLE)		X	X	X		X	X	
HYDRAULIC POWER SUPPLY FROM ENGINE			X	X	X	X		
ZERO NPSH PUMPS		X	X	X	X	X	X	X
IDLE MODE OPERATION	X	X			X	X	X	X
HIGH MIXTURE RATIO		X	X	X	X	X	X	X
OPERATIONAL COST REDUCTION	X	X	X	X	X	X	X	X
REDUCED ENGINE WEIGHT	X	X	X	X	X	X	X	X

Table 33 Main Engine Technology Benefits Analysis

Normal technology
(No additional funding)

- 2.8% landing gear
- 3.45×10^7 Pa (5000 psi) hydraulics
- Flight control actuators
- LSI circuitry
- Laser radars
- Micro processor
- Solid state displays
- Bubble memories
- Solid state power conditioning and switching equipment
- Boron aluminum composites (non cryogenic application)

Focused technology
(Redirected funding)

- Titanium honeycomb
- Rene' 41 honeycomb
- SSME 2-position nozzle
- SSME idle mode operations
- LO_2/LH_2 APU
- Zero NPSH pump
- SSME operations cost reduction

Perturbed technology
(Additional funding–new starts)

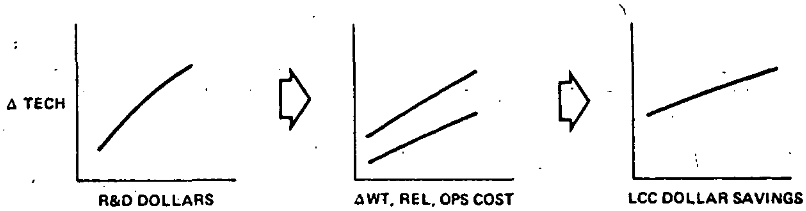
- Linear engine
- Tri-propellant engine
- Slush/triple point hydrogen
- Slush/triple point oxygen
- Slush/triple point hydrogen/oxygen
- SSME hydraulic power
- Increased chamber pressure
- Increased mixture ratio
- Increased engine thrust
- Metallic/atomic hydrogen
- Integrated subsystems
- Flight control actuators
- All movable tail
- Advanced landing gear
- Advanced composites
- 866 K (1100°F) titanium

Table 34 Advanced Technology Classification

Figure of Merit Methodology

Once the R&D cost estimates were made, the technology programs were ranked based on the ratio of the change in life cycle costs to the dollar investment. This ranking was made with both 1976 and 10% discounted dollars. The rationale and methodology for the "figure of merit" is shown on Figure 122. The life cycle cost reduction trending lines were developed by separately costing a .771 X 10^6kg (1.7 million lb) GLOW vehicle. The cost trending line between .771 X 10^6kg and .998 X 10^6kg (1.7 and 2.2 million lb) was assumed to be nearly linear for this analysis (see Figure 123).

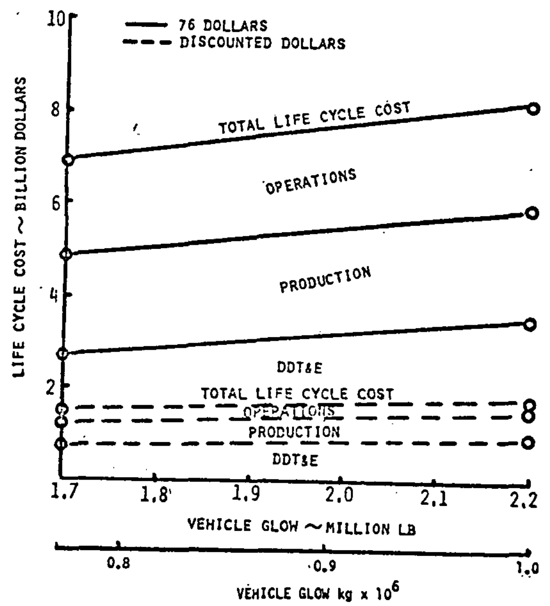
$$\text{FIGURE OF MERIT (FOM)} = \frac{\Delta\text{ LIFE CYCLE COST DOLLARS*}}{\text{R\&D PROGRAM DOLLARS**}}$$



*SAVINGS IN DDT&E, PROD OR OPERATIONS AS A RESULT OF TECHNOLOGY IMPROVEMENT

**ESTIMATED R&D (TECHNOLOGY IMPROVEMENT PROGRAM COST) TO DEMONSTRATE TECHNICAL FEASIBILITY

Figure 122 Figure of Merit Analysis



TRENDING ANALYSIS

76 DOLLARS	\$/kg (\$/lb)
DDT&E	3240 (1470)
PROD	1173 (532)
OPS	1380 (626)
TOTAL	5793 (2648)

DISCOUNTED DOLLARS	\$/kg (\$/lb)
DDT&E	904 (410)
PROD	247 (112)
OPS	128 (58)
TOTAL	1279 (580)

Figure 123 HTO SSTO Cost Trending Analysis

Focused Development

The assumption was made during the early portion of the study that several technology items would be classified under normal development. As discussed previously, it was decided that these development programs should be placed in a slightly more advanced category called focused technology. The difference is that although normal funding was available within the existing and projected NASA budgets for these programs, "emphasis" or "focus" was required to get them off the ground. As for example in the case of the two-position nozzle, Rocketdyne indicated that several studies had investigated and proven the performance benefits of such a development, but at present no "real" effort was directed at funding such a program. This led to a review of all postulated normal technology programs, and as a result several were reclassified in a new "focused technology" program grouping as discussed in detail in the following paragraphs.

Two-Position Nozzle. The two-position nozzle was a former SSME design concept. The SSME currently under development will have a fixed nozzle with an expansion ratio, ϵ, of 77:5 to 1. Design development of the two-position translating nozzle before its elimination was sufficient to prove its feasibility, including development testing of translating hardware. (See performance trades for two-position nozzles).

There was no requirement to translate the Space Shuttle orbiter SSME nozzle extension during engine operation and no testing was directed toward development of this feature. However, ground tests of the SSME design concept at the 1.12 MN (250K lb) thrust level (Pratt and Whitney XCR129) demonstrated nozzle translation from a stowed to an extended position after engine start and the reverse translation prior to engine shutdown, thus establishing the feasibility of operation under thrust loading.

SSME Idle Mode Operations. The trapped propellants on the HTO are 1018 kg (2244 lb) of LO_2 and 934.4 kg (2060 lb) of LH_2. These are trapped due to bubble ingestion and cutoff of the engines. If the engine had an idle mode and could burn these propellants, the injected, and hence, the lift-off weight could be reduced. The usable propellant load would be increased.

For the resultant mixture ratio (1.09) and the idle mode operation, the performance is assumed to be about 1/2 of the nominal engine performance. This degraded performance should also account for flight performance losses incurred by this mode. The following weight changes result:

(1) Residual propellants decrease 1952 kg (4304 lb)
(injected weight savings)

(2) Usable propellant weight increases 976 kg (2152 lb)
(the last 1952 kg (4304 lb) propellant are 1/2 as efficient)

(3) The lift-off weight (2-1) then decreased 976 kg (2152 lb)

Zero NPSH Pumps. The assumption was made in the study that the SSME will operate satisfactorily with zero NPSH at the LH_2 pump inlet. The current design requirement for the SSME hydrogen pump inlet is 1.38×10^4 Pa (2.0 psi) NPSH at normal power level and 1.72×10^4 Pa (2.5 psi) at emergency power level. However, tests on the Rocketdyne J-2 and Nerva liquid hydrogen pumps, plus tests by Pesco and others, have demonstrated that hydrogen pumps operating at zero NPSH are feasible under some inlet temperature and pressure conditions.

There is no current funded program developing zero NPSH pumps for LO_2 and LH_2, and therefore continued R&D is required to support the SSTO. Alternate development programs were considered but eventually discarded.

The SSME low pressure pump could be redesigned for zero NPSH operation or a tank-mounted zero NPSH boost pump could be developed to be included in the LH_2 feed system ahead of the SSME low pressure pump. In addition to pump design, the design of the feed system upstream of the pumps must consider minimizing pressure drop (short lines, no prevalves or sharp bends, etc.) to be compatible with pump inlet requirements.

An analysis has been conducted to determine the weight penalty as a function of NPSH requirements of the LH_2 and LO_2 main engine pumps. Using the SSME pump requirements as state of the art, development of a zero NPSH LH_2 pump for the engines saves 1588 kg (3500 lb) of weight and a 6894.8 Pa (1.0 psi) NPSH LO_2 pump saves 907 kg (2000 lb) of weight. A 6894.8 Pa (1.0 psi) LH_2 pump is as low as the NPSH requirement need be for the SSTO self-pressurization concept. These NPSH requirements are based on 50% flow rate which is the approximate burnout (critical period) flow rate.

The analysis considered stratification of the propellant to determine residual quantities as a function of NPSH and tank pressure. Pressurization systems are required to satisfy higher NPSH requirements. The reduction in pressurant gas weight was also included, and the lower unit on residuals was provided by a draining "pull through" analysis. The largest penalty results from increased tank weight due to increased pressures. The LO_2 tanks are assumed to have a 100.9 kg/(Pa) (222.5 lb/psi) sensitivity and the LH_2 tank is 294.8 kg/(Pa) (650.3 lb/psi).

SSME Operations Cost Reductions. Current operational costs associated with the Space Shuttle and proposed future transportation systems are impacted significantly by main engine operational costs. These include normal flight servicing, major overhaul, unscheduled maintenance, flight support, and propellant and transportation costs. The present Space Shuttle program quotes a cost of 1/2 million dollars per flight for main engine operations. In order to significantly reduce these costs, it is recommended that a program be implemented to examine component life, accessibility, logistic procedures, etc., to determine the reduction possible in refurbishment operations costs.

LO_2/LH_2 APU. A prototype 27.6 kW (37 hp) H_2/O_2 APU was developed by Sunstrand for the X-20. Although the program was cancelled, sufficient work was accomplished to demonstrate preliminary feasibility. Later work on the early Space Shuttle and related contracts involved NASA Lewis, Vickers, AW Research, and others in investigating the O_2/H_2 auxiliary concept. Development funding constraints ended this type of research activity and a hydrazine APU was selected for the shuttle orbiter. Selection of an APU concept for a particular application is dependent upon the total hp-hr requirements of the system. Low hp-hr favor the storable hydrazine type system whereas high hp-hr demands tend to favor the lower fuel consumption (high I_{SP}) H_2/O_2 power plants. The HTO baseline SSTO falls near the cross-over for the different fueled APU concepts. If power requirements were to increase, as in the case of a VTO configuration, an H_2/O_2 fuel would offer substantial benefits. This is predicated on (1) H_2/O_2 offering much lower fuel consumption than typical monopropellant or hypergolic fuels, (2) use of H_2/O_2 results in commonality of fuel with other onboard systems resulting in minimization of GSE and potential integration with other onboard systems such as the RCS and OMS, and (3) propellant and exhaust gases of a H_2/O_2 APU being less corrosive and damaging to equipment resulting in reduced maintenance problems.

Titanium Honeycomb. The development of aluminum brazed titanium honeycomb was accomplished by the Supersonic Transport program under funding from the Department of Transportation. This activity provided data from 164 K (-165°F) to 644 K (+700°F). This type of honeycomb could be used on the upper surfaces of the Boeing all-metallic reusable structural/thermal concept if development work were directed at additional cryogenic allowables development, and compatibility characterization, panel buildup, joints assembly and test.

Rene'41 Honeycomb. Basic Rene'41 materials technology was developed by NASA and the Air Force during the X-20 program in the early 1960's. This material has reuse capability up to 1144K (1600°F). The low planform loading philosophy of the Boeing concept allows usage of a Rene'41 honeycomb sandwich on the lower surface. The honeycomb core and face sheets would be joined by a nickel braze alloy. In addition to providing a reusable metallic structure to withstand the reentry temperatures, the closed cell honeycomb also provides adequate insulative properties at cryogenic temperatures to prevent air liquification. The proposed development program starts with a basic braze alloy process development (currently underway with NASA/Boeing contract) produces small panel samples for allowables development, then graduates to panel and joint assembly and test, and ultimately major subscale component assembly and test.

The development program, for developing the braze process to large scale vehicle section fabrication and test, is shown on Figure 124.

Perturbed Technology

Technology programs which require additional funding (based on not having to redirect other programs) are discussed in the following paragraphs.

Advanced Landing Gear (Servo Oleo). A significant weight savings benefit appears likely through the use of a servo oleo landing gear (see Figure 125) The servo oleo eliminates high lateral and torsional loads from the air/hydraulic support cylinder permitting a more efficient actuator design. The

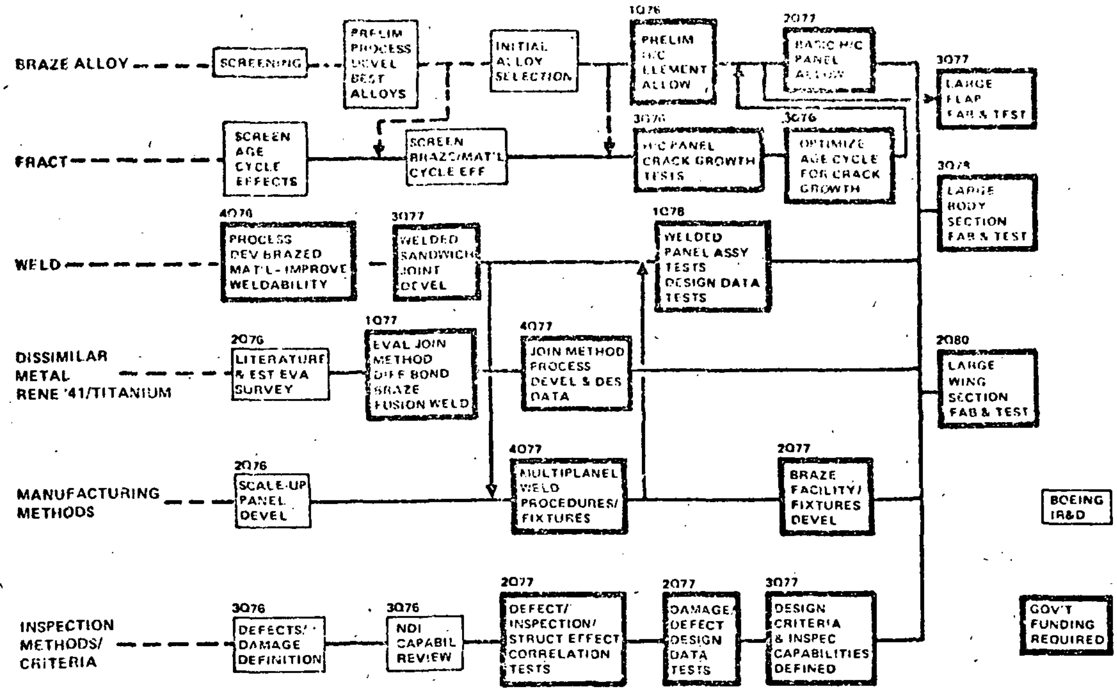


Figure 124 Rene'41/Titanium Honeycomb Development Program

STRUT	391 kg (862 lb)
PISTON	174 kg (384 lb)
INTERNAL VALVING	107 kg (236 lb)
TORSION	51 kg (112 lb)
OIL	38.1 kg (84 lb)
	761 kg (1678 lb)

SERVO ACTUATOR 27.6×10^6 Pa (4000 psi)	181 kg (400 lb)
VERTICAL	363 kg (800 lb)
SWING LINK	136 kg (300 lb)
	680 kg (1500 lb)

11% WEIGHT SAVINGS BY DESIGN CHANGE TO SERVO

SERVO STRUTS MORE COMPATIBLE WITH BORON/ALUMINUM CONSTRUCTION

20% TO 25% WEIGHT SAVINGS

 Figure 125 Servo Oleo Landing Gear

servo oleo permits higher pressure loading in the cylinder without distortion concerns normally experienced with the standard oleo. A second area of significant savings occurs as a consequence of constraining the usual structures loads to the structures elements, verticals and swing arms. This saving is further enhanced by the use of lightweight, high stiffness composite structure. In addition, the elements can be more efficiently designed for the imposed loads without the constraint of a circular cross-section suitable for the hydraulic seals.

The development of this type of landing gear would necessitate the design and development of the servo oleo, the vertical and swing link for the specified application. This program would proceed from an initial computer loads analysis through design and fabrication of the elements to dynamic testing of the assembly.

<u>All Moveable Vertical Fin</u>. The SSTO vehicle flight profile is such that throughout much of the flight the vertical fin and rudder contribute little to the control of the vehicle and, in fact, complicate the control problem. On ascent yaw control is provided by the main engine TVC. On entry, due to the high angle of attack, the vertical fin and rudder are blanked by the fore body. Therefore, it would seem that for the short flight period these surfaces are significant contributors, a major effort should be to assure that these surfaces have the maximum effectivity or efficiency. This would be satisified by the use of an all moveable vertical fin.

Since the entire lateral area is used with the all moveable surface, the required area is reduced by almost 50%. Further, because the aspect ratio is similar, lateral hinge moment loads produce only half the fin bending loads. A larger actuation system is required, but, due to the large structural weights normally associated with conventional tails, the sizable structures weight savings more than balance with an overall saving approaching 40% with no loss of effectivity.

The development program would include computer dynamic analysis, wind tunnel testing, and a dynamic model facility of the system to assure adequate control and stability margins.

<u>Slush and Triple Point Propellants</u>. The difference in slush and triple point are the conditions under which they are stored. The triple point is a

particular temperature and pressure at which all three phases (solid, liquid, and gas) exist. Slush can exist at other temperature and pressure conditions, but the gas phase does not exist at that condition. Gas will exist in the ullage space at a different temperature. Essentially, the solid and liquid phase equilibrium is at a constant temperature, hence slush is at the same temperature as triple point, the pressure in the tank is different. Actually, due to gravity head, even at triple point only the surface is triple point and the lower levels are slush. For use in a propulsion system their performance is identical once tank expulsion begins. The boiling pressure, at which self-pressurization is the operating mode, is the triple point pressure. The liquid/solid mixture cannot convert to gas at any other condition. The drawback is that if flash boiling is used, the tank will collapse. The triple point pressures are 7032 Pa (1.02 psia) for hydrogen and 144 Pa (.021 psia) for oxygen. The net result is that pressurization is required to maintain tank pressure above ambient.

The use of a pressurization system with the triple point propellant still can offer an advantage. The advantage occurs if the pressurization system operates in a manner that assures a positive pressure relative to ambient, but allows the tank pressure to decay. That is, the tank pressure is maintained between 0 and + (some control margin) relative to ambient. Preliminary estimates indicate a 13790 Pa (2 psig) control band with a 17390 Pa (2 psig) relief band could be achieved providing a 27580 Pa (4 psig) design pressure band. The pressurant flow would be stored between 40 and 50 seconds and not required any more. The utilization of the pressurant will save weight of residual pressurant, some of which will be cancelled by the pressurization system weight. The big weight benefit results from structural weight savings due to the lower design pressures. It is estimated that this could result in a 2268 or 4536 kg (5,000 or 10,000 lb) reduction in vehicle weight. Detailed analysis to more precisely determine the weight savings would be quite involved and should not be done. Consideration of minimum gage, thermal requirements, etc. do not permit a straight forward analysis.

For establishment of a figure of merit for slush or triple point technology, a two-step approach is recommended:

(1) Use tank weight savings due to volume reduction; and

(2) Use(1) plus a 2266 kg (5,000 lb) weight saving due to pressure design benefits.

The benefits due to pressurization (reduced pressurant weight) are small in comparison and probably cancelled by the pressurization and vent system hardware weight.

The technology of manufacturing or conditioning LO_2 and LH_2 to their respective triple points is available today. It can be accomplished either by subcooling LH_2 to slush/solid state and using as a heat exchanger or developing a liquid helium refrigeration system, both of which have been demonstrated in laboratory conditions. The technical problems are associated with increased facilities costs and propellant gaging. Density variations in the propellant would be sensitive to any thermal gradients from heat leaks. Large variations could negate a large portion of the potential weight reduction.

Linear Engine Systems. The linear-nozzle type engine is a versatile design which uses available power cycle and propellant flow systems. The multiplicity of packaging options results in extensive flexibility in terms of geometrical configuration and system thrust level. Development is simplified since a high thrust system can be provided from an assembly of lower thrust, independent engine modules, all operating at the same chamber pressure; the thrust chamber assembly can be made up of standardized combustor segments. The design results in a minimum overall length because of the shortness of the nozzle and the ability to package the power system components within the nozzle compartment.

The linear nozzle is a truncated ideal nozzle of the in-flow or linear-spike type with a secondary (base) flow used to incrase the nozzle's base pressure and thus compensate for the shortened length. With an in-flow or spike-type nozzle, the combustion gas is exhausted from the nozzle throat in an inward direction. The primary gases expand against the nozzle wall producing thrust. In operation, the primary flow continues to expand beyond the nozzle exit plane, and encloses the base region. Recirculating gases in the base region provide a pressure on the base. An added, or secondary flow into the base increases this base pressure and thus increases thrust. The outer surface of the primary flow is a free-jet boundary, which is influenced by the ambient pressure. In low altitude operation, the ambient pressure compresses the primary flow against the nozzle wall and increases the nozzle wall static pressure. As a result, the low pressure ratio (i.e., low altitude) performance of a high area-ratio truncated in-flow type nozzle is significantly higher than an equivalent high-area-ratio conventional bell or conical nozzle operating under the same conditions. Also, the compression of the primary flow against the nozzle walls at low altitude conditions eliminates separation, and the side load effects which occur in a conventional high-area-ratio nozzle operating at low-altitude conditions do not occur.

Thrust vector control can be obtained by hinging of the combustor banks. In a typical installation, this would provide pitch and roll control. When moving the hinged banks, effective gimbal angles of 8 or 9 degs. can be achieved with no difficulty.

The present major concern with the linear engine is credible parametric data, especially weights. This problem is being addressed by Rocketdyne under a current study for the NASA Lewis Research Center (NAS3-20114). Present estimates show very little improvement in performance and packaging is promoted as potential weight savings area.

SSME Hydraulic Power. The Space Shuttle main engine in its present form requres a separate hydraulic power source for the thrust vector control system. The F-1 engine provided its own power source in terms of pumped RP-1 propellant to drive the TVC actuators. The J-2 engine similarily contained an accessory pad on which the user could mount a pump or conversion source to power the TVC system. The advantage of using this type of system on the SSTO vehicle is that it significantly reduces the APU size. The APU size is based on the TVC hydraulic flow requirement which in most cases is much greater than the control surface demands during reentry. Sizing the APU for ascent TVC thus provides a lower specific fuel consumption when operated at reduced power level for the flight control loads. This difference is accented at higher power levels, as is the case of the SSTO vehicle in which the APU powers both the hydraulic pumps and the electrical alternators. This is not necessarily true for the Space Shuttle which obtains electrical power from the fuel cells which are weight effective over the APU driven alternators for the longer duration missions.

Tri-Propellant Engine. Some previous developments in tri-propellant combinations indicate a potential for reduced Space Transportation systems costs, both for reusable and expendable vehicles. These propellant combinations are characterized by high I_{SP} (500/s) and low bulk density.

Several tri-propellant propulsion systems have been investigated by engine companies and the Government. The bulk of the work was done on the fluorine/

lithium/hydrogen trio. The general procedure is to have the "main" reaction between the fluorine and lithium and "afterburn" the hydrogen. The exhaust products of the first reaction include gaseous lithium fluoride which is cooled with the hydrogen. Vacuum specific impulses greater than 500 seconds appear attainable.

Rocketdyne reported experimental firings in NASA CR-72325. A separate furnace supplies molten lithium to the main reaction chamber. An additional F_2/H_2 preburner supplies heat for gasifying all the hydrogen needed for the "afterburner". In addition, it has been found desirable to use high pressure hydrogen as an acid to atomize the liquid fluorine and thereby promote high combustion efficiency. Combustion chamber tests at the 11120N (2500 lb) thrust level have demonstrated combustion efficiencies of 94-99%. Maximum run duration has been 10 seconds.

Reported data were collected at various mixture ratios and at a very low expansion ratio; the highest vacuum specific impulse measured was 389 seconds. The I_{SP} efficiency under these conditions was 94.2%. This work has been funded at about the $200,000/year level and was completed. A paper presented at the Miami CPIA Propulsion Conference in September 1969 was said to have documented a vacuum I_{SP} of "over 510 seconds" at an expansion ratio of 60.

The predicted specific impulse of these systems is uniformly high. Laboratory scale combustion tests indicate very efficient combustion is realizable. However, the two-phase flow losses, due to the utilization of metal as a propellant constituent, will make I_{SP} efficiencies above 96% rare. Even so, a specific impulse improvement over LOX/LH_2 of greater than 40 seconds could be attainable. Combustion stability, which has plagued liquid engine development, is said to be no problem with the hydrid-type combustion and was not reported with the Rocketdyne three-liquid work.

The biggest problem , from a performance standpoint, is the extreme difficulty in maintaining the lithium in a liquid state or introducing power into the combustion chamber coupled with the extremely low bulk density. The lithium storage system and the excessive bulk of the propellant tanks will have an impact on tank weight, pressurization system weight, insulation, and consequently on vehicle mass fraction. Furthermore there are problems of toxicity of propellants, complexity and engine weight.

Increased Chamber Pressure. Higher chamber pressure capability is dependent on turbopump capability. Higher pressures require more pump power which in turn requires higher preburner combustion temperatures. Current SSME technology has been demonstrated to 22.5 MPa (3270 psia) capability and could probably allow growth to 24 MPa (3500 psia) but not much more. Improved materials with additional technology in turbopump design should enable an increase to the 26 to 27.6 MPa (3800 to 4000 psia) range for staged combustion cycles. Seals and bearing would require further technology development to achieve these pressures.

Increasing to higher chamber pressures does not improve vacuum specific impulse performance of the engines. The potential benefit to be obtained is in improved thrust to installed weight. At the significantly increased pressures, 26 MPa (3800 psia), it is not clear if there is such an improvement. The studies have assumed some improvement in engine thrust to weight, but significant savings in installed thrust to weight result from reducing the number of engines.

Increased Engine Thrust. Engine thrust can be increased by raising chamber pressure, discussed above, increasing the engine size or both. As engine size is increased, the thrust to weight of the engine is not significantly changed. The benefit is from using fewer engines, reducing vehicle weights. Increasing engine thrust by a factor of two is not considered a technical risk and would be accomplished with a straightforward engine development program. At some factor greater than two, technical questions begin to arise and development programs at component levels would also be required. The engine development program would become more complex, approaching that of a totally new engine program.

Boron Aluminum. Boron aluminum development to produce improved design allowable stresses for high operating loads in the 21K to 589K (-423°F to 600°F) range, combined with the ability to withstand temperature exposure to 700K (800°F), will offer weight savings on the SSTO of at least 1633 kg (3600 lb). Development of large diameter tubes with swaged titanium end fittings for application to body, thrust structure, and fin struts offer weight savings of 998 kg (2200 lb). Application of diffusion bonded boron aluminum to titanium flanges in body, fin, and thrust structure frame, spar and truss member flanges will yield a minimum of 635 kg (1400 lb) weight savings. These savings are based on achieving a minimum of 17 MPa (250,000 psi) tension ultimate stress at room temperature. This strength level is currently available on a development basis and on specific components. Continued development should yield further improvement in strength.

Titanium Matrix Composites. Application of titanium matrix composite tubes to SSTO wing spar structure would yield a minimum of 239 kg (526 lb) weight savings. Titanium matrix composites would have the advantage of withstanding exposure to temperatures of 811K to 866K (1000 to 1100°F), and offer significant strength/weight improvement over titanium tubes.

Brazed Titanium Sandwich. Brazed titanium sandwich that utilizes the full capability of titanium to withstand temperature exposure to 811K (1000°F) would save 567 kg (1250 lb) in SSTO application to forward areas of the fin and wing and body upper surfaces. Weight savings to 923 kg (2034 lb) would be obtained by using an advanced titanium alloy with 866K (1100°F) temperature exposure capability.

Improving titanium sandwich temperature exposure limits from 700K (800°F) to 811K to 866K(1000°F to 1100°F) requires the development of a new brazing alloy system different from the present aluminum braze material. Extension of the use of titanium 866K to (1100°F) from 811K (1000°F) will require further titanium alloy development and materials testing. Efforts in braze alloy and titanium development have been initiated to improve elevated temperature capability.

Increased Mixture Ratio. SSME mixture ratio (O/F ratio) is 6:1. Increasing the mixture ratio of oxygen to hydrogen to values approaching stoichiometric conditions will increase the amount of oxygen required and reduce the amount of hydrogen. Liquid oxygen is 16 times as dense as liquid hydrogen, and, as a result, the volume requirement for the total propellant is reduced, resulting in a reduction in overall dry weight. However, the increase in mixture ratio to 7:1 causes a slight degradation in I_{SP} of 2.7 sec.

Metallic/Atomic Hydrogen. This fuel was originally proposed as a Task III perturbed technology development program, but discussions with Rocketdyne and a review of the Lewis Research Center program with Cornell Laboratories indicate the fuel is not available to support an SSTO development program start in 1987. As a result the figure of merit analysis on this program was terminated.

Integrated Subsystem (OMS-RCS-APU). Individual trades on certain subsystems for a particular set of requirements indicate a variety of different types of concepts are the least weight. As is the case with the HTO SSTO APU, the trade indicated for the power level profile developed, based on ascent and entry hydraulic and electrical requirements, hydrazine shuttle APU was the least weight system. Although the specific fuel consumption or I_{SP} of the cryogenic APU is significantly higher than the storable fuels, the weight penalties associated with the storage and conditioning of the cryogenics cannot be ammortized over the mission at the lower power profiles. However, when a number of subsystems are coupled to a common storage source, then the weight can be distributed in a proportional ratio.

Development of an O_2/H_2 APU and O_2/H_2 RCS has been studied by contractors for NASA Lewis. An RL-10 or reusable space tug engine based on SSME technology is considered sufficiently mature enough to initiate DDT&E. Integrating all three systems would require a detailed examination of the various usage profits and the impact of the systems on each other. Also, an actual demonstration of inter-connecting valving and lines and the verification of hardware feasibility for the APU and RCS would be required.

Flight Control Actuators. The proposed normal technology growth area for the flight control actuators is based on a 34.5 MPa (5000 psi) hydraulics system. The high pressure fluids result in a reduced piston/cylinder volume and in combination with the use of advanced composites significantly reduce overall actuator weight. Other potential concepts for reducing actuator weight or possibly reducing flight control system weight are electro-mechanical or hot gas actuators.

GLOW/Inert Weight-I_{SP} Sensitivity

Sensitivity trades were developed for the HTO vehicle to determine the impact on GLOW of changes in inert weight.

The total effect on resizing the vehicle to maintain a fixed payload of 29484 kg (65,000 lb) due to changes in vehicle inert weight is presented for three engine configurations on Figure 126 . The original baseline with three engines and nozzle ratios $\epsilon = 50/180$, was very sensitive to inert weight increases. The other engine configurations did not have this extreme sensitivity as discussed in the section on performance sensitivity.

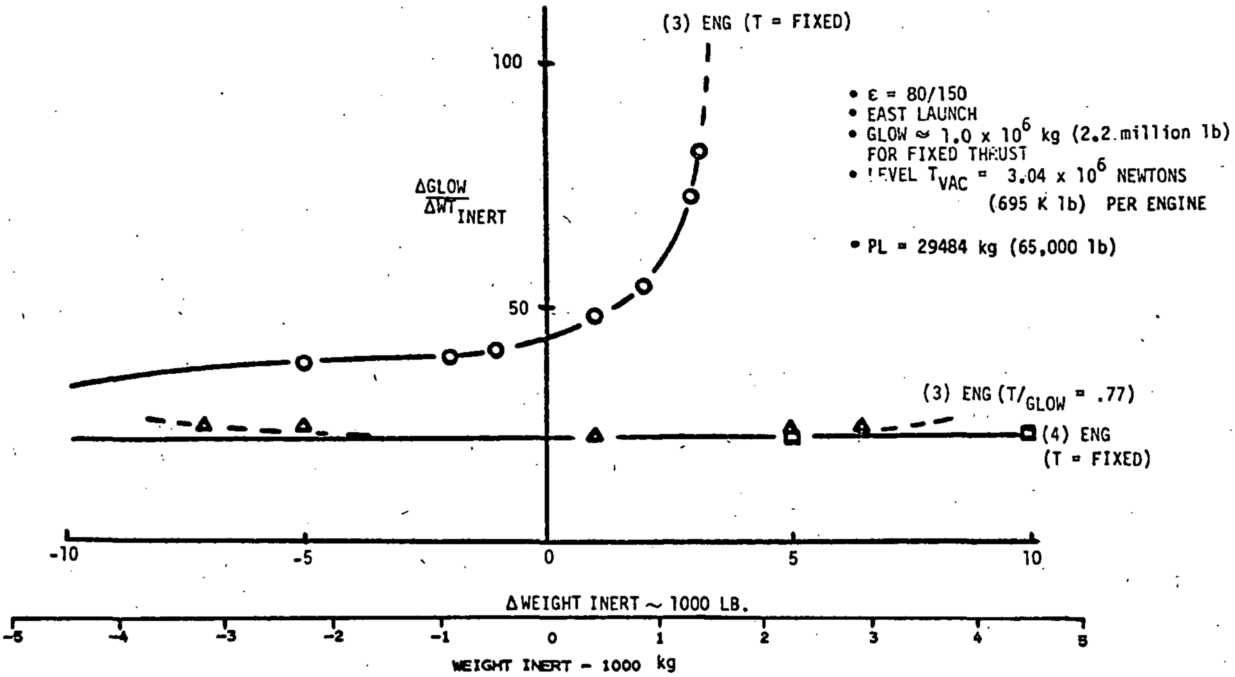


Figure 126 HTO SSTO Inert Weight Sensitivity

Sensitivity trades were developed for the HTO vehicle to determine the impact on GLOW of changes in engine specific impulse I_{SP} on Figure 127. Like the inert weight sensitivity trades, these total effects were determined by resizing the vehicle to maintain a fixed payload of 29484 kg (65,000 lbs.) for two engine nozzle configurations of ϵ = 80/150 and 53/190. The original baseline with a 2-position nozzle of ϵ = 80/150 was very sensitive to small decreases in specific impulse. The optimum nozzle configuration overcame this extreme sensitivity, as shown in Figure 127 and as discussed in section on performance sensitivity.

Use of one or both figures provides the GLOW change resulting from an inert dry weight or engine performance (I_{SP}) technology program improvement. The GLOW change is then transcribed to changes in program cost.

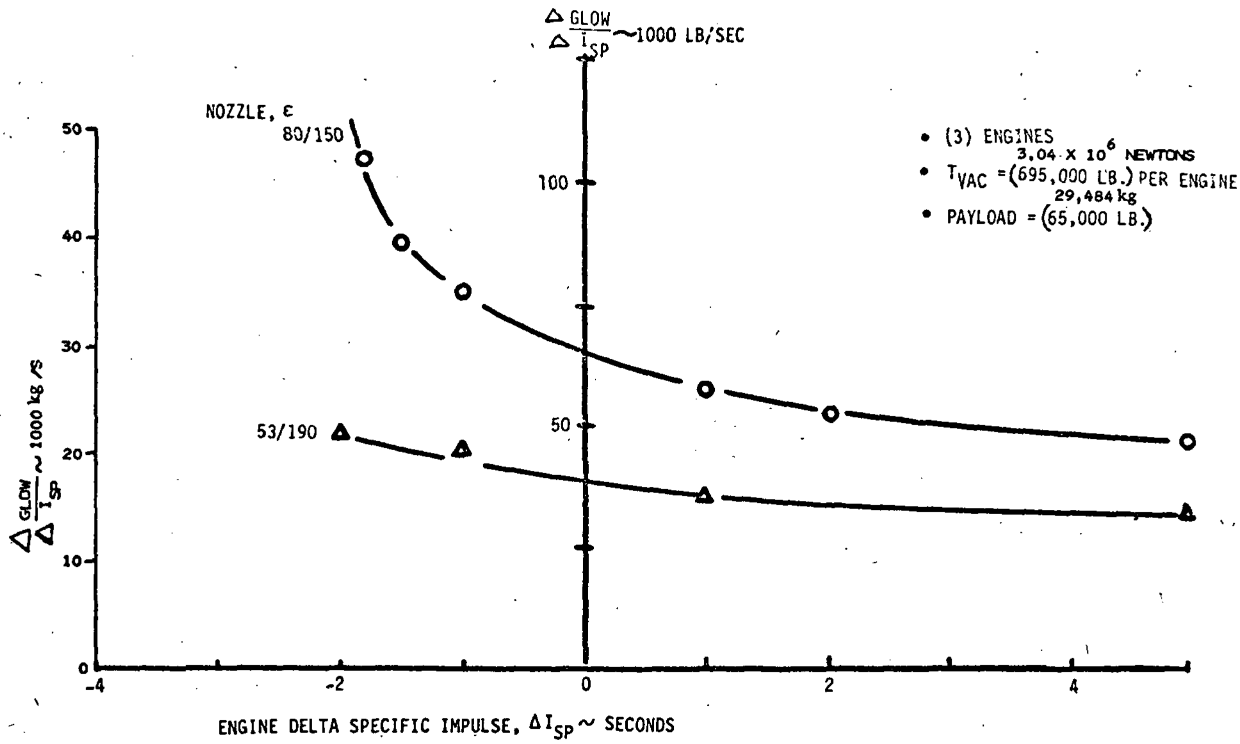


Figure 127 HTO SSTO I_{SP} Sensitivity

Figure of Merit Analysis

Discussed up to this point has been the recategorization of advanced technology programs, the figure of merit methodology which includes the R&D program cost and schedule, the life cycles cost changes as a function of weight and brief discussions of the actual R&D programs themselves.

The technology increments associated with each area differ. A subsystems weight reduction program for example can be directly related to payload and GLOW. However, in the case of engine I_{SP}, the performance improvement is estimated and then translated into a payload gain. Some operations improvements can be directly related to cost savings, either on a per flight or life cycle basis. Tables 35 through 39 illustrate the actual figure of merit analysis for both the "focused technology" and "perturbed technology" development programs. The technology development program column lists the programs which were previously outlined with the associated rationale under their respective categories. The R&D cost in most cases expresses a range of dollars estimated by subcontractors, vendors or qualified personnel in the field who were consulted during the study. The weight columns are self-explanatory and in the majority of cases represent a change in dry weight which can be directly associated with a change in the GLOW, as illustrated in Figure 127. In some instances, performance improvements are combined with dry weight additions to provide an overall payload gain. The cost savings (in most cases) are broken down between DDT&E production and operations and cumulated in the life cycle cost column. The FOM column shows the LCC change over the R&D cost forecast in the first column. Savings are shown in 1976 dollars (no brackets), and dollars discounted from 1976 at 10% per year (with brackets).

In all cases, a detailed examination was made of the technology program to define the weight savings and life cycle cost impact. In certain areas (slush/triple point propellants), although the technology improvement resulted in weight savings which could be related directly to a LCC savings, some additive costs associated with the program reduced the overall savings. A typical figure of merit analysis is detailed in Figure 128 which is provided to illustrate the depth and level of analysis behind each calculation. The example provided outlines the analysis involved with modifying the SSME engine to operate on an idle mode, which would enable complete usage of all of the liquid propellant above the main engine valving. A detailed analysis of residuals in the HTO vehicle tankage indicated that approximately 1018 kg (2244 lb) of oxygen and 934 kg (2060 lb) of hydrogen were trapped in lines between the tank sumps and main engine. This does not include the 819 kg (1806 lb) of propellant trapped within the engine itself. The estimated cost of this program, provided by Rocketdyne, is 7.5 million dollars. The actual weight derivation is illustrated by the tank illustrations on the left hand side of the figure. The upper tank which utilizes the existing SSME contains usable propellant (derived from the performance analysis) reserve and gaseous residuals, trapped liquid propellants, and propellant gaging errors and bias propellant.

TECHNOLOGY DEVELOPMENT PROGRAM	R & D COST $	INERT Δ WEIGHT (LBS)/KG	PERF. Δ WEIGHT (LBS)/KG	P/L Δ WEIGHT (LBS)/KG	DRY Δ WEIGHT (LBS)/KG	GLOW Δ WEIGHT (LBS)/KG	DDT&E COST Δ $	PROD COST Δ $	OPS. COST Δ $	L.C.C. Δ $	FIGURE OF MERIT LCCΔ$/R&D	RNK
Rene'41 H/C Develop Program + Alum. Brazed T. H/C Develop Prog.	14-18M 10M (7.09)*	(-36,000) -16,329	--	(+36,000) +16,329	(-36,000) -16,329	(188,000) 538,865	1746M	632M	742M	3120M	3120M / 16 = 195 (66)*	
SSME with 2-Pos. Nozzle	5-10M 7.5M (4.89)*	(- 5,455) -2474	(+ 7,105) 3223	(+ 5,455) 2474	(+ 1,650) 748	(180,000) 81,646	264M	96M	113M	473M	-473M / 7.5 = 63 (21.3)*	
0 NPSP Pumps	1-3M 2M (1.21)*	(- 5,500) -2495	--	(+ 5,500) 2495	(- 5,500) -2495	(181,500) 83,327	267	97	113M	477M	477 / 2 = 238 (69.7)*	
SSME Operations Cost Reductions	1M (.789)*	--	--	--	--	--			20% 81M 40% 162	81M 162M	81M / 1 = 81 (10.4) 162 / 1 = 162 (20.8)*	
SSME Engine Idle Mode Operations	5-10M 7.5M (5.34)*	(-2152) -976	--	(+ 2,152) 976	--	(71,016) 32,212	104M	38M	44M	186M	186M / 7.5 = 24.8 (7.7)*	
LO_2/LH_2 APU Supercritical Subcritical	2M (1.21) 5M (3.02)*	APU (-625) -283 FUEL (-1964) -891			(+ 625) 283 (- 1964) -891	(20,625) 9,355 (64,812) 29,398	30 95	11 34	13 41	54 11.96 170 (12)*	54 / 2 = 27 170 / 5 = 34 (12)*	
Alum Brazed Titanium H/C Development Program	8-12M 10M (7.09)*	(-18,000) -8165	--	(-18,000) -8165	(-18,000) -8165	(594,000) 269,432	873M (243)	316M (66)	371M (34)	1560M (343)*	1560M / 10M = 15 = 156 (57)*	

( )*INDICATES 1976 DOLLARS DISCOUNTED AT 10%/YEAR

Table 35 Figure of Merit Analysis

TECHNOLOGY DEVELOPMENT PROGRAM	R & D COST $	INERT Δ WEIGHT (LBS)/KG	PERF. Δ WEIGHT (LBS)/KG	P/L Δ WEIGHT (LBS)/KG	DRY Δ WEIGHT (LBS)/KG	GLOW Δ WEIGHT (LBS)/KG	DDT&E COST Δ $	PROD COST Δ $	OPS. COST Δ $	L.C.C. Δ $	FIGURE OF MERIT LCCΔ$/R&D	RNK
Linear Engine M S/L Thrust	20-30M 25M (14.7)*	Struct (-3448) -1564 ENGINE	(-2000) -907	(+552) 250	(-1448) -657	(47784) 21,674	- 70M + 97M +27M	25M 25M	30M 30M	-125M + 97 -28M	- 28.3M / 25M = 1.1 (.02)*	
Hydraulic Power Supply From Main Engine	.5-2M 1.25M (.710)*	(-786) -357	NEG	(+786) 357	(- 786) -357	(25938) 11,765	38M	14M	16M	68M	68M / 1.25M 54.4M (20.1)*	
Slush Hydrogen 50% Quality 13.3% Vol. Reduction	4-6M 5M (3.02)*	(-3865) -1753		(+3865) 1753	(-3865) -1753	(127,545) 57,853	187M - 16M 171M	68M 68M	80M -112M - 32M	335M -119M 217M	217M / 2.5 = 43.4 (20)*	
Slush Oxygen 50% Quality 14.8% Vol. Reduction	4-6M 5M (3.02)*	(-3559) -1614		(+3559) 1614	(-3559) -1614	(117,747) 53,409	173M - 20M 153M	63M 63M	74M - 8M 66M	310M - 28M 282M	282M / 5M = 56 (21)*	
Slush Oxygen + Slush Hydrogen 50% Quality 13.3% O_2 Vol. Reduct. 14.8% H_2 Vol. Reduct.	8-12M 10M (6.04)*	(-7424) -3367		(+7424) 3367	(-7424) -3367	(244,922) 111,094	360M - 36M 324M	130M 130M	153M -120M 33M	643M -156M 487M	487M / 10M = 48.7 (20)*	
Triple PT Hydrogen 8% Vol. Reduction	2-3M 2.5M (1.47)*	(2325) -1055		(+2325) 1055	(-2325) -1055	(-76,725) 34,802	113M - 16 97M	41M 41M	48M -112M 64M	202M -128M 74M	74M / 2.5 = 30 (20)*	

( )*INDICATES 1976 DOLLARS DISCOUNTED AT 10%/YEAR

 Table 36 Figure of Merit Analysis

TECHNOLOGY DEVELOPMENT PROGRAM	R & D COST $	INERT Δ WEIGHT (LBS)/KG	PERF. Δ WEIGHT (LBS)/KG	P/L Δ WEIGHT (LBS)/KG	DRY Δ WEIGHT (LBS)/KG	GLOW Δ WEIGHT (LBS)/KG	DDT&E COST Δ $	PROD COST Δ $	OPS. COST Δ $	L.C.C. Δ $	FIGURE OF MERIT LCCΔ$/R&D	RNK
Triple PT Oxygen 12.9% Reduction	2-3M 2.5M (1.47)*	(-3102) -1407	--	(+3102) 1407	(-3102) -1407	(-102,360) -46,432	150M - 20 130M	54M 54M	64M -8 56M	268M - 28M 240M	240M / 2.5 = 96 (37)*	
Triple Pt. Oxygen + Triple Pt. Hydrogen 8% H_2 Vol. Reduction 12.9% O_2 Vol. Reduct.	4-6M 5M (3.02)*	(-5427) -2462	--	(+5427) 2462	(-5427) -2462	(-179,097) -81,234	263M - 36 227M	95M 95M	112M ÷120M ÷ 8M	470M -156M 314M	314M / 5 = 63 (28)*	
Increase Chamber Pressure (3000-3800)	20-30M 25M (14.06)					(-70,603) -32,025	112M	40M	48M	200M	200M / 25 = 8 (3.1)*	
Increase Mixture Ratio (6:1 to 6:7) (6:1 to 5:1)	.5-2M 1.25M (.925)*	(-1799) -816 (+2570) 1166	(-2500) -1134			(- 59,367) 26,928 (+100,000) 45,359 (40,633) 18,431 (84,810) 38,469				107M 224M	-107M / 1.25 = -86 (-25.4)* -224M / 1.25 = -179 (-53)*	
Increase Engine Size (680K to 1.02M) (680K to 1.91M)	3-5M (1.8)* (3.0)*					(- 70,888) -32,154 (+136,450) 61,893	109M -200 -91M 201M -1M -200M	39M 39M 73M 73M	46 46 85M 85M	194M -200 -6M 359M -631M -272M	- 6M (100M loss)* 272M loss (93.3M loss)*	

( )* INDICATES 1976 DOLLARS DISCOUNTED AT 10%/YEAR

Table 37 Figure of Merit Analysis

TECHNOLOGY DEVELOPMENT PROGRAM	R & D COST $	INERT Δ WEIGHT (LBS)/KG	PERF. Δ WEIGHT (LBS)/KG	P/L Δ WEIGHT (LBS)/KG	DRY Δ WEIGHT (LBS)/KG	GLOW Δ WEIGHT (LBS)/KG	DDT&E COST Δ $	PROD COST Δ $	OPS. COST Δ $	L.C.C. Δ $	FIGURE OF MERIT LCCΔ$/R&D	RNK
Tri Propellant Engine Develop.	100-200M		40 sec			(1,279,000) 580,145	1,867M -1,500 367M	676M 676M	795M -795M	3338M 1500M 1838	1838 / 150 = 12.2 (3.44)*	
Integrated Subsystems OMS-RCS-APU	5M (3.02)*	(-2000) -907	--	(+2000) 907	(-2000) -907	(66,000) 29,937					174 / 5 = 34.7 (10.9)*	
Flight Control Actuators			DISCONTINUED									
Fly By Wire Digital Control		INCORPORATED AS NORMAL TECHNOLOGY										
LLV (All Moveable Tail)	6M (4.14)*	(-3058) -1387	--	(+3058) 1387	(-3058) -1387	(-100,914) -45,774	148.3 - 10.0 138.3	53.7M 53.7M	63.2M 63.2	265.2 10.0 255.2	255.2 / 6 = 47.1 (13.4)*	
Landing Gear	.75M	(- 810) -367	--	(+ 810) 367	(- 810) -367	(- 26,730) -12,125	39.3M - 5.0M 34.3M	14.2M 14.2M	16.7M 16.7M	70.2 5.0 65.2	65.2 / .75 = 86.9	

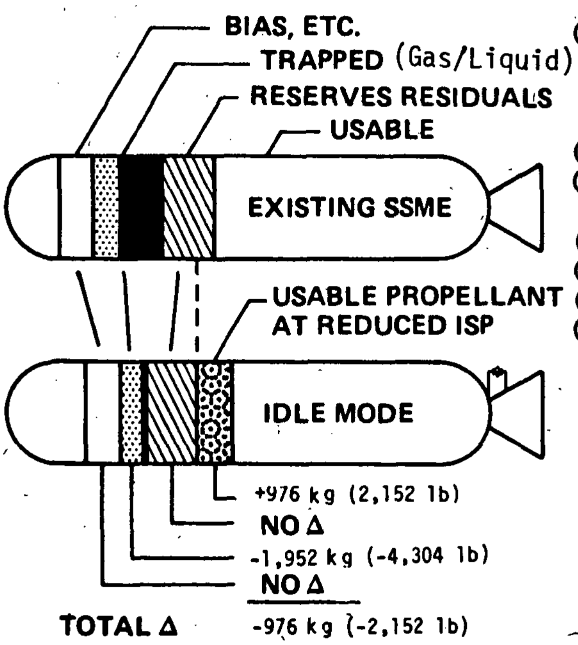
( )* INDICATES 1976 DOLLARS DISCOUNTED AT 10%/YEAR

Table 38 Figure of Merit Analysis

TECHNOLOGY DEVELOPMENT PROGRAM	R & D COST $	INERT Δ WEIGHT (LBS)/KG	PERF. Δ WEIGHT (LBS)/KG	P/L Δ WEIGHT (LBS)/KG	DRY Δ WEIGHT (LBS)/KG	GLOW Δ WEIGHT (LBS)/KG	DDT&E COST Δ $	PROD COST Δ $	OPS. COST Δ $	L.C.C. Δ $	FIGURE OF MERIT LCCΔ$/R&D	RNK
Advanced Composites 1) Boron Alum	8-12M 2M (1.4)*				(-3576) -1622	(-118,008) -53,528	173M	63M	74M	310M	310M/2 = 155 (49.6)*	
2) Ti Matrix	8M (5.5)*				(-526) -239	(-17,358) -7873	26M	9.2M	11M	46M	46M/8 = 5.75 (1.8)*	
1100°F Brazed Titanium Sandwich	1-3M 2M (1.4)				(-2034) -923	(-67,122) -30,446	99M	36M	42M	177M	177M/2 = 88.5 (28.2)*	
Ground Accelerator* Increase L/O Speed to 700 ft/sec	10M					(-200,000) -90,718					521M/10 = 52.1	
Decrease Speed to 500 ft/sec	2M					(+200,000) 90,718					+525M/-26.3 = 6	
Air Cushion Veh.	20-30M			(DISCONTINUED)								
*Applicable to HTO concept only Used for internal design trade												

( )* INDICATES 1976 DOLLARS DISCOUNTED AT 10%/YEAR

Table 39 Figure of Merit Analysis



① **Present HTO analysis**
Trapped propellant = 1,017.9 kg (2,244 lb) **of LO_2**
= 934.4 kg (2,060 lb) **of LH_2**
Engine propellant = 819.2 kg (1,806 lb) **of LH_2/LO_2**

② **Add idle mode to engine (R&D = \$7.5 M)**

③ **Burn propellant at 1.09 mixture ratio (~1/2 nominal engine performance)**

④ **Total inert weight savings** = 976 kg (2,152 lb)

⑤ **Total GLOW reduction** = 32,206 kg (71,000 lb)

⑥ **Total LCC reduction = 186 M**

⑦ **Figure of merit**

$$\$76 = \frac{186}{7.5} = 24.8$$

$$10\% \text{ discount} = \frac{41.4}{5.34} = 7.7$$

Figure 128 Idle Mode Operations Figure of Merit Analysis

The addition of an idle mode allows burning of the trapped propellant at a much lower flow rate and off design mixture ratio. The resultant ratio of 1.09 is estimated to achieve a performance of about 1/2 the nominal. As a result, the 1952 kg (4304 lb) of trapped propellant are burned completely but require an additional 976 kg (2152 lb) of loaded propellant due to the performance degradation. Overall inert weight change is 976 kg (2152 lb). This would be applicable to a case in where the reserves have already been utilized due to dispersions, etc. in the ascent trajectory. Savings of 976 kg (2152 lb) result in a GLOW reduction of 32205 kg (71,000 lb). Using the cost trending curve illustrated previously as Figure 123 yields a program cost saving of 186 million dollars. The 186 million dollar life cycle cost saving over the estimated 7.5 million dollar research and development program cost gives a figure of merit value of 24.8. Discounted dollars at 10% per year yield a FOM of 7.7.

Technology Improvement Sensitivity

It is interesting to note that the technology improvements cannot be considered on a linear relationship as the sensitivity of GLOW impacts vary considerably depending upon the assumptions made. For instance, shown in Figure 129 are the structural weight improvements as a function of GLOW for alternate propulsion advances. The existing technology for structural unit weight is assumed to be 34 Pa (7 lb/ft^2). As noted on the chart, propulsion improvements have a rather large improvement as in the case of (1) the 2-position nozzle for the SSME, and (2) in addition to the 2-position nozzle an increase in thrust, with a resultant decrease in the installed weight/thrust ratio. However, if technology gains are made in the area of structural unit weights as represented by the HTO SSTO 253 Pa (5.3 lb/ft^2) on the figure, the propulsion improvements are not nearly as significant or critical. However, if the goals of the structural program are not completely attained, then the propulsion system advances have an increasing impact.

Another way of expressing this non-linear relationship is shown on Figure 130. The same trending data are used to show the impact on GLOW of advancements in structure, propulsion and subsystems when developed separately. Advanced structures are reducing the average unit weight from 33 to 25.4 Pa (6.8 to 5.2 lb/ft^2). Advanced propulsion is an SSME type 3.04 MN (695K lb) vacuum thrust engine with a 2-position nozzle. Advanced subsystems reflect

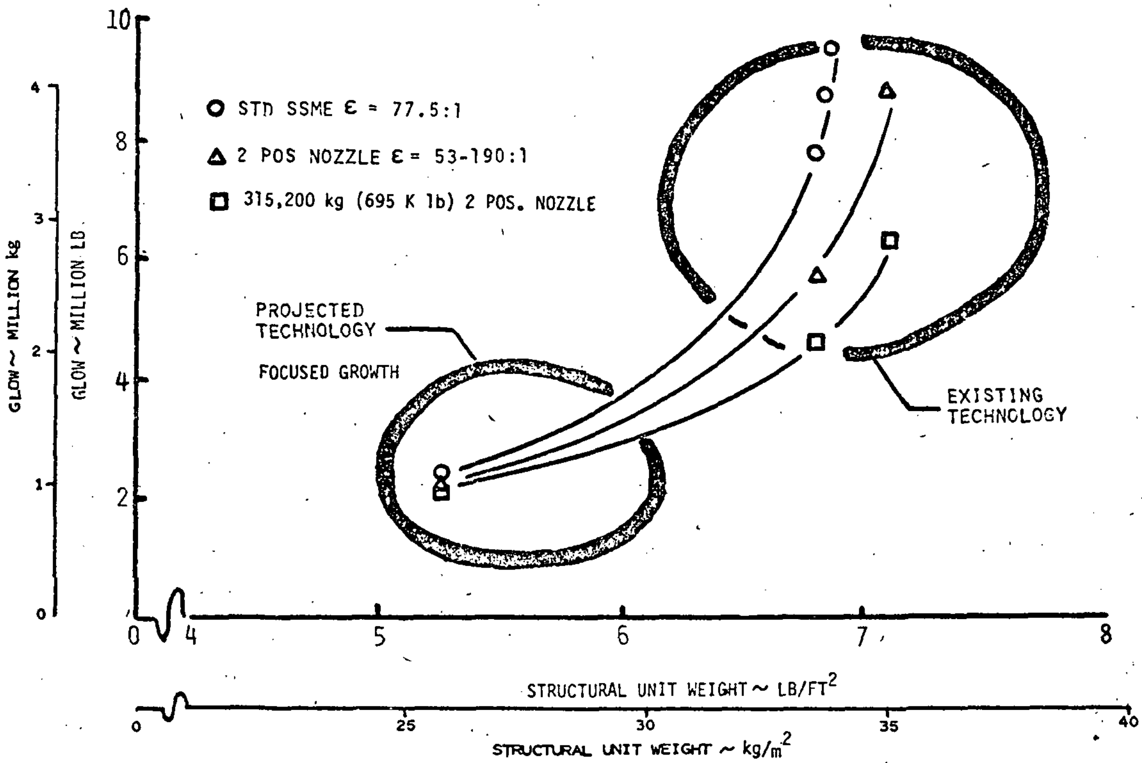
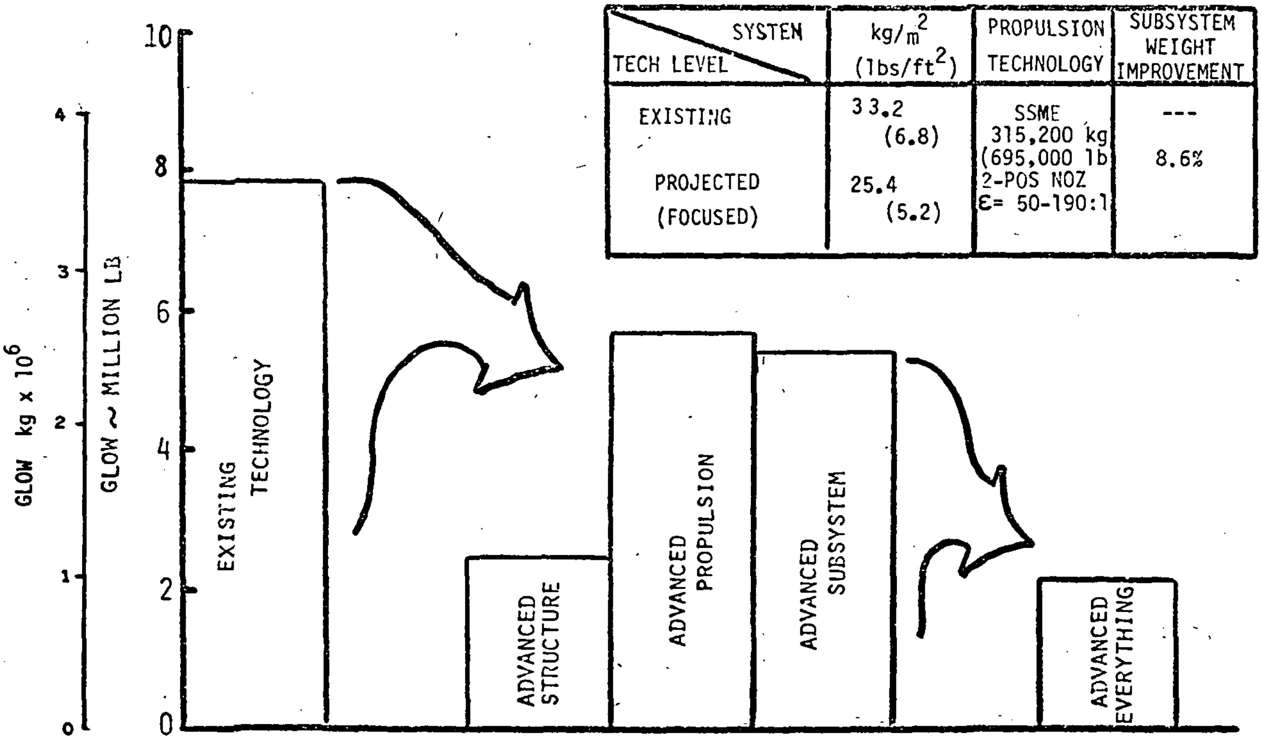


Figure 129 SSTO Structural Unit Weight Sensitivity



TECH LEVEL \ SYSTEM	STRUCT kg/m² (lbs/ft²)	PROPULSION TECHNOLOGY	SUBSYSTEM WEIGHT IMPROVEMENT
EXISTING	33.2 (6.8)	SSME 315,200 kg (695,000 lb 2-POS NOZ ε= 50-190:1	---
PROJECTED (FOCUSED)	25.4 (5.2)		8.6%

 Figure 130 SSTO Technology Advancement Sensitivity

an 8.6% reduction in inert weight. Improvements in all areas combined at once provide a GLOW of 957075 kg (2.11 million lb). For this analysis it was assumed that each development was separate. This is not a true picture of several technology gains, but it is used to show each relative to the other.

Advanced Technology Ranking

Figure 131 shows the relative ranking of the "focused technology" development programs. The total bar represents FOM in 1976 dollars. The shaded portion represents 10% discounted dollars. The zero NPSH pumps have the highest value for FOM. This results from the weight savings associated with reducing the overall design operating pressure limit due to the reduction in ullage pressure. The Rene'41 and titanium honeycomb programs follow in ranking again due to the significant weight impact they

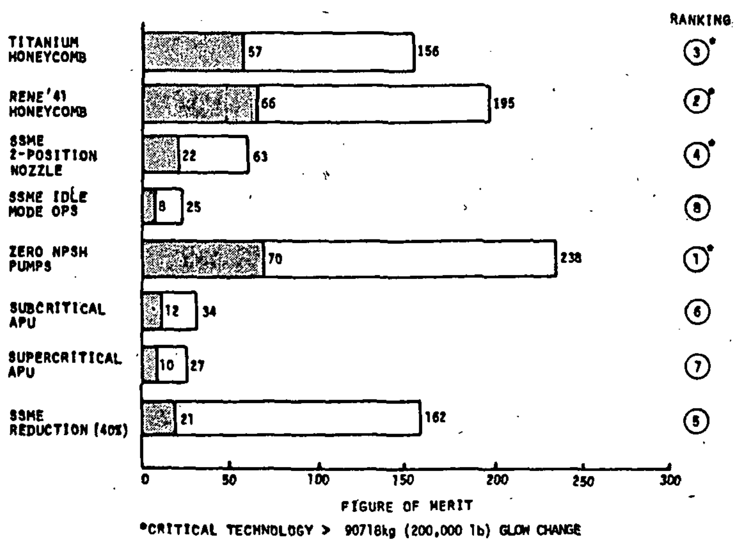


Figure 131 Focused Technology Program Ranking

have on inert weight. The Rene'41 honeycomb reflects the combined usage of both Rene'41 and titanium honeycomb on the lower and upper vehicle surfaces respectively. The titanium honeycomb program reflects a complete overall surface utilization of titanium with the addition of insulation system on the lower surface to prohibit temperatures in excess of 700K ($800^{\circ}F$). Reductions in main engine operation costs of up to 40% show a relatively high figure of merit. It is interesting to note that when discounted, the overall rating of the operations cost reduction is reduced. This is attributed to the cost savings occuring in the later stages of the program when the discounted rate tends to drive the savings to a lower value. The SSME 2-position nozzle program, when analyzed with projected improvements in structural technology, has a relatively high ranking. Ranking of the subcritical and supercritical APU's and the SSME idle mode operations follow.

Figure 132 shows the relative ranking of the perturbed technology development programs. Again the difference between 1976 dollars and dis-

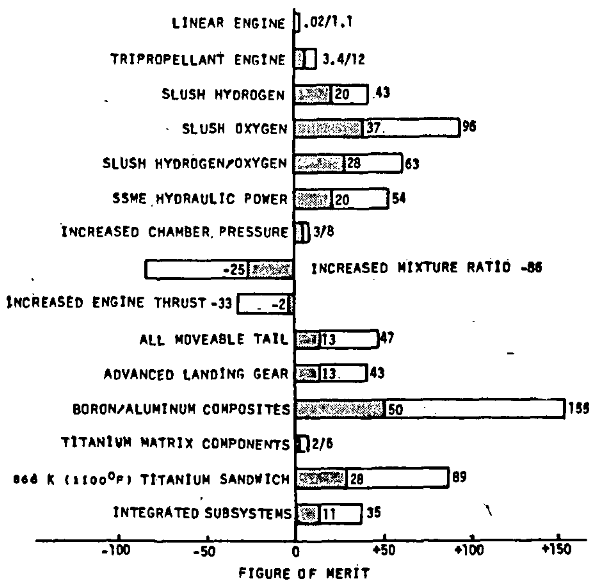


Figure 132 Perturbed Technology Program Ranking

counted dollars is indicated by the plain and shaded portions of the bar respectively. The boron aluminum composite work, in addition to the 867K (1100°F) titanium sandwich, show high yields in terms of figure of merit. The slush and triple point propellant programs offer a high potential for reducing the life cycle costs in relationship to the R&D investment. The programs to the left of the vertical centerline (a FOM ranking of zero) indicate that the R&D program investment yielded an increase in life cycle costs.

Table 40 summarizes the technology development program ranking. As noted on the figure, the programs are arranged in subsets which are based on changes in GLOW. This is intended to show the criticality of the various programs and their overall impact on the SSTO program. The dual values shown in the table reflect the range of figure of merit. The lower is if the structural program (Rene H/C) is successful, the higher is if present structural technology is used.

Technology developments > 90,720 kg (200,000 lb)			
0 NPSP pumps	70	SSME with 2-position nozzle	21/200
René 41 H/C development	66	Slush hydrogen/oxygen	20/28
Aluminum brazed titanium	57	Tri propellant	3.4
Technology developments 90,270 - 45,360 kg (200,000 - 100,000 lbs)			
Boron aluminum	50	Slush/triple point oxygen	21/37
Slush/triple point hydrogen	20/20	All movable tail	13
Technology developments ≤ 45,360 kg (100,000 lb)			
866K (1100°F) titanium	28	Titanium matrix composite	1.8
Landing gear	22	Linear engine	.02
Hydraulic power	20	Increased mixture ratio	-25
Super critical APU	12	Decreased mixture ratio	-53
Integrated subsystems	11	Increased engine size 8.45 MN (1.9 M)	-20
Subcritical APU	10	Increased engine size 4.5 MN (1.02 M)	-33
SSME idle mode operations	7.7		
Increased chamber pressure	3.1		

Table 40 Advanced Technology FOM Summary

Qualitative Observations. In summary, the Rene'41 and titanium honeycomb programs, combined with zero NPSH pumps are critical to development of an all metallic reusable (hot structure) SSTO concept. Propulsion is a critical development item especially when considering that the structural unit weight goals forecast are not obtained.

Within the propulsion constraints of the study, little or no gain is evidenced in developing a new LO_2/LH_2 propulsion system. The study excluded hydrocarbon propellants and combinations of hydrocarbon and LO_2/LH_2 propellants (dual mode). The linear engine system analysis indicated a relatively low FOM ranking. This analysis based on preliminary data showed a net loss in performance when compared to the 2-position nozzle on the SSME type engine. A potential for better performance or decreased engine weight in terms of a constant installed thrust is possible with the linear engine. Net savings in thrust structure and installation weights did not have a significant impact. However, the installation was made more or less on a one to one basis with the SSME (i.e. similar to the aerospike design) and did not take advantage of the fuel capabilities of the linear engine design. It is felt that a more detailed study of the linear engine is warranted in that it is sensitive to the configuration. Integration of the engine with a new HTO design could offer reduced engine weights resulting in a more stable vehicle in addition to providing the potential for added lift during ascent.

The technology programs associated with modifications and/or improvements to the existing SSME show relatively high gains. The 2-position nozzle would not only benefit the SSTO program, but could provide performance gains to shuttle derivative and heavy lift programs as well.

The slush/triple point cryogenic propellant programs indicated a potential for reducing the overall volume requirements. It is felt however, that a more detailed analysis of this option is required due to the limited depth of the analysis. Of particular concern are the added cryogenic transfer cool down losses, the specific facility requirements including lines, refrigeration equipment, etc., and the propellant gaging tolerances. The later problem is associated with thermal gradients within the vehicle tankage, how they are impacted with delays or hold times, and the variations in density which could negate some of the volumetric reductions.

Several technology programs indicate a rather high yield FOM because of the low R&D funding required to demonstrate feasibility of rather moderate weight savings. Typical of these programs are the LO_2/LH_2 APU, the engine driven hydraulic pumps and integration of cryogenic propulsion systems such as the OMS, RCS, and APU. All of these programs have had some effort directed at them in the past, whereas the forecasted R&D program does not reflect a new start.

Quantitative Considerations. Several technology programs which are not direct hardware developments, but have a significant impact on hardware systems, require continued support. In most cases the ability to actually attribute specific weight savings or life cycle cost reduction to these activities would be arbitrary. However, their importance and broad application in terms of Advanced Space Transportation systems analysis cannot be discounted.

The requirement of several iterations of the vehicle configuration has revealed the importance of computer aided design as a vital tool in future configuration and system studies. The ability to associate several key technology disciplines and determine the interactions and constraints of each on vehicle design not only results in massive labor savings, but assures a complete and total analysis of vehicle design changes.

Control configured design offers a potential solution to the age old stability problem associated with rocket powered flight vehicles. Its application is not concept oriented as both the HTO and VTO vehicle configurations could benefit from this design technique. The HTO inherently due to its lower thrust to weight ratio has less of an airframe balance problem. However, it spends more time in the horizontal flight regime where flight control is required.

Mold line tankage and integrated equipment packaging and installation are extremely important when considering proper volume utilization of the total SSTO vehicle.

Additional data are required to understand boundary layer transition and interference heating. Additional trajectory analysis and flight data are also required to reduce performance margins and conservatism.

EXTENDED PERFORMANCE STUDIES - TASK IV

Sensitivity and trade studies were performed on the HTO vehicle system selected by the Government in Task II to define the impact of the focused programs established in Task III on the vehicle characteristics and mission performance. Using these results, the characteristics and performance of the systems offering the optimum potential for resource investment were identified. Critical and high yield technology items which have been identified and included in this section are the areas of technology which should be vigorously pursued.

Performance Sensitivity

The use of partial derivatives to obtain sensitivity effects can often result in misleading trends if the assumptions are not adhered to. In place of these partials, the total effect on resizing the vehicle to maintain fixed payload due to changes in inert weight and specific impulse were determined by a mini-computer performance program. The program was calibrated against results obtained from AS 2530 detailed trajectories. This approach permitted non-linear effects of the sensitivity variables to be evaluated as shown previously in Figures 126 and 127. For the original baseline, the change in GLOW with a change in inert weight ratio increases rapidly as the inert weight increases beyond 907.2 kg (2000 lb), remembering that the vehicle is resized to hold payload fixed. This increase in sensitivity can be overcome by changing the baseline configuration from 3 to 4 engines, or else permitting the thrust level to grow for the three engine vehicle in order to keep the thrust loading at 0.77. This same type of non-linear sensitivity occurs with changes in specific impulse of the engine and launch velocity as shown on Figure 133. Launch velocity sensitivities include necessary change in wing area for initial lift-off and buffet limits. In selecting the final baseline vehicle configuration for the extended performance, these sensitivity trade study results were considered. These sensitivity values were also used as inputs to the system cost and figure of merit analysis.

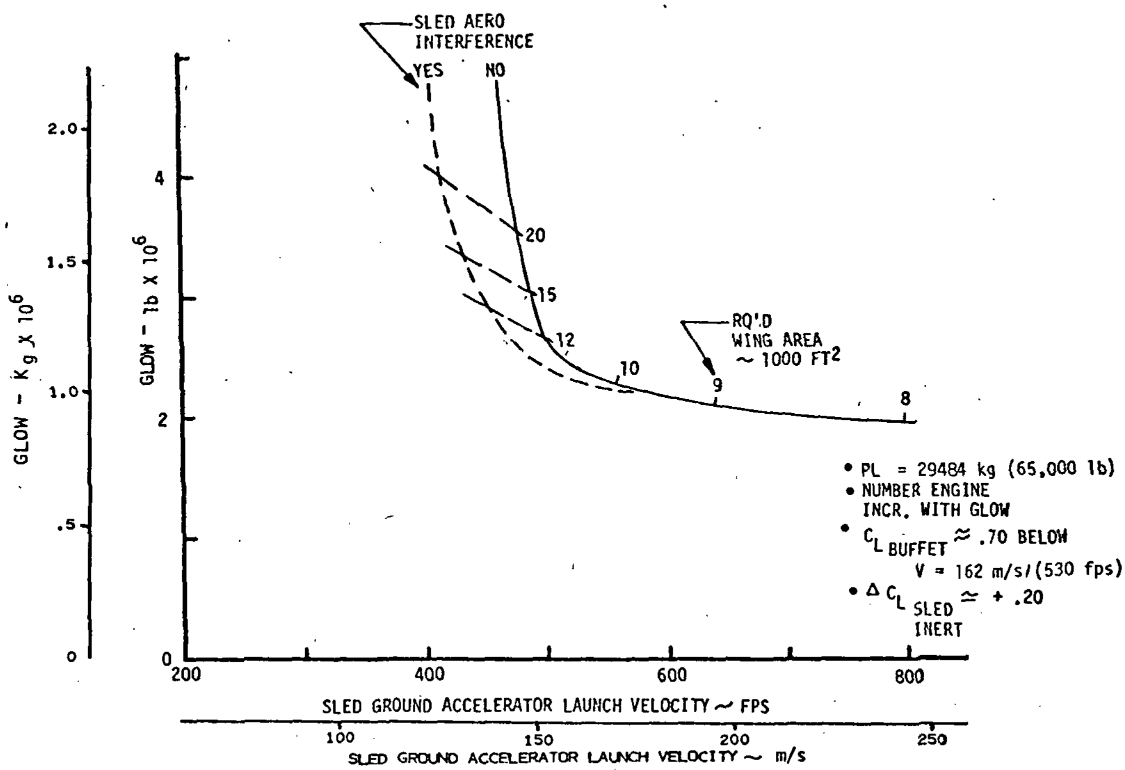


Figure 133 SSTO Launch Velocity Sensitivity

Technical Application

Figure 134 summarizes the technology areas which are recommended for application to the Task II vehicle. A combination of normal technology growth (avionics 3.45×10^7 Pa (5000 psi) hydraulics and 2.8% landing gear as examples) and focused technology growth (zero NPSH pumps, Rene'41 honeycomb, titanium honeycomb, and the SSM 2-position nozzle as examples) yield a vehicle GLOW of 997920 kg (2.2 million lb.). Engine trades discussed under vehicle performance reduce vehicle GLOW over 45360 kg (100,000 lb). The inert weight decrease associated with technology programs recommended is 4135 kg (9,248 lb). This provides an additional reduction to GLOW of nearly 90720 kg (200,000 lb). Shown on the figure as additional technology programs which could provide additional benefits but not recommended for incorporation into the final extended performance vehicle design, are triple point cryogens, all moveable tail, increased chamber pressure, linear engine, etc.

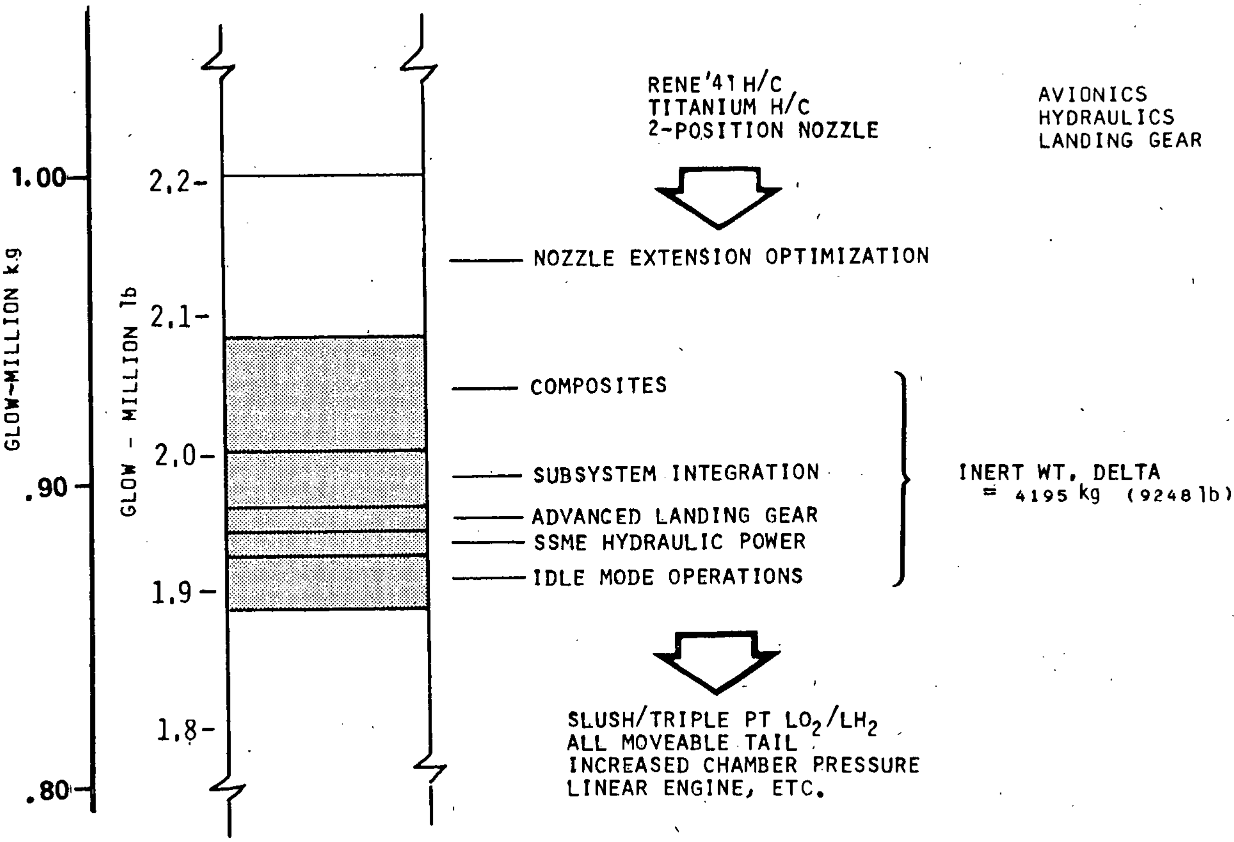


Figure 134 Advanced Technology Program Impact

Extended Performance

Vehicle Performance. The trade and sensitivity studies given in previous sections provided a major input for selecting the final updated baseline vehicle configuration. The 2-position nozzle expansion ratios were changed to 53/190 with three engines at a vacuum thrust level of 3.09×10^6N (695,000 lb) per engine for the extended position. Also, high figure of merit subsystems technology growth accounted for an estimated inert weight reduction of 4195 kg (9,248 lb.). The usable propellant was revised to meet the requirements of placing a 29484 kg (65,000 lb) payload in a low earth orbit with an east launch from ETR. This resulted in a sea level thrust loading of .93 and an injected weight of 127,915 kg (283,000 lb). The ascent trajectory characteristics are similar to those presented previously. The entry trajectory characteristics are similar to those previously shown in Figure 74 except for minor differences in entry weight.

Vehicle Configuration. Figure 135 lists the characteristics of the extended performance vehicle as a result of incorporating the recommended technology developments discussed in Task III. The overall vehicle GLOW is reduced from 997,920 kg (2.2 million lb) to 861,840 kg (1.9 million lb). Body length is shortened to 59m (194 ft.) from 62.8m (206 ft.) and wing span is decreased from 42.7m (140 ft.) to 39.9m (131 ft) wing area (reference) is reduced from 882.5m^2 (9500 ft^2) to 789.6m^2 (8500 ft^2).

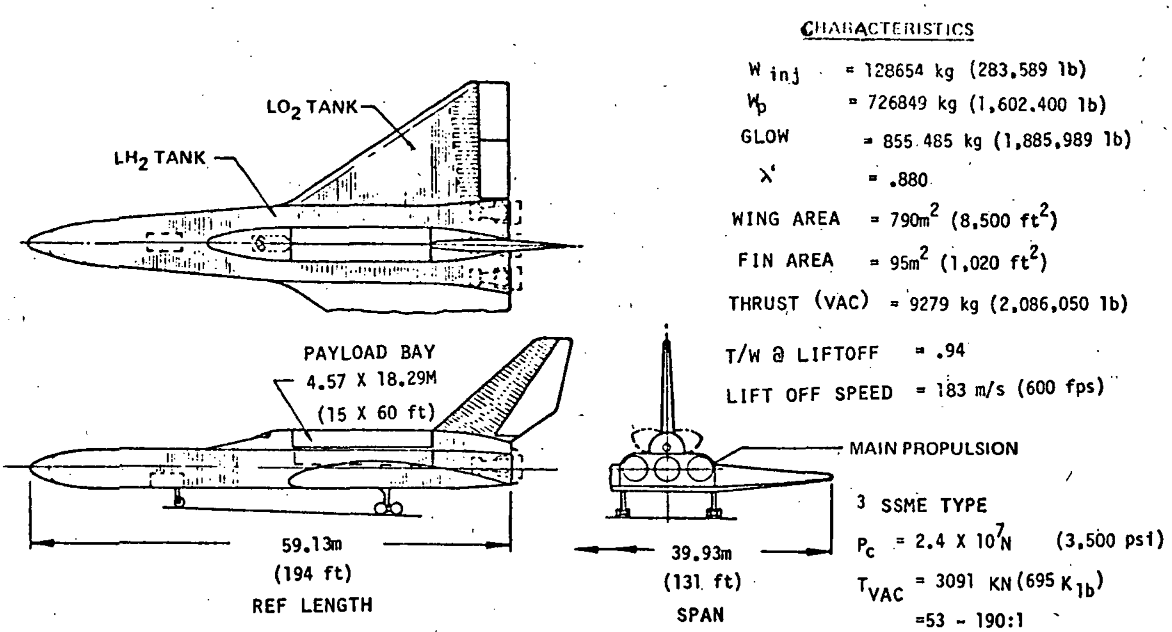


Figure 135 Extended Performance Vehicle Configuration

Vehicle System Cost. Resultant vehicle system costs are shown on Table 41 both for 1976 dollars and 10% discounted dollars. Overall life cycle costs (1976 dollars) are reduced by 652 million dollars to 7510 million. The projected cost per flight is 1.26 million dollars or a transportation cost of slightly under 44 dollars per kg (20 dollars per lb) based on full payload load factor.

DOLLARS IN MILLIONS

	1976 DOLLARS	10% DISCOUNT
DDT&E	3,030	533
PRODUCTION	2,195	386
OPERATIONS	2,285	401
TOTAL	7,510	1,320
COST PER FLIGHT	1.26	
TRANSPORTATION COST (\$/kg)(\$/lb)	(8.79)(19.39)	

Table 41 Extended Performance Vehicle Cost Summary

STUDY CONCLUSIONS AND RECOMMENDATIONS

Study Conclusions

This study demonstrates that major technology advances in the areas of the high pressure LO_2/LH_2 rocket engines, titanium and superalloy airframe systems and operational concepts now provide a sound basis for projecting development of transportation systems capable of operating single stage to space orbital conditions and then return to earth, completing the flight with an aerodynamically controlled horizontal landing. Furthermore this type of earth-to-space operation is achievable without major perturbation of basic technology.

However systematic advances in selected technologies will permit development of smaller take-off-weight transportation systems which should eventually result in lower operating costs. The very significant reduction in operating costs and improved operational flexibility of the Single-Stage-to-Orbit system over staged systems warrants initiation of technology development activities with the objective of providing the basis for development of the National Space Transportation systems for the period of 1990 and on.

Of the three systems (concepts) studied, the Horizontal Take-Off concept is most closely aligned with conventional air transportation. This system appears to offer the operational characteristics that are desired in an Advanced Space Transportation System. Key characteristics are its lower operating cost, ground handling features (i.e. kept on landing gear throughout ground operation), and rapid turnaround.

Therefore, it is recommended that major emphasis be placed in development of the technologies and supporting system studies associated with the Horizontal Take-Off/Horizontal Landing Low Wing Loaded Earth to Orbit Transportation System. Primary emphasis should be in development of the (1) Rene'41 Honeycomb, Titanium Honeycomb Airframe Systems (2) Two-position nozzle for the main ascent engine and (3) Low pressure NPSP cryogenic pumps.

Several technology development programs are universally applicable to several transportation concepts and should be initiated. The all-metallic, completely reusable thermal structural concept proposed by Boeing has direct application on either a HTO or VTO type launch. In addition, this approach can be applied to selected portions of the Space Shuttle (body flap), hypersonic research aircraft, commercial aircraft engines and other proposed space transportation systems. The same is true for the SSME 2-position nozzle program, advanced composites and key subsystem elements previously discussed.

A complete understanding and evaluation of the design impact of operational costs requires additional study effort. It is felt that the ultimate successor to the Space Shuttle must operate in a transportation mode approaching commercial aircraft. To minimize turnaround/launch operations costs of future programs, it is apparent that the SSTO vehicle should be designed for processing from recovery through the next succeeding launch with a minimum of vehicle-to-ground interfaces, ground operations and ground processing time. Operational costs must be driven down to where fuel costs dominate the costs per flight element.

In summary, the SSTO type Program appears to be a viable candidate for the low cost transportation system required for future earth orbital operations. However, the sensitivity of this type of concept requires proceeding with conceptual design, analysis and technology programs in systematic and logical increments taking full advantage of the time available between now and a future program go-ahead.

Study Recommendations

Follow-on effort recommended as a result of this study should concentrate on the following areas.

New Structural Development Program. The Rene'41 honeycomb development program should be pursued. The proposed program as discussed in Task III has been initiated in the area of braze alloy development and allowables definition. Follow-on effort would fund this program through large vehicle section fabrication and test. Ultimate capability would provide low density, insulative structure with -21K $(-423^{\circ}F)$ to 1144K $(1600^{\circ}F)$ operational capability.

Existing Structural Development Program. The aluminum brazed titanium honeycomb program developed during and after the SST program should be advanced. The proposed program (similar and in conjunction with the Rene'41 development program) would provide a low density/insulative structure with 21K $(-423^{\circ}F)$ to 811K $(1000^{\circ}F)$ operational capability.

The existing programs working with the application of metal matrix composites should be expanded to cover operational environments from cryogenic temperatures up to 811K $(1000^{\circ}F)$.

Existing Propulsion Development Program. The SSME program as forecast will provide a significant step in propulsion system performance. Further advances and modifications to this engine are possible and should be initiated. They include the 2-position nozzle, the zero NPSH pump, addition of idle mode, addition of an accessory pad for hydraulic power supply, further weight reduction and operational maintenance cost reduction programs.

New/Existing Subsystem Programs. Several subsystem programs should be pursued which offer potential for weight reductions as well as operational costs. The primary subsystem development program should address an integrated cryogenics auxilliary propulsion system. Reduced weight resulting from higher I_{SP}'s and reduced maintenance costs resulting from the less severe fluid environment indicate the value of R&D funding.

Existing Research Programs. For the reasons given in the study conclusions, a most pressing need is for heat transfer data obtained in the actual reentry environment. The Space Shuttle orbiter provides a unique opportunity to obtain such data. When properly instrumented, the orbiter would provide aerodynamic heating data of immense value to future spacecraft that could not be obtained in any other way.

Studies should be conducted to determine suitable instrumentation and techniques for obtaining accurate heat transfer data during reentry. Possible methods include infra-red measurements (upper surfaces only), thermocouples imbedded in the insulation, and/or high temperature heat transfer gages bonded to the skin but with the sensing surface flush with the insulation surface. It is essential that the measurements be sensitive enough to detect the onset of transition, as well as provide an accurate determination of the heating levels. Bond line temperature measurements will not provide this information.

In addition, more trajectory analysis and dispersion analysis effort is recommended with the potential of reducing margins and typical conservatism.

Study Programs. A linear engine program study which uses the performance data developed during a recently funded NASA Lewis contract is recommended. This study would integrate the linear engine with the airframe to actually delineate the structural and vehicle performance benefits.

A study of the manufacturing transfer and gaging aspects of triple point hydrogen and oxygen is warranted.

Additional configuration and system studies are recommended in the areas of body, wing, and tail shaping (total vehicle structural integration), and control configured design. Both of these areas are somewhat configuration dependent but have a broad application to all SSTO type vehicles.

☆U.S. GOVERNMENT PRINTING OFFICE:1978-735-078/14

1. Report No.
NASA CR-2879

2. Government Accession No.

3. Recipient's Catalog No.

4. Title and Subtitle
Technology Requirements for Advanced Earth Orbital Transportation Systems
Volume 2: Summary Report

5. Report Date
July 1978

6. Performing Organization Code

7. Author(s)
A. K. Hepler and E. L. Bangsund

8. Performing Organization Report No.
D180-19168-4

10. Work Unit No.

9. Performing Organization Name and Address
Boeing Aerospace Company
P. O. Box 3999
Seattle, Washington 98124

11. Contract or Grant No.
NAS 1-13944

13. Type of Report and Period Covered
Contractor Report

12. Sponsoring Agency Name and Address
National Aeronautics and Space Administration
Washington, D. C. 20546

14. Sponsoring Agency Code

15. Supplementary Notes
Langley Technical Monitor: Charles H. Eldred
Volume 2 of Final Report

16. Abstract

The fundamental objective of this study was to identify those areas of technology associated with future earth-to-orbit transportation systems which are either critical to the development of such systems or which offer a significant cost and performance advantage as a result of their development. Additional objectives were to determine the most efficient operational mode for such systems and to define performance potential as a function of technology growth. The intent was to utilize vehicle system studies as a means of identifying critical and high yield technology areas upon which to base the planning for and the development of advanced technology programs.

Normal technology requirements applicable to Single Stage to Orbit (SSTO) systems were projected to the 1985 time period in Task 1. These technology projections were then incorporated in a vehicle design analysis of three different operational concepts resulting in four configurations of a Single Stage to Orbit system in Task 2. The resultant performance, weights and costs of each concept were then compared and a system concept selected. A "figure of merit" was developed for advanced technology programs based on a cost/performance basis in Task 3. The selected advanced technology programs were then used to reassess the Task 2 vehicle to determine the impact on performance, weight and cost in Task 4.

Based on study results, recommendations are provided in the two above mentioned categories of technology areas associated with future earth orbit transportation systems. The recommendations address advanced space transportation system design considerations, both hardware and software technology program requirements and suggested areas of future effort. This work is covered in detail in Volume 2: Summary Report. Results of an additional investigation dealing with dual mode propulsion are reflected separately in Volume 3: Summary Report - Dual Mode Propulsion.

17. Key Words (Suggested by Author(s))
Single Stage to Orbit Vehicles
Space Vehicles
Hypersonic Flight
Advanced Technology

18. Distribution Statement
Unclassified - Unlimited

Subject Category 16

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	197	$9.00

* For sale by the National Technical Information Service, Springfield, Virginia 22161